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CMH-17-6
Volume 6 of 6
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COMPOSITE MATERIALS HANDBOOK

VOLUME 6. STRUCTURAL SANDWICH COMPOSITES

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FOREWORD

The Composite Materials Handbook, CMH-17, provides information and guidance necessary to design and fabricate structural components from composite materials. Its primary purposes are a) the standardization of engineering data development methodologies related to testing, data reduction, and data reporting of property data for current and emerging composite materials, b) guidance on material and process specifications and procedures for utilization of the material data presented in the handbook, and c) methodologies for the design, analysis, certification, manufacture, and field support of composite structures. In support of these objectives, the handbook includes composite materials properties that meet specific data requirements. The Handbook therefore constitutes an overview of the field of composites technology and engineering, an area that is advancing and changing rapidly. As a result, the document will be continually revised as sections are added or modified to reflect advances in the state-of-the-art.

CMH-17 Mission

The Composite Materials Handbook organization creates, publishes and maintains proven, reliable engineering information and standards, subjected to thorough technical review, to support the development and use of composite materials and structures.

CMH-17 Vision

The Composite Materials Handbook will be the authoritative worldwide focal point for technical information on composite materials and structures.

Goals and Objectives to Support CMH-17 Mission

- To periodically meet with experts from the field to discuss critical technical issues for composite structural applications, with an emphasis on increasing overall product efficiency, quality and safety.
- To provide comprehensive, practical engineering guidance that has proven reliable for the design, fabrication, characterization, test and maintenance of composite materials and structures.
- To provide reliable data, linked to control of processes and raw materials, thereby being a comprehensive source of material property basis values and design information that can be shared within the industry.
- To provide a resource for composite material and structure education with examples, applications and references to supporting engineering work.
- To establish guidelines for use of information in the Handbook, identifying the limitations of the data and methods.
- To provide guidance on references to proven standards and engineering practices.
- To provide for periodic updates to maintain the all-inclusive nature of the information.
- To provide information in formats best-suited for user needs.
- To serve the needs of the international composites community through meetings and dialogue between member industries, which use teamwork and the diverse member engineering skills to provide information for the handbook.

Notes

1. Every effort has been made to reflect the latest information on polymer (organic), metal, and ceramic composites. The handbook is continually reviewed and revised to ensure it is complete and current.

2. CMH-17 provides guidelines and material properties for polymer (organic), metal, and ceramic matrix composite materials. The first three volumes of this handbook currently focus on, but are not limited to, polymeric composites intended for aircraft and aerospace vehicles. Metal matrix composites (MMC), ceramic matrix composites (CMC) including carbon-carbon composites (C-C), and sandwich composites are covered in Volumes 4, 5, and 6, respectively.

3. The information contained in this handbook was obtained from materials producers, industry companies and experts, reports on Government sponsored research, the open literature, and by contract with research laboratories and those who participate in the CMH-17 coordination activity. The information in this handbook has undergone vigorous technical review and was subject to membership vote.

4. Beneficial comments (recommendations, additions, deletions) and any pertinent data which may be of use in improving this document should be addressed to: CMH-17 Secretariat, Materials Sciences Corporation, 135 Rock Road, Horsham, PA 19044, by letter or email, handbook@materials-sciences.com.

ACKNOWLEDGEMENT

Volunteer committee members from government, industry, and academia develop, coordinate and review all the information provided in this handbook. The time and effort of the volunteers and the support of their respective departments, companies, and universities make it possible to insure the handbook reflects completeness, accuracy, and state-of-the-art composite technology.

Support necessary for the development and maintenance of the Composite Materials Handbook (CMH-17) are provided by the handbook Secretariat, Materials Sciences Corporation. The primary source of funding for the current Secretariat contract is the Federal Aviation Administration.

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CHAPTER 1 GENERAL INFORMATION

1.1 INTRODUCTION TO THE HANDBOOK

Standardized, statistically-based material property data are essential to the development of composite structures; such data are needed by material suppliers, design engineering personnel, manufacturing organizations, and structure end-users alike. In addition, reliable, proven design and analysis methods are essential to the efficient development and application of composite structures. This handbook is intended to provide these through standardization of:

1. Methods used to develop, analyze, and publish property data for composite materials.
2. Statistically-based, material property datasets for composite materials.
3. General procedures for designing, analyzing, testing, and supporting composite structures that utilize the property data published in this handbook.

In many cases, this standardization is intended to address the requirements of regulatory agencies, while providing efficient engineering practices for developing structures that meet the needs of customer organizations.

Composites is an evolving and growing technical field, and the Handbook Coordinating Committee is continuously working to incorporate new information and new material properties data as it becomes available and is proven acceptable. While the source and context for much of this information has come from experience with aerospace applications, all industries utilizing composite materials and structures, whether commercial or military, will find the handbook useful. This latest revision includes information related to broader applications from non-aerospace industries, and incorporation of non-aerospace data will increase as development of the handbook continues.

Composite Materials Handbook-17 (CMH-17) has been developed and is maintained as a joint effort of the Department of Defense and the Federal Aviation Administration, with considerable participation and input from industry, academia, and other government agencies. Although initial structural applications of composites tended to be military, recent development trends have seen increasing use of these materials in commercial applications. In part because of these trends, the formal administration of the handbook passed from the Department of Defense to the Federal Aviation Administration in 2006 and the handbook title was changed from Military Handbook-17 to Composite Materials Handbook-17. The organization of the Coordinating Committee and the purpose of the handbook did not change.

1.2 OVERVIEW OF HANDBOOK CONTENT

Composite Materials Handbook-17 is composed of a series of six volumes.

Volume 1: Polymer Matrix Composites - Guidelines for Characterization of Structural Materials

Volume 1 contains guidelines for determining the properties of polymer matrix composite material systems and their constituents, as well as the properties of generic structural elements, including test planning, test matrices, sampling, conditioning, test procedure selection, data reporting, data reduction, statistical analysis, and other related topics. Special attention is given to the statistical treatment and analysis of data. Volume 1 contains guidelines for general development of material characterization data as well as specific requirements for publication of material data in CMH-17.

Volume 2: Polymer Matrix Composites - Materials Properties

Volume 2 contains statistically-based data for polymer matrix composites that meets specific CMH-17 population sampling and data documentation requirements, covering material systems of general interest.

As of the publication of Revision G, data published in Volume 2 are under the jurisdiction of the Data Review Working Group and are approved by the overall CMH-17 Coordinating Committee. New material systems will be included and additional material data for existing systems will be added as data becomes available and are approved. Selected historical data from previous versions of the handbook that do not meet current data sampling, test methodology, or documentation requirements, but that still are of potential interest to industry are also included in this Volume.

Volume 3: Polymer Matrix Composites - Materials Usage, Design, and Analysis

Volume 3 provides methodologies and lessons learned for the design, analysis, manufacture, and field support of fiber-reinforced, polymeric-matrix composite structures. It also provides guidance on material and process specifications and procedures for utilization of the data presented in Volume 2. The information provided is consistent with the guidance provided in Volume 1, and is an extensive compilation of the current knowledge and experiences of the engineers and scientists who are active in composites from industry, government, and academia.

Volume 4: Metal Matrix Composites

Volume 4 publishes properties on metal matrix composite material systems for which data meeting the specific requirements of the handbook are available. In addition, it provides selected guidance on other technical topics related to this class of composites, including material selection, material specification, processing, characterization testing, data reduction, design, analysis, quality control, and repair of typical metal matrix composite materials.

Volume 5: Ceramic Matrix Composites

Volume 5 publishes properties on ceramic matrix composite material systems for which data meeting the specific requirements of the handbook are available. In addition, it provides selected guidance on other technical topics related to this class of composites, including material selection, material specification, processing, characterization testing, data reduction, design, analysis, quality control, and repair of typical ceramic matrix composite materials.

Volume 6: Structural Sandwich Composites

Volume 6 is an update to the cancelled Military Handbook 23 (Reference 1.2), which was prepared for use in the design of structural sandwich polymer composites, primarily for flight vehicles. The information presented includes test methods, material properties, design and analysis techniques, fabrication methods, quality control and inspection procedures, and repair techniques for sandwich structures in both military and commercial vehicles.

1.3 INTRODUCTION

The Sandwich Structures Volume of Composite Materials Handbook-17 (CMH-17), Volume 6, contains seven chapters. Chapter 1, General Information, provides the objective, background, introduction and notation utilized in sandwich structures. Chapter 2, Guidelines for Property Testing discusses property testing of sandwich constituent materials; core materials, core-to-face sheet bonds, and face sheets; as well as sandwich panels, inserts and fasteners, and other sandwich details such as ramps and close-outs. Chapter 3, Material Data, contains core, face sheet, adhesive, and self-adhesive face sheet properties. Chapter 4, Design and Analysis, provides structural design, sizing and analysis methods for critical failure modes of sandwich structures. Chapter 5, Fabrication, discusses sandwich materials and processes, and lessons learned. Chapter 6, Quality Control, discusses in-process and end-article inspection, material properties verification, and process controls as applicable to sandwich structures. Chapter 7, Supportability, discusses design practices to improve damage tolerance and repair aspects of sandwich structures. Discussions of basic design principles and fundamental formulas are included in all sections as appropriate.

1.4 NOMENCLATURE AND DEFINITIONS

The following notation is used throughout this volume. Additionally, portions of the volume devoted to a particular component define the symbols used for the first time in that portion. An occasional symbol not in general use will appear in specific areas and not be included in this notation. Figure 1.4 shows notation for sandwich construction.

Units of dimensions, forces, stresses, constants, and other quantities are not specified unless they are employed in formulas wherein numerical coefficients are not non-dimensional. In applying formulas for which units are not specified, correct results will not be obtained unless units are consistent – for example: If thicknesses are given in inches and forces in pounds, then the length and width of a panel must be in inches (not feet) to give stresses in pounds per square inch.

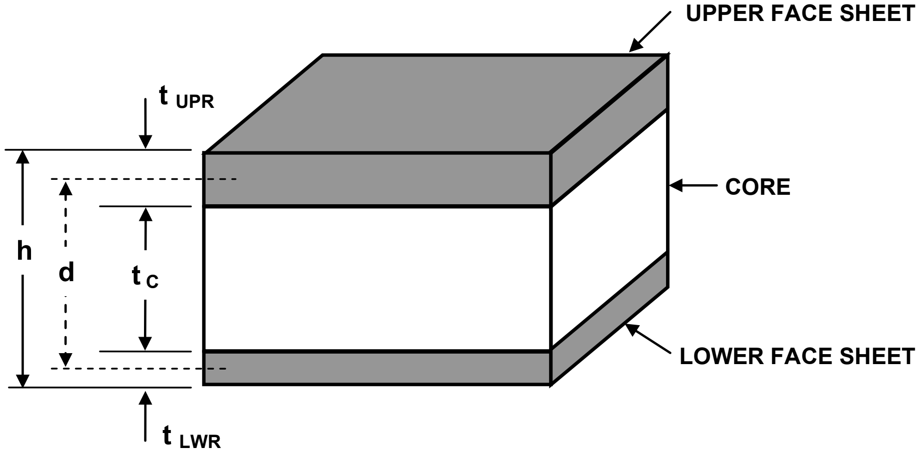

FIGURE 1.4 *Sandwich construction notation.*

1.4.1 Loads, geometry, and material properties

The following variables are used throughout this volume to represent the loads applied to a flat sandwich panel. Figure 1.4.1 gives a pictorial representation of the general load state for a sandwich panel.

N_x, N_y, N_{xy}	- Distributed in-plane panel forces
M_x, M_y, M_{xy}	- Applied edge moments
Q_x, Q_y	- Out-of-plane edge reactions
q	- Uniform pressure

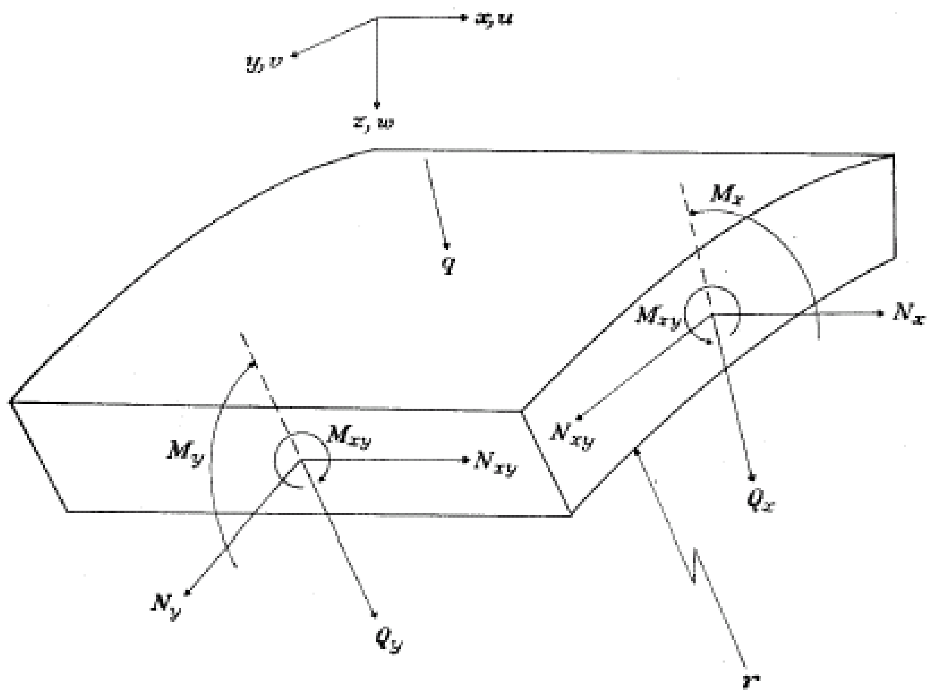

FIGURE 1.4.1 *Generalized load state for a sandwich panel.*

The following general symbols and abbreviations are considered standard for use in this volume. Where exceptions are made, they are noted in the text and tables.

A_c - core solidity, $A_c = {}^{w_c}/_{w_o}$

A_{ij} - extensional and shear stiffnesses

a - length of panel edge, parallel to the loading direction (mm, in)

B_{ij} -extensional-bending coupling stiffnesses

b - (1) length of panel edge, transverse to the loading direction (mm, in)
- (2) unsupported width of face sheet element (mm, in)
- (3) width of beam (mm, in)
- (4) width of face sheet in corrugated core (mm, in)

D - beam bending stiffness or twisting stiffness (N-m,lbf-in)

D_c - bending stiffness of the core (N-m,lbf-in)

D_f - bending stiffness of the face sheets about their individual neutral axes (N-m,lbf-in)

D_{ij} - bending and torsional stiffnesses

D_o - bending stiffness of the face sheets about the middle axis of the sandwich beam (N-m,lbf-in)

d - (1) distance between the face sheet midplanes at any point in the sandwich panel (mm, in)
- (2) mathematical operator denoting differential

E - modulus of elasticity in tension, average ratio of stress to strain for stress below proportional limit (GPa, Msi); for orthotropic face sheets: $E = [E_a E_b]^{1/2}$

E' - (1) effective modulus of elasticity; for orthotropic face sheets $E' = [E_a' E_b']^{1/2}$
- (2) core modulus in Z direction (GPa,Msi)

E_{cx} - stiffness of the core, in the direction of the beam long axis (GPa, Msi)

E_x, E_y- face sheet elastic moduli parallel to the X and Y directions, respectively (GPa, Msi)

e - distance from the midplane of the lower face sheet to the neutral axis of the beam (mm, in)

F - (1) allowable stress (MPa,ksi)
- (2) Fahrenheit

f - (1) internal (or calculated) stress (MPa,ksi)
- (2) stress applied to the gross flawed section (MPa,ksi)
- (3) creep stress (MPa,ksi)
f_{Xflat}, f_{Yflat} - flatwise tension or compression stresses at the ramp radii.
F_{12} - geometric view factor between sandwich facings
G - modulus of rigidity (shear modulus) (GPa,Msi); with subscripts G_{ab} is the modulus of rigidity associated with shear distortion of the ab plane, with subscript G_c is the modulus of the core
G' - effective modulus of rigidity
Gxy - face sheet shear modulus in the XY plane (GPa, Msi)
Gzx, Gzy - core shear moduli in the XZ and YZ planes, respectively (GPa, Msi)
GPa - gigapascal(s)
H/C - honeycomb (sandwich)
H - extensional stiffness
h - total thickness or depth of the sandwich (mm, in)
in. - inch(es)
K - (1) a coefficient
- (2) Kelvin
K_e - effective conductivity
K_o - conductivity of core ribbon material
k - conductivity (Btu in. /hr ft2 F°)
L - (1) length (mm, in)
- (2) core ribbon axis direction
lb - pound(s)
M - applied bending moment or couple (N-m,in-lbf)
m - (1) half width of corrugation
- (2) number of half waves
- (3) cos(θ) when used in coordinate transformations
MPa - megapascal(s)
MS - military standard
M.S. - margin of safety
N - (1) design load per unit length of panel edge
- (2) Newton(s)
- (3) exponent in core shear interaction criterion
n - (1) number of half waves
- (2) sin(θ) when used in coordinate transformations
NA - neutral axis
P - applied load (N,lbf)
pcf - pounds per cubic foot
psi - pounds per square inch
Q - face sheet dissimilarity index $Q = 1/[1+(E_{LWR}'t_{LWR}/E_{UPR}'t_{UPR})]$
q - (1) normal pressure (Pa, psi)
- (2) intensity of distributed load
R - (1) ramp radius (mm, in)
- (2) the ratio of the applied stress or load under combined loading to the buckling stress or load under separate loading
- (3) radius of curvature (mm, in)
- (4) ratio of lower to upper face sheet stiffness and thickness, $E_{LWR}\,t_{LWR}$ / $E_{UPR}\,t_{UPR}$
r - radius (mm,in)
S - (1) shear bending stiffness
- (2) shear load normal to surface of panel
s - core cell size (diameter of inscribed circle) (mm, in)
T - (1) temperature (°C,°F)
- (2) applied torsional moment (N-m,in-lbf)
- (3) core axis direction (through the thickness of the sandwich panel)

T_m - mean temperature
t - thickness (mm,in)
t_{UPR}, t_{LWR} - thickness of the upper and lower face sheets, respectively (mm, in)
tc - core depth (mm, in)
te - edgeband thickness (mm, in)
U - transverse sandwich shear stiffness
u - deflection in direction of x axis (mm, in)
V - (1) volume (mm^3,in^3)
- (2) shear force (N,lbf)
- (3) parameter relating shear and bending stiffness
V_2 - special parameter relating shear and bending stiffness for sandwich with corrugated core
V_t - parameter relating shear and bending stiffness for sandwich strip with triangular or trapezoidal section
v - deflection in direction of y axis (mm, in)
W - (1) sandwich weight per unit area (N,lbf)
- (2) core axis direction perpendicular to the ribbon axis L
- (3) special parameter relating shear and bending stiffness for sandwich
W_t - parameter relating shear and bending stiffness for sandwich strip with triangular or trapezoidal section and corrugated core
w - (1) transverse deflection (mm, in)
- (2) density
x - distance along a coordinate axis
y - (1) deflection (due to bending) of elastic curve of a beam (mm,in)
- (2) distance from neutral axis to given point
- (3) distance along a coordinate axis, perpendicular to x-axis
Z - parameter for trapezoidal sandwich strips: $Z = (b/h) \tan \alpha$
z - distance along a coordinate axis, perpendicular to x-y plane
α - (1) $[E_b'/E_a']^{1/2}$
- (2) angle giving the rise of a trapezoidal or triangular sandwich strip
α_1, α_2 - scale factors relating curvilinear and Cartesian coordinates
β - $\alpha\nu_{ab}+2\gamma$
δ - elongation or deflection (mm,in)
ε - (1) compression or extension strain
- (2) emissivity
- (3) mid-plane strains
κ - curvatures
λ - (1) load factor
- (2) one minus the product of two Poisson's ratios ($\lambda = 1 - \nu_{ab}\nu_{ba}$)
η_X, η_Y - (1) Percent rotational fixity for panel edges normal to the X and Y directions, respectively
= 1.0 for fully fixed
= 0.0 for simply supported
- (2) Plasticity coefficient
- (3) convective heat transfer coefficient
ρ - radius of gyration
ν - Poisson's ratio
ν_{XY}, ν_{YX} - face sheet Poisson's ratios. The term, ν_{XY}, is defined as the absolute ratio of strain in the Y direction to strain in the X direction when load is applied uniaxially in the X direction ($E_X \nu_{YX} = E_Y \nu_{XY}$)
ν_{ab}, ν_{ba} - face sheet Poisson's ratios, in directions aligned with panel sides
γ - shear strain; elastic property parameter $\gamma = \lambda Gba'/[Ea'Eb']^{1/2}$
Σ - total, summation
σ - Stefan-Boltzmann constant

σ_{ij}, τ_{ij} - stress in j direction on surface whose outer normal is in i direction (i, j = 1, 2, 3 or x, y, z) (MPa,ksi)
τ - transverse shear stress (MPa, ksi)
T - applied shear stress (MPa,ksi)
ω - angular velocity (radians/s)
∞ - infinity
θ - (1) angle of twist
- (2) angle between material axes and loading axes
- (3) local angle between bag side and tool side in the ramp region
- (4) angle between corrugation element and face sheet in corrugated core
Ψ - (1) deflection or stress due to edgewise load combined with normal load, (ψ_0 is deflection or stress due to normal load only)
- (2) rotation about the midplane of a curved shell
Ψ_x, Ψ_y - rotations about the midplane
ξ_i - shell reference coordinates
ϕ, ζ - parameters for buckling of flat panels supported by beams
ϕ – ramp angle
$\phi_m(x)$, $\phi_n(y)$ - displacement functions

1.4.1.1 Subscripts

The following subscript notations are considered standard in this volume.

UPR - upper face sheet of a sandwich
LWR - lower face sheet of a sandwich
1, 2 - principal material directions
45f - elastic flexural in 45 degree direction
a - parallel to direction of loading
B - (1) bending
- (2) bond between face sheet and core
b - (1) bag side
- (2) perpendicular to direction of loading
- (3) buckling
C - compression
c - (1) core
- (2) crush
cr - critical
dimple - intracell buckling
e - Euler buckling
F - face sheets when applied to buckling coefficients
f - (1) face sheet
- (2) flexural
i - ply number
L - ribbon direction
M - denotes behavior of sandwich with thin face sheets (when applied to buckling coefficients)
max - maximum
min - minimum
n - (1) n^{th} (last) position in a sequence
- (2) normal
0 - denotes V = 0
o - denotes honeycomb core ribbon or core corrugation sheet
r - reduced
s - shear (when applied to stress) or secant (when applied to moduli)
sc - core shear
T - transverse

t - tangent (when applied to moduli)
u - ultimate (when applied to stress)
W - (1) wrinkling
- (2) transverse
x, y, z - general coordinate system
φ - wave angle
Σ - total, or summation
o - initial or reference datum

1.4.1.2 Superscripts

The following superscript notations are considered standard in this volume.

c - core
max - maximum
s - shear
scr - shear buckling
sec - secant (modulus)
so - offset shear
t - tension
tan - tangent (modulus)
u - ultimate
y - yield
' - (1) secondary (modulus), or denotes properties of honeycomb core when used with subscript c
- (2) effective

1.4.1.3 Assumptions and definitions

The analysis of sandwich panels is made easier when a few assumptions are made. The following statements are considered true for every sandwich analysis method, unless otherwise noted.

- The in-plane stiffness of the core is negligible compared to that of the face sheets.

$$E_{Xcore} = E_{Ycore} = G_{XYcore} = 0$$

- The point at which in-plane load is applied to the edges of the sandwich is considered to be halfway between the face sheet centroids (midplanes).
- Normal (out-of-plane) shear forces are carried exclusively by the core, and are distributed evenly through the core thickness.
- The panels in this chapter are rectangular, and the load coordinate system (X, Y, Z) is coincident with the core material system (L, W, T). Exceptions are the circular panels in Section 4.7.2.2, and the core shear interaction criterion in Section 4.6.2.

- The *midplane* of a sandwich element is located halfway between the exterior surfaces of the two face sheets:

$$y_{midplane} = \frac{d}{2}$$

where d is the total thickness of the sandwich.

- The *centroid* of an object is located at its geometric center. The centroid of the whole sandwich element is considered to be coincident with the centroid of the two face sheets by themselves (neglecting the core).

- When the face sheets have unequal thicknesses, the centroid does not coincide with the mid-plane of the sandwich.

$$y_{centroid} = \frac{t_{UPR} \cdot y_{UPR} + t_{LWR} \cdot y_{LWR}}{t_{UPR} + t_{LWR}}$$

  where t_{UPR}, t_{LWR}, and y_{UPR}, y_{LWR} are the thickness and centroid position of the upper and lower face sheets, respectively.

- The face sheets are thin, and the moments of inertia about their own centroids are negligible.

$$\left(t_{UPR}\right)^3 = \left(t_{LWR}\right)^3 = 0$$

- The centroid of each face sheet is coincident with its midplane. Therefore, the distance between the face sheet midplanes is considered to be equal to the distance between the face sheet centroids.

- The *neutral axis* of a sandwich element is located at the point where zero strain is induced under the influence of bending loads. For a flat panel, the neutral axis coincides with the centroid, but for a curved panel, the neutral axis does not coincide with the centroid.

The equations in Sections 4.7 through 4.11 and 4.13 are presented for isotropic face sheets. In most cases, an equation is given that applies to a sandwich with face sheets of different materials and/or thicknesses, and then a second equation is given for the simplified case where the two face sheets are the same. In general, the equations can be adapted for orthotropic face sheets, by using $E = [Ea\ Eb]^{1/2}$ in place of elastic modulus, E, and by using $\lambda = 1 - \nu_{ab}\,\nu_{ba}$ in place of $\lambda = 1 - \nu^2$, where Ea and Eb are elastic moduli, and ν_{ab} and ν_{ba} are the in-plane Poisson's ratios, in the directions aligned with the panel sides. In some cases, such as face sheet wrinkling calculations, it is more appropriate to use face sheet bending stiffness in place of elastic modulus, E.

1.4.2 System of units

To comply with Department of Defense Instructive 5000.2, Part 6, Section M, "Use of the Metric System," dated February 23, 1991, the data in CMH-17 are generally presented in both the International System of Units (SI units) and the U. S. Customary (English) system of units. IEEE/ASTM SI 10, American National Standard for Use of the International System of Units (SI): The Modern Metric System provides guidance for the application for SI units which are intended as a basis for worldwide standardization of measurement units (Reference 1.4.2(a)). Further guidelines on the use of the SI system of units and conversion factors are contained in References 1.4.2(b) through (e).

English to SI conversion factors pertinent to CMH-17 data are contained in Table 1.4.2.

TABLE 1.4.2 *English to SI conversion factors.*

To convert from	to Multiply by	
Btu (thermochemical)/in^2-s	watt/meter^2 (W/m^2)	1.634 246 E+06
Btu-in/(s-ft^2-°F)	W/(m K)	5.192 204 E+02
degree Fahrenheit	degree Celsius (°C)	T = (T - 32)/1.8
degree Fahrenheit	kelvin (K)	T = (T + 459.67)/1.8
foot	meter (m)	3.048 000 E-01
ft^2 m^2	9.290 304 E-02	
foot/second	meter/second (m/s)	3.048 000 E-01
ft/s^2	m/s^{2}3.048 000 E-01	
inch	meter (m)	2.540 000 E-02
in.^{2}	meter^2 (m^2)	6.451 600 E-04
in.^{3}	m^{3}1.638 706 E-05	
kilogram-force (kgf)	newton (N)	9.806 650 E+00
kgf/m^2	pascal (Pa)	9.806 650 E+00
kip (1000 lbf)	newton (N)	4.448 222 E+03
ksi (kip/in^2)	MPa6.894 757 E+00	
lbf-in	N-m1.129 848 E-01	
lbf-ft	N-m1.355 818 E+00	
lbf/in^2 (psi)	pascal (Pa)	6.894 757 E+03
lb/in^2	gm/m^2	7.030 696 E+05
lb/in^3	kg/m^3	2.767 990 E+04
Msi (10^6 psi)	GPa6.894 757 E+00	
pound-force (lbf)	newton (N)	4.488 222 E+00
pound-mass (lb avoirdupois)	kilogram (kg)	4.535 924 E-01
torr	pascal (Pa)	1.333 22 E+02

∗ The letter "E" following the conversion factor stands for exponent and the two digits after the letter "E" indicate the power of 10 by which the number is to be multiplied.

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1.2 MIL-HDBK-23A, Military Handbook 23A, Structural Sandwich Composites, Notice 3, June, 1974 (Cancelled by Notice 4, February, 1988).

1.4.2(a) IEEE/ASTM SI 10-02, "American National Standard for Use of the International System of Units (SI): The Modern Metric System," Annual Book of ASTM Standards, Vol. 14.04, American Society for Testing and Materials, West Conshohocken, PA.

1.4.2(b) Brown, James, "Metric Conversion Guide: Engineering Design Handbook", University Press of the Pacific, October, 2004.

1.4.2(c) NIST Special Publication 330, "The International System of Units (SI)," National Institute of Standards and Technology, 2008 edition.

1.4.2(d) NBS Letter Circular LC 1035, "Units and Systems of Weights and Measures, Their Origin, Development, and Present Status," National Bureau of Standards, November 1985.

1.4.2(e) NASA Special Publication 7012, "The International System of Units Physical Constants and Conversion Factors," 1973.

CHAPTER 2 GUIDELINES FOR PROPERTY TESTING

2.1 INTRODUCTION

A sandwich structure consists of face sheets, core, and some means of connecting the two, such as an adhesive or brazing. Face sheet and core properties must be known in order to design the sandwich, and the sandwich is tested to ensure the face sheets are adequately connected to the core. This chapter discusses mechanical properties, environmental effects, test methods, and data reduction and presentation.

The basic core design properties are compressive strength and modulus and shear strength and modulus. Standard test methods for obtaining core properties are given in Section 2.3. Chapter 3 contains additional information on properties of various kinds of core.

The face sheets of a sandwich panel take the bending loads (one face sheet in compression and the other in tension) or in some cases the in-plane shear loads. The major face sheet properties are the compressive, tensile, and shear strengths and moduli. Some ASTM test methods for composite face sheets are given in Section 2.5.

Evaluation of the core-to-face sheet bond, of sandwich panels as a whole, and of inserts and fasteners are covered in Sections 2.4, 2.6, and 2.7, respectively.

The face sheet properties, data reduction, and presentation are in Volume 2, therefore, this chapter will deal primarily with the sandwich core properties. Core properties given in the Chapter 3 tables include density, compressive strength and modulus, shear strength and modulus, and tensile strength. The values in Chapter 3 are from published test values from material suppliers. The property values given are not design allowables, but are intended to illustrate the different types of core.

2.2 DATA REDUCTION AND PRESENTATION

While core properties have been published by individual suppliers, and some core materials are purchased according to industry specifications, a standardized approach to data reduction and presentation has not yet been established for inclusion into this Volume of CMH-17. The near-term approach to address the issue of core data is to provide material property values in Chapter 3, solely for illustration and comparison so that the user can get a feel for what kind of properties to expect from a given material. The data is clearly labeled regarding its source and published value interpretation (i.e., whether the data represents typical or minimum values), and is not to be considered as certifiable for official design qualification (or submittable as “Type Data” for FAA Certification, for example) unless agreed to explicitly by that governing agency.

The ultimate goal of CMH-17 is to establish protocols and standards for inclusion of core data for official design use, including thorough peer review and scrutiny with the statistical rigor that is currently applied to fiber-reinforced polymer composite material data presented in Volume 2.

2.3 EVALUATION OF CORE MATERIALS

2.3.1 Introduction

The three main types of cores used in sandwich constructions are honeycomb, foam, and balsa wood. Each type has its advantages and disadvantages.

2.3.2 Mechanical properties

The mechanical properties of interest for core materials include compressive strength and modulus, tensile strength, and shear strength and modulus. Section 2.3.4 describes the most common test methods.

2.3.3 Environmental effects

The hygrothermal stability of sandwich panels is of special interest to the integrity of solar panels, antenna structures, aircraft, and payload fairings. In addition to aluminum cores, a variety of materials are in use, including alloys such as stainless steel and titanium, KRAFT paper (softwood), KEVLAR® (para-aramid), NOMEX® (meta-aramid), TYVEK® (high density polyethylene), plastic foams, and fiberglass. Bias or straight carbon fabrics with phenolic or polyimide resin represent developmental honeycomb materials. Their properties, such as shear modulus, vary with core density and cell dimensions and directions.

Tables in Chapter 3 give room temperature properties of various honeycomb, foam, and balsa wood cores in English units (SI conversion factors also given). These values were obtained from supplier publications. They are not design allowable values.

The core mechanical properties generally published are the room temperature values. The ASTM Standard Test Methods state the testing should be done at 23 ± 3°C (73 ± 5°F) and a relative humidity of 50 ± 5%. All the cores, including honeycomb, balsa, and foam, will lose some of their strengths and moduli at elevated temperatures and when wet. The amount of degradation depends on the core material. As an example, aluminum honeycomb if submersed in water for 24 hours would not lose any of its strength or stiffness (no time for corrosion), but some non-metallic honeycomb could lose 50 percent or more of its properties. In general, all the core strengths and moduli will be slightly higher when tested cold.

In selecting a core, the maximum service temperature, flammability, moisture pickup, corrosion resistance, impact resistance, and heat transfer all can be important depending on the application. Table 2.3.3 qualitatively summarizes these characteristics for some honeycomb core types.

Figure 2.3.3 gives mechanical property percent retention percentages for some commercially available honeycomb cores when tested at the indicated temperature. Foam and balsa wood cores are likely to exhibit similar characteristics and should be evaluated for their respective design environmental conditions.

TABLE 2.3.3 Honeycomb *core environmental characteristics.*

Attributes	**Aluminum**			**Glass**			**Aramid**		**Carbon**
	5052 – Aerospace Grade	5056 – Aerospace Grade	Commercial Grade	Phenolic	Bias Weave/ Phenolic	Bias Weave/ Polyimide	Meta/Phenolic Aerospace Grade	Para/Phenolic Aerospace Grade	Bias Weave/ Polyimide
Relative Cost	Mod Low	Med	Very Low	Mod High	High	Very High	Med	High	Very High
Maximum Long-Term Temperature	175 °C	175 °C	175 °C	175 °C	175 °C	260 °C	175 °C	175 °C	260 °C
Flammability Resistance	E	E	E	E	E	E	E	E	E
Impact Resistance	G	G	G	F	G	F	E	E	F
Moisture Resistance	E	E	E	E	E	E	G	E	E
Fatigue Strength	G	G	G	G	G	G	E	E	E
Thermal Conductivity	High	High	High	Low	Low	Low	Very Low	Very Low	Med
Corrosion Resistance	G	E	G	E	E	E	E	E	E

E = Excellent
G = Good
F = Fair
P = Poor
Mod = Moderately
Med = Medium

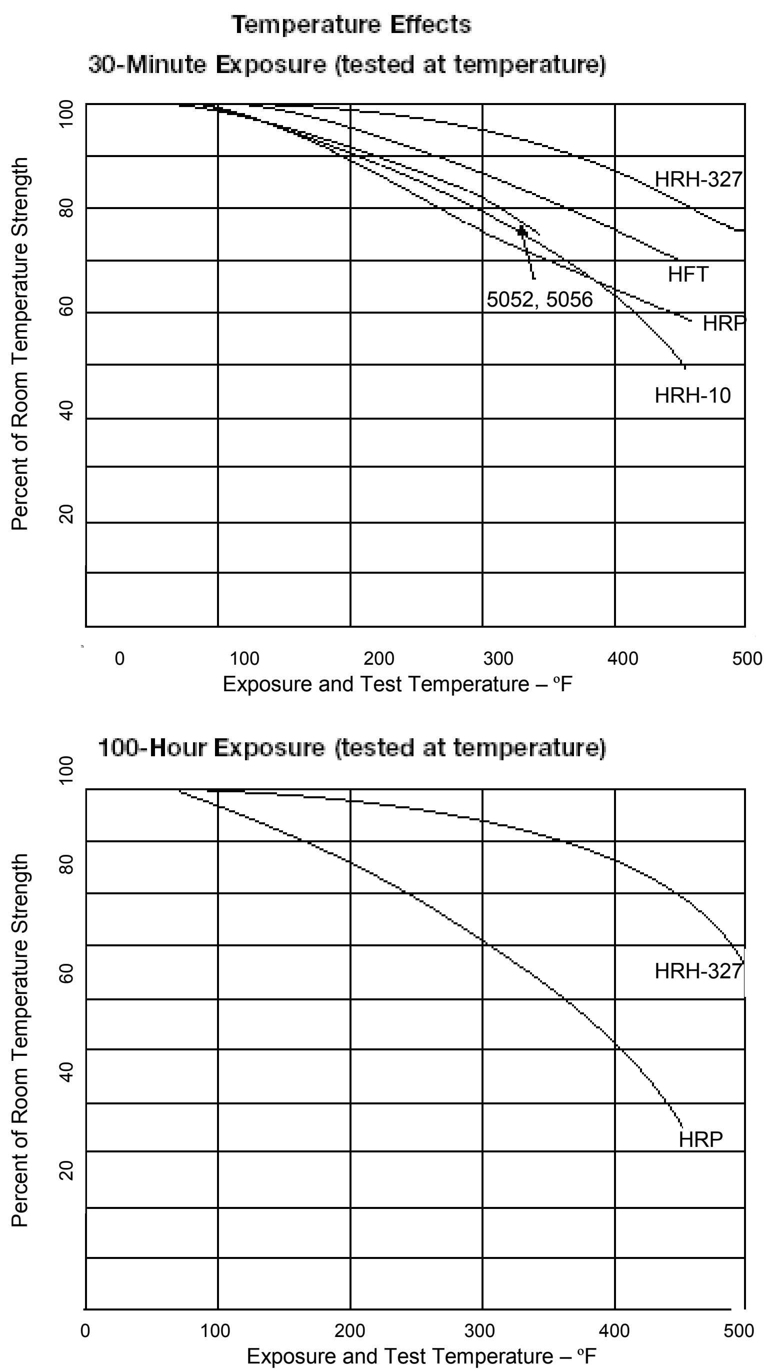


FIGURE 2.3.3 *Honeycomb percent retention at temperature (Reference 2.3.3).*

HRH, HFT and HRP are Hexcel nomenclatures for non-metallic core material types. 5052 and 5056 are aluminum alloy designations.

2.3.4 Test methods

The ASTM Volume 15.03 contains most of the core test methods. The main test methods are listed below.

Core Property	ASTM Standard Test Method	
Density	C271	Density of Sandwich Core Materials
Water Absorption	C272	Water Absorption of Core Materials
Shear Strength and Modulus	C273	Shear Properties of Sandwich Core Materials
Flatwise Tensile Strength	C297	Flatwise Tensile Strength of Sandwich Constructions
Core Node Tensile Strength	C363	Node Tensile Strength of Honeycomb Core Materials
Compressive Strength and Modulus	C365	Flatwise Compressive Properties of Sandwich Cores
Core Thickness	C366	Measurement of Thickness of Sandwich Cores
Shear Fatigue Strength	C394	Shear Fatigue of Sandwich Core Materials
Aging of Core	C481	Laboratory Aging of Sandwich Constructions
Flexural Strength and Modulus	C393	Core Shear Properties of Sandwich Constructions by Beam Flexure
	D7250	Determining Sandwich Beam Flexural and Shear Stiffness
Energy Absorption	D7336	Static Energy Absorption Properties of Honeycomb Sandwich Core Materials
Water Migration	F1645	Water Migration in Honeycomb Core Material

Shear properties of balsa and foam cores are about the same in both directions. The shear properties for honeycomb cores are different in the L (ribbon) and W (transverse) directions depending on the cell geometry. For hexagonal honeycomb core, the L shear properties are about two times the W shear properties. The honeycomb core shear strengths also vary with core thickness as shown below (Figure 2.3.4). The normal thickness for testing honeycomb cores is 0.625" for aluminum honeycomb and 0.500" (15.875 mm and 12.7 mm) for non-metallic honeycomb, so the effective shear strength to be used for design should be reduced by the associated K factor for the thicker core as indicated in Figure 2.3.4.

The bare compressive strength is obtained on the core without any face sheets. This test is primarily used as a quick quality control test for honeycomb, but can be used for balsa and foam design properties. When the core has face sheets bonded on, the test is called a stabilized compression test. This test specimen is used to obtain the honeycomb compressive modulus. The adhesive fillets on the honeycomb core-to-face sheet bond support the honeycomb cell edges and give slightly higher strength and stiffness values than the unsupported bare compression test values. The modulus for balsa and foam cores can be obtained from either test.

The shear strength and modulus of cores are usually obtained from the C273 plate shear test. This consists of bonding the core to steel plates and then loading the plates in either tension or compression with the load line going through the diagonal corners of the core. This core property is probably the most important as in a sandwich beam or plate the core takes the shear load.

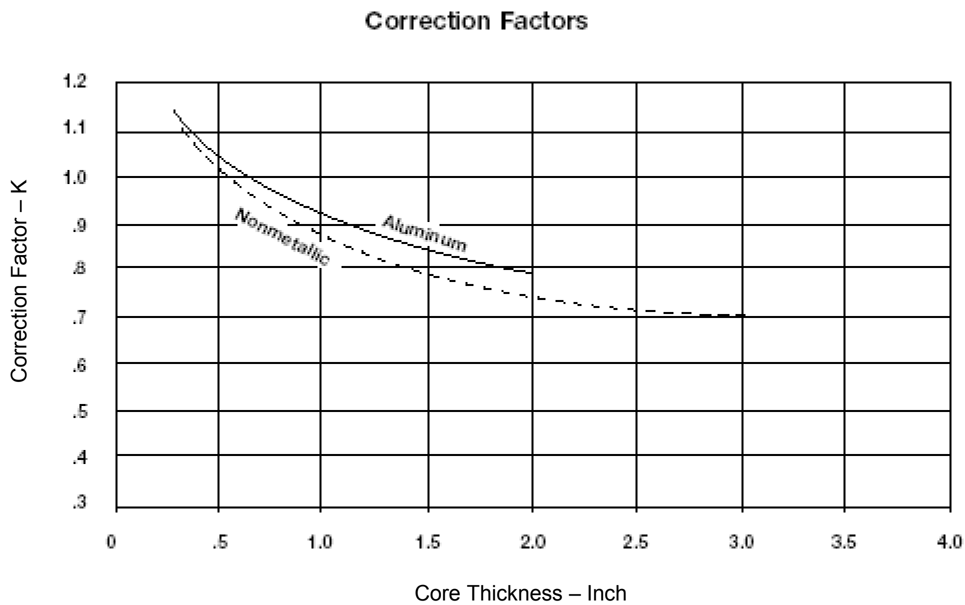


FIGURE 2.3.4 *Honeycomb core shear strength versus core thickness (Reference 2.3.3).*

2.4 EVALUATION OF CORE-TO-FACE SHEET BONDS

2.4.1 Introduction

The core-to-face sheet bond is extremely important in a sandwich structure. It enables the face sheets and core to work together, producing an extremely light and effective structure. Generally, failure of this bond is not the failure mode desired for the structure. There are several methods of attaching the face sheets to the core; adhesive bonding, self-adhesive prepregs, brazing, and fusion methods. The best method for evaluating the bond is actually testing the sandwich panel. When testing honeycomb core panels, the cell size used is very important.

2.4.2 Mechanical properties

Chapter 3 discusses different types of adhesives, and provides some data on material properties that may be used to guide selection of appropriate adhesives.

2.4.3 Environmental effects

Integrity of core-to-face sheet bonds is discussed by Epstein (Reference 2.4.3) who points out that moisture present during cure can produce porous bonds and also can retard curing of the adhesive. Adhesive bonds are also sensitive to environmental conditions (temperature, moisture) during service.

2.4.4 Test Methods

Military Specification Mil-A-25463, “Adhesive, Film Form, Metallic Structural Sandwich Construction”, contains adhesive requirements for core-to-face sheet bonds. The basic tests this specification requires are the following: climbing drum peel, flatwise tensile, flexural strength, and creep. The tests are conducted at room temperature and at elevated temperature. Also, some tests require the specimens to be conditioned in humidity or soaked in various fluids. More information regarding this specification can be found in Reference 2.4.4(a).

Federal specification MMM-A-132, "Adhesives, Heat Resistant, Airframe Structural, Metal to Metal", is a general specification for adhesives. The basic tests this specification requires are the following: tensile lap shear, creep rupture, T-peel, and blister detection. Again these tests are conducted at various temperatures and conditioning. More information regarding this specification can be found in Reference 2.4.4(b).

The ASTM standard test methods for core to face sheet bond tests are listed below.

Panel Property	ASTM Standard Test Method	
Flatwise Tensile Strength	C297	Flatwise Tensile Strength of Sandwich Constructions
Flexural Strength and Modulus	C393	Core Shear Properties of Sandwich Constructions by Beam Flexure
Flexure Creep	C480	Flexure Creep of Sandwich Constructions
Peel Strength	D1781	Climbing Drum Peel Test for Adhesives
Cleavage Strength	E2004	Face sheet Cleavage of Sandwich Panels

The face sheets must be well bonded to the core for the sandwich to work together. Two of the main tests used to evaluate the core-to-face sheet bond are the flatwise tensile and the climbing drum peel tests. A cleavage test has also been developed for thicker faced panels.

The flatwise tensile test consists of cutting a small specimen from the panel, usually 2" by 2" (50 mm by 50 mm), and bonding it to metal blocks. The specimen is then fixtured into a test machine and pulled apart with the maximum load and mode of failure recorded. The failure mode is extremely important as it can tell if the panel was made properly. The modes of failure are the following: core tearing, cohesion failure of the core-to-face sheet adhesive, and adhesion failure (either as adhesion failure between the core and interfacial adhesive, adhesion failure between the face sheet and interfacial adhesive, or adhesion failure between the core and face sheet for self-adhesive face sheets). If the bond between the block and the panel fails, this is not considered a valid failure and the test should be repeated. If the failures are adhesion to core or face sheet, this suggests that the core or face sheet is contaminated and the cleaning process should be reviewed. In some cases a core failure may simply not be achievable due to extremely high-strength core or in cases where the test temperature has exceeded the practical use temperature for the adhesive.

The climbing drum peel test consists of peeling off one face sheet from the panel. This test only works well on relatively thin face sheets, and the modes of failure are the same as above. For thicker face sheets, the cleavage test may give better results, as the face sheet does not have to be bent around a drum.

2.5 EVALUATION OF FACE SHEET PROPERTIES

2.5.1 Introduction

In general, a sandwich structure is designed so that the face sheets carry the in-plane loading, while the core resists out-of-plane shear loads. The face sheets of a sandwich panel develop the running axial loads for a panel in bending, one in compression and the other in tension, or for a column both face sheets in compression. Also, some sandwiches are shear panels where the face sheets take the in-plane shear loads.

2.5.2 Mechanical properties

Usually the most important mechanical properties for the face sheets are compressive and tensile strengths and moduli. Volume 2 contains test data on polymer composite materials that can be used as face sheets. Mechanical property reductions may be required for co-cured or co-bonded face sheets depending on the core geometry and/or processing techniques. Factors will need to be established through testing as applicable.

Metallic materials such as aluminum alloys are often utilized as face sheets in sandwich structures. Refer to the various published metal property data sources such as Metallic Materials Properties Development and Standardization (MMPDS) (Reference 2.5.2) handbook for design values.

2.5.3 Environmental effects

In-plane hygrothermal expansion of polymer composite face sheets depends mainly on ply orientation, fiber coefficient of thermal expansion (CTE), modulus, and volume fraction. Face sheet behavior can be analyzed in terms of classical laminate theory coupled with Fourier's law of heat flow or Fickian diffusion for moisture equilibrium. Zero CTE's and coefficients of moisture expansion (CME's) are possible in selected directions but not at the same time. Negative in-plane CTE's and CME's are also possible.

2.5.4 Test methods

Volume 1 Chapter 6 contains extensive information on composite material testing. The following ASTM standard test methods are commonly used to obtain face sheet properties

Face Sheet Property	**ASTM Standard Test Method**	
Face Sheet Compressive Strength	C364	Edgewise Compression Strength of Sandwich Panels
	D6641	Compressive Properties of Polymer Matrix Composite Laminates Using a Combined Loading Compression (CLC) Test Fixture
Tensile Strength and Modulus	D3039	Tensile Properties of Polymer Matrix Composite Materials
In-Plane Shear Strength	D3518	In-Plane Shear Response of Polymer Matrix Composite Materials by Tensile Test of a ±45° Laminate
	D7078	Shear Properties of Composite Materials by the V-Notched Rail Shear Method
Face Sheet Tensile/Compressive Strength	D7249	Facing Properties of Sandwich Constructions by Long Beam Flexure
Face Sheet Stiffness	D7250	Determining Sandwich Beam Flexural and Shear Stiffness

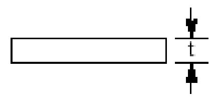
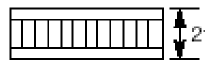
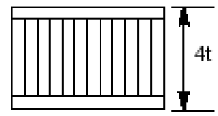
2.6 EVALUATION OF SANDWICH PANELS

2.6.1 Introduction

Sandwich panels can be designed to be extremely lightweight and be very stiff and strong. Table 2.6.1 illustrates this property.

The bending stiffness and strength of a sandwich panel consisting of face sheets bonded to a core can be significantly increased over those of a laminate or plate with the same material as the sandwich face sheets for a very small weight increase. The additional bending stiffness and strength will depend on the face sheet thickness, the core depth and density, and adhesive bond strength. Typically a laminate or plate structural panel will need stiffening with bonded or mechanically fastened stiffeners to avoid or reduce buckling, so the sandwich panel can achieve the same overall panel performance without the additional stiffening elements and fabrication costs.

TABLE 2.6.1 *Sandwich structure characteristics (Reference 2.3.3).*

	Solid Metal Sheet 	Sandwich Construction 	Thicker Sandwich 
Relative Bending Stiffness	100	700 7 times more rigid	3700 37 times more rigid!
Relative Bending Strength	100	350 3.5 times as strong	925 9.25 times as strong!
Relative Weight	100	103 3% increase in weight	106 6% increase in weight

2.6.2 Mechanical properties

The basic sandwich panel mechanical properties are the following:

Bending strength -- determined by the thickness and compressive and tensile strength properties of the face sheets and how far apart they are spaced
Bending stiffness -- determined by the thickness and compressive and tensile modulus properties of the face sheets and how far apart they are spaced
Out-of-plane shear strength -- determined by the thickness and shear strength property of the core
Out-of-plane shear rigidity -- determined by the thickness and shear modulus property of the core

2.6.3 Environmental effects

Environmental conditions, including temperature, moisture and fluids, can affect the strength and stiffness of a sandwich panel. The effects of environment on polymer composite materials are discussed elsewhere in CMH-17, including general discussion in Volume 3 Sections 2.2.4 and 3.4.3, and discussion of testing in Volume 1 Sections 2.2.7 and 6.5. These same effects will be seen when the material is used as a sandwich panel face sheet. The core will also be affected by the same environmental conditions, as will the core-to-face sheet bond. In general, high temperature, moisture, and fluids all degrade the sandwich properties.

Common problems with sandwich structures subjected to temperature and moisture/fluids are those associated with poor core sealing and porous or easily damaged face sheets. Face sheets that have been damaged can provide moisture paths to the core, which then may become degraded. Sandwich panels that are well sealed and with face sheets that are durable and nonporous are typically affected only slightly by environment, similar to the effects on laminate structure.

2.6.4 Damage resistance

Damage resistance is defined in Section 12.5 of Volume 3 as "a structure's resistance to various forms of damage occurring from specific events. Considering potential threats for commercial and military aircraft, this covers a large range of damage. Based on the specific structural configuration and design details, some damage types pose a more serious threat to structural performance than others". Sand-

wich panels, because they are usually designed to be lightweight, may not be damage resistant to a variety of damage types including impacts. The face sheet material properties and thickness, adhesive, and core type and density can play interdependent roles in sandwich panel damage resistance. Thin faced lightweight sandwich panels are susceptible to impact damage; therefore, in areas subject to numerous impacts, thicker and tougher face sheets and a tougher core will provide better impact resistance.

2.6.5 Damage tolerance

Damage tolerance is more difficult to define for a sandwich structure than for stiffened laminate or plate structure with multiple load paths. Damage tolerance is defined in Section 12.2 of Volume 3 as "Damage tolerance provides a measure of the structure's ability to sustain design loads with a level of damage or defect and be able to perform its operating functions. Consequently, the concern with damage tolerance is ultimately with the damaged structure having adequate residual strength and stiffness to continue in service until the damage can be detected by scheduled maintenance inspection and be repaired". Built-up metal and composite laminate stiffened structures have multiple load paths, so they have varying degrees of "fail safety" which was the starting point for damage tolerant structure. Until recently, most sandwich structural components were of the secondary structure or control surface nature, and were not required to be designed damage tolerant. Now, some control surfaces, such as flaps on commercial transports, are deemed primary structure and they are designed to be damage tolerant. For example, the 777 flaps are designed such that they are able to carry regulatory loads (design limit load) with one complete panel bay between ribs delaminated or damaged. This interpretation of the transport category aircraft 14 CFR 25.571 damage tolerance requirements requires design features to restrict the area of potential damage growth to an acceptable size.

2.6.6 Repair

Repairing sandwich structures can usually be done easily. Military Handbook 337 "Adhesive Bonded Aerospace Structure Repair" (Reference 2.6.6) is a very good source. This manual describes the entire process of repairing composite and metallic sandwich panels.

With sandwich construction, as with any other type of construction, it is inevitable that a certain amount of damage will occur. During the manufacturing stages, where hazards of dropped tools and equipment are encountered, serious damage to sandwich parts may be eliminated by protecting exposed corners and by using temporary protective covers. In service, unavoidable instances will occur that will damage the sandwich panel. Proper precautions will minimize damages; but when damage does result, acceptable methods of repair are available.

Repair procedures are developed with the objective of equaling, as nearly as possible, the strength and stiffness of the original part, with a minimum of increase in weight or change in aerodynamic characteristics and electrical properties where applicable. This can only be accomplished by replacing damaged material with identical material or an approved substitute. In order to eliminate dangerous stress concentrations, abrupt changes in cross-sectional areas must be avoided whenever practicable by tapering joints, by making small patches round or oval-shaped instead of rectangular, and by rounding corners of all large repairs. Smoothness of outside surfaces of high-speed aircraft is a necessity for proper performance, and consequently patches that protrude from the original surfaces must be avoided if at all possible. When this is impossible, the edges must be generously tapered to fair the repair into the original contour.

Special aspects of sandwich repair are discussed in Chapter 7 Supportability, as well as in Chapter 14 of Volume 3, especially paragraphs 14.7.4 Bonded Repairs and 14.7.5 Sandwich (Honeycomb) Repairs.

2.6.7 Test methods

The following ASTM Standard Test Methods can be used to test sandwich panels:

Panel Property	ASTM Standard Test Method	
Flatwise Tensile Strength	C297	Flatwise Tensile Strength of Sandwich Constructions
Face Sheet Compressive Strength	C364	Edgewise Compressive Strength of Sandwich Constructions
Compressive Strength and Modulus	C365	Flatwise Compressive Properties of Sandwich Cores
Flexure Creep	C480	Flexure Creep of Sandwich Constructions
Aging of Core / Sandwich	C481	Laboratory Aging of Sandwich Constructions
Peel Strength	D1781	Climbing Drum Peel Test for Adhesives
Flexural Strength and Modulus	D6416	Two-dimensional Plate Flexure
	D7249	Face sheet Properties of Sandwich Constructions by Long Beam Flexure
	D7250	Determining Sandwich Beam Flexural and Shear Stiffness
Damage Resistance	D7766	Standard Practice for Damage Resistance Testing of Sandwich Constructions
Cleavage Strength	E2004	Face sheet Cleavage of Sandwich Panels

2.7 EVALUATION OF INSERTS AND FASTENERS

2.7.1 Introduction

Sandwich components are often attached to other parts, resulting in the introduction of concentrated stresses at the attachments. The degree of local reinforcement required at these attachments varies depending on the magnitude of the loading and the configuration of the components. For very lightly stressed parts, unreinforced fastener holes or subsequently inserted reinforcements will suffice, but in most structural applications local reinforcements must be incorporated during fabrication. Figure 2.7.1(a) provides some examples of reinforcements that are incorporated during fabrication of the sandwich structure. These reinforcement details enable the core and face sheets to maintain load path continuity, but provide additional local reinforcement to introduce stresses into those constituents.

There are also numerous commercial inserts and attachment schemes that are suitable for incorporation into a sandwich component as a secondary operation (after the sandwich panel has been fabricated). These attachment details may or may not require additional reinforcement, depending on the magnitude of the loading and the configuration of the panel. Some examples of these types of attachments are shown in Figure 2.7.1(b).

2.7.2 Environmental effects

As discussed in the previous section for sandwich panels, the environment (temperature, moisture, fluid exposure, etc.) can also affect the strength and stiffness of an installed insert or attachment scheme for a sandwich panel. The environmental conditions must be determined for the application, and the properties must then be characterized in terms of failure modes and magnitudes for those conditions.

2.7.3 Test methods

Inserts and attachments are generally characterized using sub-element or sub-component tests due to their dependence on the specific configuration. Refer to Volume 1, Chapter 7, for test methods of bonded and bolted joints in composites. Environmental effects are generally incorporated into the testing conducted at the sub-element and/or sub-component level.

2.7.4 Mechanical properties

The most important mechanical properties for inserts and fasteners are generally core shear strength and modulus and face sheet compression (local bearing) strength and modulus. Tables in Chapter 3 con-

tain room temperature properties for some of the common core materials. Volume 2 contains test data on some materials that can be used as face sheets.

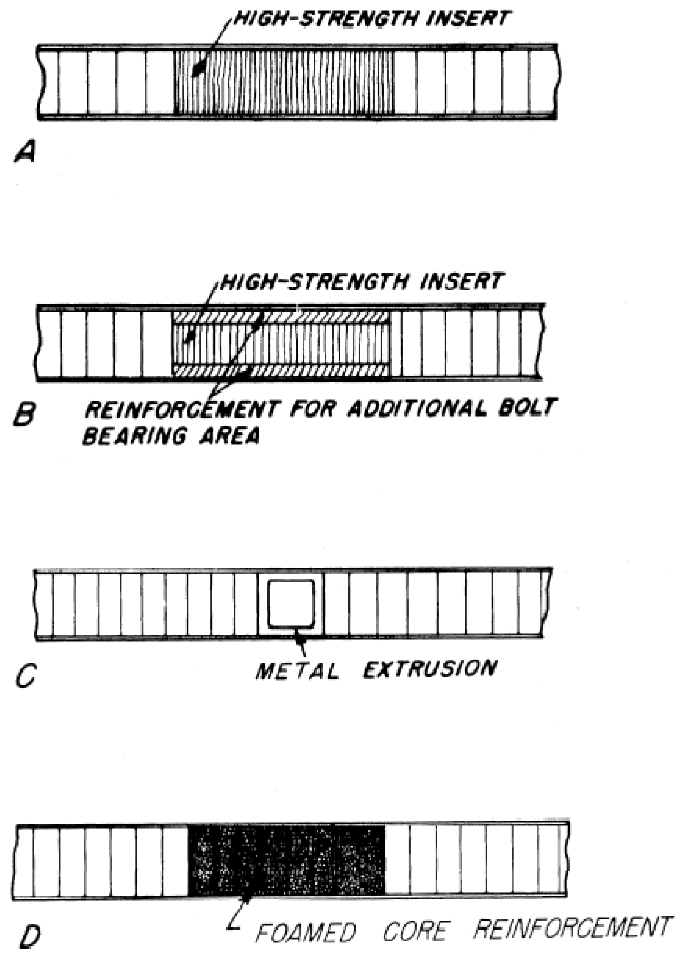


FIGURE 2.7.1(a) *Examples of sandwich reinforcement incorporated during fabrication.*

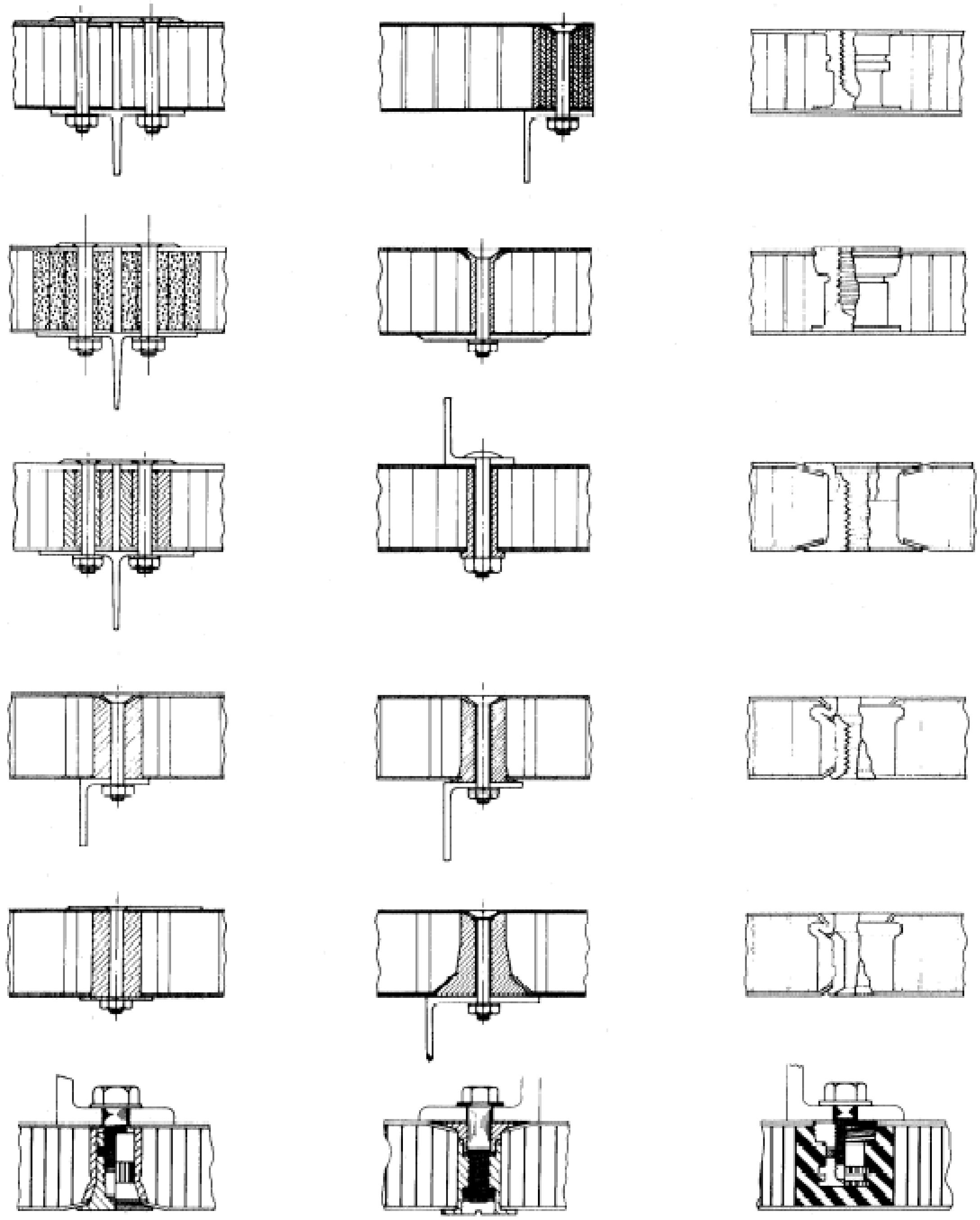


FIGURE 2.7.1(b) *Examples of sandwich attachments incorporated as a secondary operation.*

2.8 EVALUATION OF OTHER FEATURES

2.8.1 Introduction

Sandwich parts are often joined to framing members or other structure, and it is common practice to incorporate a continuous-edge reinforcement to facilitate the transfer of stresses. There are many ways of accomplishing satisfactory edge reinforcing so that such details as loads to be transferred, type of face sheets and core, attachment fittings, and importance of smoothness of surface, should be considered before selection is made. Typical edge reinforcements for some sandwich structures are shown in Figure 2.8.1. Areas of crushed low-density honeycomb core should be resin stabilized to prevent disintegration under sonic environment.

Some edge treatments serve as an effective moisture seal in addition to providing reinforcement. Others depend upon edge coating to seal out moisture and miscellaneous fluids. High-strength inserts may be of a variety of materials, including end-grain mahogany or spruce, plywood (flat or on edge), or reinforced plastics. Additional bolt-bearing area may be provided by reinforcements or by increasing face thickness.

Openings in sandwich parts for inspection, filler holes, or adjustment of fittings must often be provided. Tests have demonstrated that, with certain ratios of opening to panel size, there is a concentration of stress around the cut-outs that may require consideration in design. Experience has shown that these increased stresses can often be carried by high-strength core inserts or edge treatments around the opening. If the cut-out requires a cover, the means of attachment must be considered in choosing the proper edge reinforcement around the cut-out. Reference 2.8.1 provides more information on the design and analysis of these sandwich features.

2.8.2 Mechanical properties

The most important mechanical properties for edge reinforcements are generally the same as those for the panel and attachment areas, but are largely dependent on the specific configuration under consideration. Tables in Chapter 3 contain room temperature properties for some of the common core materials. Volume 2 contains test data on some materials that can be used as face sheets.

2.8.3 Environmental effects

As discussed in the previous section for sandwich panels, the environment (temperature, moisture, fluid exposure, etc.) can also affect the strength and stiffness of edge reinforcement for a sandwich panel. The environmental conditions must be determined for the application, and the properties must then be characterized in terms of failure modes and magnitudes for those conditions.

2.8.4 Test methods

Edge reinforcements are generally characterized using sub-element or sub-component tests due to their dependence on the specific configuration. Refer to Volume 1, Chapter 7, for test methods of bonded and bolted joints in composites. Environmental effects are generally incorporated into the testing conducted at the sub-element and/or sub-component level.

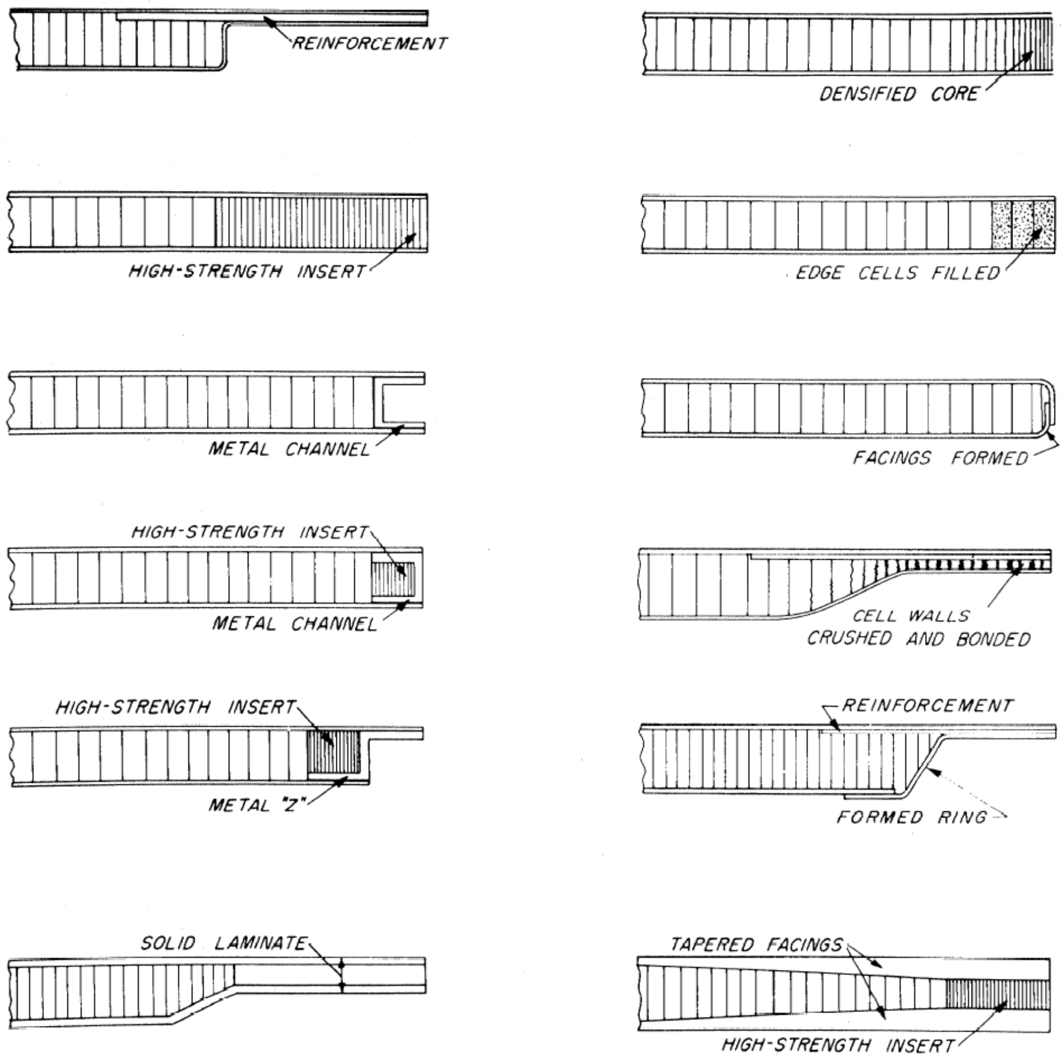


FIGURE 2.8.1 *Typical edge reinforcements for some sandwich structures*.

REFERENCES

2.3.3 "HexWeb™ Honeycomb Attributes and Properties: A comprehensive guide to standard Hexcel honeycomb materials, configurations, and mechanical properties," Hexcel Composites, Pleasanton, CA, Nov. 1999 (available at http://www.hexcel.com/Resources).

2.4.3 Epstein, G., *The Composites & Adhesives Newsletter*, T/C Press, Los Angeles, Vol. 13, No. 3, p. 4, 1997.

2.4.4(a) MIL-A-25463, Adhesives, Film Form, Metallic Structural Sandwich Construction.

2.4.4(b) MMM-A-132, Adhesives, Heat Resistant, Airframe Structural, Metal to Metal.

2.5.2 "Metallic Materials Properties Development and Standardization (MMPDS)", formerly MIL-HDBK-5, 2012, MMPDS-07.

2.6.6 Military Standardization Handbook, "Adhesive Bonded Aerospace Structure Repair," MIL-HDBK-337, 1 December 1982.

2.8.1 "The Handbook of Sandwich Construction", D. Zenkert, Editor, EMAS Publishing, 1997, Warrington England.

CHAPTER 3 MATERIAL DATA

The design of a sandwich structure requires an understanding of the roles of each constituent material employed and the respective mechanical properties, as well as factors which influence the performance of the constituents when they work together as a system. Typical properties of sandwich structure constituents are discussed in this chapter. Cores, face sheet materials, and adhesives are addressed in the sections that follow with examples of typical mechanical properties tabulated.

It should be emphasized that the mechanical property data provided in these sections has no statistical significance, nor has it been scrutinized with the technical rigor necessary for use in final designs. It is simply provided to enable the reader to obtain a feel for how broad categories of materials compare, and how the properties are used in design.

3.1 CORES

The core of a sandwich structure serves primarily to separate, support, and stabilize the face sheets such that the desired bending rigidity is attained. In nearly all cases, the core carries the major out-of-plane or transverse shear loads in a sandwich shell or panel structure. Other functions such as thermal and acoustic insulation are also largely dependent on the core material properties. To perform these functions at a minimum weight, the core material is generally of relatively low density in comparison to the face sheets and adhesive constituents. In many instances the core material needs special handling or interim processing to maintain dimensional stability due to its relative fragility or instability in the stand-alone state.

A few of the core materials typically used are discussed briefly in the following sections.

3.1.1 Description of cores

To permit a sandwich structure to perform satisfactorily, the core of the sandwich must have certain mechanical properties, thermal and acoustical characteristics, and dielectric properties under conditions of use, and still conform to weight limitations. Cores of densities ranging from 1.6 to 23 pounds per cubic foot have found use in airframe sandwich structures, but the usual density range is 3 to 10 pounds per cubic foot. Specifications for cores intended for use in airframes are listed in Section 3.1.2. Various core properties are given in US, Customary Units. Conversion to the International System of Units can be made by using factors given in notation at the front of this Volume of the Handbook (Section 1.4.2).

The core types that may be incorporated into a sandwich structure are numerous. The core types discussed in this chapter include the following:

Honeycomb
Cross-banded
Corrugated
Waffle-Type
Foams
Natural (Balsa wood and other wood)

3.1.2 Core specifications

Table 3.1.2 lists various core material specifications. A number of the specifications are cancelled or inactive, but relevant for reference purposes. For these specifications, the date listed is the date of last publication, not the date of cancellation or inactivation.

TABLE 3.1.2 *Core material specifications.*

Specification	Title	Status (Publication Date)
SAE AMS 3711	Core, Honeycomb Fibrous, Aramid Base, Phenolic Coated	Current
SAE AMS 3712	Core, Honeycomb, Glass/Polyimide	Current
SAE AMS 3713	Core, Flexible Honeycomb, Polyamide Paper Base, Phenolic Coated	Current
SAE AMS 3714	Core, Overexpanded Honeycomb, Polyamide Paper Base, Phenolic Coated	Current
SAE AMS 3715	Core, Honeycomb, Glass/Phenolic	Current
SAE AMS 3716	Core, Honeycomb, Glass/Phenolic, Bias Weave Fiber Construction	Current
SAE AMS 3725	Core, Polyurethane Foam (Polyether), Rigid, Cellular	Noncurrent (1993)
SAE AMS 4177	Core, Flexible Honeycomb, Aluminum Alloy, for Sandwich Construction, 5056, 350 (177)	Current
SAE AMS 4178	Core, Flexible Honeycomb, Aluminum Alloy, Treated for Sandwich Construction, 5052, 350 (177)	Current
SAE AMS 4348	Core, Honeycomb, Aluminum Alloy, Corrosion Inhibited, for Sandwich Construction, 5052, 350 (177)	Current
SAE AMS 4349	Core, Honeycomb, Aluminum Alloy, Corrosion Inhibited for Sandwich Construction, 5056, 350 (177)	Current
SAE AMS 5850	Steel, Corrosion and Heat-Resistant, Honeycomb Core, Resistance Welded, Square Cell	Current
SAE AMS-C-7438	Core Material, Aluminum, for Sandwich Construction	Current
SAE AMS-C-8073	Core Material, Plastic Honeycomb, Laminated Glass Fabric Base, for Aircraft Structural and Electronic Applications	Current
SAE AMS-C-81986	Core Material, Plastic Honeycomb, Nylon Paper Base; for Aircraft Structural Applications	Current
SAE-AMS-PRF-46194	Foam, Rigid, Structural, Closed Cell	Current
MIL-C-8087	Core Material, Foamed-in-Place, Urethane Type	Cancelled (1968)
MIL-L-7970	Lumber, Hardwood, Mahogany, Aircraft Quality	Inactive (1953)
MIL-S-7998	Sandwich Construction Core Material, Balsa Wood	Inactive for New Design (1978)
MIL-S-25395	Sandwich Construction, Plastic Resin, Glass Fabric Base, Laminated Facings and Urethane Foamed-in-Place Core, for Structural Applications	Inactive for New Design (1968)

3.1.3 Honeycomb Cores

Honeycomb core is also called cellular core or open-cell core. A wide variety of core materials can be constructed of thin sheet materials or ribbons formed to honeycomb-like configurations. By varying the sheet material, sheet thickness, cell size, and cell shape, cores of a wide range in density and properties can be produced. Various core configurations, many of which are readily commercially available, are

shown in Figure 3.1.3(a). Most honeycomb cores can be formed (bent) to moderate amounts of single curvature. Cores with specialized cell configurations have been developed for forming with compound curvature, or severe single curvature. For some cores, particularly nonmetallic cores, forming can also be facilitated by the use of heat (hot forming).

The honeycomb cores Types A and B shown in Figure 3.1.3(a) are less formable. These are preferred for applications with flat to slight curvature. The over-expanded core (Type C), the under-expanded core (Type D), and flexible core (Type E) are more formable than the others shown in this figure, and can be easily formed to fairly severe single curvature, and to moderate compound curvature (spherical) shapes. Other unique core shapes and configurations have also been used to meet specific performance requirements but are not shown here as they are less common.

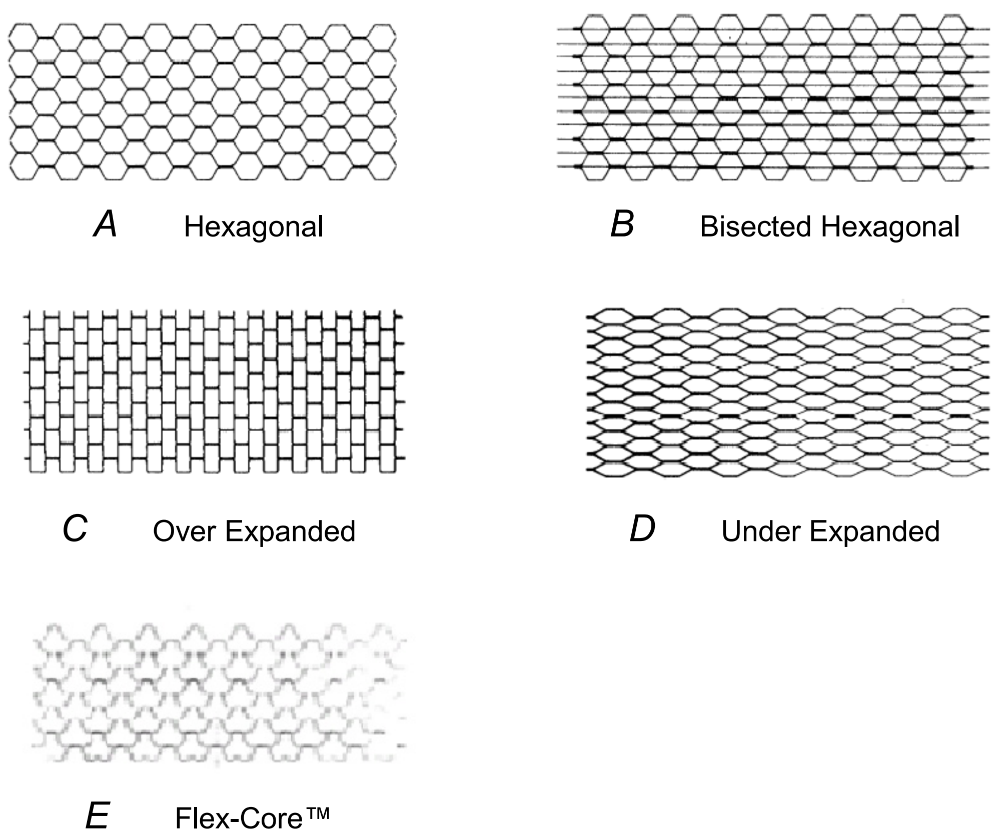


FIGURE 3.1.3(a) *Honeycomb core cell configurations.*

Honeycomb core cell size is determined by the diameter of a circle that can be inscribed in a cell. Figure 3.1.3(b) shows the ribbon (L), transverse (W), and thickness (T) for typical hexagonal honeycomb core. Honeycomb core cell sizes used in aircraft range from about 1/16 to 7/16 inch, usually in multiples of 1/16 inch. The cell configurations that can be obtained from a particular sheet material are limited by its thickness and stiffness.

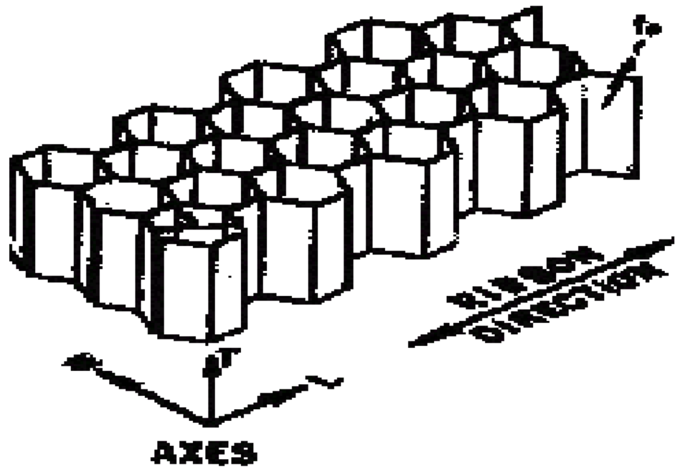


FIGURE 3.1.3(b) *Honeycomb core axes and cell size notation.*

For special applications, such as at an insert, honeycomb cores can be densified locally by under-expanding or by crushing the cells together. Such densified core has properties increased approximately in proportion to its density increase. Cores for sandwich construction are routinely fabricated from thin sheets of aluminum alloys, resin-treated fabrics, resin-treated paper, stainless steel alloys, titanium alloys, and refractory metals.

Sheets of corrugated metal foil are usually assembled with the corrugations parallel to form honeycomb cores. The foil may be perforated for use in core where solvents or gases must be vented. Perforated foil in sandwich panels that are not sealed or are poorly sealed will allow penetration of moisture, etc., which, depending on the core material, may cause severe deterioration of the core performance.

Honeycomb cores fabricated from nonmetallic materials have better thermal insulating characteristics than metallic honeycomb cores, even though both allow transmission of heat by radiation in the open cells. In considering thermal effects in sandwich structure, it should be understood that the sandwich could act as a reflective thermal insulator.

The effective thermal conductivity of a honeycomb core depends upon conduction of the material of which the core is made, radiation between face sheets, and convection within the core cell (References 3.1.3(a) and (b)) and can be computed approximately with the formula:

$$K_e = K_o A_c + \frac{4\sigma t_c \left(1 - A_c\right) T_m^3}{\frac{1}{\varepsilon_{UPR}} + \frac{1}{\varepsilon_{LWR}} - 2 + \frac{2}{1 + F_{12}}} + t_c \left(1 - A_c\right) \eta$$

where

K_e - effective conductivity

K_o - conductivity of core ribbon material

A_c - core solidity, $A_c = {}^{w_c}\!/\!_{w_o}$

w_c - core density

w_o - core ribbon material density

- σ - Stefan-Boltzmann constant
- t_c - core thickness
- T_m - mean absolute temperature of the two sandwich face sheets
- ε_{UPR} - emissivity of inside of sandwich upper face sheet
- ε_{LWR} - emissivity of inside of sandwich lower face sheet
- F_{12} - geometric view factor between face sheets (Reference 3.1.3(c))
- η - convective heat transfer coefficient inside core cell

3.1.4 Cross-banded core

If the sheets are assembled with the corrugations in adjacent sheets perpendicular to each other, a well-vented cross-banded core is produced. Cross-banded cores may be cut so that the corrugation flutes are at an angle of 45 degrees to the sandwich face sheets, giving the core a trussed appearance. A picture of cross-banded core with a description of the ribbon direction is shown in Figure 3.1.4.

Cross-banded cores are not as strong in compression in the T direction or in shear in the TL or TW planes (see Figure 3.1.3(b)) as honeycomb cores of the same density. Honeycomb cores, however, have negligible compressive strength in the W and L directions and shear strength in the WL plane, whereas cross-banded cores have considerable strength in these directions.

Because of the many cross connections between core flutes, cross-banded core is particularly adapted to construction of airframe sandwich panels with integral fuel tanks. Cross-banded core is not readily formed to curved surfaces because of its relatively high stiffness in all directions. Curved parts and parts of unusual shape could, however, be machined from blocks of cross-banded core.

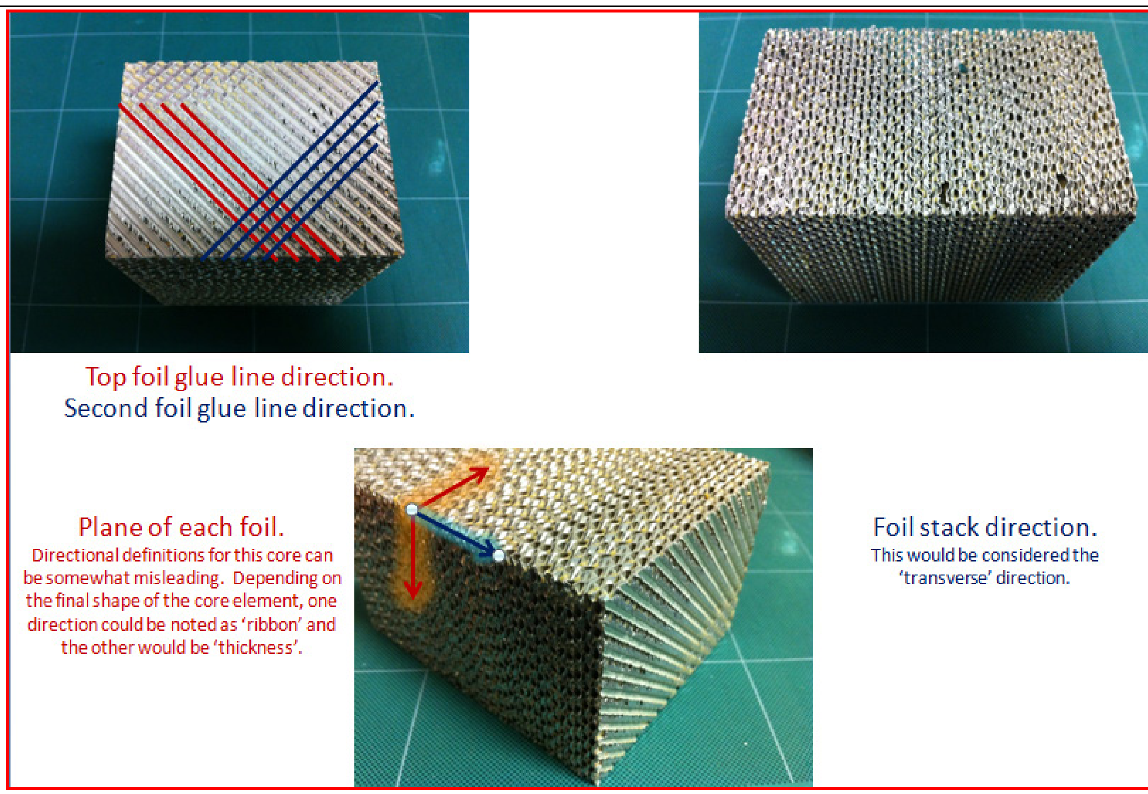


FIGURE 3.1.4 *Cross-banded core.*

3.1.5 Corrugated core

Corrugated cores are produced by forming a sheet of metal foil or resin-treated glass cloth to a series of sine wave corrugations. Figure 3.1.5 shows sketches of sandwich having single and double rows of corrugations. The corrugation flutes run parallel to the face sheets, whereas honeycomb core cell axes are normal to the face sheets. Corrugated cores may be formed to single curvature. Approximate thermal conductivity expressions for corrugated core are given in (Reference 3.1.3(b)).

3.1.6 Waffle-type core

Cores of sheet material fabricated into a configuration resembling a waffle have been produced. Sheets of resin-treated glass-fiber mat have been formed to waffle-type core for use in radomes. Thin metal sheets embossed or dimpled into a waffle configuration of rows of square or triangular flat surfaces on either side have been manufactured. The waffle-type core does not lend itself well to sandwich constructions that require tapered core thickness.

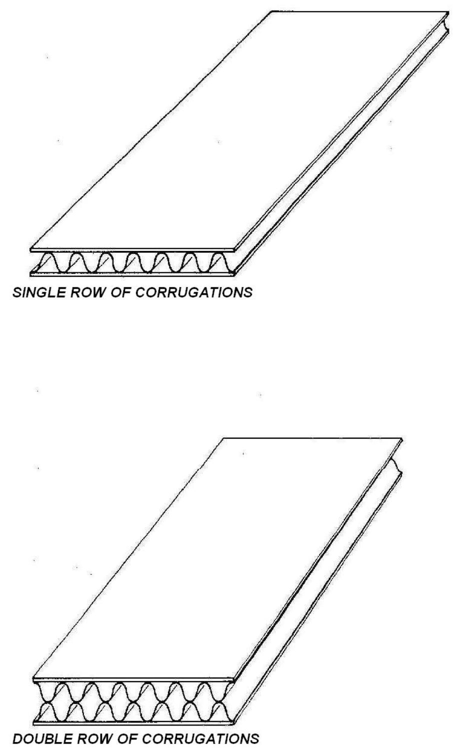


FIGURE 3.1.5 Sandwich with corrugated cores.

3.1.7 Foam cores

Foam cores have been developed to overcome the principal disadvantages of natural (wood or paper) core materials, particularly undesirable variation in density and moisture absorption. Plastic cores are foamed, expanded, or processed by other means to reduce the apparent density of the plastic to a practical range for core material. Desired core properties can be achieved by controlling the expansion process for these cores. Metallic foamed cores can be produced by mixing molten aluminum-magnesium alloys with suitable foaming agents and cooling the mixture to form a porous solid. Glass foams can also be produced.

In order to obtain necessary radiation-transmission characteristics, certain types of radomes require sandwich construction with homogeneous face sheets and core, together with tapering thickness of the sandwich and close control of thickness throughout (References 3.1.7(a) through (c)). To obtain all these characteristics, foamed-in-place core materials have been developed that will adhere to accurately pre-molded face sheets. This type of core material, while not as strong or stiff as honeycomb core of the same density, offers these advantages: Core joint elimination; thin and uniform bonding layer between face sheets and core; use of accurately pre-molded, void-free face sheets that can be readily inspected before assembly; good electrical properties; and flexibility in manufacture.

Foam cores with uniform density of 3 to 30 pounds per cubic foot have been produced, but materials of a density of 10 to 16 pounds per cubic foot are most commonly used (References 3.1.7(d) and (e)). In addition to their use in radomes, some foams have also been used to a limited extent to stabilize the skins of hollow steel propeller blades and aluminum alloy control surfaces, particularly for complex configurations in which honeycomb core materials cannot readily be used. Due to their generally lower cost than honey comb cores, foam cores have been commonly used in cost-sensitive applications such as general aviation and marine applications.

The thermal insulation of foamed-in-place core is generally good compared to honeycomb core. Metallic foam, glass foam, and metal-filled foams of various densities have also been produced and used for specific applications.

3.1.8 Wood cores

The selection of natural core materials for sandwich structure is confined principally to balsa wood, with mahogany, spruce, and poplar being used for inserts and edgings.

The thermal conductivity of wood across the grain can be computed by the formula (Reference 3.1.8(a)).

$$k = (1.39 + 0.028\ M)\ S + 0.165$$

where

k = conductivity (Btu in. /hr ft^2 F°)
M = moisture content, (percent)
S = specific gravity of wood

The thermal conductivity parallel to the grain is about 2-1/2 times that across the grain (Reference 3.1.8(a)).

Balsa wood with typical density ranging from 6 to 15 pounds per cubic foot can be used with the grain direction oriented parallel or perpendicular to the sandwich face sheets. The present practice is to place the grain perpendicular to the face sheets, which is called “end-grain” application (References 3.1.8(b) through (f)).

Certain portions of sandwich panels, such as points of attachment and exposed edges, require a high strength insert (References 3.1.8(g) through (i)). End-grain mahogany has been historically used at these points, but has more recently been replaced with aluminum extrusions or non-metallic details. The density of mahogany, determined by weight and measurement of planed boards at a moisture content of 8 to 12 percent, is normally between 25 and 35 pounds per cubic foot. As a substitute for mahogany, end-grain spruce has sometimes been used for inserts. Its relatively poor machinability across the grain and difficulty in bonding to the end-grain surface have limited its use to an occasional experimental or emergency application. End-grain poplar is also reported to have found limited use for core inserts.

3.1.9 Core properties

Typical mechanical properties of various core materials are given in Tables 3.1.9(a) through (f). Axes notation for core properties in the different directions specified in the tables are illustrated in Figure 3.1.3(b) and Figure 3.1.9.

For each material, typical average density and mechanical properties are given. Statistical data is not currently available for the core materials shown, but the values provided represent typical industry vendor information on commercially available products. Accurate sizing of core for a given application should be established using the specific approved properties based on the methods discussed in Chapter 2 for actual materials of construction.

TABLE 3.1.9(a) *Typical properties for non-metallic honeycomb core material.*

Core Material	Cell size	Density	Paper Thickness	Bare Com-pressive Strength	L-Shear Strength	L-Shear Modulus	W-Shear Strength	W-Shear Modulus
	in	lb/ft^3	in	psi	psi	ksi	psi	ksi
Nomex®	1/8	1.8	0.0015	131	87	3.9	44	1.7
	1/8	3.0	0.0020	334	203	7.0	131	3.8
	1/8	4.0	0.0020	595	290	9.1	174	5.5
	3/16	1.8	0.0015	87	73	3.3	44	1.9
	3/16	2.0	0.0020	160	102	4.4	58	2.3
	3/16	3.0	0.0020	305	174	6.2	102	3.8
	3/16	4.0	0.0020	522	247	8.0	160	5.1
	1/4	1.5	0.0015	87	58	2.9	29	1.5
	1/4	2.0	0.0015	145	102	3.9	58	2.2
Kevlar™	1/8	2.5	0.0014	261	203	15.2	116	7.8
	1/8	3.0	0.0018	377	276	18.9	160	10.0
	1/8	4.0	0.0018	595	363	21.0	203	12.3
	1/8	4.0	0.0028	493	406	29.7	247	14.5
	1/8	4.5	0.0018	725	377	19.1	218	11.0
	1/8	4.5	0.0028	653	479	37.7	276	14.5
	1/8	5.0	0.0018	841	464	21.8	261	13.1
	1/8	5.0	0.0028	827	551	37.7	319	16.0
	1/8	5.0	0.0023	841	508	29.7	290	14.5
	1/8	6.0	0.0018	1102	522	20.3	305	12.0
	1/8	6.0	0.0028	1146	682	37.7	392	16.0
	5/32	2.5	0.0014	261	203	14.8	131	9.3
	5/32	4.5	0.0018	754	348	20.3	247	13.1
	5/32	4.5	0.0028	740	450	26.1	276	14.5
	5/32	5.0	0.0028	899	551	28.3	334	16.0
	5/32	6.0	0.0028	1233	638	29.7	392	18.1
	5/32	6.0	0.0039	1088	725	29.0	348	13.8
	3/16	2.0	0.0014	218	145	10.2	87	4.9
	3/16	2.0	0.0018	145	145	11.2	87	7.1
	3/16	2.5	0.0014	319	189	11.5	116	6.2
	3/16	2.5	0.0018	276	203	14.5	116	8.0
	3/16	3.0	0.0018	406	232	14.9	160	9.7
	3/16	4.0	0.0018	667	334	20.3	247	13.6

Kevlar™	3/16	4.0	0.0028	653	392	23.2	232	12.6
	3/16	4.5	0.0018	827	392	23.2	276	15.2
	3/16	4.5	0.0028	870	464	22.6	319	15.5
	3/16	6.0	0.0018	1218	522	31.9	392	19.6
	3/16	6.0	0.0028	1320	580	24.9	435	18.7

Data is typical data based on a survey of manufacturers' published data.
For specification values, consult AMS3711 or the appropriate end user specification.

TABLE 3.1.9(b) *Typical properties for glass phenolic honeycomb core material.*

Cell size	Density	Stabilized Compressive Strength	Compressive Modulus	L-Shear		W-Shear	
				Strength	Modulus	Strength	Modulus
in	lb/ft^3	psi	ksi	psi	ksi	psi	ksi
3/16	4	595	51	285	14	160	7
3/16	4.5	725	61	305	17	188	9
3/16	5.5	900	82	450	20	245	11
3/16	6	1,050	90	520	25	300	13.5
3/16	7	1,300	110	650	30	370	16
3/16	8	1,600	130	730	35	470	20
3/16	9	1,900	150	800	41	535	23
3/16	12	2,600	230	1,000	49	680	29
1/4	3.5	440	55	260	11	140	6.5
1/4	4.5	640	70	350	15	200	8
3/8	3.2	390	38	205	11	110	5
3/8	3.5	410	41	210	11.3	120	5.5
3/8	4.5	635	67	330	14.5	200	8
3/8	6	1,050	105	475	21	300	11

Data is typical data based on a survey of manufacturers' published data.
For specification values, consult AMS3715 or the appropriate end user specification.

TABLE 3.1.9(c) *Typical properties for 5052 aluminum honeycomb core material.*

Cell size	Density	Foil Thick-ness	Stabilized Compressive Strength	Compressive Modulus	L-Shear		W-Shear	
					Strength	Modulus	Strength	Modulus
in	lb/ft^3	in	psi	ksi	psi	ksi	psi	ksi
1/8	3.1	0.0007	292	75	212	38.5	131	19.0
1/8	4.5	0.0010	557	150	342	60.5	222	28.0
1/8	6.1	0.0015	973	240	543	87.5	335	39.0
1/8	8.1	0.0020	1,512	350	778	123.5	488	52.0
1/8	10	0.0025	2,063	-	1,028	157.5	580	62.5
5/32	2.6	0.0007	243	55	168	30.5	101	15.5
5/32	3.8	0.0010	413	110	273	48.5	167	23.0
5/32	5.3	0.0015	725	195	423	74.0	273	33.5
5/32	6.9	0.0020	1,135	285	593	103	378	44.0
5/32	8.4	0.0025	1,608	370	765	128	478	53.5
3/16	2.0	0.0007	178	34	121	22.0	71	11.5
3/16	3.1	0.0010	315	75	212	38.5	128	19.0
3/16	4.4	0.0015	535	145	332	59.0	217	27.0
3/16	5.7	0.0020	833	220	462	80.0	302	36.0
3/16	6.9	0.0025	1,147	285	593	103	377	44.0
3/16	8.1	0.0030	1,618	350	728	124	475	52.0
1/4	1.6	0.0007	96	20	86	17.0	50	8.5
1/4	2.3	0.0010	197	45	142	26.5	86	13.5
1/4	3.4	0.0015	355	90	233	42.5	145	21.0
1/4	4.3	0.0020	522	140	322	57.0	205	27.0
1/4	5.2	0.0025	733	190	412	72.0	267	33.0
1/4	6.0	0.0030	1,020	235	520	85.5	335	38.5
1/4	7.9	0.0040	1,452	340	703	119.0	443	51.0
3/8	1.0	0.0007	47	10	45	9.5	30	5.0
3/8	1.6	0.0010	97	20	87	17.0	51	8.5
3/8	2.3	0.0015	190	45	138	26.5	82	13.5
3/8	3.0	0.0020	295	70	202	36.5	126	18.0
3/8	3.7	0.0025	410	105	255	47.5	165	23.0
3/8	4.2	0.0030	528	135	312	56.0	202	26.0
3/8	5.4	0.0040	777	200	432	76.0	282	34.5
3/8	6.5	0.0050	1,008	265	550	94.0	355	42.0
1/2	3	0.0030	315	-	240	30.0	125	15.0
1/2	6	0.0040	1,000	-	640	75.0	375	36.0

Data is typical data based on a survey of manufacturers' published data.
For specification values, consult AMS4348 or the appropriate end user specification.

TABLE 3.1.9(d) *Typical properties for 5056 aluminum honeycomb core material.*

Cell size	Density	Foil Thick-ness	Stabilized Compressive Strength	Compressive Modulus	L-Shear		W-Shear	
					Strength	Modulus	Strength	Modulus
in	lb/ft^3	In	psi	ksi	psi	ksi	psi	ksi
1/8	3.1	0.0007	348	97	252	38.5	157	18.0
1/8	4.5	0.0010	673	185	438	60.5	257	26.5
1/8	6.1	0.0015	1,137	295	677	89.5	393	37.5
1/8	8.1	0.0020	1,700	435	935	118	552	50.5
1/8	10	0.0025	2,200	-	1,190	140	700	60.0
5/32	2.6	0.0007	268	70	203	30.5	118	14.5
5/32	3.8	0.0010	505	140	338	49.0	198	22.0
5/32	5.3	0.0015	870	240	555	74.5	330	32.0
5/32	6.9	0.0020	1,345	350	763	105	435	42.5
3/16	2.0	0.0007	187	45	142	22.0	86	11.0
3/16	3.1	0.0010	390	97	263	38.5	153	18.0
3/16	4.4	0.0015	648	180	423	59.0	247	25.5
3/16	5.7	0.0020	973	270	573	82.0	335	35.0
3/16	6.9	0.0025	1,250	-	765	91.0	450	42.0
3/16	8.1	0.0030	1,625	-	925	112	550	50.0
1/4	1.6	0.0007	108	30	91	16.5	61	8.3
1/4	2.3	0.0010	247	58	178	26.5	103	13.0
1/4	3.4	0.0015	455	115	293	42.5	177	20.0
1/4	4.3	0.0020	610	172	403	56.0	235	25.5
1/4	5.2	0.0025	813	230	497	67.0	303	31.0
1/4	6.0	0.0030	1,000	-	640	75.0	375	36.0
1/4	7.9	0.0040	1,580	-	900	108	540	49.0
3/8	1.0	0.0007	52	15	57	11.0	36	4.9
3/8	1.6	0.0010	108	30	91	16.5	61	8.3
3/8	2.3	0.0015	220	58	172	26.5	100	13.0
3/8	3.0	0.0020	337	92	247	34.0	147	17.0
3/8	3.7	0.0025	450	-	325	40.0	190	20.0
3/8	4.2	0.0030	550	-	395	47.0	225	23.0
3/8	5.4	0.0040	850	-	565	66.0	325	32.0
3/8	6.5	0.0050	1,135	-	710	83.0	420	40.0
1/2	2.6	0.0025	230	-	190	24.0	100	12.0
1/2	3.0	0.0030	315	-	240	30.0	125	15.0
1/2	6.0	0.0040	1,000	-	640	75.0	375	36.0

Data is typical data based on a survey of manufacturers' published data.
For specification values, consult AMS4349 or the appropriate end user specification.

TABLE 3.1.9(e) *Typical properties for foam core materials*
Testing directions are parallel and perpendicular to the direction of the foam rise.

Density	Compressive Strength		Compressive Modulus		Tensile Strength		Tensile Modulus	
	Parallel	Perp.	Parallel	Perp.	Parallel	Perp.	Parallel	Perp.
lb/ft^3	psi	psi	psi	psi	psi	psi	psi	psi
3	63	45	1,250	1,410	59	50	1,750	1,510
4	95	73	2,090	2,320	88	77	2,830	2,520
5	130	106	3,130	3,410	119	108	4,110	3,730
6	169	144	4,360	4,660	152	142	5,570	5,150
7	211	186	5,750	6,080	187	179	7,200	6,760
8	255	232	7,320	7,650	224	219	8,990	8,560
10	351	336	11,000	11,200	302	306	13,000	12,700
12	491	474	15,400	11,600	388	420	17,100	17,000
15	749	732	22,600	17,800	538	586	25,100	24,800
18	1,060	1,040	30,900	25,200	701	769	34,400	33,700
20	1,290	1,280	37,100	30,800	818	899	41,200	40,300
25	1,970	1,980	54,400	47,300	1,130	1,250	60,600	58,700
30	2,780	2,820	74,500	67,100	1,480	1,650	82,900	79,800
35	3,720	3,810	97,200	90,200	1,850	2,070	108,000	103,000
40	4,780	4,940	122,000	116,000	2,250	2,520	136,000	129,000

Density	Shear Strength		Shear Modulus	
	Parallel	Perp	Parallel	Perp
lb/ft^3	psi	psi	psi	psi
3	39	48	477	709
4	60	72	730	1,020
5	84	97	1,020	1,340
6	111	125	1,330	1,690
7	140	155	1,670	2,050
8	171	186	2,030	2,420
10	239	253	2,820	3,200
12	315	326	3,950	4,240
15	453	457	5,990	6,150
18	608	601	8,420	8,340
20	721	705	10,300	9,950
25	1,040	987	15,600	14,400
30	1,390	1,300	21,900	19,600
35	1,780	1,640	29,200	25,300
40	2,210	2,010	37,400	31,700

Data is typical data based on a survey of manufacturers' published data.
For specification values, consult AMS3725, SAE-AMS-PRF-46194, or the appropriate end user specification.

TABLE 3.1.9(f) *Typical properties for balsa core materials*
Testing direction is perpendicular to the plane.

Density	Compressive Strength	Compressive Modulus	Tensile Strength	Tensile Modulus	Shear Strength	Shear Modulus
lb/ft^3	psi	psi	psi	psi	psi	psi
6.2	992	312,000	1,140	336,000	277	15,900
9.4	1,840	569,000	1,890	510,000	427	22,800
15.2	3,740	1,137,000	3,360	825,000	703	43,800

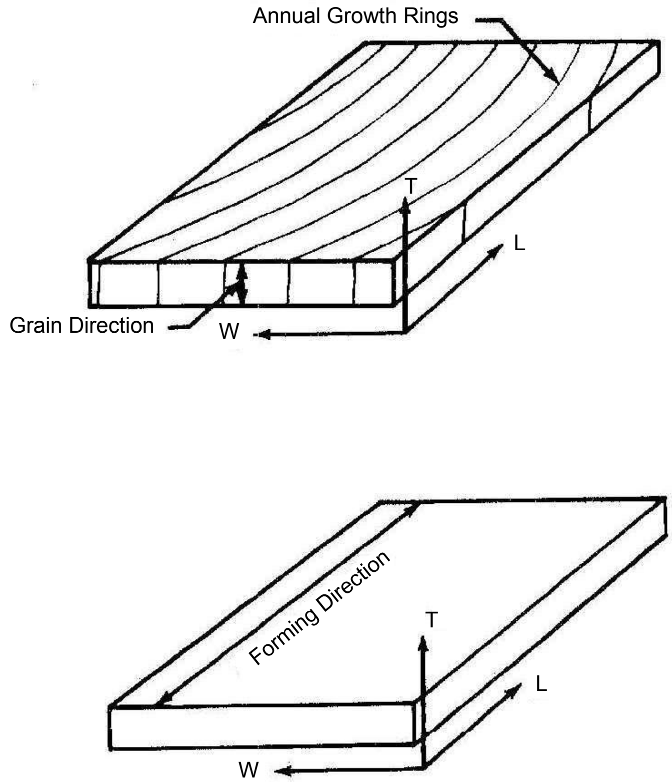


FIGURE 3.1.9 *Orthotropic axes notation for sandwich cores: (a) Wood Core and (b) Foam Core.*

3.1.9.1 Estimation of core properties

If core properties have not been established by test, it is possible to obtain reasonable estimates by consideration of core material, density, and configuration. Conversely, if a design requires a core with certain properties, it is possible to estimate the density, material, and configuration of core needed to obtain a satisfactory sandwich construction. The elastic moduli and strength of cores of a particular material generally increase as core density increases. Thus if properties at a certain density are known, a linear extrapolation can be used to obtain estimates of properties at a different density. If properties of cores of several densities are known, a curvilinear relationship may be exhibited between property and density, and this should be used rather than a linear extrapolation.

For cellular (honeycomb) metallic cores, the density and elastic moduli can be estimated from properties of the foil or ribbon material, thickness of the ribbons, and core cell size and shape. Estimates of metallic honeycomb core density can be made by determining the amount of foil or ribbon material per unit volume of core. Results for hexagonal core (Figure 3.1.3(b)) is given by the formula:

$$w_c = \frac{8t_o}{3s} w_o \quad \text{for core with hexagon cells}$$

where

w_c is core density;
w_o is density of foil or ribbon material;

t is thickness of foil or ribbon material; and
s is cell size (diameter of inscribed circle).

The core modulus of elasticity in the through-thickness direction, E_C, can be estimated by multiplying the modulus of elasticity of the foil or ribbon material, E_O, by the ratio of core density to foil or ribbon density:

$$E_c = \frac{w_c}{w_o} E_o$$

3.2 FACE SHEETS

The face sheets of a sandwich structure serve many purposes, depending upon the application, but in all cases they carry the major applied loads. The stiffness, stability, configuration, and to a large extent, the strength of the part are determined by the characteristics of the face sheets as stabilized by the core. To perform these functions the face sheets must be adequately bonded to a core of acceptable quality. Face sheets sometimes have additional functions, such as providing a profile of proper aerodynamic smoothness, a rough non-skid surface, or a tough wear-resistant floor covering. To better fulfill these special functions, one face sheet of a sandwich is sometimes made thicker or of slightly different construction than the other.

Any thin, sheet material can serve as a sandwich face sheet. In many cases, the face sheet material serves to seal the core from environmental effects such as moisture, mandating the use of impervious materials for face sheets. However, there are applications where face sheet materials do not seal the core materials, such as in some acoustic panel applications. A few of the face sheet materials typically used are discussed briefly in the following sections.

3.2.1 Description of face sheets

Face sheet materials are presented in this section corresponding to the general method in which the face sheet is incorporated into the sandwich structure resulting in three basic categories: adhesively-bonded pre-fabricated face sheets, co-cured or co-bonded face sheets with adhesive, and self-adhesive face sheets (prepreg with no separate adhesive or liquid molding construction).

3.2.1.1 Adhesively-bonded pre-fabricated face sheets

Adhesive-bonded pre-fabricated face sheets enable the designer to use the sheet material properties of the face sheet material for structural sizing, as those in-plane material properties are generally not dependent upon the bonding operation. This category applies to all sandwich structures employing metallic face sheets, as well as pre-cured composite face sheets. The adhesive used is generally a film adhesive but may also be a paste adhesive.

3.2.1.2 Co-cured or co-bonded face sheets with adhesive

Many composite face sheet sandwich structure configurations require adhesive between the face sheet and core in addition to the resin in the constituent face sheet material. Sandwich co-cure is defined as an approach where the adhesive and both face sheets are assembled to the core in the uncured state and the adhesive and face sheets are cured in a single cure cycle.

Sandwich co-bond refers to an approach where one of the face sheets is pre-cured and the other face sheet is cured simultaneously with the adhesive that bonds the core to that face sheet (and even perhaps the adhesive that bonds the core to the pre-cured face sheet). This alternate approach is used for instance, to fabricate doors where the overall thickness and finish of both face sheets must be controlled; in this method one face sheet and its adhesive are cured to the core in one cycle; the core is then

machined to the required profile, and finally the second face sheet plus adhesive is cured to the core in a second cure cycle.

Face sheet properties for this category are highly dependent on processing parameters and core configuration and geometry. Curing the face sheet materials directly to a honeycomb core may result in waviness and/or dimpling of the face sheet plies adjacent to the core. Resulting properties of the sandwich should be established by testing representative sections employing production processing techniques and representative tooling approaches.

It should be noted that in some cases, the film adhesive is applied and cured to sections of core that have been shaped or will be shaped to a contour or a specific shape; this is done in order to stabilize the core and not necessarily to enhance the strength of the core-to-face sheet bond.

3.2.1.3 Self-adhesive face sheets

Self-adhesive face sheets refers to a category of composite face sheet materials that are assembled in their uncured state with the core during layup operations, relying exclusively on the face sheet resin material to bond the face sheet to the core. As implied by the category title, no separate adhesive is used to establish face sheet-to-core bonding.

For sandwich structures employing prepreg face sheets, this approach may require prepreg with special resin formulation and/or higher resin content, compared to prepregs that are used for solid laminates. This category also applies to sandwich structures fabricated using liquid molding processes such as resin transfer molding (RTM), vacuum-assisted RTM (VARTM), and even wet lay-up.

As previously discussed, the mechanical properties of self-adhesive face sheets may be highly dependent on processing parameters and core configuration and geometry. Specifically, waviness and/or dimpling of face sheet materials may result due to unsupported spans between cell walls of a honeycomb or corrugated core. Actual properties of the sandwich should be established by testing representative sections employing production processing techniques and representative tooling approaches.

3.2.2 Face sheet properties

Metallic face sheet material properties are available in commercial sources such as Metallic Materials Properties Development and Standardization (MMPDS) (Reference 3.2.2) handbook for design values. Composite face sheet properties are generally determined using traditional analytical techniques as described in Volume 3 with values provided for some materials in Volume 2 of this handbook, but consideration must be given to properties that may be affected by processing as discussed above. In all cases, properties should be adjusted based on testing of representative components fabricated using representative production materials, processing, and tooling.

3.3 ADHESIVES

Adhesives are used for numerous purposes in sandwich structures. Their primary purpose is to structurally attach the face sheets to the core, reacting shear and peeling forces between the face sheets and the core, and enabling the materials to work together as a system. Adhesives are also used to splice or stabilize cores, and to bond face sheets to fittings, reinforcing plates, edge strips, and other inserts.

Adhesives are varied in their forms and types, but also in their chemistries, each resulting in different ranges of operating conditions and performance standards as described in the following sections.

Adhesives are first described in general terms, with industry specifications listed as applicable in Table 3.3.2, and then the various forms and types of adhesives are discussed including their typical uses. The chemistries of various adhesives are then presented with discussions regarding their use along with various applications for each. Typical adhesive properties are provided in Table 3.3.5 for reference pur-

poses only at the end of this section, and processing parameters are dependent upon the specific formulation.

3.3.1 Description of adhesives

Adhesives are compounds that are used to adhere (bond) or structurally join items together. The types of materials that can be structurally joined using adhesives are virtually limitless, but they are especially useful for bonding thin materials and are, therefore, invaluable in sandwich construction. Adhesives used in sandwich construction are typically synthetic and polymer-based, usually require a controlled temperature to cure or set, and are typically controlled through procurement and processing specifications depending on the materials and the application.

3.3.2 Adhesive specifications

Adhesive industry specifications are listed in Table 3.3.2.

TABLE 3.3.2 *Adhesive material specifications*

Specification	Title	Status (Publication Date)
SAE AMS 3686A	Adhesive, Polyimide Resin, Film and Paste High Temperature Resistant, 315 Mdc (519 Mdf)	Current
SAE AMS 3688B	Adhesive, Foaming, Honeycomb Core Splice, Structural, -55 to +82 Mdc (-65 to +180 Mdf)	Current
SAE AMS 3689B	Adhesive, Foaming, Honeycomb, Core Splice, Structural, -54 to +177 Mdc (-65 to +350 Mdf)	Current
SAE AMS 3690C	Adhesive Compound, Epoxy, Room Temperature Curing	Current
SAE AMS 3692C	Adhesive Compound, Epoxy Resin, High Temperature Application	Current
SAE AMS 3695	Adhesive Film, Epoxy Base, for High Durability Structural Adhesive Bonding	Current
SAE AMS 3695/1	Adhesive Film, Epoxy Base, High Durability, for 95 Mdc (200 Mdf) Service	Current
SAE AMS 3695/2	Adhesive Film, Epoxy Base, High Durability, for 120 Mdc (250 Mdf) Service	Current
SAE AMS 3695/3	Adhesive Film, Epoxy Base, High Durability, for 175 Mdc (350 Mdf) Service	Current
SAE AMS 3695/4	Adhesive Film, Epoxy Base, High Durability, for 215 Mdc (420 Mdf) Service	Current
SAE AMS 3705A	Epoxy Resin, Cycloaliphatic Liquid	Current
SAE AMS 3710C	Sandwich Structures, Glass Fabric-Resin, Low Pressure Molded, Heat Resistant	Current
SAE AMS 3728A	Epoxy Cresol Novolac Resin Low Molecular Weight	Current
SAE AMS 3729A	Epoxy Resin Matrix, Thermosetting Moderate Temperature Resistant Unfilled	Current
SAE AMS A25463	Adhesive, Film Form, Metallic Structural Sandwich Construction	Current

3.3.3 Adhesive forms/types and uses

Adhesives used in sandwich construction are available in various forms and types with some better suited than others, depending on the application and the fabrication processing for the component. Film adhesive supported by a fibrous carrier is normally used to attach face sheets to honeycomb core in composite structural honeycomb sandwich assemblies.

Primary considerations for selecting an adhesive for sandwich structures are:

- Strength requirements
- Service temperature range
- Ability to form a suitable fillet at the cell wall – face sheet interface (for a honeycomb core)
- Compatibility of cure parameters for face sheet and adhesive (for co-cured or co-bonded face sheet)

Adhesives in sandwich structures are discussed in the following sections in terms of their original material forms prior to processing as part of the sandwich structure. The form of adhesive for a given sandwich structure design is often selected based on economic and weight considerations. For example, if adequate bonding between the face sheets and the core can be accomplished using the base resin from the prepreg face sheets as part of a co-cure process, then the added cost and weight of a separate film adhesive is eliminated.

Foaming adhesives are typically used in sandwich construction to splice core sections and fill otherwise void areas within the sandwich structure, but are generally not used to bond the core to face sheets.

3.3.3.1 Resins from self-adhesive face sheets

Self-adhesive prepreg materials discussed in earlier sections provide their own resin to bond the face sheets to the core material, by using the resin that flows from the prepreg as the processing temperature is elevated and the resin viscosity is lowered. More information regarding optimum fillet formation at the cell wall-face sheet interface is provided in Chapter 5 of this volume. This approach is often preferred from an economic standpoint as minimal material forms are used in the process, and the resulting weight of the structure is lower as added layers of adhesive are eliminated.

3.3.3.2 Film adhesives

Film adhesives are a family of adhesives that are provided in a semi-solid state and typically procured with a minimal-weight fibrous carrier (often knitted or non-woven mat fiber). This form is easy to handle and control during lay-up, and it provides a controlled thickness or amount of adhesive during lay-up. For sandwich with carbon fiber face sheets and metallic core, the film adhesive carrier can provide some degree of separation between the carbon and metallic constituents to minimize galvanic corrosion problems, but galvanic protection remains a design concern in aggressive operating environments. In terms of structural performance, film adhesives with knitted carriers can provide higher peel strength in a sandwich structure, while film adhesives with a mat carrier limit the co-mingling of the prepreg resin with the film adhesive, and may result in lower peel strength.

3.3.3.3 Paste adhesives

Paste adhesives are a family of adhesive materials that are provided in a semi-solid state and typically procured as either a single-part or two-part compound. Paste adhesives generally have a very short working life and require careful planning in order to ensure that the desired physical properties are maintained throughout processing. Paste adhesives are fairly inexpensive in general, but require some means to control bondline thickness. While they may be used to bond various parts into a sandwich assembly (e.g., metallic inserts) or to fill areas of core for stiffening/strengthening purposes, paste adhesives are generally not used for bonding face sheets to core.

3.3.3.4 Liquid resins

Liquid resins may be used to bond face sheets to core materials during liquid resin molding processes, provided that the core is relatively solid and non-porous such as balsa and foam materials. Liquid resin has been used in honeycomb core applications to bond the face sheets to the core, but only after the core has been sufficiently sealed at the face sheet interface such that the cells are not filled with neat resin.

3.3.3.5 Foaming adhesives

Foaming adhesives are used to splice core sections when the size of the part exceeds that which is available in standard core sheet stock sizes, or when an area within the sandwich will otherwise result in a void or unsupported section of face sheet material. Foaming adhesives are also used for bonding replacement sections of damaged core for repairs. Foaming adhesives contain blowing agents that produce gases (e.g., nitrogen) during heating to provide the expansion necessary to fill gaps in the core sections as well as other void areas and provide adhesion between the open cell walls of adjacent core sections.

3.3.4 Adhesive chemistries

Adhesives are available as organic or synthetic materials, but the present discussion is limited to polymeric (synthetic) adhesives. These compounds vary in chemistry as discussed in the following section, with each offering unique characteristics as applicable to sandwich construction. This list is not intended to be an exhaustive representation but is provided to depict the most commonly used adhesives. More information regarding adhesive chemistries is available in ASM Volume 21 Composites (Reference 3.3.4).

3.3.4.1 Epoxy

Epoxy resins or epoxies, so called because of the epoxirane functional group responsible for chemical conversion of liquid resin to hardened adhesive, are a class of thermosetting resins used extensively in structural composite applications.

Epoxy-based adhesives are most common as two-part paste and one-part frozen film adhesives. The two-part paste systems are typically curable under ambient conditions, while films require elevated temperature for activation. Their availability in such a wide variety of physical forms (ranging from low-viscosity liquid to high-melting solids) adds to their adaptability to numerous processing techniques and applications.

Epoxies offer relatively high strength and modulus, low levels of volatiles, low cure shrinkage, good chemical resistance, ease of processing, and excellent adhesion to a range of substrate materials. Among the disadvantages of epoxy adhesives are the mixing requirements (two-part systems), limited pot life (both two-part paste and one-part films), relative brittleness, and a reduction of properties with continued exposure to moisture. The processing or curing of epoxies is often slower than polyester resins and the cost of the resin is also higher than the polyesters. Typical cure temperatures for epoxies range between 250° and 350°F (120° and 180°C). The use temperatures of the cured structure will vary with the cure temperature; higher temperature cures generally yield higher glass transition temperatures (Tg's) and, therefore, higher end use temperatures.

3.3.4.2 Bismaleimide

Bismaleimide (BMI) resins, so called because of the two maleimide chemical moieties responsible for cure conversion, are a class of addition-type polyimide thermosetting resins. They are used most often in high temperature applications due to their excellent physical property retention at elevated temperatures and in wet environments. The processibility and balance of thermal, mechanical, and electrical properties provided by BMI resins have made them popular in advanced composites and electronics applications.

Increased toughness, thermal stability beyond the range of typical epoxies, and reduced moisture absorption are provided by newer, advanced BMI resin systems.

Typical cure temperatures for BMI resins range between 350° and 400°F (175° and 205°C). However, in order to achieve optimal properties, a secondary higher temperature post cure up to 440°F (225°C) is essential to obtain a higher end use temperature, with a higher Tg of 540°F (282°C). The post cure results in an increase in degree of conversion in the matrix that significantly improves thermal stability (Tg), but at the expense of compliance (toughness) in the resulting system.

3.3.4.3 Phenols

Phenols, sometimes called phenolics because of the hydroxyl-group bonded directly to an aromatic hydrocarbon, are a class of thermosetting resins that are used in applications that require their excellent fire resistance, high-temperature performance, long-term durability, and resistance to hydrocarbon and chlorinated solvents. Phenolics are broadly used in the walls, ceilings and floors of aircraft interiors, so that passengers may have increased evacuation time during an airplane fire as mandated by certification agency flammability related airworthiness regulations. Phenolics have gained increased popularity with non-aerospace applications such as mass-transit, marine, mine ducting, and offshore structures where stringent fire resistance requirements have become common. Phenolic resins are also used extensively in the fabrication of honeycomb core materials where fire resistance is required.

Typical cure temperatures for phenolic resins have a large range (between 75° and 475°F (25° and 245°C), depending on the specific formulation and required usage temperatures. Among the disadvantages of phenolics are their relative brittleness, volatiles generated during cure for some types, and modest health and safety issues for some formulations.

3.3.4.4 Polyester

Polyester resins, so called due to the unsaturated polyester resin forming the back-bone of the reactive unsaturated hydrocarbons, are a class of thermosetting resins that are relatively inexpensive and fast processing compared to epoxies. Polyester resins have good fatigue resistance, UV stability, and retain good performance in the presence of moisture. Polyester resins have excellent adhesion to glass fibers, but lack equivalent compatibility with carbon fibers. They are the most commonly used resins and adhesives in commercial marine and wind-power applications.

Polyester resins are very easily processed by mixing the resin with a small amount of catalyst to initiate the free-radical conversions and cure temperatures range from room temperature ambient to 350°F (180°C). Due to the long-chain intermediate polyester functionality, converted polyester adhesives are generally tougher than epoxies for a given thermal stability. However, due to the free-radical cure of the polyesters, the presence of oxygen can inhibit cure conversion, especially at the surface, resulting in a tacky surface if exposed to air during cure. In general, polyester resins are used for lower-cost applications and are preferred when quick processing is needed.

3.3.4.5 Polyimide

Polyimide resins may be either thermoset or thermoplastic depending on their formulation and processing. Polyimide matrix composites are used in high temperature applications where their thermal resistance, oxidative stability, low coefficient of thermal expansion, and solvent resistance justify their higher cost and processing difficulties. Their primary use is in circuit boards, high temperature structures, and aerospace applications. Polyimide resins typically require cure temperatures in excess of 550°F (290°C), consequently demanding special, higher-temperature bagging films, bleeder and breather cloths, and steel (or other) tooling that can accommodate the higher processing temperatures; standard lower cost nylon bagging films and polytetrafluoroethylene (PTFE) release films used with epoxies will not survive the processing temperatures required for polyimide resins.

3.3.5 Adhesive properties

Typical mechanical properties of various adhesives are given in Table 3.3.5. Statistical data are not currently available for the adhesive materials shown, but the values provided represent typical industry vendor information on commercially available products. Accurate design for a given application should be established using the specific approved properties based on the methods discussed in Chapter 2 for actual materials of construction.

TABLE 3.3.5 *Typical properties for some adhesive materials.*

Adhesive Type	Cure Temp.	Max. Service Temp.	Test Environment	Shear Strength	Shear Modulus	Tensile Strength	Modulus of Elasticity	T-Peel Strength
	°F	°F		ksi	ksi	ksi	ksi	lb/in
Phenolic Film	300 - 350	180	RTD (75°F)	2.8 – 5.4	7 - 13	2.7 – 4.3	20 - 40	35 – 62
			ETD (180°F)	1.6 – 3.0				25
Epoxy Film	250 - 350	180 - 300	RTD (75°F)	4.9 - 6.8	77 - 142	6.0 - 7.5	102 -345	30 - 43
			ETD (180°F)	3.2 - 5.7	25 - 84			20 - 34
			ETW (180°F, 85% RH)	0.9 - 5.3	2 - 83			
Bismaleimide Film	350+ 475 post-cure	450-550	RTD (77°F)	2.0 - 3.5				
			ETD (450°F)	1.5 - 1.7				
			ETW (200°F, 100% RH)	1.4 - 2.8				
Cyanate Ester Film	350 + 440 post-cure	375-450	RTD (77°F)	2.7 - 3.0				
			ETD (350°F)	2.4 - 4.2				
			ETW (160°F, 95% RH)	2.8 - 3.9				
Two-Part Epoxy Paste	RT up to 200	180-450	RTD (77°F)	3.2 - 6.2	50-212	4.7-6.7	100-620	25-50
			ETD (180°F)	0.9 - 4.1	30-135			
			ETW (145°F, 85% RH)	0.7 - 4.1	13-99			
Epoxy Foaming Film	250-350	250-350	RTD (75°F)	1.1 – 1.7				
			ETD (180°F)	1.1 – 1.9				

Data is typical data based on a survey of manufacturers' published data.
For specification values, consult specifications listed in Table 3.3.2, or the appropriate end user specification.

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CHAPTER 4 DESIGN AND ANALYSIS OF SANDWICH STRUCTURES

4.1 INTRODUCTION

A structural sandwich is a layered composite consisting of two or more thin face sheets, bonded to a comparatively thick, low-density core, usually a cellular material or corrugated sheet. This results in an efficient construction in which the face sheets react nearly all of the in-plane loads and flatwise bending moments, and provide nearly all of the bending rigidity to the construction. The core material resists shear forces resulting from loads normal to the sandwich plane and provides most of the shear rigidity for the sandwich. The choice of constituents depends mainly on the specific application and its design criteria. The design of a structural sandwich involves an integrated process of geometrical design and materials selection.

The sandwich is analogous to an I-beam, which is an efficient structural shape because as much material as possible is placed in the flanges situated farthest from the center of bending or neutral axis. Only enough material is left in the connecting web to make the flanges act in concert and to resist shear and buckling. In a sandwich the face sheets take the place of the flanges and the core takes the place of the web. The difference is that the core of a sandwich is of a different material from the face sheets and it is spread out as a continuous support for the face sheets, thereby stabilizing the face sheets against buckling or wrinkling. This analogy is illustrated in Figure 4.1.

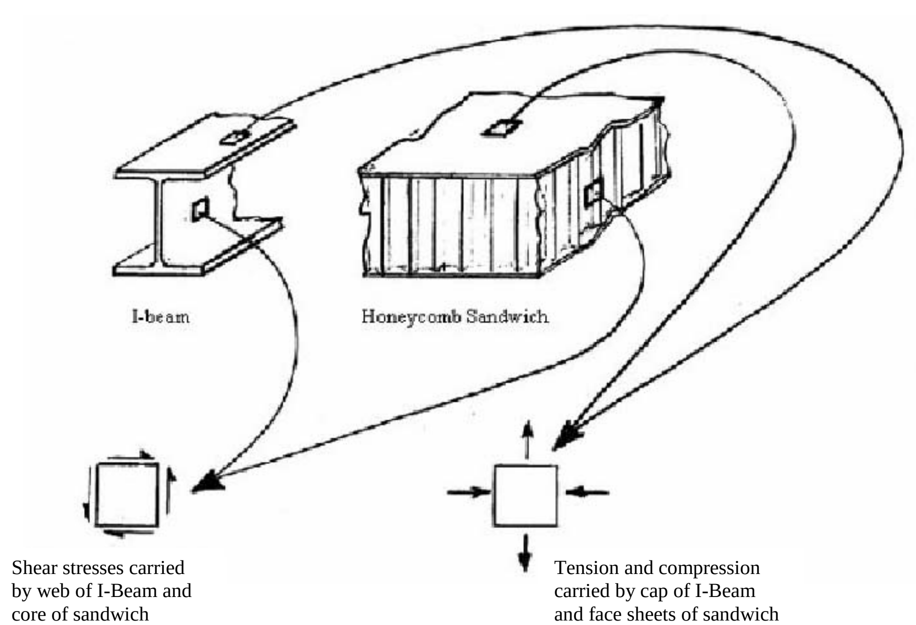


FIGURE 4-1. *Analogy between I-beam and sandwich structure.*

As a consequence of using a lightweight core, which generally has a low effective shear modulus, design methods must account for core shear deformation. One of the main differences in design analyses for sandwich versus solid laminate material elements is the inclusion of the effects of core shear properties on deflection, stress, and instability of the sandwich.

The bond between the face sheets and core must be strong enough to resist the shear and tensile stresses set up between them. Core materials are generally selected to be compatible with the face sheet material and the face sheet-to-core attachment methods used in fabrication (typically bonding or brazing). The face sheet-to-core attachment method is generally based on structural and environmental requirements for the sandwich system.

Assumptions, definitions, and nomenclature related to the analysis of sandwich structure are given in Section 1.4 of this Volume.

Many of the equations in Sections 4.7 through 4.11 and 4.13 were originally taken from Mil-Hdbk-23 (Reference 4.1), and are presented for isotropic face sheets. In most cases, an equation is given that applies to a sandwich with face sheets of different materials and/or thicknesses, and then a second equation is given for the simplified case where the two face sheets are the same. In general, the equations can be adapted for orthotropic face sheets, by using $E = [E_a E_b]^{1/2}$ in place of elastic modulus, E, and by using $\lambda = 1 - v_{ab} v_{ba}$ in place of $\lambda = 1 - v^2$ where E_a and E_b are elastic moduli, and v_{ab} and v_{ba} are the in-plane Poisson's ratios, in the directions aligned with the panel sides. In some cases, such as face sheet wrinkling calculations, it is more appropriate to use face sheet flexural stiffness in the loading direction in place of elastic modulus, E. See additional discussion in Section 4.5.2.

4.2 DESIGN AND CERTIFICATION

4.2.1 Basic design principles

The basic design concept for sandwich structures is to space strong, thin face sheets far enough apart to achieve a high ratio of bending stiffness to weight. The lightweight core between the face sheets must provide the required strength to resist the design shear, compression, and tension loads, while also being stiff enough to stabilize the face sheets in the desired configuration. The face sheets and core are joined through a bonding medium (i.e., adhesive, welding, or brazing), which must be capable of supporting and transferring all design loads.

Design of sandwich structures generally proceeds along the same basis as for composite laminate structures, but several failure modes and design issues specific to sandwich construction must be considered. Some of the more important issues specific to sandwich design are:

- Core shear
- Core crushing
- Core buckling
- Face sheet dimpling
- Face sheet wrinkling
- Face sheet buckling
- Strength of core-to-face sheet attachment
- Hardpoints (inserts and attachment points)
- Ramps (areas of transition from sandwich to solid laminate)

While the rest of this chapter provides detailed methods to address these and other sandwich-specific design issues, the basic principles for successful sandwich design can be summarized as follows:

1. Sandwich face sheets should be thick enough and strong enough to withstand design stresses under chosen design loads.
2. The core should be thick enough and/or dense enough to have sufficient shear rigidity and strength so that overall sandwich buckling, excessive deflection, and shear failure do not occur under design loads.

3. The face sheet and core should be stiff enough, and the sandwich should have great enough flatwise tensile and compressive strength, so that wrinkling of either face sheet will not occur under design loads.
4. If dimpling of the face sheets is not permissible and the core is a coarse cellular material (e.g., honeycomb) or of corrugated material, the cell size or corrugation spacing should be small enough so that dimpling of either face sheet into the core spaces will not occur under design loads.
5. The attachment between the core and the face sheets should be strong enough to withstand the chosen design loads. This is typically accomplished by an adhesive layer, but other options include welding or brazing of metallic core and face sheets, or self-adhesive composite face sheet material. The materials and fabrication processes should be selected so that the core, rather than the bond, is the weak link.

The choice of materials, methods of sandwich assembly, and material properties used for design must also be compatible with the expected service conditions. For example, face sheet to core bonding must have sufficient flatwise tensile and shear strength to develop the required sandwich strength in the expected service environment. Included in the service environment are effects of temperature, moisture, corrosive atmosphere or fluids, fatigue, creep and any other condition that may affect material properties.

Another major service issue that must be addressed during design is impact damage. Lightweight sandwich structures have been found to be susceptible to impact damage, and some types of impacts may leave significant damage that is not visible on the surface, so the degraded state of the structure is not apparent to the user. Design of the structure must take this into account through choice of materials, redundant load paths, etc.

Certain additional characteristics, such as thermal conductivity, resistance to surface abrasion, dimensional stability, permeability, and electrical properties of sandwich materials should also be considered in developing a sandwich design for the intended purpose.

4.2.2 Design process

The design and certification process for sandwich structures is very similar to that for solid laminate composite structures, although there are some unique aspects to sandwich structure development and certification that need to be addressed. The following paragraphs provide a summary of this process, with an emphasis on the unique elements for sandwich structure.

Many considerations must be taken into account in developing a successful composite structure. To address these, the designer must make decisions in a specific order, and the development program must follow a path that supports this decision-making. Table 4.2.2 gives a high-level overview of a typical design and development sequence for a generic composite structure. This is certainly not the only order that can be followed, and frequently some of these steps are done in parallel to save time. As well, there can be a wide variety of factors that constrain the decisions the designer makes. But this table provides a general “flavor” for the steps that must be followed.

Design decision-making for sandwich structures can be more complex than for other types of composite structure (e.g., solid laminate) because of the addition of cores and adhesives to the material/process selection list. As well, development testing needs to include an emphasis on in-service effects to which sandwich structures tend to be more susceptible, such as impact damage and fluid ingress. Impact damage is a concern because sandwich structures, especially light weight designs, can be easily damaged and the damage may not be apparent to users of the structure. For example, a localized impact such as a tool drop may crush an area of core, but the face sheet may rebound to its original position with no indication that damage has occurred. Fluid ingress is a concern because thin face sheets can develop microcracks over time, due to flexing, low energy impacts, etc., and these can allow fluids such as water, fuel, hydraulic fluid, and de-icing fluid to penetrate the face sheet. These fluids can attack the face sheet/core bond and, for open cell materials such as honeycomb, collect in the core. Collected water is a

particular problem for aircraft that fly to higher altitudes, e.g., 30 or 40 thousand feet, where it will freeze and expand, potentially debonding the face sheets from the core.

The final result of most development programs is formal certification (a.k.a. qualification) of the new structure. For aircraft structures, certification is a complex process with many steps, which overlays the development program, and requires the certification authority to oversee and, in many cases, witness the development steps listed in Table 4.2.2. Typically, it requires approval of the composite materials and the production processes to be used, as well as approval of the design of the structure. Section 4.3 below discusses the certification aspects of sandwich structure in greater depth. Volume 3, Chapter 3 of this Handbook provides an in-depth review of aircraft structure certification, highlighting the aspects that are unique to the certification of composite structures.

TABLE 4.2.2. *Typical composite structure design/development sequence.*

Development Steps	Design Decisions
Understand requirements (intended use, environment, geometry, loads, weight, cost, etc.)	
Understand available material choices and typical material properties	Preliminary selection of materials (face sheets, core, and adhesive) *
Understand available process choices (in-house, new process to be acquired, subcontract, etc.) and their effect on performance (properties, weight, cost, etc.)	Preliminary selection of processes * This includes attachment of core to face sheet (bonding, welding, brazing). For composite face sheets and adhesively bonded sandwiches, it also includes the curing process (autoclave, oven, press, resin transfer molding, etc.)
Perform preliminary design analyses	Preliminary selection of construction (number of face sheet plies, ply orientations, core density, core thickness, core ribbon orientation) * Selection of mold tooling approach (1-sided vs. 2-sided, composite vs. metallic tools, etc.)
Manufacture and test coupons to finalize material/design properties	Finalize material and process selections; qualify materials
Perform detailed design analyses	Finalize construction and minor design choices (fasteners, adhesives, seals, protective materials, etc.)
Manufacture and test prototypes (elements or small parts)	Refine design to optimize (for weight, cost, durability, etc.)
Manufacture and test full-scale part	Compare results to analyses and decide if acceptable
Finalize documentation (design definition, process documentation, design substantiation, etc.)	

* May result in a single choice or a short list of candidates.

4.2.3 Aircraft damage tolerance

This section focuses on aspects of damage tolerance that are unique to sandwich structure. For additional information on damage tolerance, see CMH-17, Volume 3, Chapters 3, 12, 13, and 17.

Sandwich structures can be very susceptible to damage. Hence, demonstrations of the damage levels that can be sustained at limit and ultimate loads are typically required for damage tolerance substantiation in the process of aircraft certification. Structures with realistic damage that may never be discovered either during manufacturing or during in-service inspections have traditionally been required to sus-

tain fatigue cycles without significant growth. A demonstration of such fatigue resistance usually culminates with a static strength test taken to ultimate loads. More severe composite damage, which is detectable using in-service inspection procedures, but which may escape regular simple inspections such as pilot walk-around checks, are required to withstand the more familiar damage tolerance practices of sustaining limit load for inspection intervals. Finally, structures with reasonable in-flight damage that occurs with knowledge to the aircraft operator are expected to sustain continued safe flight loads. Guidelines for achieving damage tolerance of sandwich structures are presented below.

Guidelines for Impact Damage Characterization Tests - A threat assessment is needed to identify impact damage probability, severity, and detectability for design and maintenance. A threat assessment usually includes damage data collected from service plus an impact survey. An impact survey consists of impact tests performed on a representative sandwich panel, which is subjected to boundary conditions characteristic of the real structure. Many different impact scenarios and locations are typically considered in the survey, which has as its main goal to attempt to identify the most critical impacts, i.e., those causing the most serious damage but that are least detectable. Until sufficient service experience exists to make good engineering judgments on energy and impactor variables, impact surveys should consider a wide range of conceivable impacts, including runway or ground debris, hail, tool drops, and vehicle collisions. Service data collected over time can better define impact surveys and design criteria for subsequent products as well as establish more rational inspection intervals and maintenance practices.

A typical method that is used to determine the ability of structure to withstand impact damage is to perform Tension, Compression, and Shear After Impact (TAI, CAI, SAI) residual strength tests using representative structural elements (see CMH-17, Volume 1, Chapter 7). Investigations based on test specimens impacted with a range of energies and impactor tip diameters (i.e., 1 inch and 3 inches) have found that CAI residual strength degradation curves possibly approach an asymptote as damage areas become large (see Figure 4.2.3(a) and Reference 4.2.3(a)). Test and analysis efforts should evaluate the potential for specimen size to effect the shape and asymptotic value of the CAI strength degradation curve.

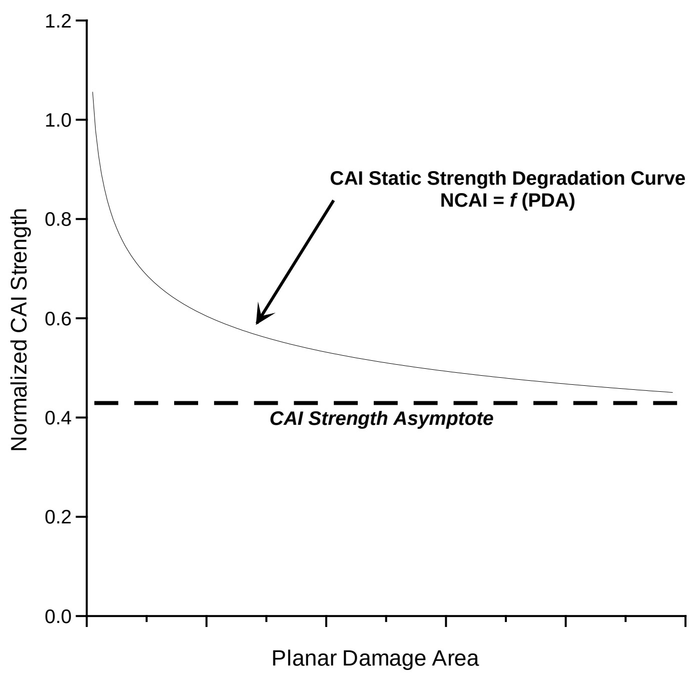


FIGURE 4.2.3(a). *CAI static strength degradation curve.*

Impact damage in sandwich panels caused by blunt impactors has been found to lead to one of two contrasting final failure modes under in-plane compressive loads. As load is applied, the impact damage,

which manifests in the form of a surface dimple, will propagate well before final failure occurs. The amount of this damage propagation will be dependent on the flexural properties of the face sheet, transverse compressive properties of the core, and the damage metrics. A thin face sheet with negligible flexural stiffness will promote either a strain concentration adjacent to the impact damage and a local compressive failure mechanism in the face sheet, or an outward buckling of the delamination area followed by either delamination growth or a local compressive failure mechanism in the face sheet. A thicker face sheet, given enough flexural stiffness and local damage to the core, will drive the impact dent through a characteristic sequence of events leading to a progressive core crush and a flexural face sheet failure mechanism. Both the strain concentration and stability-based failure mechanisms should be characterized in a damage tolerance test program whenever the threat assessment and sandwich design parameters indicate the potential for both mechanisms exists.

After performing a threat assessment and impact survey, CAI strength degradation curves should be developed for a representative range of impact damage threats. In combination, this data provides the information on damage detectability and residual strength needed to establish rationale for design criteria. As discussed previously, non-detectable damage should not lower the residual strength below ultimate load levels unless a probabilistic assessment can be made indicating that the impact scenario for such damage is extremely improbable. Large diameter impactors and some combinations of sandwich design parameters can lead to non-detectable damage with an apparent asymptote of the residual strength curve. When configuration and impact variables for a specific design can result in this scenario, the asymptotic value can be used as the allowable design strain level associated with ultimate static strength. This approach provides maximum airworthiness assurance of the sandwich structure under a variety of damage threats that may result in both undetectable and detectable damage. If a reliable NDI method is available and used in service, this rule of thumb can be relaxed.

Based on limited data, the fatigue threshold limit for 150,000 cycles has been found to be approximately at 65% of the static CAI strength for carbon/epoxy face sheet honeycomb panels and 75% for fiberglass/epoxy face sheet foam panels (Reference 4.2.3(b)). This data was derived for constant-amplitude testing. The design guideline is that the maximum compressive load in a fatigue spectrum should not exceed these thresholds. Repeated loads above the thresholds may lead to early fatigue failures, especially if such loads impinge on the static CAI strength distribution. For fatigue cycling below these limits, life will exceed 150,000 cycles of constant-amplitude fatigue, which is a very severe spectrum for most airplanes. Such cycling generally also has very little effect on residual CAI strength. This data should only be used for preliminary structural evaluations and should not be relied upon for structural certification. Fatigue data for the specific materials, layups, etc., should be obtained to validate no-growth of damages and post-fatigue residual strength for certification.

Guidelines for NDI of Impact Damage - Impact results from some experimental investigations with sandwich structure have indicated that larger-diameter impactors can produce significantly different damage states when compared to smaller impactors (at the same impact energy level) and that visual detection alone cannot be used to assess damage (reference 4.2.3(a)). The results indicated that the larger-diameter impactor produced a very benign-appearing damage state, wherein no surface fracture or cracks or visually perceptible levels of indentation existed, but NDI did indicate a large damage region, which was also evident through an observed reduction in strength. This damage scenario proved to be the most elusive when the impacted specimens were inspected using a typical visual inspection protocol. The results of these investigations showed that visual inspection methods can be misleading and that for some types of impacts, residual indentation cannot be used as a reliable damage metric for static ultimate strength and damage tolerance criteria for sandwich structures.

Under the assumption that additional field inspection techniques must be used to quantify the extent of damage in the structure, various field techniques for NDI were also investigated in the Reference 4.2.3(b) study. Based on the experimental results, it was concluded that the detection of impact damage in honeycomb and foam core sandwich panels cannot be accomplished to the same level of accuracy using a single field inspection technique when compared to laboratory techniques. The experimental data suggests that impact damage in honeycomb core sandwich panels is better detected by a technique that measures the local stiffness of the sandwich, while damage in foam core panels can be best assessed

with a technique relying on the measurement of acoustic impedance. The trends observed for foam core panels may be biased by the normalization procedure due to the inability to corroborate the damage size using destructive sectioning.

Using the honeycomb core panels as a baseline, it was also concluded that the reliability of detecting damage using field inspection techniques is generally reduced when compared to laboratory or manufacturing production C-scan methods (Reference 4.2.3(a)). Figure 4.2.3(b) shows the extent of damage that was detected using various field inspection instruments as a function of face sheet thickness. The error bars shown in the figure indicate the range of reported measurements. As the face sheet thickness increased, the ability of each field inspection technique to locate and characterize the extent of damage reduced, as shown in Figure 4.2.3(b). It should also be noted that the maximum number of plies in this study was six fabric plies. For thicker face sheets, the measurement reliability may decrease further. Thus, the measured damage in the laboratory may actually be two times greater than that measured in the field, which needs to be taken into account in the damage tolerance and inspection plans developed based on allowable damage limit and/or critical damage threshold size.

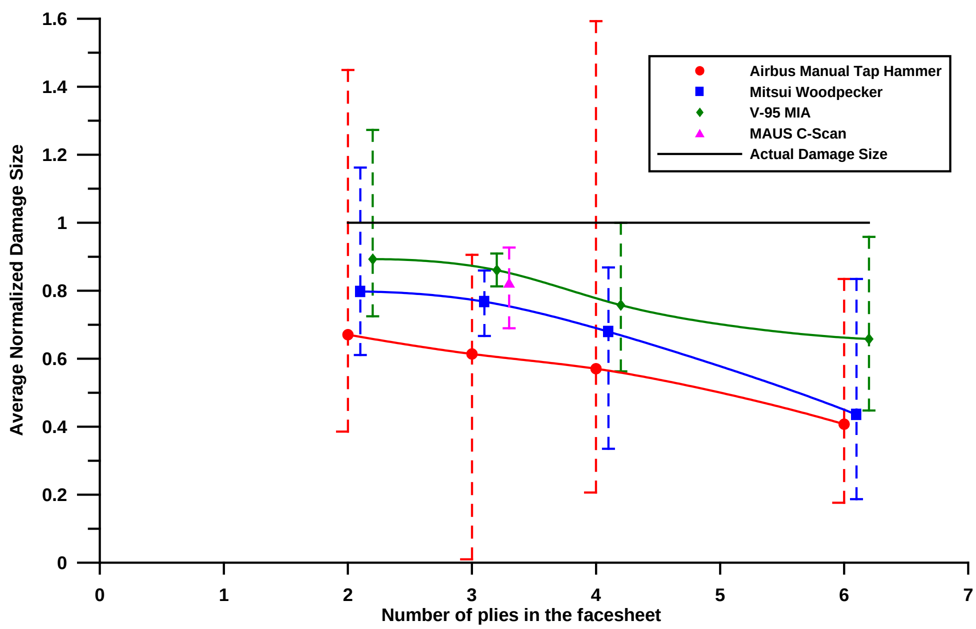


FIGURE 4.2.3(b). *Detected damage size based on damage diameter using different field detection instruments as a function of face sheet thickness.*

In the beginning of a damage tolerance test program, it is advisable for the designer to define a similar detectability ratio plot, as shown in Figure 4.2.3(b), to account for the sensitivity of the field inspection technique used for the specific design detail of a given aircraft product. All efforts in establishing reliable NDI for field inspection should be coupled with work on the threat assessment, impact survey, residual strength, and fatigue testing. The relationships developed should then be incorporated into an appropriate inspection plan for damage throughout the service life of the structure.

Guidelines for Analysis of Impact Damage - Analysis, whether performed by closed form solutions or finite element models, can be useful in design and certification of impact-damaged composite sandwich structures. The current state of the art is such that analysis cannot be reliably applied without supporting test data. However, it can be useful in directing and analyzing test results and in expanding test data to untested configurations by semi-empirical methods. The analytical work completed and documented in

Reference 4.2.3(c) showed that careful modeling can describe the structural response of the impacted panel under compressive load, but some adjustments were needed to predict failure. The analysis performed also showed that damage progression prediction capability is vital to the fidelity of the analysis.

Statistical response surfaces can be used as guidance to determine what size of damage is expected in terms of the pertinent impact variables. This will reduce hunt and peck testing to find the energy levels for barely visible damage or planar damage when interpolating within the variables studied. Furthermore, response surface damage metrics can be used as input in consequent structural analysis methods.

4.3 CERTIFICATION

The following paragraphs provide an introduction to the subject of certification as it relates to composite structures, with a particular emphasis on issues relevant to sandwich structures. It is written from the perspective of certification of aircraft structure, since that is the experience base of the authors, though many of the issues and considerations presented here apply to certification of any type of composite structure. The following discussions only cover the highlights of the subject. For a more detailed review of composite certification issues, refer to Volume 3, Chapter 3, of this Handbook. For a definitive understanding of the regulations and certification issues applicable to a specific structure, refer to the appropriate certifying agency.

4.3.1 Introduction to certification issues

Certification is the process by which an applicant formally shows to a certifying agency that a given design, or product, has satisfied all of the applicable requirements, and culminates with a certificate being issued by the certifying agency indicating that the design, or product, is fit for production and/or use. Certifying agencies typically are government agencies. Certification is applicable to design, production, and modification (including continued airworthiness issues). Typical requirements (or regulations) for aircraft are found in: FAR's (US), CS standards (Europe, previously JAR's), and CAR's (Canada). The certifying agencies have also produced a number of guidance documents providing general information regarding the issues and approaches for certification of composite aircraft structures. Principal amongst these are the US Federal Aviation Administration's AC 20-107 (Reference 4.3.1(a)), the European EASA's AMC 20-29 (Reference 4.3.1(b)), and Transport Canada's AMA 500C/8 (Reference 4.3.1(c)). A useful discussion of the generic issues associated with composite structure airworthiness and certification is provided in Reference 4.3.1(d) Chapter 13.

The major issues for composite structures that need to be addressed to the satisfaction of the certifying agency are: suitability and control of materials and manufacturing processes; substantiation of the design; allowance for effects of the in-service environment and damage events; and plans for ensuring continued airworthiness for the life of the structure. All of these issues are common to both solid laminate and sandwich composite structures. However, a specific concern for sandwich constructions is control of the manufacturing processes used to bond the face sheets to the core, since this is a principal load-path for the structure, and inspection of this bond can be difficult. As well, damage and environmental effects are issues of particular concern since sandwich structures tend to use relatively thin face sheets for their main load-bearing members and the adhesive bonds joining the core to the face sheets can be sensitive to these effects.

Composite materials have tended to be more brittle than metallic materials and more sensitive to moisture and temperature. These characteristics can contribute to much greater scatter in static strength and fatigue results. Impact damage and its resulting (often invisible) effects, can seriously weaken composite structure. In particular, some sandwich constructions have shown a tendency for face sheets to rebound after impact events, hiding face sheet-to-core disbonds or core crush damage that may have occurred. Micro-cracking of face sheet laminates due to impact events or repeated load cycles can allow moisture ingress that can compromise the face sheet-to-core bond or result in significant water accumulation in porous core (e.g., honeycomb). Accumulated water can result in serious internal pressure problems for sandwich constructions when frozen.

Certification and the showing of compliance for sandwich structure should include significant focus on these issues, although other cases should not be forgotten. Also, application-specific effects may need to be addressed, for example fatigue can be critical for rotorcraft and certain other structures.

4.3.2 Approach to certification testing

To a large extent, these certification issues can be managed through the design allowables development process by use of a Building Block approach (see CMH-17, Volume 3, Chapter 4) based upon testing statistically significant numbers of specimens of varying complexity, e.g., simple coupons through to complete components, both statically and in fatigue, while including the effects of humidity, temperature, damage, and other in-service conditions. Various sandwich-specific test methods have been developed to determine the mechanical behavior of these constructions and should be included in a Building Block program. To support design substantiation, the selection of specimens should be justified, showing clearly that all likely ‘hot spots’, e.g., joints, ply drops, cutouts, areas of high stress, etc., have been addressed.

Both static and fatigue substantiation should include consideration of moisture and temperature effects. Ideally, the most critical structures should be tested at the critical environment (often hot/wet) for all critical load cases. However, it is common to test the higher level components at room temperature with appropriate knockdown factors, Load Enhancement Factors (LEFs), etc.

When showing that fatigue and damage tolerance requirements have been satisfied, it is usual for composite structures to be configured with both manufacturing flaws and in-service damage. These test articles are typically required to show no damage growth during the appropriate test period, due to the limited existing data and limited confidence in predictable and repeatable damage growth behavior in composites.

Another certification issue of general concern for composites and particular concern for sandwich constructions is exposure to fluids other than water. The different types of fluids that may come in contact with parts in service, e.g., fuel, anti-icing fluids, hydraulic fluid, lubricants, etc., and their effect on strength degradation, should be evaluated. Available information on chemical compatibility of candidate materials with these fluids should be reviewed at the time of material selection. Material qualification tests should be used to complete the assessment during the certification program.

4.3.3 Analysis validation

Since it is impractical to test for all load conditions that occur in service, full-scale structural tests are generally conducted for the most critical conditions and for validation of the applicant’s analysis method/results (generally finite element method for full airframes). The certification applicant bears the responsibility to the regulating authority to demonstrate that it can determine the internal load distribution to an agreed upon accuracy level for certain load conditions. Once the analysis is correlated through model adjustments, and the analysis results match up to test results, the analysis is accepted as a validated method of compliance supported by test data. Afterwards, it may be used for Proof of Static Strength for various load conditions with minimum material strengths established by the Building Block test program.

4.3.4 Conformity oversight

In order to demonstrate compliance to certification regulations to the regulating authority, it is common practice for all test articles (coupons, sub-elements, elements) to be built to the Type Certificate design. This is performed for multiple reasons such as:

- To demonstrate to regulators that the article can be built to the type design configuration.
- To assure that conforming type design articles are used for tests.
- To validate the applicant’s process, material, NDT, or other special manufacturing procedures, specifications, and controls.
- To assist in establishing a fabrication and quality assurance program.

4.3.5 Nondestructive testing (NDT)

It is required to validate that the selected NDT methods can detect defects (i.e., cracks, voids, porosity, face sheet-to-core disbonds, and foreign materials) within the limits of type design. Test articles are inspected to verify the state of the articles prior to test, including any simulated manufacturing flaws, as well as to ensure readiness of the selected NDT methods for use in manufacturing production.

Additional information on inspection and NDT may be found in Sections 5.5, 6.3.2, and 7.2.2 of this Volume.

4.3.6 Documentation requirements

Required documents to support certification include drawings or Computer Aided Design (CAD) models of the parts and tools, material specifications, process specifications, manufacturing instructions, structural repair manual, Material Review Board (MRB) system procedures, and engineering operating procedures. The drawings must include all the details of part lay up (e.g., number of plies, orientation of plies, types of materials, dimensions of each ply, and where plies are located with respect to each other). The drawings must also list notes dealing with critical features of the particular part and specify geometric tolerances. Material and process specifications, and manufacturing instructions used to produce composite structures, must contain sufficient information on critical parameters and inspections to facilitate repeatable production. Similarly, maintenance manuals should define acceptable inspection procedures and repair methods.

4.3.7 Continued airworthiness

The objective of maintaining continued airworthiness is to ensure that the aircraft continues to satisfy, or exceed, the requirements met at certification throughout its life, taking into account the environment, repairs, Supplemental Type Certificates (STC's), etc.

Design organizations, including those responsible for STC's, are expected to show that continued airworthiness will be maintained for the aircraft, e.g., per 14 CFR 25.1529, Part 21 Subpart H. In particular, it is the responsibility of any STC holder to show that any STC installation will not adversely affect the continued airworthiness of the original structure, or any other STC structure (e.g., through changes to loads, through inflicting damage or reducing damage tolerance, through reducing access for inspection, etc.).

It may prove difficult to confirm that continued airworthiness issues have been fully addressed at the time of original certification because the only evidence that is likely to be available is a plan, e.g., draft Structural Repair Manual (SRM), Maintenance Schedule, etc., and/or possibly evidence of the approach taken from previous programs by that Original Equipment Manufacturer (OEM). Again, STC holders should pay particular attention to satisfying the certifying agency regarding this issue.

Note that maintaining continued airworthiness is the responsibility of all parties involved with the aircraft during its life, including design, production, maintenance, and operating organizations.

4.4 SANDWICH PANEL FAILURE MODES

Sandwich panels can fail in several ways, each mode giving one constraint on the load bearing capacity of the sandwich. Depending on the geometry of the panel and the loading, different failure modes become more critical and set limits on the performance of the structure. Failure of the sandwich may be driven by the strength of the face sheet, core, or adhesive, by a local instability mode such as face sheet wrinkling or dimpling, or by general instability such as general buckling or shear crimping. These failure modes are illustrated in Figure 4.4(a) and are briefly described as follows:

Face sheet failure occurs when one or both of the face sheets fails by yielding or fracture. The criterion for failure is that the face sheet material exceeds its allowable stress or strain.

Core shear failure occurs when the core fails in shear, usually resulting in cracks inclined at 45 degrees to the midplane. The core material is mainly subjected to shear since it carries almost the entire transverse load, and very little in-plane load. Honeycomb core may fail by cell wall buckling, which may not be visible after load is removed.

Core crushing is when the face sheets move towards each other under the influence of bending or through-thickness loads. This failure mode occurs when the core has insufficient compressive strength.

Core tensile failure occurs when the core has insufficient flatwise tensile strength.

Face sheet-to-core debonding occurs when the face sheet-to-core bond has insufficient shear, peel, or tensile strength.

Local indentation occurs at concentrated loads, such as fittings, corners, and joints. When point loads are applied, the face sheet acts as a plate on an elastic foundation. The loaded face sheet bends independently of the opposite face sheet, and if the stress induced in the core exceeds the core's compressive strength, the core will fail. This failure mode may be avoided by spreading the load over a sufficiently large area.

Face sheet wrinkling is a local instability characterized by buckling of the face sheet, often accompanied by core crushing, core tearing, or face sheet-to-core debonding. This failure is most prevalent with thin face sheets and low density core. One or both face sheets may wrinkle, depending on the loading, materials, and thickness of core and face sheets. Figure 4.4(a) shows the cases where both face sheets are wrinkling, in the symmetric and antisymmetric modes.

Face sheet dimpling*, also known as* ***intracell buckling***, is a local instability characterized by the buckling of a face sheet into or out of the confines of a single cell. This failure can occur when the face sheets are thin and cell size is large.

General buckling of a sandwich panel resembles the classical buckling of plates or columns. The face sheets and core remain intact in this type of failure.

Shear crimping is an instability that can occur if the wavelength of each buckle is of the same order as the cell size. The crimping phenomenon is characterized by a local core shear failure and the lateral dislocation of the face sheets. Since the wavelengths are so short, shear crimping appears to be a local instability failure, but it is really a form of general instability. This failure mode can occur when the core shear modulus is low.

Each of these failure modes is discussed in more detail in Section 4.6.

Flatwise tension or compression is out-of-plane loading, i.e., loading through the thickness of the sandwich panel. A flatwise tensile stress can cause core tensile failure or face sheet-to-core debonding, while flatwise compressive stress can cause core crushing. Flatwise stresses may be due to applied out-of-plane loads, especially at inserts or joints. Flatwise stresses also occur in ramp areas where core thickness changes and one face sheet changes direction, inducing flatwise, or interlaminar, stresses at the ramp radii.

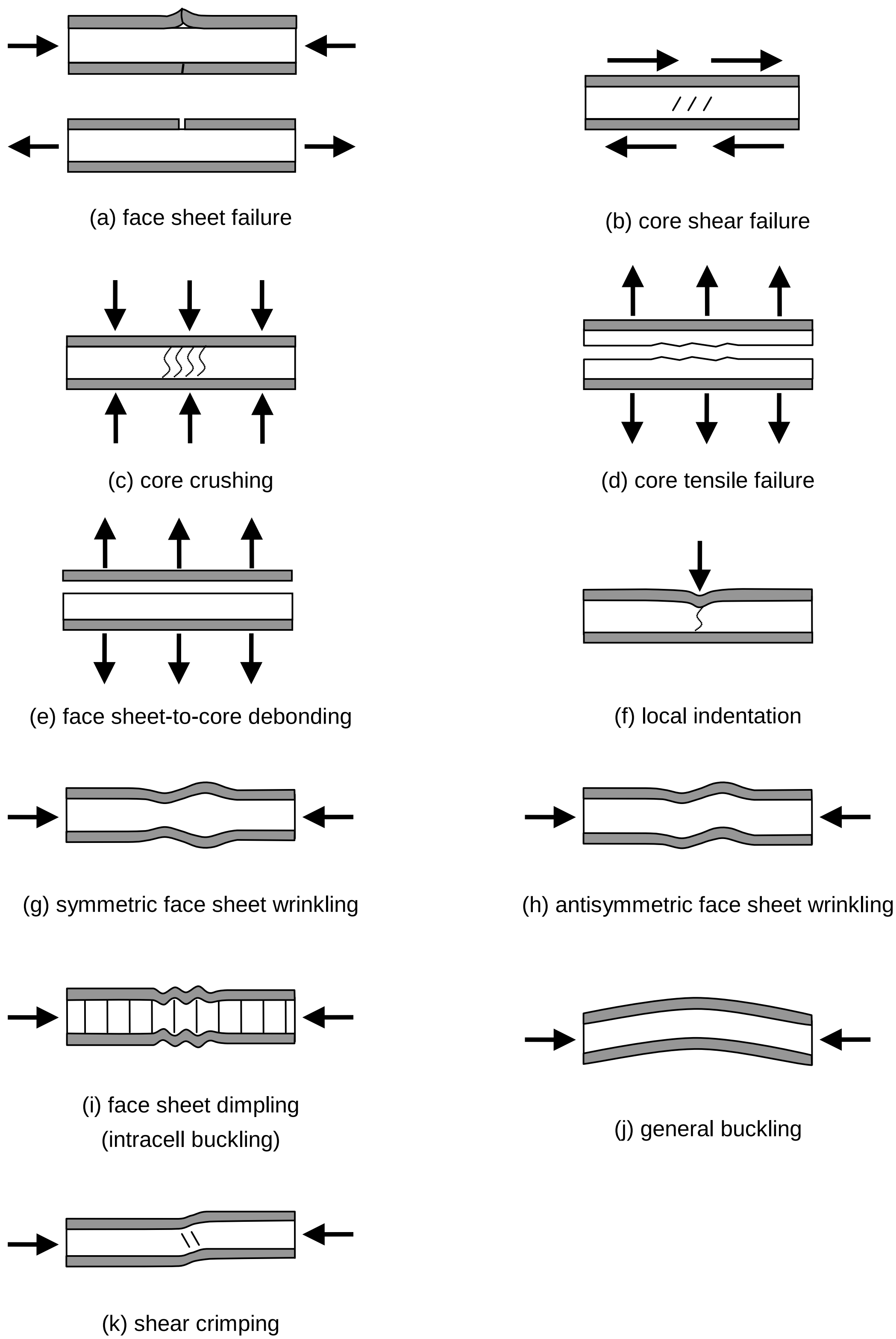


FIGURE 4.4(a). *Sandwich failure modes.*

Areas where the sandwich transitions to a solid laminate are shown in Figure 4.4(b). In the area where one face sheet changes direction, the core is subject to flatwise tension and/or compression. In the case of flatwise tension, the face sheet-to-core adhesive must provide an adequate bond, and the core must have sufficient flatwise tensile strength to prevent failure. In the case of flatwise compression, the core must have adequate crushing strength. The shear force normal to the panel reaches a maximum at the fastener centerline. In the ramp region, the bag-side face sheet and the core both react the shear load. The core is checked for adequate shear strength in the entire ramp region, but it is often critical at the top of the ramp when pressure is the only applied load. At this location, the core is assumed to carry 100% of the out-of-plane shear. Flatwise tension and compression in ramp regions is discussed in Section 4.6.3.

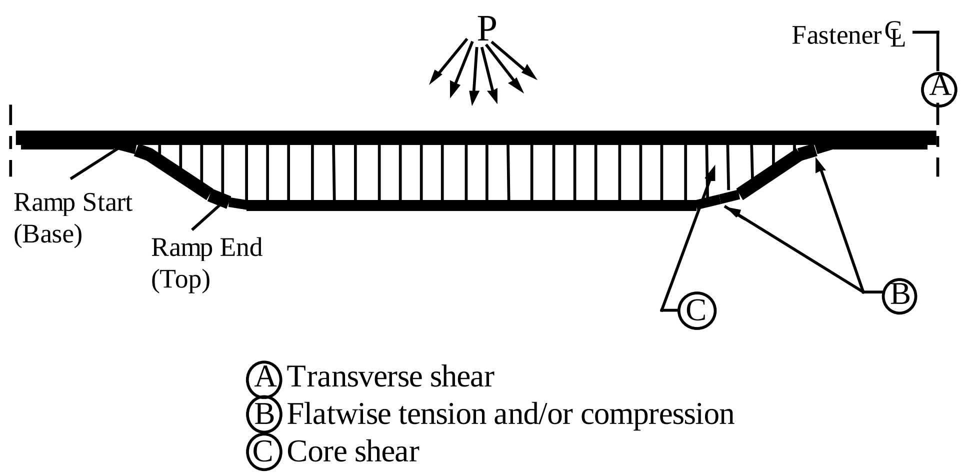


FIGURE 4.4(b) *Critical strength check for sandwich panel transitions.*

4.5 STIFFNESS AND INTERNAL LOADS

4.5.1 Beam stiffness analysis

The theory for sandwich beams is very similar to the ordinary engineering beam theory but with the addition of shear stresses and transverse shear deformations. This theory is usually referred to as Timoshenko beam theory. For simplicity, in the following all beams are assumed to have unit width, so all stiffnesses are also given per unit width.

Many methods are available to calculate the deflections, bending moments and shear force distribution in beams under various types of loading and boundary conditions. All these equations require the use of the beam bending stiffness, D. Note that D may be a function of direction if face sheets are anisotropic. D can be calculated from

$$D = \int E(z)\, z^2 dz \qquad 4.5.1(a)$$

For a symmetrical cross-section where the face sheets are of the same thickness and material, this becomes

$$D = 2D_f + D_o + D_c = \frac{E_f t_f^3}{6} + \frac{E_f t_f d^2}{2} + \frac{E_{cx} t_c^3}{12} \quad 4.5.1(b)$$

where $2D_f$ is the bending stiffness of the face sheets about their individual neutral axes, D_o is bending stiffness of the face sheets about the middle axis of the sandwich beam, D_c is the bending stiffness of the core, and d is the distance between the centroids of the face sheets. t_f and t_c are the thickness of the face sheet and core, respectively. E_f and E_{cx} are the stiffness of the face sheet and core, respectively, in the direction of the beam long axis.

If the face sheets are thin compared to the core, that is,

$$\text{If} \quad 3\left(\frac{d}{t_f}\right)^2 > 100 \quad \text{or} \quad \frac{d}{t_f} > 5.77, \quad \text{then} \quad \frac{2D_f}{D_o} < 0.01 \quad 4.5.1(c)$$

Similarly, if core has low stiffness compared to the face sheets, that is,

$$\text{if} \quad \frac{6E_f t_f d^2}{E_{cx} t_c^3} > 100, \quad \text{then} \quad \frac{D_c}{D_o} < 0.01 \quad 4.5.1(d)$$

If both Equations 4.5.1(c) and (d) are true, then the bending stiffness can be approximated by

$$D = D_o = \frac{E_f t_f d^2}{2} \quad 4.5.1(e)$$

For a non-symmetrical cross-section, that is, one with dissimilar face sheets, the bending stiffness includes terms to account for the position of the neutral axis being off the midplane of the beam. Using e, the distance from the midplane of the lower face sheet to the neutral axis of the beam, the bending stiffness becomes

$$D = \frac{E_{UPR} t_{UPR}^3}{12} + \frac{E_{LWR} t_{LWR}^3}{12} + \frac{E_{cx} t_c^3}{12} + E_{UPR} t_{UPR} (d-e)^2 + E_{LWR} t_{LWR} e^2 + E_{cx} t_c \left(\frac{t_c + t_{LWR}}{2} - e\right)^2 \quad 4.5.1(f)$$

where UPR and LWR indicate the upper and lower face sheets, respectively.

If the stiffness of the core is low compared to the stiffness of the face sheets, this becomes

$$D = \frac{E_{UPR} t_{UPR}^3}{12} + \frac{E_{LWR} t_{LWR}^3}{12} + \frac{E_{UPR} t_{UPR} E_{LWR} t_{LWR} d^2}{E_{UPR} t_{UPR} + E_{LWR} t_{LWR}} \quad 4.5.1(g)$$

And if the core has low stiffness and the face sheets are thin, then the rigidity can be approximated as

$$D = D_o = \frac{E_{UPR} t_{UPR} E_{LWR} t_{LWR} d^2}{E_{UPR} t_{UPR} + E_{LWR} t_{LWR}} \quad 4.5.1(h)$$

Note that for wide beams $(b/d>6)$, one should use $D = D_{11}$ where D_{11} is the plate bending stiffness in the axial direction. Calculation of this quantity will be discussed in the Section 4.5.2.

Note also that in general the transverse shear deformation of the core cannot be neglected. For a sandwich beam, the shear stiffness must be computed using an energy balance equation. The shear stiffness S, is found from the differential equation relating transverse deflection, w_s, and transverse shear force, V_x:

$$\frac{d^2 w_s}{dx^2} = \frac{1}{S}\frac{dV_x}{dx} \quad \text{where} \quad V_x \frac{dw_s}{dx} = \int \tau_{xz}(z)\gamma_{xz}(z)dz \qquad 4.5.1(i)$$

Using the approximations for a sandwich where the face sheets are thin (as in Equation 4.5.1(c)) and the core stiffness is low (Equation 4.5.1(d)), and assuming the shear modulus of the core, G_c, to be large,

$$S = \frac{G_c d^2}{t_c} \qquad 4.5.1(j)$$

For most sandwich applications, the difference between this approximation and the exact value is less than 1 percent.

The effect of the different degrees of approximation can be illustrated by considering the direct and shear stress distributions in the face sheets and core due to bending, as shown in Figure 4.5.1 for a symmetrical sandwich. The results reflect the principle of a sandwich construction: the face sheets carry the bending moments as tensile and compressive stresses and the core carries the transverse forces as shear stresses.

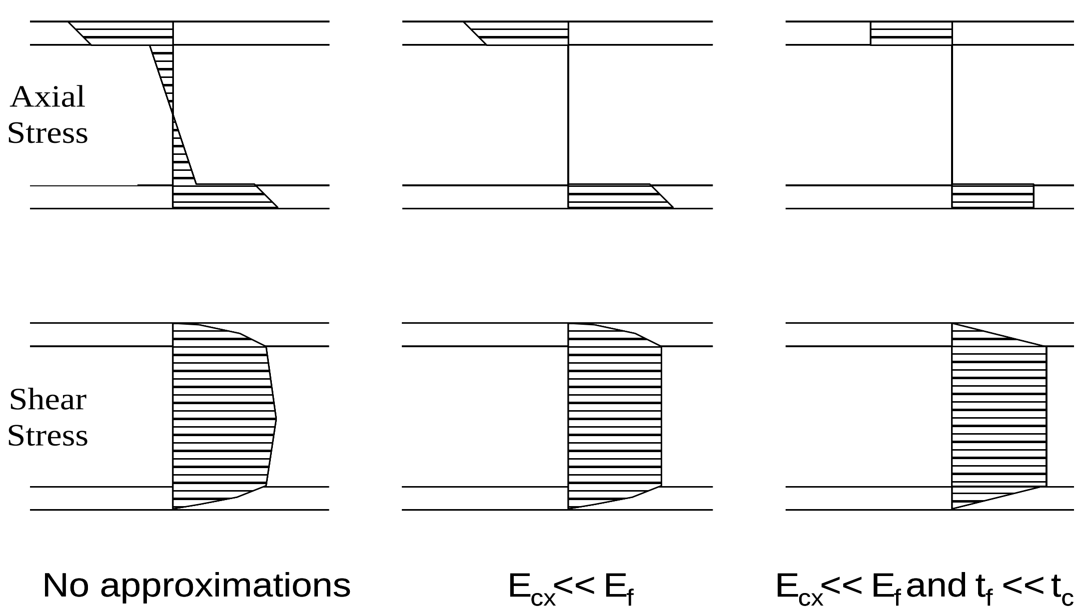


FIGURE 4.5.1 *Effect of different degrees of approximation for sandwich beam.*

A consequence is that for any sandwich construction, the deformation always consists of two parts: deformation due to bending and deformation due to shear. For instance, the transverse deflection at the free end of a cantilevered beam of length L and width b, subjected to a concentrated load, P, can be expressed as:

$$w_{max} = -\left(\frac{PL^3}{3Db} + \frac{PL}{Sb}\right) \qquad 4.5.1(k)$$

where the first term represents the deflection due to bending and the second term represents the additional deflection due to the transverse shear deformation.

Similarly, the maximum transverse deflection of a simply-supported beam subjected to a uniform distributed load, q, can be expressed as:

$$w_{max} = -\left(\frac{5qL^4}{384Db} + \frac{qL^2}{8Sb}\right) \qquad 4.5.1(l)$$

The deflection of a general beam can be determined using the following differential equation:

$$\frac{d^2y}{d^2x} = \frac{M_x}{D} + \frac{1}{S}\left(\frac{dV_x}{dx}\right) \qquad 4.5.1(m)$$

Once the maximum bending moment and shear resultant are obtained, the designer can determine the core stresses and face sheet line loads and strains, and then check the design principles described in Section 4.2 and the potential failure modes in Section 4.4.

4.5.2 Plate stiffness analysis

The first step in the analysis is to define the stiffness properties for the whole panel and for the individual face sheets. Additional information on laminate stiffness analysis, and calculation of the [A], [B] and [D] matrices, may be found in Volume 3, Section 8.3.

One approach is to use a laminate analysis program to calculate the different stiffness matrices. Calculate first the [A], [B] and [D] matrices of the whole panel; use the individual ply properties and assume that the core is a special type of ply with no in-plane stiffness, but with the correct transverse shear stiffness. The stiffness matrices relate the resultant loads, N, and moments, M, to the midplane strains, ε, and curvature, κ:

$$\begin{Bmatrix} N \\ M \end{Bmatrix} = \begin{bmatrix} A & B \\ B & D \end{bmatrix} \begin{Bmatrix} \varepsilon \\ \kappa \end{Bmatrix} \qquad 4.5.2(a)$$

Use a program that will also determine the transverse shear stiffness coefficients, A_{44}, A_{45} and A_{55} of the plate:

$$\begin{Bmatrix} Q_y \\ Q_x \end{Bmatrix} = \begin{bmatrix} A_{44} & A_{45} \\ A_{45} & A_{55} \end{bmatrix} \begin{Bmatrix} \gamma_{yz} \\ \gamma_{xz} \end{Bmatrix} \qquad 4.5.2(b)$$

If such a program is not available, assume that the core alone provides the transverse shear stiffness (valid if the face sheets are thin compared to the core, e.g., less than 5% of the core thickness) and calculate the coefficients as follows:

$$\begin{aligned} A_{55} &= t_c\, G_{xz} \\ A_{44} &= t_c\, G_{yz} \\ A_{45} &= 0 \end{aligned} \qquad 4.5.2(c)$$

Using the same laminate analysis program or hand analysis equations, determine the stiffness properties, i.e., the [A], [B] and [D] matrices, **of each individual** face sheet:

$$\begin{Bmatrix} N \\ M \end{Bmatrix}_{UPR} = \begin{bmatrix} A & B \\ B & D \end{bmatrix}_{UPR} \begin{Bmatrix} \varepsilon \\ \kappa \end{Bmatrix}_{UPR}$$

$$\begin{Bmatrix} N \\ M \end{Bmatrix}_{LWR} = \begin{bmatrix} A & B \\ B & D \end{bmatrix}_{LWR} \begin{Bmatrix} \varepsilon \\ \kappa \end{Bmatrix}_{LWR} \qquad 4.5.2(d)$$

It is important to realize the difference between the face sheet matrices and the overall plate matrices, and to use the appropriate matrices for each analysis. For instance, when checking global buckling, use the plate [D] matrix, but when checking face sheet dimpling or wrinkling, use the face sheet [D] matrix. Also, some equations like those for face sheet wrinkling call for the face sheet extensional modulus E_x or E_y, but since wrinkling involves face sheet-bending deformation, it is more accurate to use the flexural modulus:

$$E_{x\,flex} = \frac{12D_{11}}{t_f^3} \text{ assuming that } B_{ij} = 0 \text{ and } \kappa_y = \kappa_{xy} = 0 \qquad 4.5.2(e)$$

instead of:

$$E_{x\,ext} = \frac{A_{11}}{t_f} \text{ assuming that } B_{ij} = 0 \text{ and } \varepsilon_y = \gamma_{xy} = 0 \qquad 4.5.2(f)$$

which would be appropriate if the deformation involved only a face sheet extensional deformation. Note also that these two definitions imply that the deformations (both curvatures and strains) are constrained in the transverse direction. Conversely, if the forces and moments are zero in the transverse direction, one should use:

For bending:

$$E_{x\,flex} = \frac{12}{t_f^3 D'_{11}} \text{ assuming that } B_{ij} = 0 \text{ and } M_y = M_{xy} = 0 \qquad 4.5.2(g)$$

For extension:

$$E_{x\,flex} = \frac{1}{t_f A'_{11}} \text{ assuming that } B_{ij} = 0 \text{ and } N_y = N_{xy} = 0 \qquad 4.5.2(h)$$

where D'_{11} and A'_{11} are elements of the inverse bending and extension matrices of the face sheets, [D'] and [A'], respectively.

When using any equation that calls for a face sheet modulus, use the definition of E_x that corresponds to the mode of deformation.

The equations in Sections 4.7 through 4.11 and 4.13 are presented for isotropic face sheets. In most cases, an equation is given that applies to a sandwich with face sheets of different materials and/or thicknesses, and then a second equation is given for the simplified case where the two face sheets are the same. In general, the equations can be adapted for orthotropic face sheets, by using the face sheet

bending stiffness as defined in this section in place of elastic modulus, E, and by using $\lambda = 1 - \nu_{ab}\,\nu_{ba}$ where ν_{ab} and ν_{ba} are the in-plane Poisson's ratios in the directions aligned with the panel sides.

4.5.3 Combined transverse and in-plane loadings

If a plate or beam is subjected to both transverse loads (e.g., pressure) and in-plane compression or shear, additional deflections and stresses will be created beyond the simple superposition of the deflections and stresses created by each loading. If no computer program is available to handle this type of effect, use the following procedure:

- Calculate the deflections and stresses created by the transverse loads.
- Calculate the stresses due to the in-plane loads, P.
- Calculate the critical (buckling) load under the in-plane loads, P_{cr}.
- Multiply the stresses and deflections due to transverse loads by the ratio;

$$R = \frac{1}{1 - \dfrac{P}{P_{cr}}} \qquad 4.5.3$$

where P/Pcr is the buckling ratio.

- Add together the stresses due to in-plane loads and the amplified stresses due to transverse loads.

4.5.4 Face sheet internal loads

Typically, the plate or beam stiffness properties will be input in a finite element model or in a specialized plate analysis program, which will return the internal loads in the form of the plate force, moment, and transverse shear resultants. The sandwich plate strains and curvatures are calculated with the stress-strain relations shown above or with the help of a laminate analysis program. The face sheet strains and loads can be calculated from:

$$\begin{Bmatrix} \varepsilon_x \\ \varepsilon_y \\ \varepsilon_{xy} \end{Bmatrix}_{UPR} = \begin{Bmatrix} \varepsilon_x \\ \varepsilon_y \\ \varepsilon_{xy} \end{Bmatrix}_{plate} + \left(\frac{t_c + t_{UPR}}{2}\right) \begin{Bmatrix} k_x \\ k_y \\ k_{xy} \end{Bmatrix}_{UPR} \qquad 4.5.4(a)$$

$$\begin{Bmatrix} N_x \\ N_y \\ N_{xy} \end{Bmatrix} = [A]_{UPR} \begin{Bmatrix} \varepsilon_x \\ \varepsilon_y \\ \varepsilon_{xy} \end{Bmatrix}_{UPR} + [B]_{UPR} \begin{Bmatrix} \kappa_x \\ \kappa_y \\ \kappa_{xy} \end{Bmatrix}_{plate} \qquad 4.5.4(b)$$

Similar equations may be obtained for the lower face sheet by exchanging the subscripts UPR $\leftrightarrow$ LWR.

4.6 LOCAL STRENGTH ANALYSIS METHODS

The strength analysis must check that the sandwich structure has sufficient strength in each of its possible failure modes, which are described in Section 4.4. Different failure modes may become critical depending on the materials, geometry, and loading.

4.6.1 Face sheet failure

Face sheet failure occurs when the face sheets fail by yielding or fracture. The criterion for failure is that the face sheet material exceeds its allowable stress or strain.

When face sheets are metallic, appropriate strength and other mechanical properties can be used from sources such as MMPDS (Reference 4.6.1(a)). Properties should be for the environmental condition in use (e.g., temperature).

When face sheets are composed of composite laminates, several modes of failure are possible due to the complexity of the composite structure. Numerous criteria have been proposed for calculation of onset of damage. The selection of appropriate criteria can be a controversial issue and the validity of any criterion is best determined by comparison with test data. See Volume 3, Chapter 8 for a more in-depth discussion of failure modes and strength prediction for composite materials.

In general, criteria may be grouped into two broad categories – mode-based and purely empirical. A purely empirical criterion generally consists of a polynomial combination of the three stress or strain components in a ply. Such criteria attempt to combine the effects of several different failure mechanisms into one function and may, therefore, be less representative than physically based criteria. Mode-based criteria treat each identifiable physical failure mode, such as fiber-direction tensile failure and matrix-dominated transverse failure, separately. All criteria rely on test data at the ply level to set parameters and are, therefore, at least partially empirical in nature.

Ply level stresses or strains are frequently used to predict laminate strength. The layer for which the failure criterion is reached for the lowest external load will define the loading that produces the initial laminate damage. This is generally called first-ply failure. However, using a mode-based approach, when the first ply failure is the result of fiber breakage, the resulting ply crack will introduce stress concentrations into the adjacent plies. Hence, it is reasonable to consider that first ply failure resulting from fiber strain reaching its limiting value is equivalent to laminate failure. On the other hand, first-ply transverse failure may not immediately cause laminate failure.

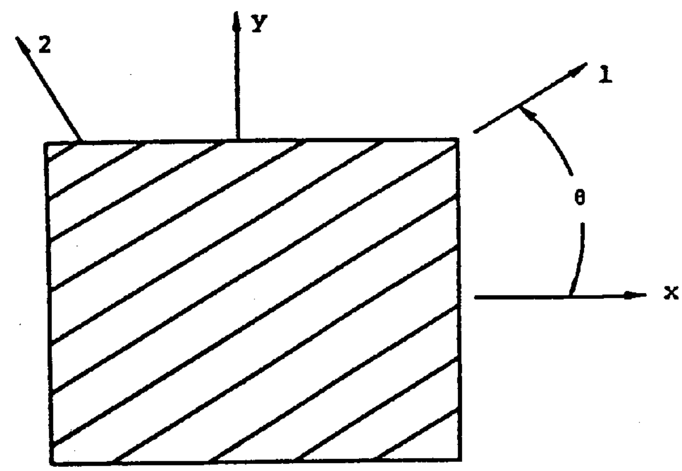
Face sheet fiber strains may be determined from face sheet strains as given in Section 4.5.4. The strains defined in Section 4.5.4 correspond to the sandwich coordinate system. To define the material response in directions other than these coordinates, transformation relations must be used to transform the strain into the material directions.

Figure 4.6.1 depicts two sets of coordinate systems. The 1-2 coordinate system corresponds to the principal material directions for the lamina, with 1 being the fiber or warp direction and 2 being the in-plane transverse direction. The x-y coordinates correspond to the sandwich coordinate system and are related to the 1-2 coordinates through a rotation about the axis out of the plane of the figure. The angle θ is defined as the rotation from the arbitrary x-y system to the 1-2 material system.

The strains can be transformed into the principal material coordinates for individual plies using the transformation

$$\begin{Bmatrix} \varepsilon_{11} \\ \varepsilon_{22} \\ 2\varepsilon_{12} \end{Bmatrix}_i = \begin{bmatrix} m^2 & n^2 & mn \\ n^2 & m^2 & -mn \\ -2mn & 2mn & m^2 - n^2 \end{bmatrix} \begin{Bmatrix} \varepsilon_{xx} \\ \varepsilon_{yy} \\ 2\varepsilon_{xy} \end{Bmatrix} \qquad 4.6.1$$

where the superscript i indicates the ply number and, therefore, which angle of orientation to use, $m = \cos\theta$, and $n = \sin\theta$.



1, 2 Principal Material Coordinates
x, y Laminate, or Arbitrary Coordinates

FIGURE 4.6.1 *Lamina coordinate transformation.*

The maximum fiber strain criterion may be written as

$$\varepsilon_{11}^{cu} \leq \varepsilon_{11}^{i} \leq \varepsilon_{11}^{tu}$$

where ply failure is indicated when this inequality is untrue.

For given loading conditions, the strains in each ply are compared to this criterion. The limiting strains, ε_{11}^{tu}, and ε_{11}^{cu}, are the specified maximum tensile and compressive fiber-direction strains to be permitted in any ply. Similar equations are used for the strains ε_{22} and γ_{12}. Generally, these quantities are specified as some statistical measure of experimental data obtained by uniaxial loading of a unidirectional laminate. See Volume 3 for further details.

4.6.2 Core shear

Transverse shear loads, Q_x and Q_y, are primarily carried by the core. A laminate analysis program or a finite element model can be used to determine the maximum transverse shear stresses τ_{xz}, τ_{yz} in the core caused by applied transverse shear forces. Otherwise, simplified relations for the computation of core transverse shear stresses are presented below.

For panels with face sheet thickness less than 5% of the core thickness, assume that the core carries the entire shear load

$$\tau_{xz} = \frac{Q_x}{t_c}; \qquad \tau_{yz} = \frac{Q_y}{t_c} \tag{4.6.2(a)}$$

If the face sheet thickness is greater than 5% of the core thickness, the face sheets will carry a portion of the shear load:

$$\tau_{xz} = \frac{Q_x}{t_c + \frac{t_{UPR} + t_{LWR}}{2}}; \qquad \tau_{yz} = \frac{Q_y}{t_c + \frac{t_{UPR} + t_{LWR}}{2}} \tag{4.6.2(b)}$$

The shear strength of honeycomb core is usually tested in the ribbon (L) and transverse (W) direction, but in actual structural applications the out-of-plane shear load may not be aligned with these directions (see Figure 4.6.2).

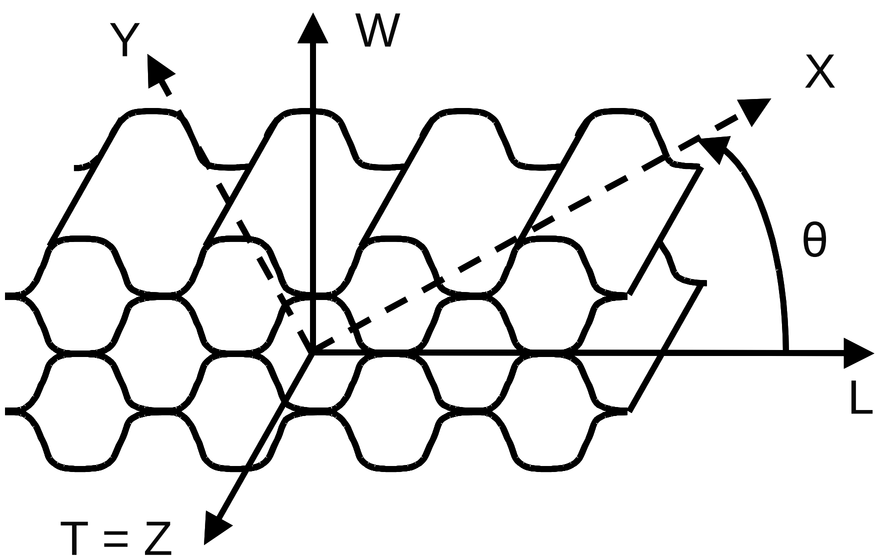


FIGURE 4.6.2. *Honeycomb core directions.*

In this case, an applied shear stress τ_{xz} can be resolved into components in the ribbon and transverse directions, and the core shear strength determined using an interaction criterion.

An empirical interaction criterion was found to fit several nonmetallic honeycomb cores used in aircraft construction (References 4.6.2(a) and 4.6.2(b)):

$$F_{s\theta} = \frac{1}{\sqrt[N]{\left(\frac{\cos\theta}{F_{sL}}\right)^N + \left(\frac{\sin\theta}{F_{sW}}\right)^N}} \tag{4.6.2(c)}$$

where $F_{s\theta}$, F_{sL}, and F_{sW} are core shear allowables in the loading, ribbon, and transverse directions, respectively, θ is the alignment angle as shown in Figure 4.6.2, and N is an exponent between 1 and 2, chosen

to fit the data for a given honeycomb core material, density, and cell geometry . The X-Y loading axes are aligned so that the applied shear stress is τ_{xz}, and the shear stress τ_{yz} is zero.

For a high-density fiberglass core with a bisected hexagonal cell geometry, an alternate approach using Lagrange Polynomials was found to provide a better fit to the test data (Reference 4.6.2(b)).

Both the exponent criterion and the Lagrange polynomial criterion require constants that must be determined by testing. The exponents that were found to fit the data for selected cores are shown in Table 4.6.2. These exponents were derived from room temperature testing (References 4.6.2(a) and 4.6.2(b)). The constants for other core materials, densities, cell geometries, and environments should be determined by testing. ASTM standard test methods C273 (plate shear) and C393 (beam flexure) are recommended (References 4.6.2(c) and 4.6.2(d), respectively).

TABLE 4.6.2. *Exponents for core shear interaction.*

Core Material	Cell Geometry	Cell Size (in)	Density (lb/in^3)	Exponent, N
Fiberglass	Hexagonal	3/8	3.5	1.34
Fiberglass	Hexagonal	3/8	4.5	1.32
Nomex™	Flex	F50	5.5	1.45
Nomex™	Overexpanded	3/16	3.0	1.5
Kevlar™	Hexagonal	1/8	3.0	1.23
Korex™	Hexagonal	1/8	6.0	1.2

Core shear modulus can also be calculated in the loading direction (Reference 4.6.2(a)):

$$G_\theta = \frac{G_L G_W}{G_L \sin^2\theta + G_W \cos^2\theta} \qquad 4.6.2(d)$$

where G_L, G_W, and G_θ are the shear modulus in the ribbon, transverse, and loading directions, respectively. Equation 4.6.2(d) is obtained from an in-plane rotation of the stiffness tensor, assuming the core behaves as an orthotropic material. It was found to provide a good fit to the test data for the cores listed in Table 4.6.2 (References 4.6.2(a) and 4.6.2(b)).

4.6.3 Flatwise tension and compression

As illustrated in Figure 4.6.3, at the edges of most sandwich panels, one face sheet, the bag-side sheet for composite construction, ramps up over the core. At the start and finish of the ramp, where the face sheet changes direction, interlaminar stresses (flatwise tension or compression) are induced at the face sheet-to-core bondline. A flatwise tension stress can cause tensile core failure or debonding between the face sheet and the core, and a flatwise compression stress can induce core crushing.

The flatwise stress should be computed at the center of each ramp radius. The following equations provide approximate magnitudes for the flatwise stresses. A finite element model may be required to provide a more accurate stress state in the ramp areas.

The equation below is based on the assumption that the flatwise stress is uniformly distributed around the radius from the start of curvature to the end of curvature. Note that the computation requires the load in the bag-side face sheet, and this is dependent on, among other things, the distance between the face sheets and the angle of the bag-side with respect to the tool-side. The gap between the face sheets and

the angle of the bag-side are both variables in the ramp region, so the analyst needs to take care in obtaining the proper geometry.

Note that the following equations use loads in the X direction. The loads in the Y direction must also be checked, and similar equations can be written.

$$f_{X\,flat} = \pm \frac{N_{xb}}{R}$$

Use "+" for concave radii and "-" for convex radii
$F_{X\ flat}$ Flatwise stress at radius
N_{xb} Load in the bag-side face sheet
R Ramp radius

Two angles are used in Figure 4.6.3:
ϕ = ramp angle
θ = local angle between bag side and tool side in the ramp region

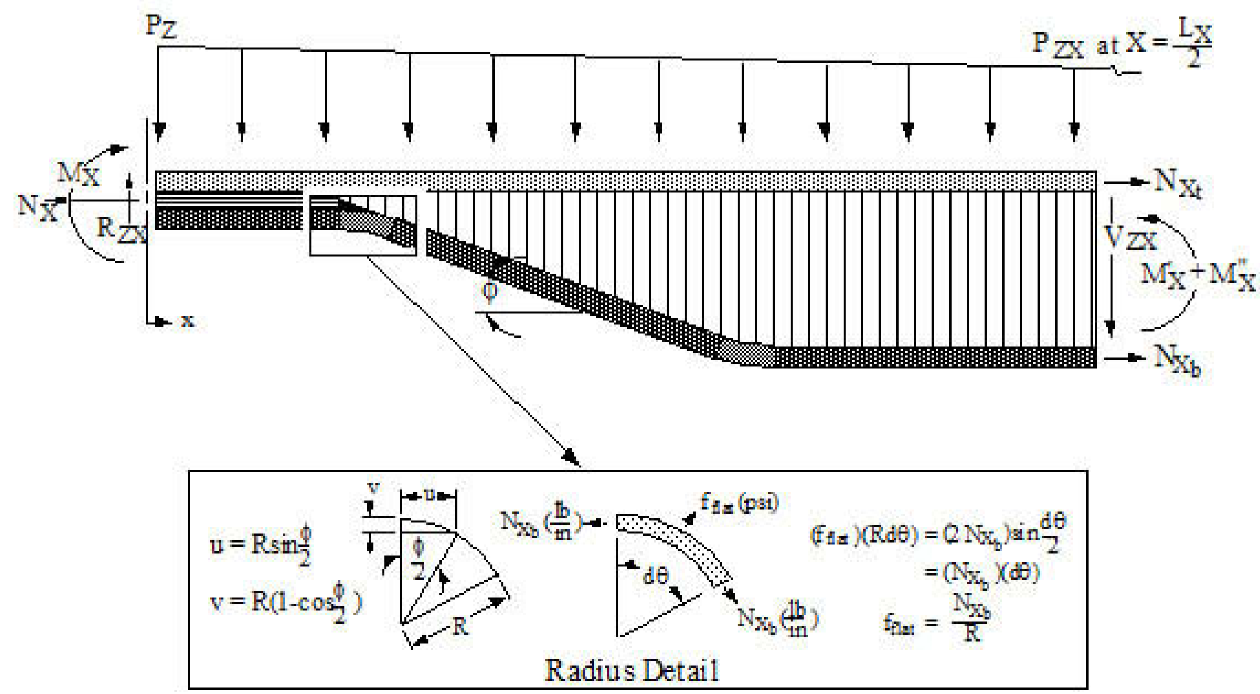


FIGURE 4.6.3 *Geometry of sandwich edge closeout .*

The internal loads, N_{Xt} and N_{Xb}, carried by top and bottom face sheet, respectively, may be determined as follows (Ref. 4.6.3):

$$N_{Xt} = \frac{N_X}{2} - \frac{M'_X + M''_X}{d}$$

$$N_{Xb} = \left[\frac{N_X}{2} + \frac{M'_X + M''_X}{d}\right]\left(\frac{1}{\cos\theta}\right)$$

The internal shear load V_{ZX} is given by:

$$V_{ZX} = \left[R_{ZX} - P_Z(x) + \left(\frac{P_Z - P_{ZX}}{L_X} \right) \left(x^2 \right) \right] - \left[N_{Xb}(\sin\theta) \right]$$

where axial loads N_X, moments M_X, and distributed pressure P_Z are as shown in Figure 4.6.3.

At the radius closest to the edgeband (on the left side of Figure 4.6.3), the critical stress is considered to be interlaminar tension between the plies of the bag-side laminate.

At the radius furthest from the edgeband (on the right side of Figure 4.6.3), the failure modes are considered to be either separation of the bag-side face sheet from the core (flatwise tension) or core crushing (flatwise compression). Flatwise compression allowables are equivalent to the crushing strength of the core material, F_{CC}.

4.6.4 Flexural core crushing

As a sandwich panel bends, the core must restrain the face sheets from moving towards each other. In this way, a compressive or crushing force is induced in the core through-thickness. This failure mode is checked at the location of the maximum bending moments. The core crushing stress is computed using:

$$f_{crush} = \frac{M_x^2}{d\,D_x} + \frac{M_y^2}{d\,D_y} + q \qquad 4.6.4$$

f_{crush} = Core compressive stress
M_X, M_y = Bending moments from in-plane and pressure loads
D_X, D_Y = Flexural stiffnesses for beams parallel to the X and Y directions, respectively
d = Distance between face sheet centroids
q = Normal pressure

4.6.5 Intracell buckling (dimpling)

If the core of a sandwich construction is of honeycomb or corrugated material, it is possible for the face sheets to buckle or dimple into the spaces between core cell walls or corrugations with a forced wavelength equal to the size of these spaces. Dimpling of the face sheets may not lead to failure unless the amplitude of the dimples becomes large and causes the dimples or buckles to grow across core cell walls and result in wrinkling of the face sheets. Dimpling that does not cause total structural failure may still be severe enough so that permanent dimples remain after removal of load.

If dimpling of the face sheets is not permissible, the core cell size or corrugation spacing shall be small enough so that dimpling will not occur under design loads. It is assumed that failure in the face sheet-to-core bond will not occur prior to dimpling. The design procedures also assume that a face sheet thickness (t) has been determined by consideration of design loads and design compressive face sheet stresses, that face sheet compressive stress, F_C, and effective compressive modulus of elasticity, E', are known, and that the core cell size or corrugation spacing is to be determined. The face sheet properties should be values at the condition of use; that is, if application is at elevated temperature, then face sheet properties at elevated temperature should be used in design. The face sheet modulus of elasticity is the effective value at the face sheet stress. If this stress is beyond the proportional limit value, an appropriate tangent, reduced, or modified compression modulus of elasticity shall be used (Reference 4.6.5(a)).

Because of uncertainties in analysis and values of material properties, it is recommended that the final design be checked by tests of a few small specimens (see References 4.6.5(b) and 4.6.5(c) for test methods).

4.6.5.1 Sandwich having cellular (honeycomb) core

This section gives the procedure for determining the core cell size such that an isotropic sandwich face sheet will not dimple (Reference 4.6.5.1(a)). The face sheet stress at which dimpling of the sandwich face sheet will occur is given by the empirical formula

$$F_C = 2\frac{E'}{\lambda}\left(\frac{t}{s}\right)^2 \qquad 4.6.5.1(a)$$

where E' is effective compressive modulus of elasticity of the face sheet at stress F_C, $\lambda = 1 - \nu^2$ when ν is Poisson's ratio of face sheets, t is face sheet thickness and s is core cell size (diameter of inscribed circle). Solving Equation 4.6.5.1(a) for s results in core cell size

$$s = t\sqrt{2}\left(\frac{\lambda F_C}{E'}\right)^{-\frac{1}{2}} \qquad 4.6.5.1(b)$$

Determine maximum core cell size from Equation 4.6.5.1(b) or by graphical solution using the chart in Figure 4.6.5.1. If the core cell size is smaller than available, it is necessary to use a thicker face sheet and a lower stress in order that dimpling will occur at the same edgewise load. The chart of Figure 4.6.5.1 can also be used to find face sheet thicknesses and stresses for dimpling of sandwich face sheets on cores of a particular cell size.

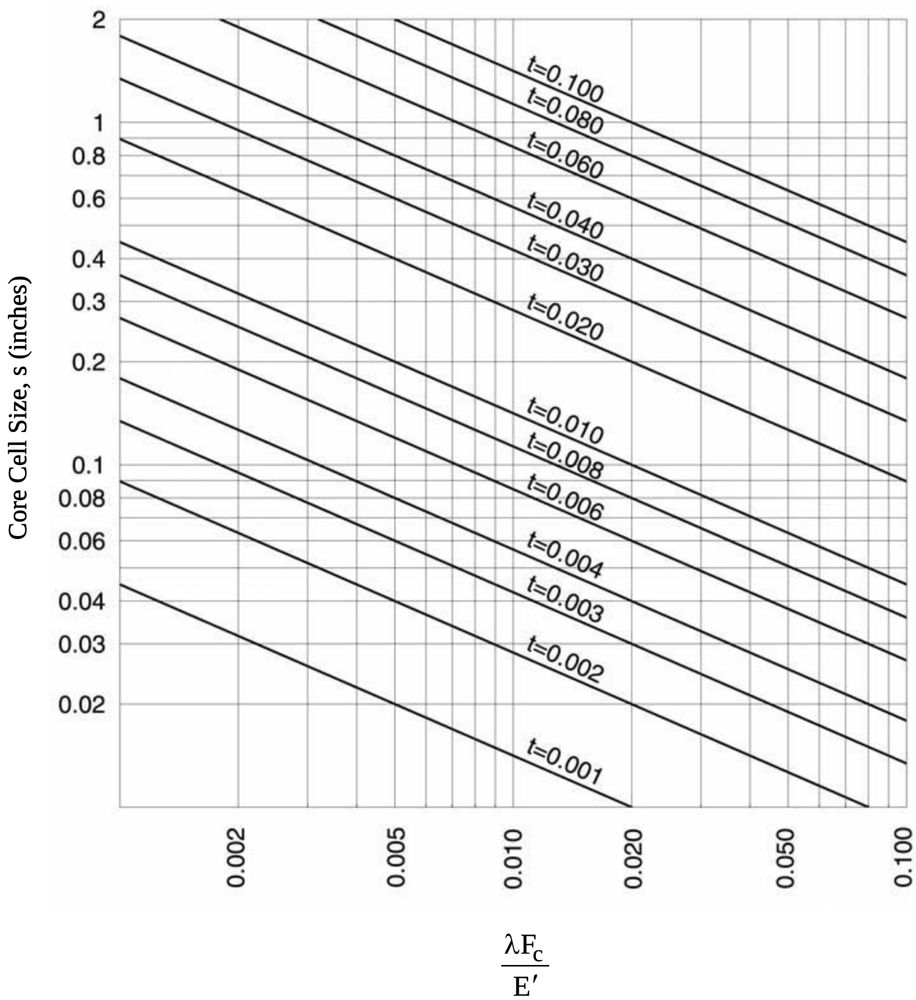


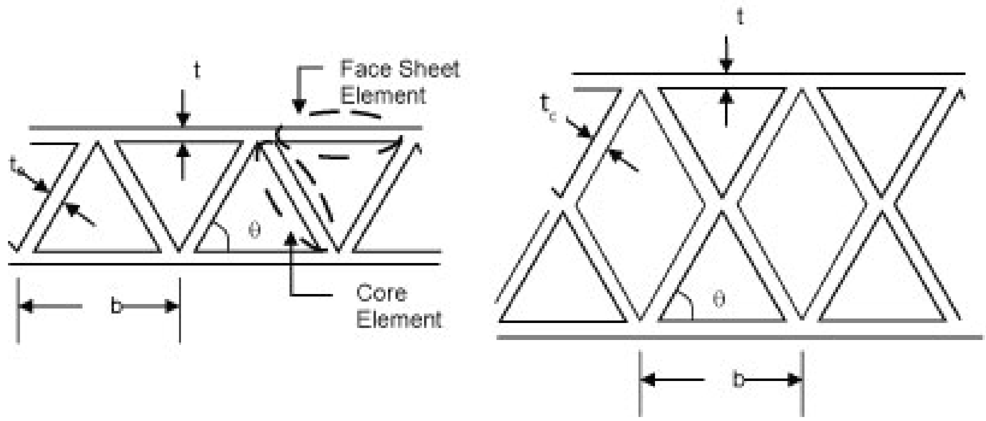
FIGURE 4.6.5.1 *Chart for determining cellular core cell size such that dimpling (intracell buckling) of isotropic sandwich face sheet will not occur.*

For face sheets that are orthotropic, the following equation may be used (Reference 4.6.5.1(b)):

$$F_c^{Fokker} = \frac{1}{t}\left(\frac{\pi}{s}\right)^2 \left\{D_{11} + 2\left(D_{12} + 2D_{66}\right) + D_{22}\right\} \qquad 4.6.5.1(c)$$

4.6.5.2 Sandwich having corrugated core

This section gives the procedure for determining the spacing between core corrugations such that sandwich face sheet or core elements will not buckle (References 4.6.5.2(a) through (c)). Figure 4.6.5.2(a) shows the geometry of a sandwich with corrugated core, for a cross section perpendicular to the core axis.



(a) Single Corrugated Core (b) Double Corrugated Core

FIGURE 4.6.5.2(a) *Illustration of corrugated core and associated dimensions.*
(core axis is perpendicular to the page)

For edge compression load in a direction parallel to the core axis, the design procedure is based on the buckling load of the unsupported face sheet element or core element, whichever is the lower, although it may be possible to utilize a sandwich in which the core elements are buckled.

For edge compression load in a direction perpendicular to the core axis, the design procedure is based on the buckling load of the unsupported face sheet element, and the core is assumed to be rigid enough to cause rotational restraint at the ends of the face sheet element. If the core is double corrugated, it may not have sufficient flatwise strength to cause the face sheet elements to buckle as assumed.

The face sheet stress at which buckling of the face sheet element or core element will occur is given by the formula

$$F_C = k\frac{E'}{\lambda}\left(\frac{t}{b}\right)^2 \qquad 4.6.5.2(a)$$

where E' is the effective compressive modulus of elasticity of the face sheet at stress F_C, $\lambda = 1 - \nu^2$, ν being Poisson's ratio of the face sheet, t is face sheet thickness, b is the unsupported width of the face sheet element (see Figure 4.6.5.2(a)), and k is a coefficient dependent upon the ratio (t_c/t) of the corrugation thickness (t_c) to the face sheet thickness (t), the angle (θ) between the corrugation element and the face sheet, and the type of material, i.e., isotropic or orthotropic. Solving Equation 4.6.5.2(a) for b results in

$$b = t\sqrt{k}\left(\frac{\lambda F_C}{E'}\right)^{-\frac{1}{2}} \qquad 4.6.5.2(b)$$

Graphical solutions for Equation 4.6.5.2(b) are given in charts of Figures 4.6.5.2(b) through 4.6.5.2(l). All charts except that of Figure 4.6.5.2(l) apply to sandwich in which the load is in a direction parallel to the core axis. The chart of Figure 4.6.5.2(l) applies to sandwich with the load in a direction perpendicular to the core axis.

Charts of Figures 4.6.5.2(b) and (c) apply to sandwich with face sheets and core of the same isotropic material, charts of Figures 4.6.5.2(d) through 4.6.5.2(k) apply to sandwich of orthotropic materials such as glass fabric laminates for which α = 2/3, 1, or 3/2, and β = 0.6 as indicated on the charts. The values of α and β depend upon the elastic properties as follows:

$$\alpha = \sqrt{\frac{E_b'}{E_a'}}\ ;\quad \beta = \frac{\lambda}{\sqrt{E_a' E_b'}}\left(\frac{E_b' \nu_{ab}}{\lambda} + 2G_{ab}'\right) \qquad 4.6.5.2(c)$$

where E_a' and E_b' are the moduli of elasticity parallel and perpendicular to the direction of loading, G'_{ab} is the shear modulus associated with those directions, ν_{ab} is the Poisson's ratio of the contraction in the b direction to extension in the a direction due to a tensile stress in the a direction. ν_{ba} is similarly defined, and $\lambda = 1 - \nu_{ab}\nu_{ba}$. Symbols subscripted with c such as α_c, β_c, etc., indicate properties of the core material.

After choosing θ and determining b from the charts, the geometry of the sandwich cross section is now fixed so that the distance, h, between the face sheet centroids is given by

$$h = (b/2) \tan \theta \text{ (single corrugated core)} \qquad 4.6.5.2(d)$$

$$h = b \tan \theta \text{ (double corrugated core)} \qquad 4.6.5.2(e)$$

Since the dimension h may be determined previously by sandwich stiffness or strength requirements, it will be necessary to determine b by solving Equation 4.6.5.2(d) or (e). Graphical solutions for determining b from Equation 4.6.5.2(d) or (e) are given in the lower portion of the charts of Figures 4.6.5.2(b) through 4.6.5.2(l). The final design will be based on the value of b determined by solution of Equation 4.6.5.2(b) or (c) and b shall be no greater than the solution of Equation 4.6.5.2(b). By iteration, it is possible to choose θ so that values of b determined by charts for Equation 4.6.5.2(b) are the same as determined by the charts for Equation 4.6.5.2(d) or (e).

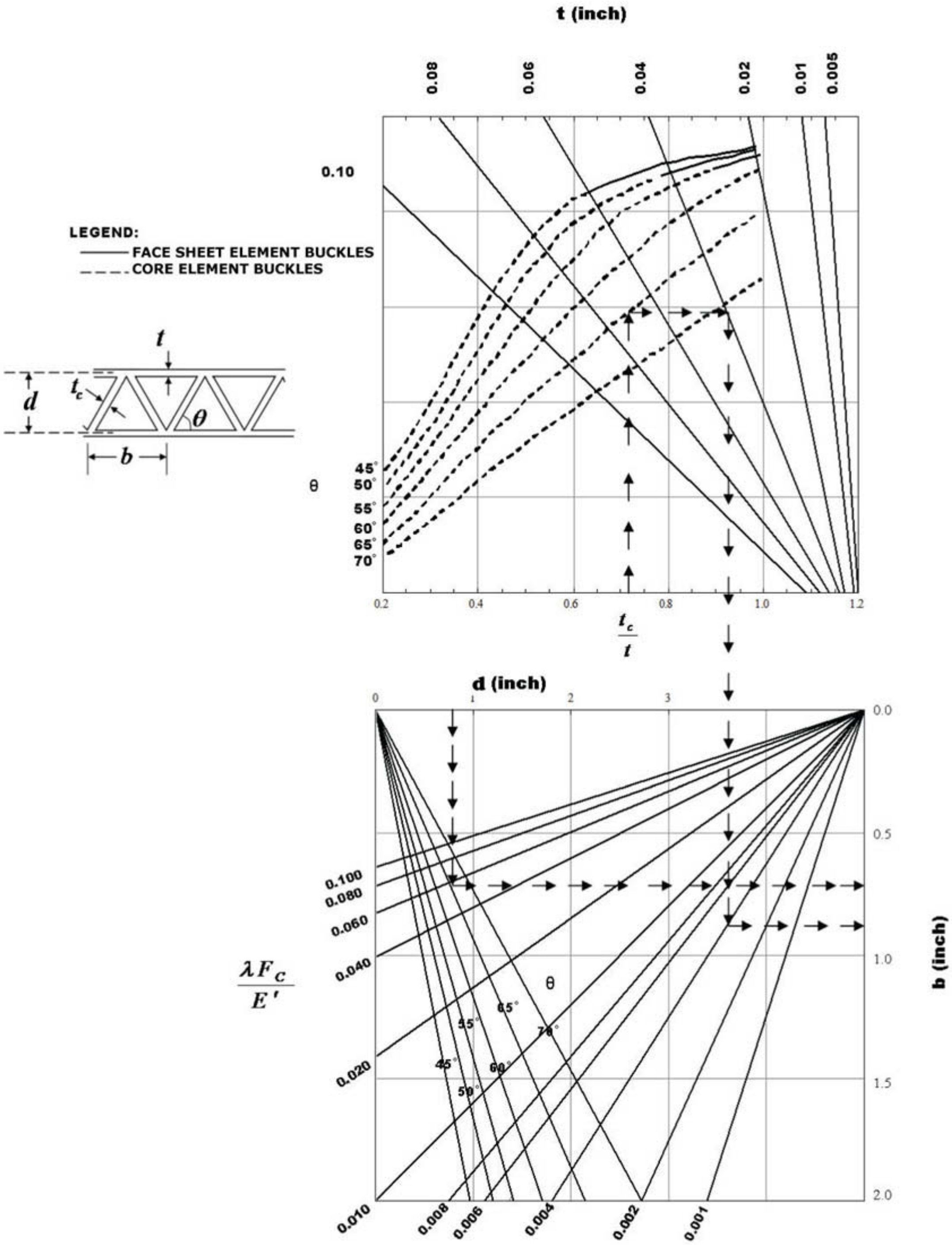


FIGURE 4.6.5.2(b) *Chart for determining width, b, of face sheet element for an isotropic corrugated core sandwich under compression edge load in a direction parallel to core axis (single corrugated core).*

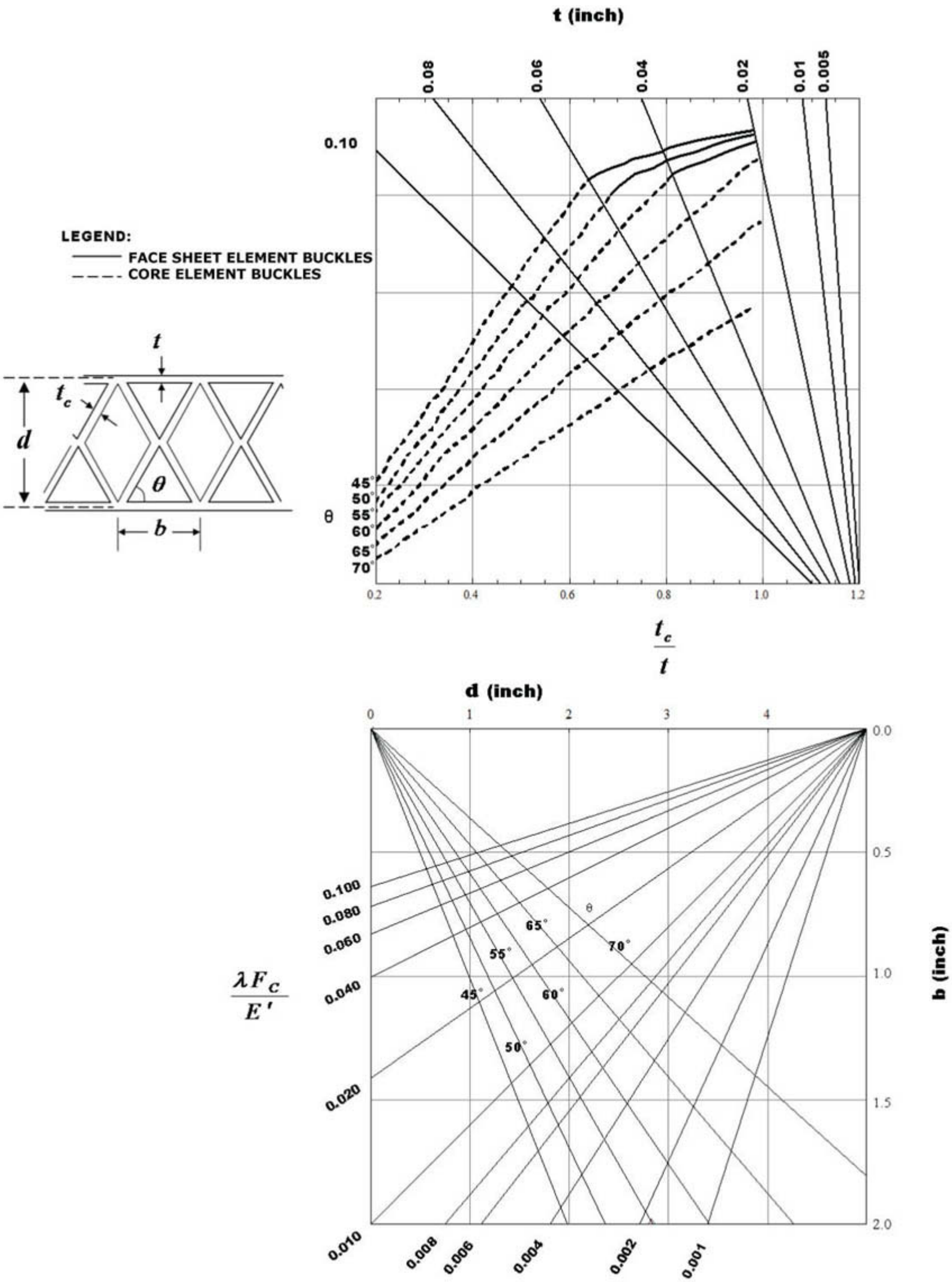


FIGURE 4.6.5.2(c) *Chart for determining width, b, of face sheet element for an isotropic corrugated core sandwich under compression edge load in a direction parallel to core axis (double corrugated core).*

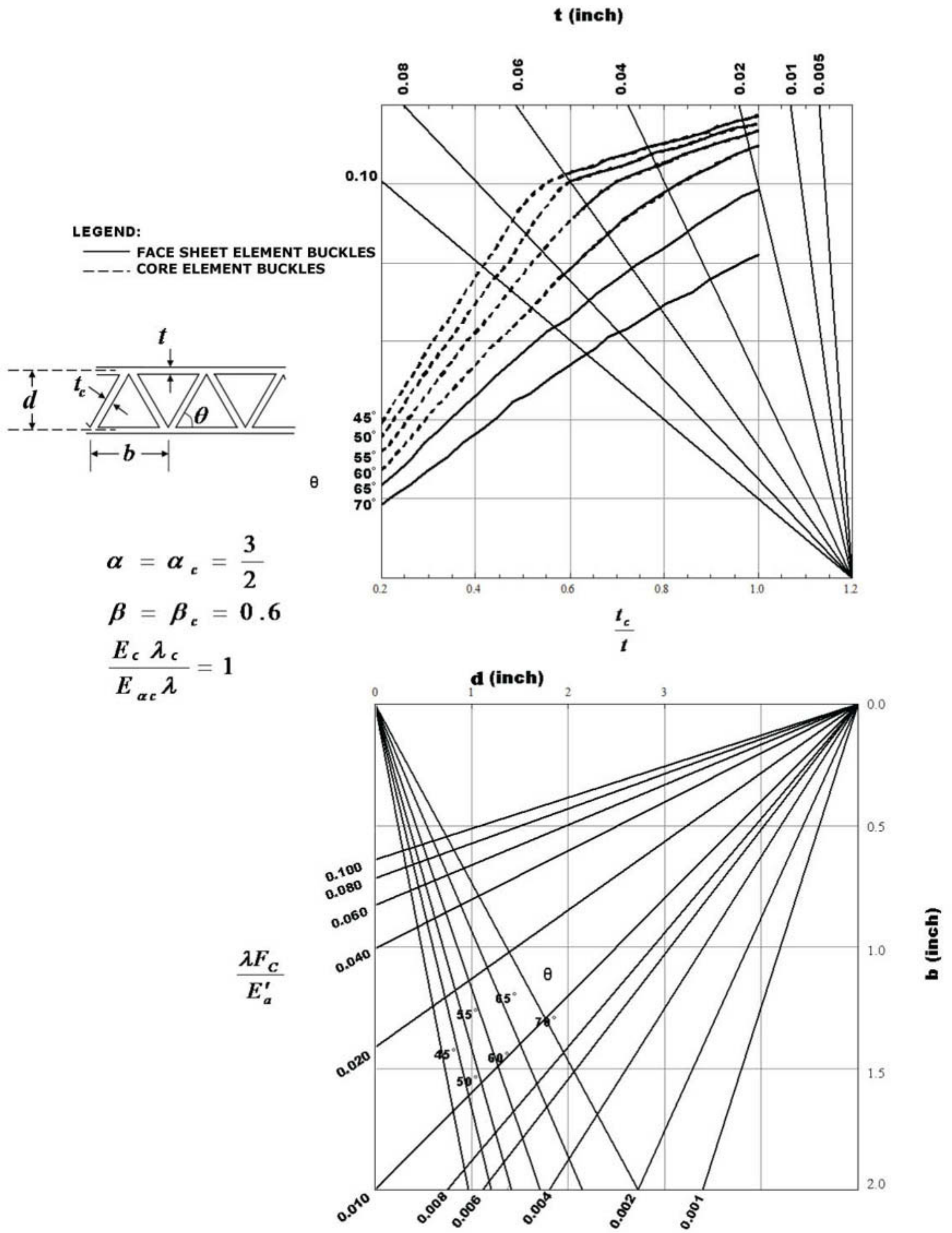


FIGURE 4.6.5.2(d) *Chart for determining width, b, of face sheet element for an orthotropic corrugated core sandwich under compression edge load in a direction parallel to core axis (single corrugated core).*

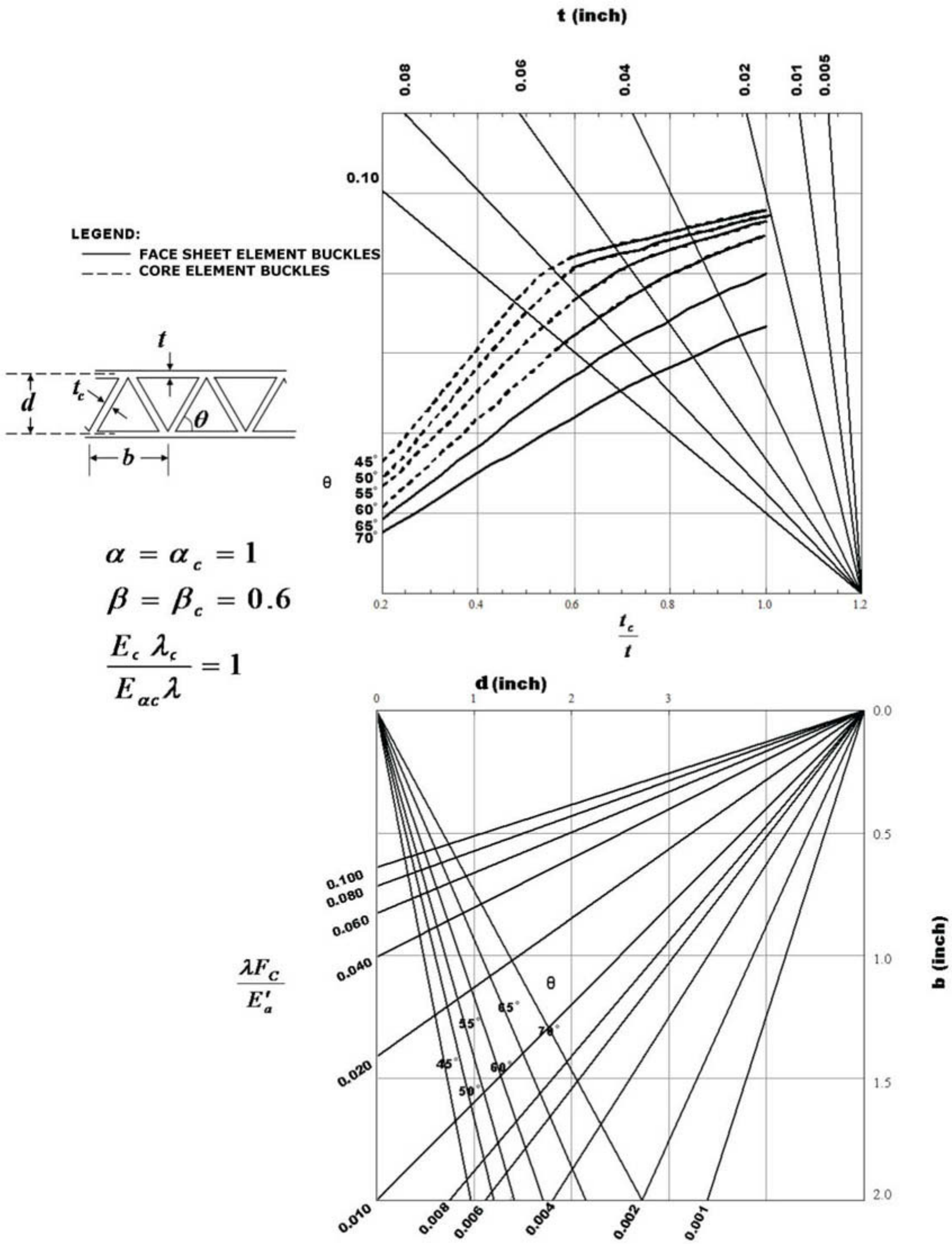


FIGURE 4.6.5.2(e) *Chart for determining width, b, of face sheet element for an orthotropic corrugated core sandwich under compression edge load in a direction parallel to core axis (single corrugated core).*

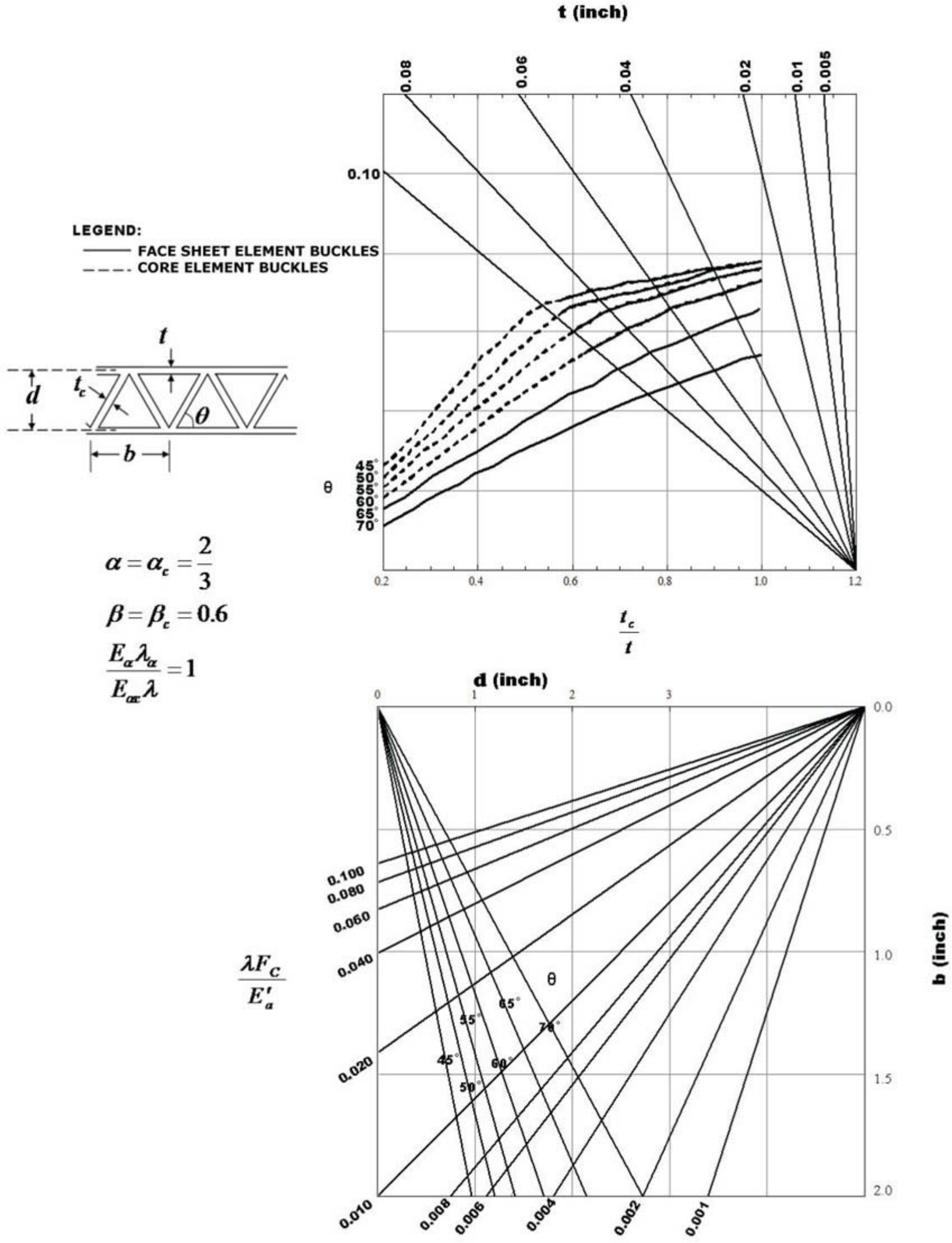


FIGURE 4.6.5.2(f) *Chart for determining width, b, of face sheet element for an orthotropic corrugated core sandwich under compression edge load in a direction parallel to core axis (single corrugated core).*

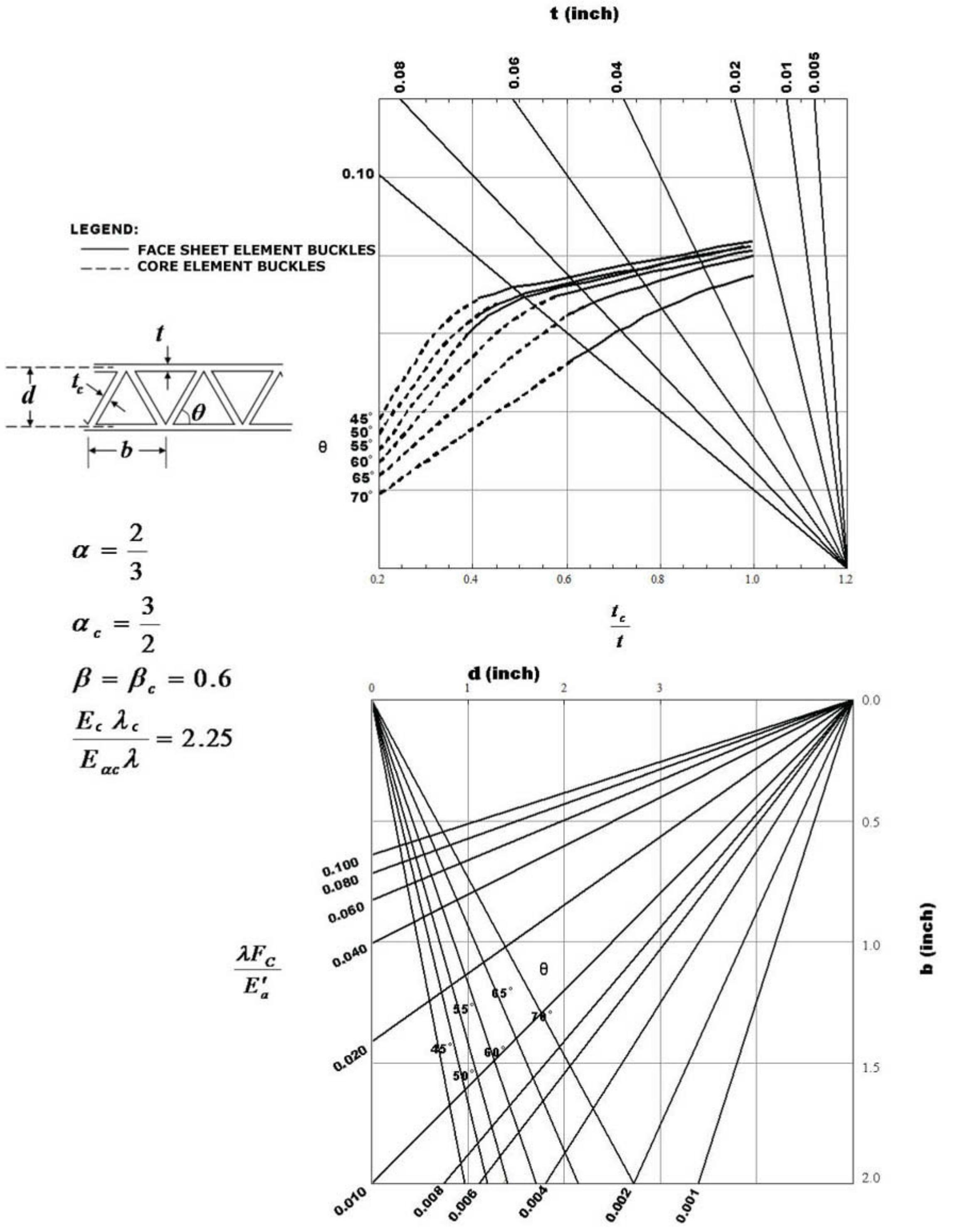


FIGURE 4.6.5.2(g) *Chart for determining width, b, of face sheet element for an orthotropic corrugated core sandwich under compression edge load in a direction parallel to core axis (single corrugated core).*

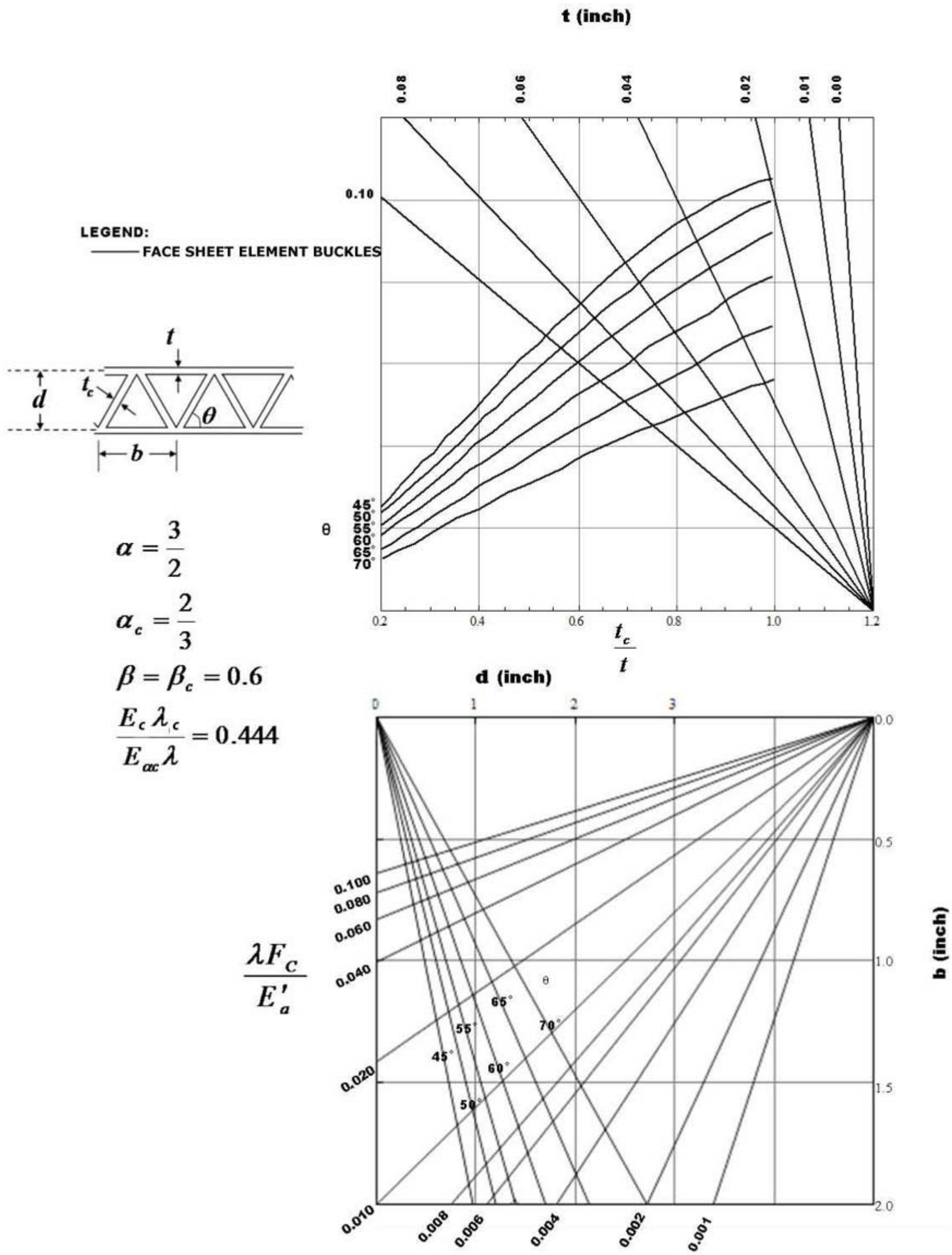


FIGURE 4.6.5.2(h) *Chart for determining width, b, of face sheet element for an orthotropic corrugated core sandwich under compression edge load in a direction parallel to core axis (single corrugated core).*

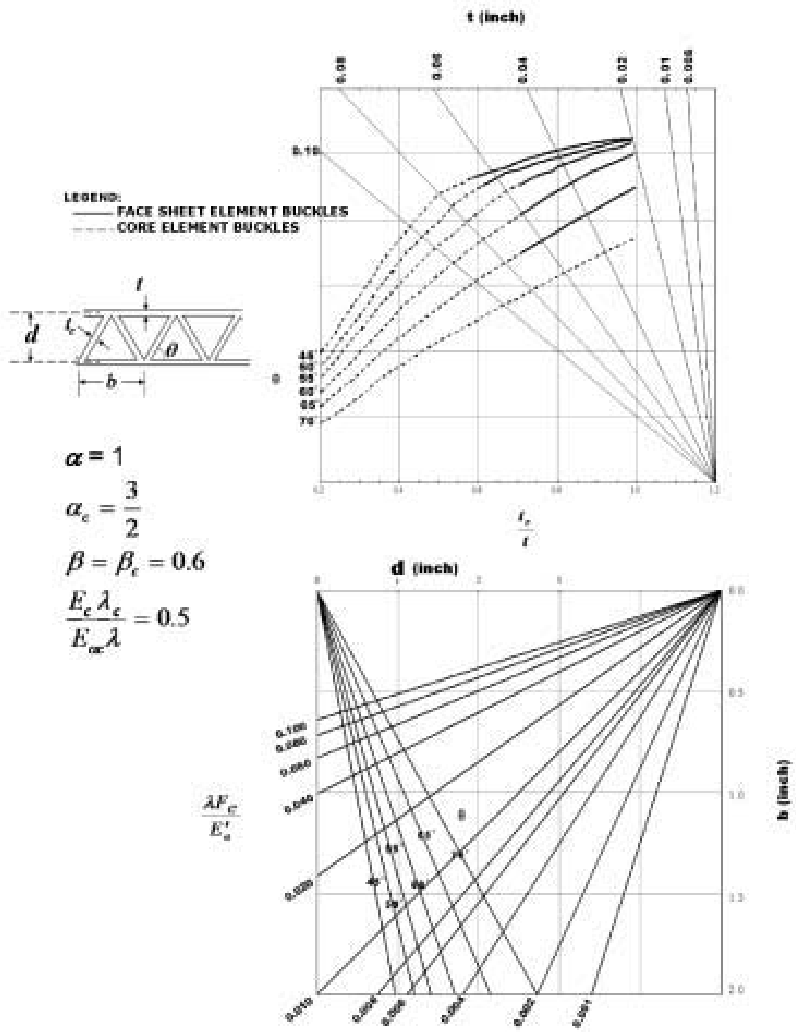


FIGURE 4.6.5.2(i) *Chart for determining width, b, of face sheet element for an orthotropic corrugated core sandwich under compression edge load in a direction parallel to core axis (single corrugated core).*

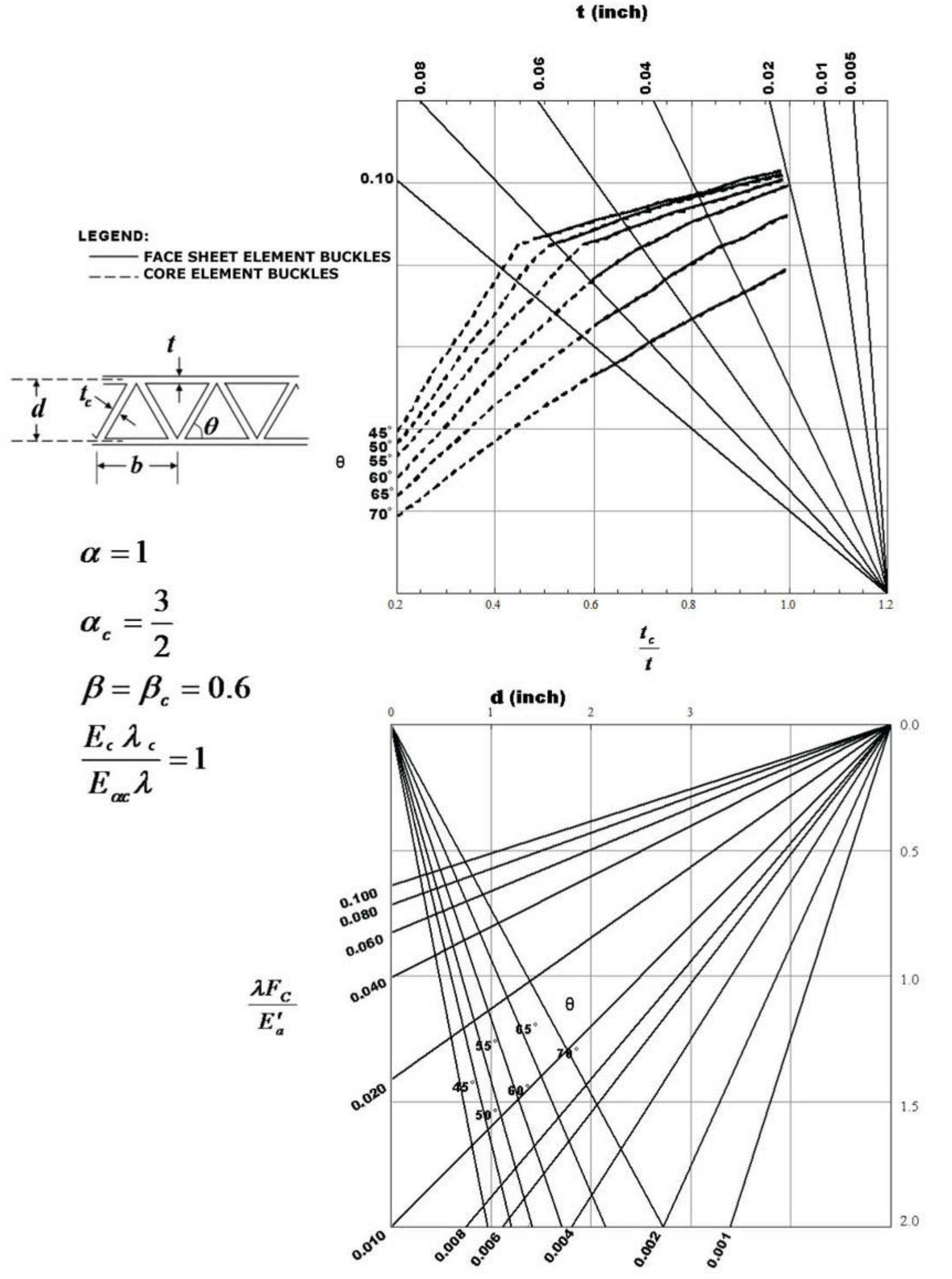


FIGURE 4.6.5.2(j) *Chart for determining width, b, of face sheet element for an orthotropic corrugated core sandwich under compression edge load in a direction parallel to core axis (single corrugated core).*

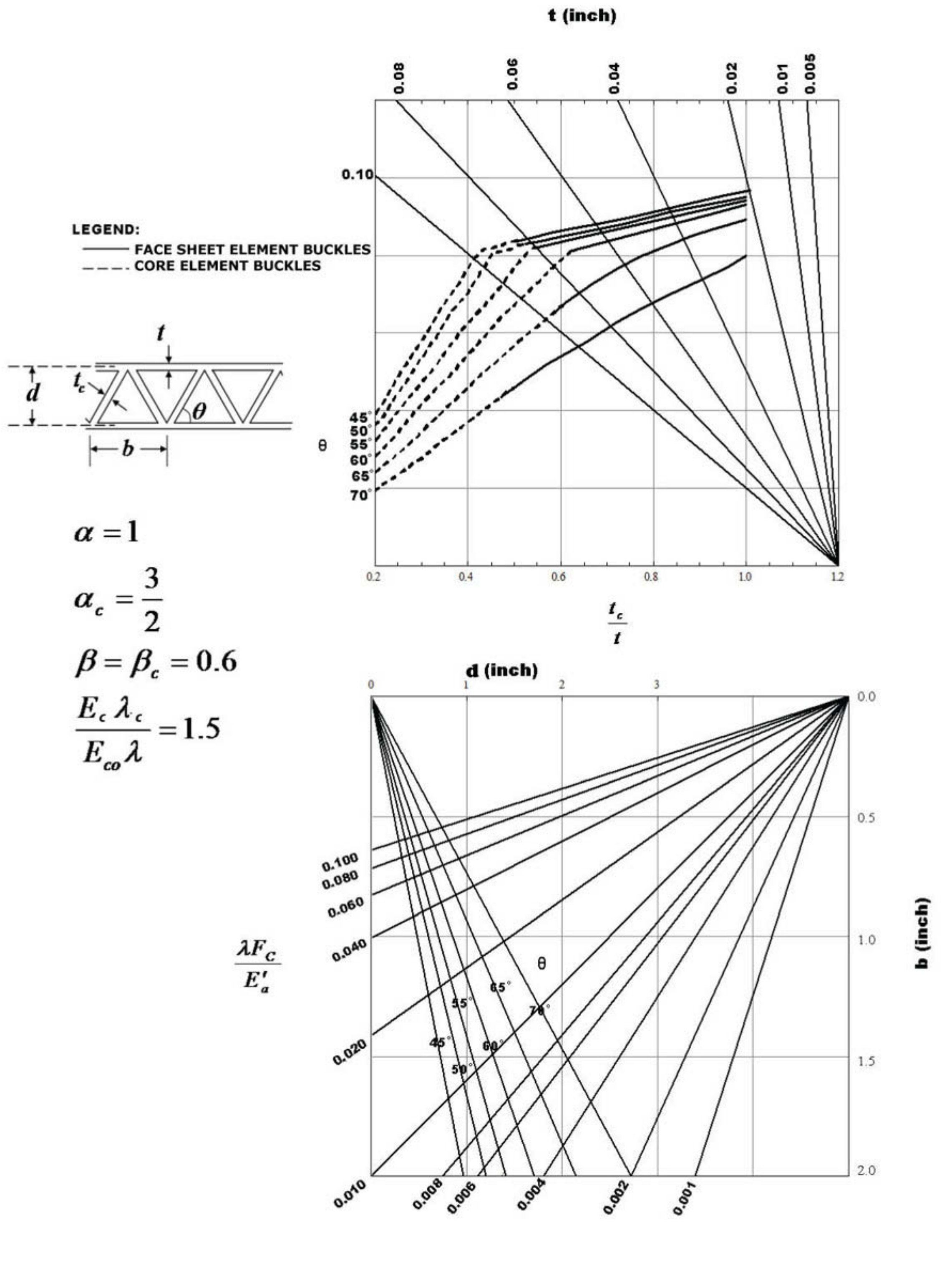


FIGURE 4.6.5.2(k) *Chart for determining width, b, of face sheet element for an orthotropic corrugated core sandwich under compression edge load in a direction parallel to core axis (single corrugated core).*

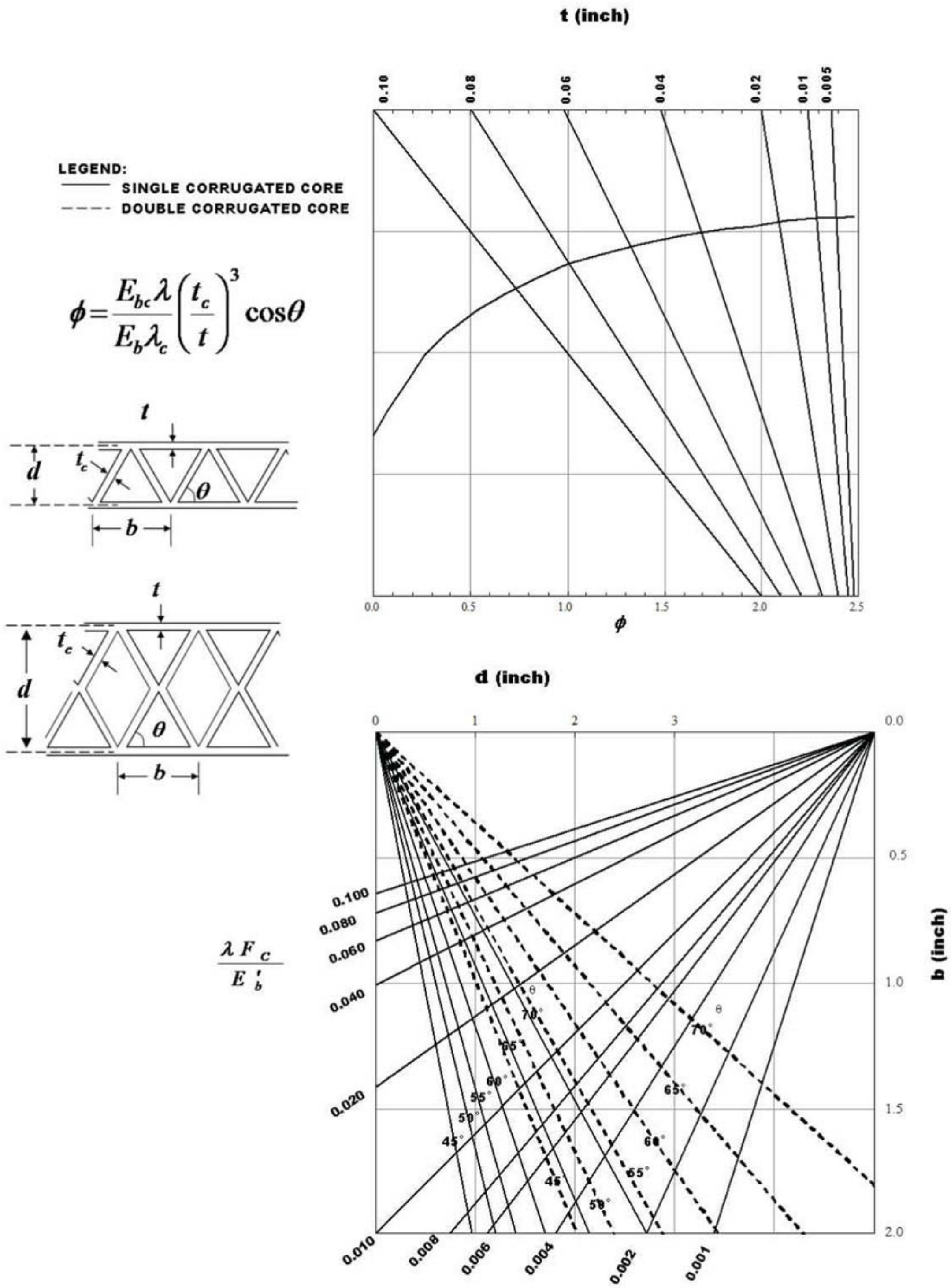


FIGURE 4.6.5.2(l) *Chart for determining width, b, of face sheet element for an orthotropic corrugated core sandwich under compression edge load in a direction perpendicular to core axis.*

4.6.5.3 Shear intracell buckling

An empirical expression for the critical stress for shear intracell buckling (also referred to as "shear dimpling") in a honeycomb sandwich panel is shown below. The allowable can be expressed in terms of the X or the Y directions. Use the lower of the two.

$$F_{s\,dimple} = (0.60)E_{xf}\left(\frac{t_f}{s}\right)^{1.5} \text{ or } F_{s\,dimple} = (0.60)E_{yf}\left(\frac{t_f}{s}\right)^{1.5} \qquad 4.6.5.3$$

$F_{S\,dimple}$ Intracell buckling critical stress for shear

E_{X_f}, E_{Y_f} Elastic moduli for the face sheet being checked

t_f Thickness of the face sheet being checked

s Cell size (in)

4.6.5.4 Combined compression and shear intracell buckling

For honeycomb sandwich face sheets with combined compression and shear loads, the following interaction equation may be used (Reference 4.6.3):

$$R_C = \frac{\left(f_x + f_y\right)}{F_{c\,dimple}}$$

$$R_S = \frac{f_{xy}}{F_{S\,dimple}} \qquad 4.6.5.4$$

$$MS_{dimple} = \frac{2}{R_C + (R_C^2 + 4R_S^2)^{1/2}} = 1$$

4.6.6 Face sheet wrinkling

4.6.6.1 Wrinkling of sandwich face sheets under edgewise load

Wrinkling failures may occur if a sandwich face sheet buckles as a plate on an elastic foundation. Wrinkling is sometimes referred to as natural wavelength buckling since it is associated with periodic waves of a wavelength that depend not on the in-plane dimensions of the sandwich, but on the material properties and thickness of the core and face sheet. Analysis of this localized buckling behavior is complicated by the unknown waviness of sandwich face sheets. Thus, the designer must, in effect, consider the buckling of a plate or column (face sheet) that is supported on an elastic foundation (core) and that is not initially straight. The initial curvature or deflection (waviness) is not easily defined or easily measured, and attempts to correlate wrinkling data, including measured face sheet waviness, with theory have not been very successful.

Increase in the out-of-plane deformation of the initial face sheet waviness causes stresses in the core and in the bond between face sheets and core. Final failure may occur suddenly, and the face sheet may buckle inward or outward, depending on the flatwise compressive strength of the core relative to the flatwise tensile strength of the bond between the face sheet and core. Wrinkling failure may be difficult to detect after the load is removed, if the failure of the core or face sheet-to-core bond does not lead to a visible failure of the face sheet.

Generally the sandwich design is not determined by face sheet wrinkling, but by other considerations such as required strength and stiffness of the face sheets and core, global buckling modes, etc. However, wrinkling failure modes can be critical for certain sandwich construction, particularly when face sheets are thin and the core material has low compressive and shear stiffness. The sandwich panel design

should, therefore, be checked to ascertain whether wrinkling of the sandwich face sheets might occur at design load. Because of uncertainties in analysis, values of material properties and waviness of the face sheets, it is recommended that the final design be validated by tests of a few small specimens (see References 4.6.6.1(a) and (b) for suitable test methods).

The face sheets of a sandwich shall not wrinkle under design load. The information given in the remainder of this section assumes that the face sheet and core properties and dimensions are known. The properties shall be values at the condition of use; that is, if application is at elevated temperature, then properties at elevated temperature shall be used in design. The face sheet modulus of elasticity is the effective value at the face sheet stress. If this stress is beyond the proportional limit value for a metallic face sheet, an appropriate tangent, reduced, or modified compression modulus of elasticity shall be used (Reference 4.6.6.1(c)).

The wrinkling stress equations are given for two types of sandwich: sandwich with continuous cores (e.g., foam or balsa wood), and sandwich with honeycomb cores for which elastic moduli in the plane of the core are very small compared with the elastic modulus in a direction normal to the core plane.

4.6.6.2 Sandwich with core supporting face sheets continuously

The stress at which wrinkling of sandwich face sheets on a continuous core will occur is given approximately by the equation (References 4.6.6.2(a) and (b)):

$$F_W = Q\left(\frac{E' E_C G_C}{\lambda}\right)^{1/3} \qquad \text{4.6.6.2(a)}$$

where F_w is face sheet wrinkling stress, E' is effective face sheet elastic modulus in the direction of the applied load, λ is one minus the product of the two in-plane Poisson's ratios ($\lambda = 1 - \nu_{ab} * \nu_{ba}$, or $\lambda = 1 - \nu^2$ for isotropic materials), E_c is core elastic modulus in a direction normal to the sandwich face sheets, G_c is core shear modulus associated with shear distortion in the plane perpendicular to the face sheets and parallel to the direction of applied load, and Q is a coefficient.

Note that for sandwiches with orthotropic laminated composite face sheets, an equivalent Young's modulus should not be used. Rather, E' should be replaced by

$$12\lambda \frac{D_f}{t_f^3} \qquad \text{4.6.6.2(b)}$$

where D_f is the face sheet laminate bending stiffness in the direction of the applied load. See discussion in Section 4.5.2.

A number of values for Q have been suggested in the literature. A comparison of proposed values and data from a large pool of experimental results indicates that a value of Q = 0.63 fits the data reasonably well (see Reference 4.6.6.2(b)). A value of 0.50 may be used for a conservative lower bound.

Alternatively, Q may be calculated as the relative minimum with respect to ζ of the expression (Reference 4.6.6.2(a):

$$\frac{\dfrac{\zeta^2}{30q^2} + \dfrac{16q}{\zeta}\left(\dfrac{\cosh\zeta - 1}{11\sinh\zeta + 5}\right)}{1 + 6.4K\zeta\left(\dfrac{\cosh\zeta - 1}{11\sinh\zeta + 5}\right)} \qquad \text{4.6.6.2(c)}$$

where

$$q = \frac{t_C}{t} G_C \left(\frac{\lambda}{E' E_C G_C} \right)^{1/3} \quad 4.6.6.2(d)$$

$$K = \frac{\delta E_C}{t_C F_C} \quad 4.6.6.2(e)$$

and t_c is core thickness, t is face sheet thickness, δ is initial deflection of face sheet waviness, and F_c is flatwise sandwich strength (the lesser of flatwise core compression or sandwich flatwise tension). The parameter ζ is proportional to the fourth root of the ratio of the core elastic moduli and to the ratio of the core thickness to the ideal buckle wavelength (see Reference 4.6.6.2(b) for details).

A graphical presentation of Q (minimum values of Equation 4.6.6.2(c)) is given in Figure 4.6.6.2(a). The graph can be entered at known values of the abscissa, q, and the ordinate, Q, may be determined after choosing an estimated K curve. Present state of the art does not permit a suitable choice for values of δ. If test values of wrinkling stresses are known, the graph of Figure 4.6.6.2(a) can be used to determine which K curve fits the data and then compute values of δ from Equation 4.6.6.2(e). Changes in design for similar sandwich can then be made by assuming δ to be a constant for that particular type of sandwich and then using the graph of Figure 4.6.6.2(a) to redesign.

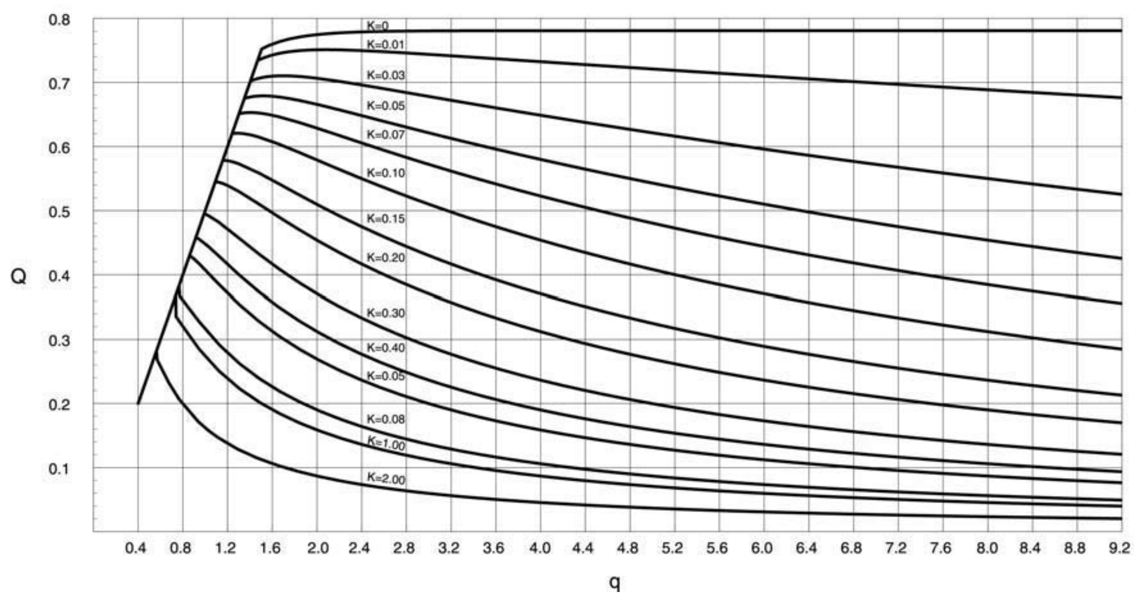


FIGURE 4.6.6.2(a) *Parameters for determining wrinkling of face sheets for sandwiches with continuous cores.*

On the left side of Figure 4.6.6.2(a), the curves terminate in a straight line represented by $Q = 1/2q$. Substitution of this into Equation 4.6.6.2(a) and replacing q by Equation 4.6.6.2(d) results in the formula

$$F_W = \frac{t_C G_C}{2t} \quad 4.6.6.2(f)$$

which is the same result as obtained for the "shear crimping" mode illustrated in Figure 4.4(a) and discussed in Section 4.6.7.

Figure 4.6.6.2(b) presents a graphical means for determining K values from the ratios δ / t_c and F_c / E_c (Reference 4.6.6.2(a)). The graph is also useful in determining δ / t values from known values of F_c/E_c and K.

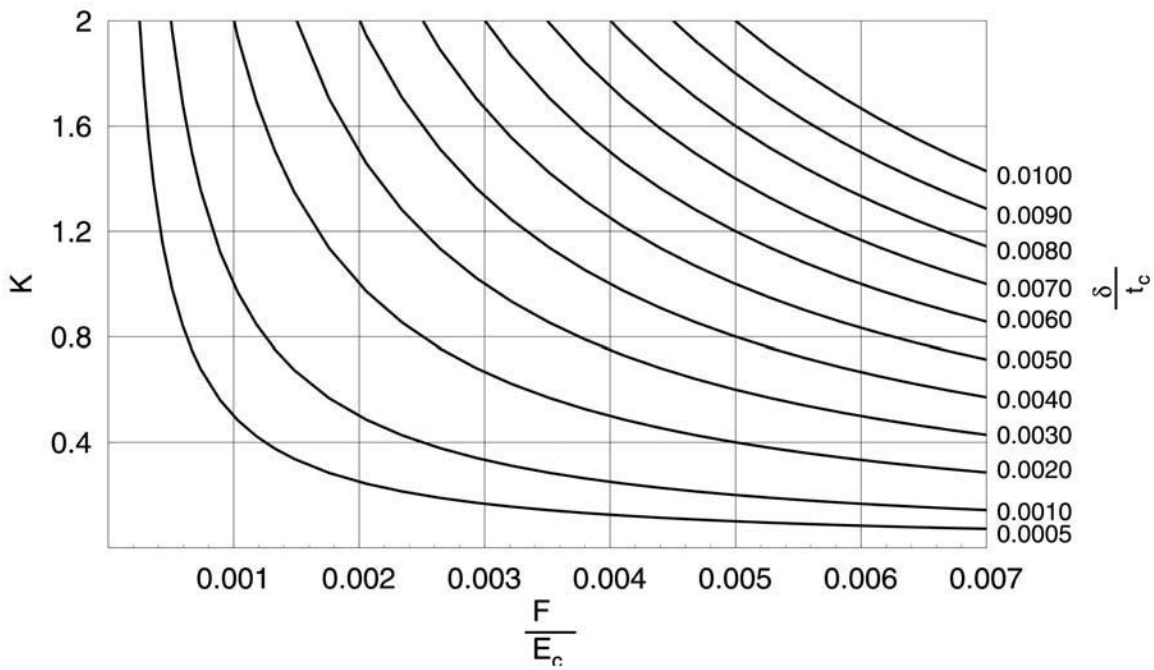


FIGURE 4.6.6.2(b) *Relationship of K to core properties (F_C/E_C) and face sheet waviness (δ/t_C).*

Examples:

1. Check Sandwich Design for Wrinkling

A sandwich has 0.067 inch (0.170 cm) face sheets on a 1 inch (2.54 cm) core. Core elastic properties are E_c = 50,000 psi (3.45 E+08 Pa) and G_c = 20,000 psi (1.38 E+08 Pa). Face sheet elastic modulus is (E'/λ) = 10,000,000 psi (6.89 E+10 Pa) at a design stress of 43,000 psi (2.96 E+08 Pa). The sandwich flatwise strength is 500 psi (3.45 E+06 Pa). Determine whether sandwich face sheets will wrinkle under this design stress.

Values of Q and q were computed from Equations 4.6.6.2(a) and 4.6.6.2(d) to be 0.20 and 1.39, respectively. The graph of Figure 4.6.6.2(a) shows the value of K to be about 1.2 and from the definition of K (Equation 4.6.6.2(e)), δ was computed to be 0.012 inch (0.030 cm) for F_c = 500 psi (3.45 E+06 Pa). This severe an amplitude of waviness for a 0.067 inch (0.170 cm) face sheet seems unlikely, hence wrinkling at design stress would be unlikely even though the equation is not exact.

2. Increase Wrinkling Stress by 50%

For a particular sandwich design, solution of Equation 4.6.6.2(a) from test values results in Q = 0.32 and solution of Equation 4.6.6.2(d) gives q = 3.86. The graph of Figure 4.6.6.2(a) shows that the curve K = 0.20 passes through the point designated by these values of Q and q.

One approach is to increase the sandwich flatwise strength, F_c, for example by changing to a stronger core or a different adhesive between the core and face sheet. In order to increase the value of the wrinkling stress by 50 percent, the value of Q must increase from 0.32 to 0.48, and then the value of K must decrease to about 0.092 for the same value of q. Thus, the core or sandwich flatwise strength, F_c would have to be increased by a factor of about 0.20/0.09 = 2.2 to raise the wrinkling stress by 50 percent.

Another option is to increase the face sheet thickness, t, without changing core or sandwich flatwise strength. This results in sliding up the K = 0.20 curve until Q = 0.48 at which point q = 1.75 and thus t increases by a factor of 3.86/1.75 = 2.2. This would be a bit conservative because it assumes the thicker face sheet has the same waviness, δ, as the thinner face sheet. From Equation 4.6.6.2(a), it is also obvious that an increase in elastic properties will also increase wrinkling stress provided Q is not decreased too much by an increase in q.

4.6.6.3 Sandwich with honeycomb core

Solution of the general expressions for wrinkling of isotropic sandwich face sheets on honeycomb cores leads to somewhat different results because for honeycomb cores the elastic moduli in the plane of the core (E_L, E_W , G_{WL}) are very small compared with elastic moduli in a direction normal to the core plane (E_T, G_{TL}, G_{TW}).

A variety of expressions have been proposed for predicting face sheet wrinkling, depending on whether the wrinkling mode is symmetric or antisymmetric, and also depending on the thickness of the core compared to that of the face sheets ("thick core" vs. "thin core"). Which expression gives the most accurate prediction will depend on the particular materials and sandwich configuration under consideration. The predictions should be validated by testing of representative sandwich coupons.

The most common expressions for face sheet wrinkling take one of two forms. For sandwich panels with "thick" cores, the equation

$$F_W = C_1\left(E' \; E_C \; G_c\right)^{1/3} + C_2 \, G_c \frac{t_c}{t_f} \qquad 4.6.6.3(a)$$

can be used to estimate the wrinkling stress. For "thin" core sandwich structure, the equation

$$F_W = C_3\sqrt{\frac{t_f}{t_c} E_c E'} + C_4 G_C \frac{t_c}{t_f} \qquad 4.6.6.3(b)$$

should be used to determine wrinkling stress. In both of these expressions, F_w is face sheet wrinkling stress, E' is effective face sheet elastic modulus in the direction of the applied load, E_c is core elastic modulus in a direction normal to the sandwich face sheets, and G_c is core shear modulus associated with shear distortion in the plane perpendicular to the face sheets and parallel to the direction of applied load. The coefficients C_1, C_2, C_3, and C_4 vary widely in the literature, and C_2 and C_4 are often shown as zero.

The determination of whether the core of a sandwich panel is considered "thick" or "thin" with regard to wrinkling can be made using the expression (Reference 4.6.6.3(a))

$$t_c \geq 1.82 t_f \sqrt[3]{\frac{E' E_c}{G_c^2}} \qquad \text{4.6.6.3(c)}$$

If t_c satisfies this inequality, then the panel can be considered as “thick” for the purposes of computing wrinkling stresses. Otherwise, the panel should be evaluated using the equation for “thin” sandwich panels.

As previously noted, the values of the coefficients in Equations 4.6.6.3(a) and (b) vary widely in the literature (References 4.6.3, 4.6.6.3(a) through (o)). For example, Equation 4.6.6.3(b) may be found with C_4 = 0 and C_3 ranging from 0.33 to 0.817. A generally conservative approach is to take the following values (References 4.6.3 and 4.6.6.3(a)):

C_1 = 0.247 C_2 = 0.078,
C_3 = 0.33 C_4 = 0

In some cases, this approach may be overly conservative. It is recommended that suitable coefficients be developed by testing sandwich coupons representative of the materials and geometry used in the sandwich structure.

An alternate approach that accounts for the orthotropic nature of a sandwich with composite face sheets is given by minimizing the following equation with respect to the wave number, m:

$$F_W = \frac{\pi^2}{t_f a}\left[D_{11} m^2 + 2\left(D_{12} + 2D_{66}\right)\left(\frac{a}{b}\right)^2 + D_{22}\left(\frac{a}{b}\right)^4 \frac{1}{m^2}\right] + \frac{2E_c a^2}{m^2 \pi^2 t_f t_c} \qquad \text{4.6.6.3(d)}$$

where D_{11}, D_{22}, D_{12}, and D_{66} are the face sheet laminate bending stiffnesses, a is the panel dimension in the direction of the applied load, and b is the panel dimension transverse to the applied load. An extension of this approach to account for shear-extension coupling and bending-extension coupling in face sheets that have asymmetric layups relative to the face sheet midplane can be found in Reference 4.6.6.3(b).

4.6.6.4 Shear face sheet wrinkling

When the sandwich panel is loaded by in-plane shear, the recommended analysis approach is to resolve the loading into principal stresses or strains, and then use the compression wrinkling equations given in Equations 4.6.6.3(a) and (b). If both in-plane principal stresses are compressive, then the combined loads should be considered as noted in Section 4.6.6.5. The analyst must use the appropriate tensor transformations to rotate the face sheet modulus and the core shear moduli into the principal directions to properly compute the wrinkling stress using Equations 4.6.6.3(a) and (b).

In general, principal stresses should be used when the face sheets are composed of homogeneous materials like metal, and principal strains will be more convenient for use when the face sheets are constructed using heterogeneous materials such as laminated composite materials.

4.6.6.5 Face sheet wrinkling - combined loads

Limited data are currently available for combined loadings. If only one of the in-plane principal stresses is in compression, then the tensile principal stress should be neglected and the problem treated as one-dimensional by rotating the sandwich panel stiffnesses into the principal stress direction as described in the previous section.

Plantema (Reference 4.6.6.5(a)) has presented results indicating that the biaxial loading case for sandwich structure with continuous core and isotropic face sheets can be treated by simply comparing the lowest wrinkling load in either of the two main directions.

Ward and Gintert (Reference 4.6.3) give an equation for calculating a margin of safety for combined loads. If the stresses in the face sheet in the x and y directions, f_{xf} and f_{yf}, are both compressive, and the stress in the x direction is larger, or $|f_{xf}| > |f_{yf}|$ then:

$$MS = \frac{1}{\left(\frac{f_{xf}}{F_{xw}}\right)^3 + \left(\frac{f_{yf}}{F_{yw}}\right)} - 1 \qquad 4.6.6.5(a)$$

If the stress in the y direction is larger, or $|f_{yf}| > |f_{xf}|$ then:

$$MS = \frac{1}{\left(\frac{f_{xf}}{F_{xw}}\right) + \left(\frac{f_{yf}}{F_{yw}}\right)^3} - 1 \qquad 4.6.6.5(b)$$

where f_{xf} and f_{yf} are the applied face sheet stresses in the x and y directions, and F_{xw} and F_{yw} are the face sheet wrinkling allowable stresses in the x and y directions.

For highly anisotropic face sheets the directions for the principal stresses do not coincide with directions for the principal strains, and the wrinkling wave may not be perpendicular to the highest compressive principal stress within the sandwich plate. In this case, the wrinkling stress can be found by finding the critical applied stress perpendicular to the wrinkling wave (see Figure 4.6.6.5). This is done by minimizing over the wave angle, φ, to obtain the load factor, λ, so that λF_φ gives the critical combined load (see References 4.6.6.5(b) and (c)):

$$\lambda = \min_{\varphi}\left(\frac{F_{cr,\varphi}}{F_\varphi}\right) \qquad 4.6.6.5(c)$$

where, for symmetrically laminated face sheets,

$$F_\varphi = \sigma_x \cos^2\varphi + \sigma_y \sin^2\varphi + \sigma_{xy} 2\sin\varphi\cos\varphi$$

$$F_{\varphi,cr} = \frac{Q_3}{t_f}\sqrt[3]{D_{11,\varphi}E_c G_c}$$

and 4.6.6.5(d)

$$D_{11,\varphi} = D_{11}\cos^4\varphi + D_{22}\sin^4\varphi + 2\left(D_{12} + 2D_{66}\right)\sin^2\varphi\cos^2\varphi + 4\left(D_{16}\cos^2\varphi + D_{26}\sin^2\varphi\right)\sin\varphi\cos\varphi$$

D_{11}, D_{22}, D_{12}, D_{16}, D_{26}, and D_{66} are the face sheet laminate bending stiffnesses in the x-y coordinate system of Figure 4.6.6.5 and Q_3 is a coefficient. A value of Q_3 = 1.2 is recommended to provide a conservative value of the predicted wrinkling load factor.

4.6.6.6 Face sheet wrinkling - curved panels

Little data and methods are currently available for curved panels. However, some test data exist (Reference 4.6.6.6) which suggest that the wrinkling load is the same for curved panels as for flat and the

influence of curvature may practically be neglected if the radii of curvature of the sandwich shell are some orders of magnitude larger than the wavelength of the wrinkles. Therefore, for shallow and moderately curved sandwich constructions, it is recommended that the above flat panel equations be used for wrinkling of curved panels.

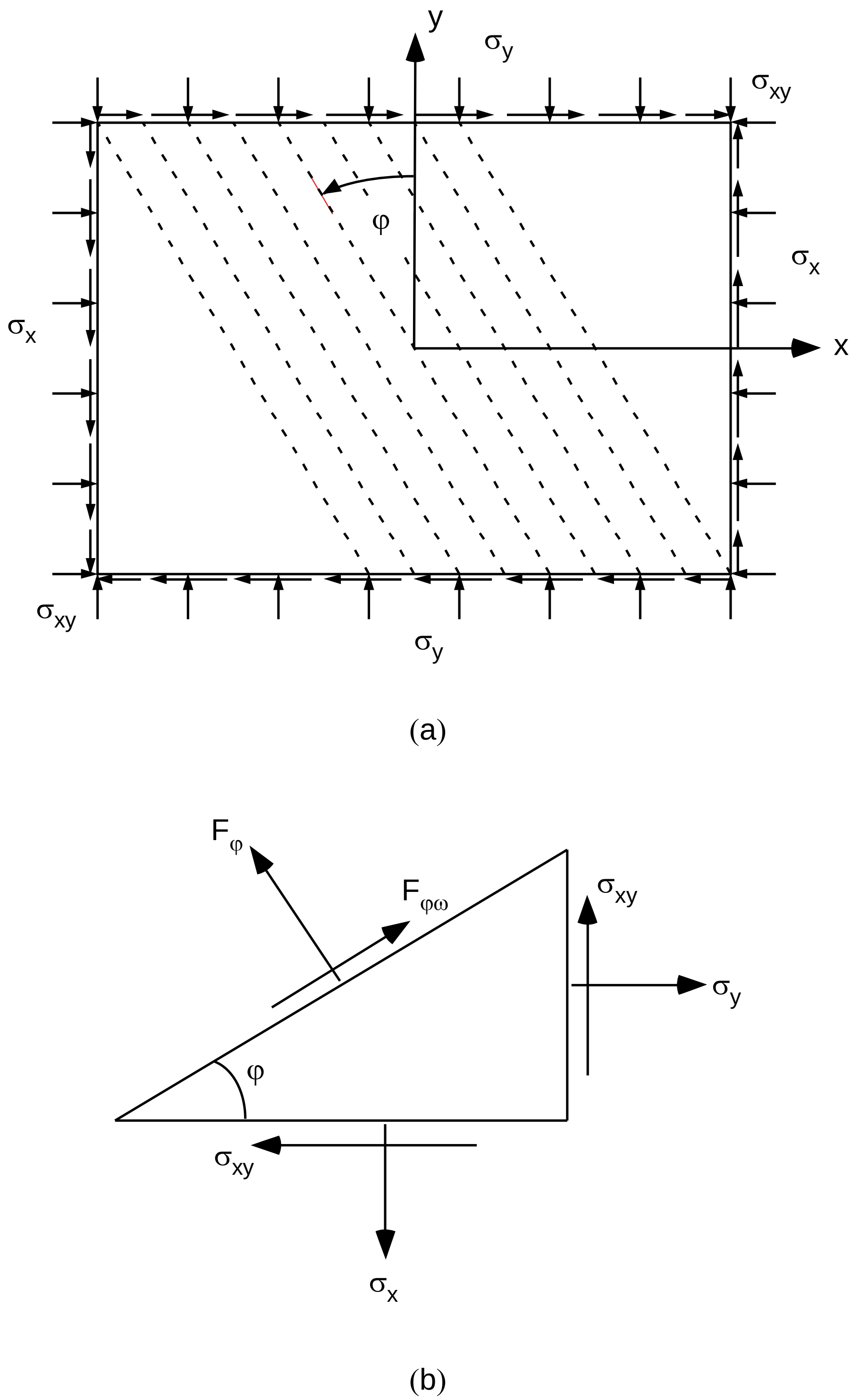


FIGURE 4.6.6.5 *(a) Face sheet subject to combined load with wrinkling wave at angle φ to the plate coordinate system, (b) load definition.*

4.6.7 Core shear crimping

Core shear crimping is a general instability failure in which the wavelength of the buckles is very small because the core shear modulus is low. The crimping of a sandwich occurs suddenly and usually causes the core to fail in shear at the crimp; it may also cause failure in the bond between the face sheet and the core.

The shear crimping mode is defined by a limit of the general buckling modes considering thin face sheets, resulting in the following equation for the critical face sheet stress (Reference 4.6.7(a)):

$$F_c = \frac{h^2 G_C}{\left(t_{UPR} + t_{LWR}\right) t_c} \qquad 4.6.7(a)$$

When the two face sheets have the same thickness, t, and the face sheet thickness is very small compared to the core thickness t_c, then $t_c \approx h$, and this equation may be written (Reference 4.6.7(b)):

$$F_c = \frac{t_c G_c}{2t} \qquad 4.6.7(b)$$

The compression and shear allowables for crimping may be computed for the X, Y and XY directions by substituting G_{xz}, G_{yz} and $(G_{xz}G_{yz})^{½}$, respectively for G_c.

When computing the shear crimping margins of safety, a suitable interaction equation for biaxial and/or shear loading is not known, so a margin of safety is computed for each direction individually. Only compressive stresses produce buckling, and all tensile stresses should be ignored.

4.6.8 Attachments and hard points

4.6.8.1 Design of flat circular sandwich panels loaded at an insert

This section presents information for the design of a sandwich panel with a rigid insert. The insert is placed in the sandwich panel to allow the introduction of load from a point outside the plane of the panel. The basic equations for deflections and stresses (Reference 4.6.8.1(a)) were derived for loads normal to the panel. Experimental data (Reference 4.6.8.1(b)), however, have shown that the equations were also satisfactory for loads inclined at angles between 45° and 90° to the panel plane provided that the equations were modified to utilize the normal component of the inclined load. Deflections at the insert and stresses in the neighborhood of the insert are the bases for design.

Although the design equations were derived for circular panels, their application to panels of other shapes would not be expected to be in great error if the insert size was relatively small compared with the panel size.

Usually the design of a panel with an insert will begin with a panel configuration based upon in-plane or normal distributed loading as covered by other sections of this chapter, thus resulting in known face sheet and core thicknesses and core shear properties. An insert is then to be placed in the panel to allow introduction of load, and the size of the insert must be determined, as well as the stresses and deflections caused by the load applied at the insert. The procedure followed here will be to determine insert size based on core shear stress limitations and then to check face sheet stresses and sandwich deflections.

Assuming that the design begins with chosen design stresses and a given load to transmit, a sandwich panel with a loaded insert should be designed to comply with the basic design principles summarized in Section 4.2.1. These conditions must be met. The other failure modes listed in Section 4.4 should be checked separately.

Detailed procedures giving theoretical equations and graphs for determining insert size, face sheet stresses, and panel deflections are given in the following paragraphs. The equations in this section were derived for isotropic face sheet and core. Where double equations are given, one equation is for sandwich with face sheets of different materials and thicknesses and a second equation is for sandwich with each face sheet of the same material and thickness. Face sheet and core elastic properties and stresses shall be values at the condition of use; that is, if application is at elevated temperature then properties at elevated temperature shall be used in design. The following procedures are limited to linear elastic behavior.

The procedure is given below for determining the insert diameter so that the shear stress in the sandwich core will not exceed allowable values. The core shear stress is given by the theoretical formula (Reference 4.6.8.1(a)):

$$F_{sc} = \frac{k_r\ P}{2\ \pi\ d\ r_i} \qquad 4.6.8.1(a)$$

where P is the normal component of the load applied at the insert, d is distance between face sheet centroids, r_i is insert radius, and k_r is a coefficient dependent upon r_i / r_p and ϕ_r where

$$\phi_r = r_p \left(\frac{dG_c}{D_F} \right)^{\frac{1}{2}} \qquad 4.6.8.1(b)$$

where r_p is the radius of the circular sandwich plate that contains the insert, G_c is core shear modulus and

$$D_F = \frac{1}{12}\left(\frac{E_{UPR}\ t^3_{UPR}}{\lambda_{UPR}} + \frac{E_{LWR}\ t^3_{LWR}}{\lambda_{LWR}} \right) \qquad 4.6.8.1(c)$$

$$D_F = \frac{Et^3}{6\lambda} \qquad \text{(for equal face sheets)}$$

where E is modulus of elasticity of face sheet, $\lambda = 1 - \nu^2$, ν is face sheet Poisson's ratio, t is face sheet thickness, and UPR and LWR are subscripts denoting the upper and lower face sheets, respectively.

The radial distribution of the core shear stress coefficient k_r is shown in Figure 4.6.8.1(a) for a panel in which $r_i / r_p = 0.04$. The maximum shear stress coefficient occurs near the insert for large values of ϕ_r and moves away from the insert and becomes smaller as values of ϕ_r decrease. Therefore, for fairly large panels, it is not wise to use a stiff core if shear stresses approach the design allowables; using a core with lower G_c will decrease the core shear stress coefficient.

The maximum core shear stress upon which the design must be based is given by the equation (Reference 4.6.8.1(a)):

$$F_{sc\,max} = \frac{k_3 P}{2\,\pi\,d\,r_i} \qquad 4.6.8.1(d)$$

where k_3 is given in the graph of Figure 4.6.8.1(b), and represents a maximum value of k_r. Solving this equation for r_i results in

$$r_i = \frac{k_3 P}{2 \pi d\, F_{sc\,max}} \qquad 4.6.8.1(e)$$

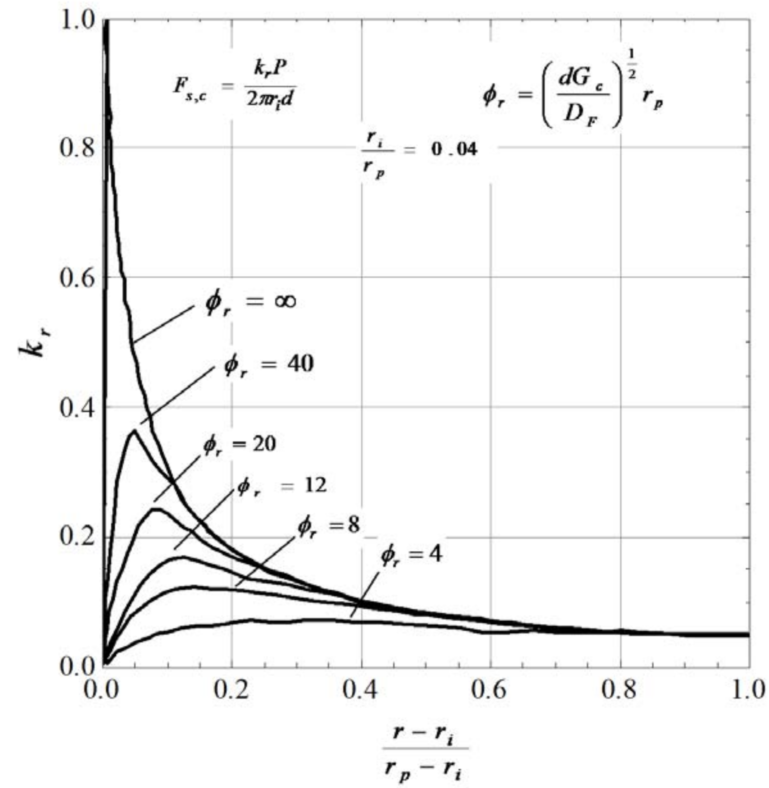


FIGURE 4.6.8.1(a) *Radial distribution of core shear stress coefficient k_r. Rim of panel simply supported. Insert size r_i / r_p = 0.04.*

This equation cannot be solved directly because the coefficient k_3 is dependent upon values of r_i as given by the abscissa, r_i / r_p, for Figure 4.6.8.1(b). A means of indirect solution of Equation 4.6.8.1(e) can be devised as follows: solving Equation 4.6.8.1(e) for k_3 in terms of r_i / r_p results in

$$k_3 = C \left(\frac{r_i}{r_p}\right) \qquad 4.6.8.1(f)$$

where

$$C = \frac{2 \pi d\, r_p\, F_{sc\,max}}{P} \qquad 4.6.8.1(g)$$

Equation 4.6.8.1(f) represents a family of straight lines having slope, C, and extending from the origin of the graph of Figure 4.6.8.1(b). The value of C is determined from Equation 4.6.8.1(g) with known dimensions, stress, and load. A solution of Equation 4.6.8.1(f) is obtained from the intersection of a straight line of slope C with the appropriate (ϕ_r) curve on the graph of Figure 4.6.8.1(b). The value of the insert radius, r_i is obtained by multiplying the abscissa of the intersection point by the panel radius, r_p. This can

be checked by reading k_3 at the intersection point and substituting into Equation 4.6.8.1(f). Examples illustrating this procedure will follow at the end of this section.

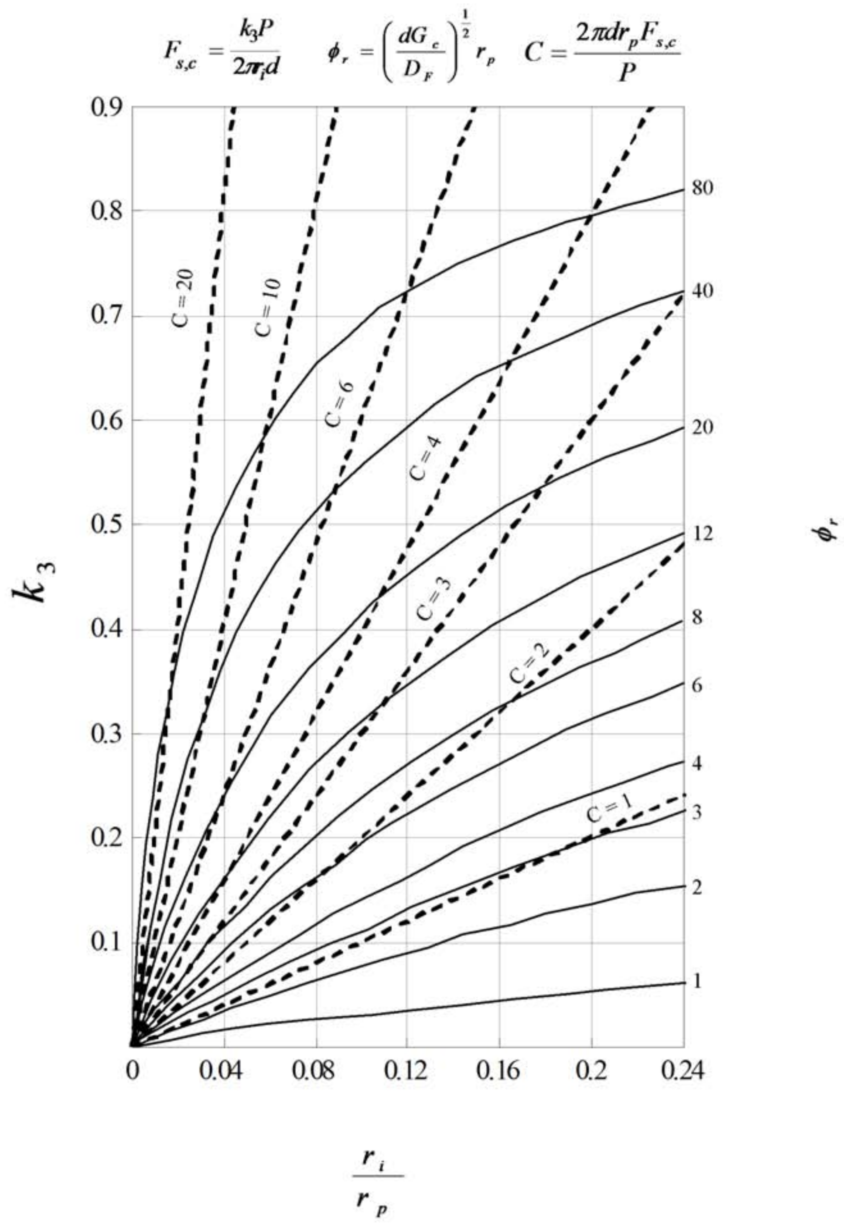


FIGURE 4.6.8.1(b) *Coefficient k_3 for determining maximum core shear stress; rim simply supported. (For a clamped rim k_3 values are conservative for $\phi r < 6$.)*

The maximum face sheet stress occurs at the insert in the radial direction and is given by the formula

$$F_{UPR} = \frac{k_4 P}{4\pi t_{UPR} d}$$

$$F_{LWR} = \frac{k_4 P}{4\pi t_{LWR} d} \tag{4.6.8.1(h)}$$

where F_{UPR} and F_{LWR} are the stresses in the upper and lower face sheet of thicknesses t_{UPR} and t_{LWR}, respectively, and k_4 is given by the graph of Figure 4.6.8.1(c). Curves are given, in Figure 4.6.8.1(c), for panels with outer rim simply supported or clamped.

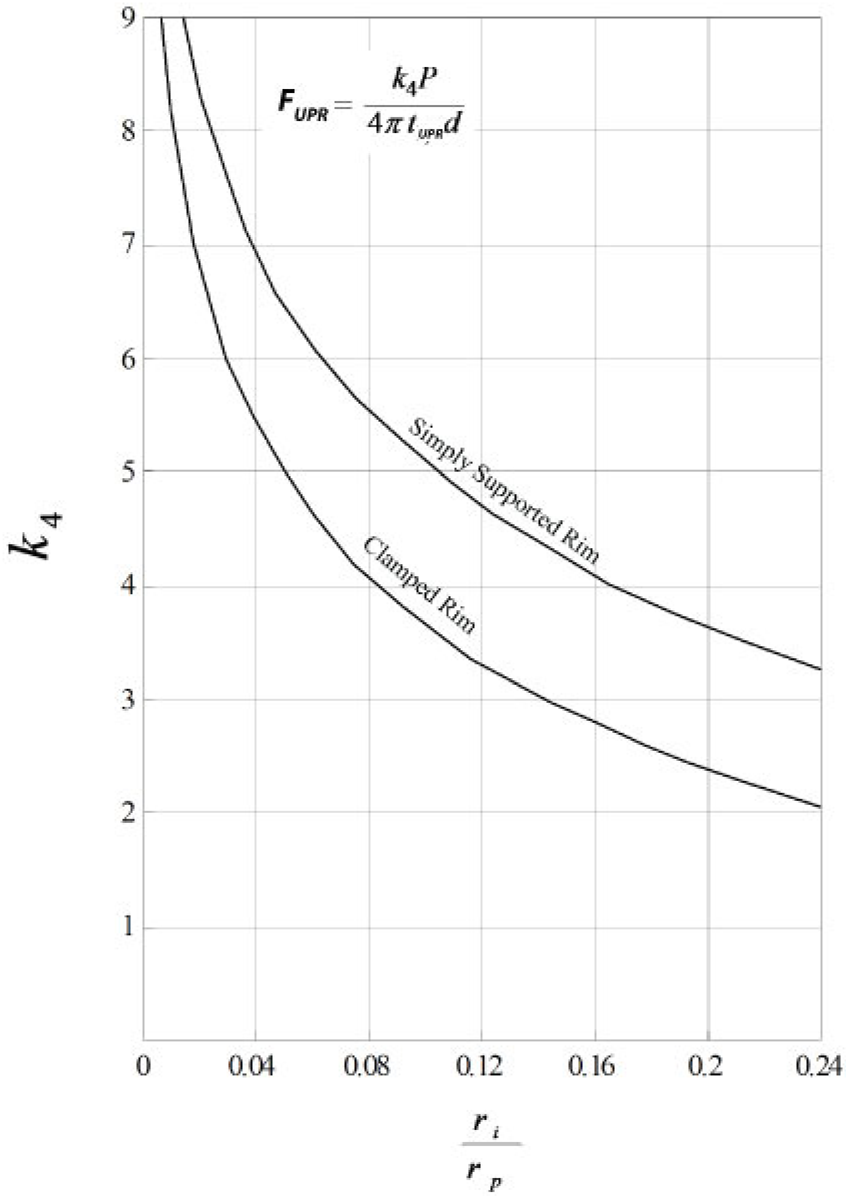


FIGURE 4.6.8.1(c) *Coefficient k_4 for determining maximum face sheet stress.*

The deflection at the insert is given in two parts, bending deflection, δ_B, and shear deflection, δ_s. The bending deflection is given by

$$\delta_B = \frac{k_1 P r_p^2}{16 \pi D} \qquad 4.6.8.1(i)$$

where k_1 is given by the graph of Figure 4.6.8.1(d) for panels with outer rim simply supported or clamped; and D is the sandwich bending stiffness given by

$$D = \frac{E_{UPR}\, t_{UPR}\, E_{LWR}\, t_{LWR}\, d^2}{\lambda\left(E_{UPR}\, t_{UPR} + E_{LWR}\, t_{LWR}\right)} \qquad 4.6.8.1(j)$$

$$D = \frac{E\, t\, d^2}{2\lambda}$$

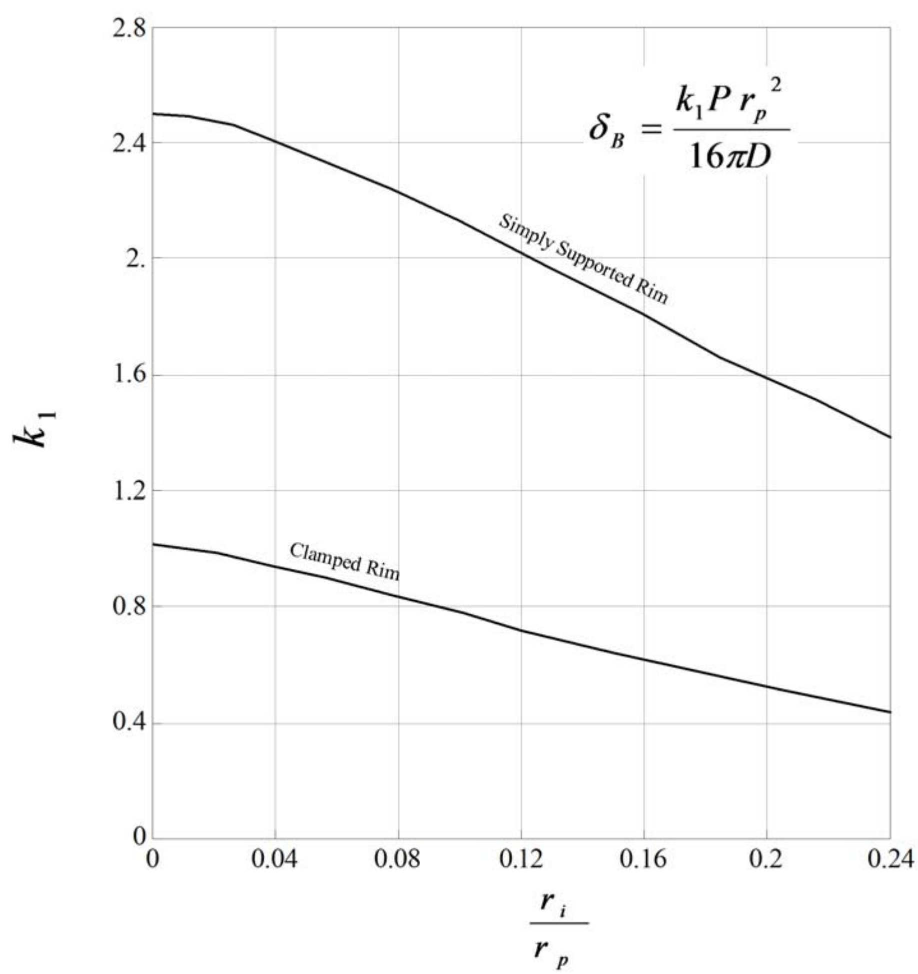


FIGURE 4.6.8.1(d) *Coefficient k_1 for determining bending deflection at insert.*

The shear deflection at the insert is given by the formula

$$\delta_S = \frac{k_2 P}{2\pi d G_c} \qquad 4.6.8.1(k)$$

where k_2 is given by the graph of Figure 4.6.8.1(e).

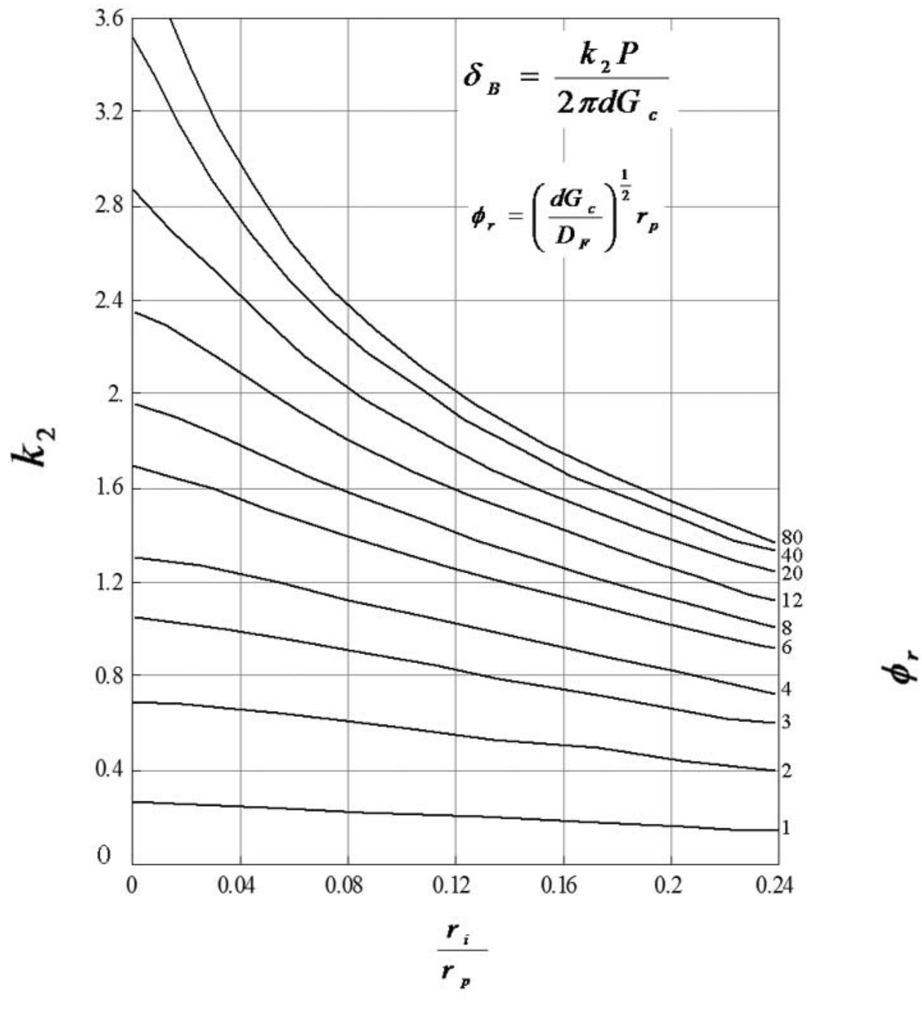


FIGURE 4.6.8.1(e) *Coefficient k_2 for determining shear deflection at insert; rim simply supported. (For a clamped rim k_2 values are conservative for $\phi r < 8$.)*

Examples:

1. A sandwich has 0.080 inch (0.203 cm) face sheets on a core 1.12 inch (2.85 cm) thick. Face sheets and core are isotropic. The face sheet modulus of elasticity is E = 10,000,000 psi (6.89 E+10 Pa) and Poisson's ratio is ν = 0.3. The core shear modulus is G_c = 2,000 psi (1.38E+07 Pa) and the core shear design stress F_{sc} = 60 psi (4.14 E+05 Pa). The plate has a radius of r_p = 25

inches (64 cm) and an insert supporting a normal load of 1,900 pounds. Determine the insert size, stresses and deflections.

From Equation 4.6.8.1(c), D_F = 940 lb-in.^{2}/in. of width, from Equation 4.6.8.1(b), ϕ_r = 40, and from Equation 4.6.8.1(g), C = 5.95. From Figure 4.6.8.1(b), the intersection of the line C = 5.95 with the curve ϕ_r = 40 occurs at r_i / r_p = 0.091 and k_3 = 0.54. From these values, r_i = 0.091(25) = 2.28 in. (5.79 cm), thus giving an insert diameter of 4.56 inches (11.6 cm). Checking by substituting k_3 = 0.54 into Equation 4.6.8.1(e) also produces the same insert size.

From Figure 4.6.8.1(c), it is determined that k_4 = 5.26 for the rim simply supported; substitution of this into Equation 4.6.8.1(h) results in a face sheet stress of 8,280 psi (5.71 E+07 Pa).

From Equation 4.6.8.1(j), the sandwich bending stiffness is found to be D = 633,000 lb-in^2/in. of width. From the graph of Figure 4.6.8.1(d), it is determined that k_1 = 2.16 for the rim simply supported and substitution into Equation 4.6.8.1(i) results in a bending deflection of 0.081 inch (0.206 cm). From Figure 4.6.8.1(e), k_2 = 2.15 and substitution into Equation 4.6.8.1(k) results in a shear deflection of 0.271 inch (0.688 cm). Thus the total insert deflection is 0.352 inch (0.894 cm).

2. A sandwich is the same size as the previous example with the same material in the face sheets, but a core 10 times as stiff (G_c = 20,000 psi (1.38 E+08 Pa)) and with core shear strength of F_{sc} = 300 psi (2.07 E+06 Pa). For these values, ϕ_r = 126 and C = 29.8, and these values are off the scale of Figure 4.6.8.1(b). A conservative solution can be made by assuming k_3 = 1. Then from Equation 4.6.8.1(e) r_i = 0.84 inch (2.13 cm), thus giving an insert diameter of 1.68 inch (4.27 cm). Proceeding as in the previous example we determine:

Face sheet stress F	= 11,460 psi (7.90 E+07 Pa)
Bending deflection δ_B	= 0.090 inch (0.229 cm)
Shear deflection δ_S	= 0.044 inch (0.113 cm)
Total deflection	= 0.134 inch (0.340 cm)

4.7 FLAT PANEL INTERNAL LOADS AND STRESSES - PRESSURE LOADING

This section discusses sandwich structure subjected to pressure loading. Section 4.7.1 discusses a general approach for flat sandwich panels subjected to pressure loading (applied in the direction through the thickness of the sandwich). Section 4.7.2 discusses calculation of stress and deflection in simply-supported panels under pressure loading, for rectangular panels in Section 4.7.2.1 and for circular panels in Section 4.7.2.2.

4.7.1 Design of flat rectangular sandwich panels under various normal loadings

The equilibrium equations given here for a flat sandwich panel under pressure loading are based on the first-order shear deformation theory for a thick, anisotropic plate. In terms of loads and bending moments, the coupled differential equations can be written

$$\begin{aligned} &\frac{\partial V_x}{\partial x} + \frac{\partial V_y}{\partial y} = q(x, y) \\ &V_x = \frac{\partial M_x}{\partial x} - \frac{\partial M_{xy}}{\partial y} \\ &V_y = \frac{\partial M_y}{\partial y} - \frac{\partial M_{xy}}{\partial x} \end{aligned} \qquad 4.7.1(a)$$

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where $q(x,y)$ is the pressure load. The shear loads and bending moments are defined as in Section 4.5.2 and the transverse shear strains and curvatures in those definitions are written in terms of the out-of-plane displacement w, and rotations about the midplane, ψ_x and ψ_y, that are independent of the slope of w:

$$\gamma_{xz} = \psi_x + \frac{\partial w}{\partial x}; \quad \gamma_{yz} = \psi_y + \frac{\partial w}{\partial y}$$
$$\kappa_x = \frac{\partial \psi_x}{\partial x}; \quad \kappa_y = \frac{\partial \psi_y}{\partial y}; \quad \text{and} \quad \kappa_{xy} = \frac{\partial \psi_x}{\partial y} + \frac{\partial \psi_y}{\partial x} \qquad 4.7.1(b)$$

These equations may be solved using an energy method in which the displacement distributions are assumed functions that satisfy the required displacement boundary conditions. For example,

$$w(x,y) = \sum_{m=0}^{m\max} \sum_{n=0}^{n\max} c_{mn} \varphi_m(x) \varphi_n(y) \qquad 4.7.1(c)$$

where c_{mn} are unknown coefficients to be determined through minimization of potential energy, and $\phi_m(x)$ and $\phi_n(y)$ are displacement functions.

4.7.2 Design of flat sandwich panels under uniformly distributed normal load

Assuming that a design begins with chosen design stresses and deflections and a given load to transmit, a flat rectangular or circular panel of sandwich construction under uniformly distributed normal load shall be designed to comply with the basic design principles summarized in Section 4.2.1. These conditions must be met. This section addresses global buckling. The other failure modes listed in Section 4.4 should be checked separately.

Detailed procedures giving theoretical equations and graphs for determining dimensions of the face sheets and core, as well as necessary core properties, are given in the following paragraphs for simply-supported flat panels. Double equations are given, one equation for sandwich with isotropic face sheets of different materials and thicknesses, and another equation for sandwich with each face sheet of the same isotropic material and thickness. Face sheet modulus of elasticity, E, and stress values, F, shall be compression or tension values at the conditions of use; that is, if the application is at elevated temperature, then face sheet properties at elevated temperature shall be used in design. For many combinations of face sheet materials, it will be found advantageous to choose thicknesses such that $E_{UPR}\ t_{UPR} = E_{LWR}\ t_{LWR}$. The following procedures are restricted to linear elastic behavior.

4.7.2.1 Determining face sheet thickness, core thickness, and core shear modulus for simply supported flat rectangular panels under uniform load

This section gives procedures for determining face sheet and core thicknesses and core shear modulus so that chosen design face sheet stresses and allowable panel deflections will not be exceeded under a given uniformly distributed normal load. The face sheet stresses, produced by bending moment, are a maximum at the center of a simply supported panel under uniformly distributed pressure load. If restraint exists at the panel edges, a redistribution of stresses may cause higher stresses near panel edges. These procedures apply only to panels with simply-supported edges.

Because face sheet stresses are caused by bending moment, they depend not only upon face sheet thickness, but also upon the distance between the face sheets, hence the core thickness. Panel stiffness, and therefore deflection, is likewise dependent on face sheet and core stiffness.

If the panel is designed so that face sheet stresses are at chosen design levels, the panel deflection may be larger than allowable, in which case the core or face sheets must be thickened and the design

face sheet stress lowered in order to meet deflection requirements. A solution is presented in the form of charts with which, by iterative process, the face sheet and core thicknesses and core shear modulus can be determined.

The average face sheet stresses, F_{UPR} and F_{LWR} (stress at the centroid of upper and lower face sheets, respectively), in the b direction[1] are given by the equations:

$$F_{UPR} = K_2 \frac{q\, b^2}{d\, t_{UPR}}$$

$$F_{LWR} = K_2 \frac{q\, b^2}{d\, t_{LWR}} \qquad 4.7.2.1(a)$$

$$F = K_2 \frac{q\, b^2}{d\, t} \quad \text{(for equal face sheets)}$$

where q is the intensity of the distributed load, b is the panel width, d is the distance between face sheet centroids, t is face sheet thickness, and UPR and LWR are subscripts denoting upper and lower face sheets, respectively.

K_2 is a theoretical coefficient dependent on panel aspect ratio and sandwich bending and shear rigidities. If the core is isotropic (shear moduli alike in the two principal directions), K_2 depends only upon panel aspect ratio. For sandwich with orthotropic core, the values of K_2 are dependent not only on panel aspect ratio but also upon sandwich bending and shear rigidities as incorporated in the parameter V:

$$V = \frac{\pi^2 D}{b^2 U}$$

Using the definition of D (see Section 4.5), V can be written as:

$$V = \frac{\pi^2\, t_c\, E_{UPR}\, t_{UPR}\, E_{LWR}\, t_{LWR}}{\left(E_{UPR}\, t_{UPR} + E_{LWR}\, t_{LWR}\right) \lambda\, b^2\, G_c} \qquad 4.7.2.1(b)$$

$$V = \frac{\pi^2\, t_c\, E\, t}{2 \lambda\, b^2\, G_c} \quad \text{(for equal face sheets)}$$

where U, the sandwich shear stiffness, has been taken to be S, as defined in Equation 4.5.1(j), E is the modulus of elasticity of the face sheet, $\lambda = 1 - \nu^2$, ν is the Poisson's ratio of the face sheets (in this equation it is assumed that $\nu = \nu_{UPR} = \nu_{LWR}$), and G_C is the core shear modulus associated with the axes parallel to the panel side of length a and perpendicular to the plane of the panel. The core shear modulus associated with axes parallel to panel side of width b and perpendicular to the plane of the panel is denoted by (RG_C). For sandwich with corrugated core having corrugation flutes parallel to panel side of length a, the parameter V is replaced by the parameter.

[1] For panels nearly square ($b/a > 0.4$), made of sandwich with orthotropic cores having greater rigidity in the a direction than the b direction, the face sheet stress may be greater in the a direction than in the b direction, and this stress is dependent upon K'_2 given in Figures 4.7.2.1.3(d) through 4.7.2.1.3(f).

$$V_2 = \frac{\pi^2 t_c E_{UPR} t_{UPR} E_{LWR} t_{LWR}}{\left(E_{UPR} t_{UPR} + E_{LWR} t_{LWR}\right) \lambda b^2 G_{cb}} \qquad \text{4.7.2.1(c)}$$

$$V_2 = \frac{\pi^2 t_c E t}{2 \lambda b^2 G_{cb}} \quad \text{(for equal face sheets)}$$

where G_{cb} is the core shear modulus associated with the axes perpendicular to direction of the corrugation flutes (parallel to panel side of length b) and perpendicular to the plane of the panel.

Solving Equation 4.7.2.1(a) for $\frac{d}{b}$ gives

$$\frac{d}{b} = \sqrt{K_2} \frac{\sqrt{\frac{q}{F_{UPR}}}}{\sqrt{\frac{t_{UPR}}{d}}} \qquad \frac{d}{b} = \sqrt{K_2} \frac{\sqrt{\frac{q}{F_{LWR}}}}{\sqrt{\frac{t_{LWR}}{d}}} \qquad \text{4.7.2.1(d)}$$

$$\frac{d}{b} = \sqrt{K_2} \frac{\sqrt{\frac{q}{F}}}{\sqrt{\frac{t}{d}}} \quad \text{(for equal face sheets)}$$

Equation 4.7.2.1(d) represents two equations, one with F_{UPR} and t_{UPR}, and the other with F_{LWR} and t_{LWR}. The value of $\frac{d}{b}$ should be the same regardless of which face sheet is used for the calculation. It is recommended to calculate the value of $\frac{d}{b}$ using the equation for one face sheet, and then use that value in Equation 4.7.2.1(a) to calculate the stress in the other face sheet, to verify that it is within acceptable limits.

A chart for solving Equation 4.7.2.1(d) graphically is given in Figures 4.7.2.1(a) through 4.7.2.1(c). The equations and charts include the ratio $\frac{t}{d}$, which is usually unknown; but by iteration satisfactory ratios of $\frac{t}{d}$ and $\frac{d}{b}$ can be found.

The deflection, δ, of the panel center is given by the equation:

$$\delta = \frac{K_1}{K_2} \frac{\lambda F_{UPR}}{E_{UPR}} \left(1 + \frac{E_{UPR} t_{UPR}}{E_{LWR} t_{LWR}}\right) \frac{b^2}{d}$$

$$\delta = \frac{K_1}{K_2} \frac{\lambda F_{LWR}}{E_{LWR}} \left(1 + \frac{E_{LWR} t_{LWR}}{E_{UPR} t_{UPR}}\right) \frac{b^2}{d} \qquad \text{4.7.2.1(e)}$$

$$\delta = 2 \frac{K_1}{K_2} \frac{\lambda F}{E} \frac{b^2}{d} \quad \text{(for equal face sheets)}$$

where K_1 is a coefficient dependent upon panel aspect ratio and the value of V or V_2. In sandwich with corrugated core, K_1 also depends on the ratio between sandwich bending stiffness parallel and perpendicular to the corrugation flutes.

Solving Equation 4.7.2.1(e) for $\frac{d}{b}$ gives

$$\frac{d}{b}=\frac{\sqrt{\frac{K_1}{K_2}}\sqrt{\frac{\lambda F_{UPR}}{E_{UPR}}}\sqrt{\left(1+\frac{E_{UPR}\,t_{UPR}}{E_{LWR}\,t_{LWR}}\right)}}{\sqrt{\frac{\delta}{d}}}$$

$$\frac{d}{b}=\frac{\sqrt{\frac{K_1}{K_2}}\sqrt{\frac{\lambda F_{LWR}}{E_{LWR}}}\sqrt{\left(1+\frac{E_{LWR}\,t_{LWR}}{E_{UPR}\,t_{UPR}}\right)}}{\sqrt{\frac{\delta}{d}}} \qquad 4.7.2.1(f)$$

$$\frac{d}{b}=\frac{\sqrt{\frac{2K_1}{K_2}}\sqrt{\frac{\lambda F}{E}}}{\sqrt{\frac{\delta}{d}}} \quad \text{(for equal face sheets)}$$

Charts for solving Equation 4.7.2.1(f) are given in Figures 4.7.2.1(d) through 4.7.2.1(h). Use of the equations and charts beyond $\frac{\delta}{d}=0.5$ is not recommended.

Figures 4.7.2.1(a) to 4.7.2.1(c) are for finding $\frac{d}{b}$, given a desired value of face sheet stress, F_{UPR} or F_{LWR}. Figure 4.7.2.1(a) includes curves for sandwich with isotropic core (R = 1), and certain orthotropic cores (R = 0.4 or 2.5), where R is the ratio of core shear stiffness in the b direction to that in the a direction ($G_{cb} = R\ G_c$). Figures 4.7.2.1(b) and (c) apply to corrugated core with the core flutes perpendicular to the panel edge of length a (G_{cb} = ∞) and parallel to the edge of length a (G_c = ∞), respectively.

Figures 4.7.2.1(d) through (h) are for finding $\frac{d}{b}$, given a desired value of deflection, $\frac{\delta}{d}$. Figure 4.7.2.1(d) applies to sandwich with isotropic cores ($R = 1$). Figure 4.7.2.1 (e) and (f) apply to sandwich with orthotropic cores for which $R = 0.4$ and $R = 2.5$, respectively. Figure 4.7.2.1 (g) applies to sandwich with corrugated core having the core flutes perpendicular to the panel edge of length a ($G_{cb} = \infty$). Figure 4.7.2.1 (h) applies to sandwich with corrugated core having the core flutes parallel to the panel edge of length a ($G_c = \infty$), and requires values of the parameter V_2 given by Equation 4.7.2.1(c) instead of values of V.

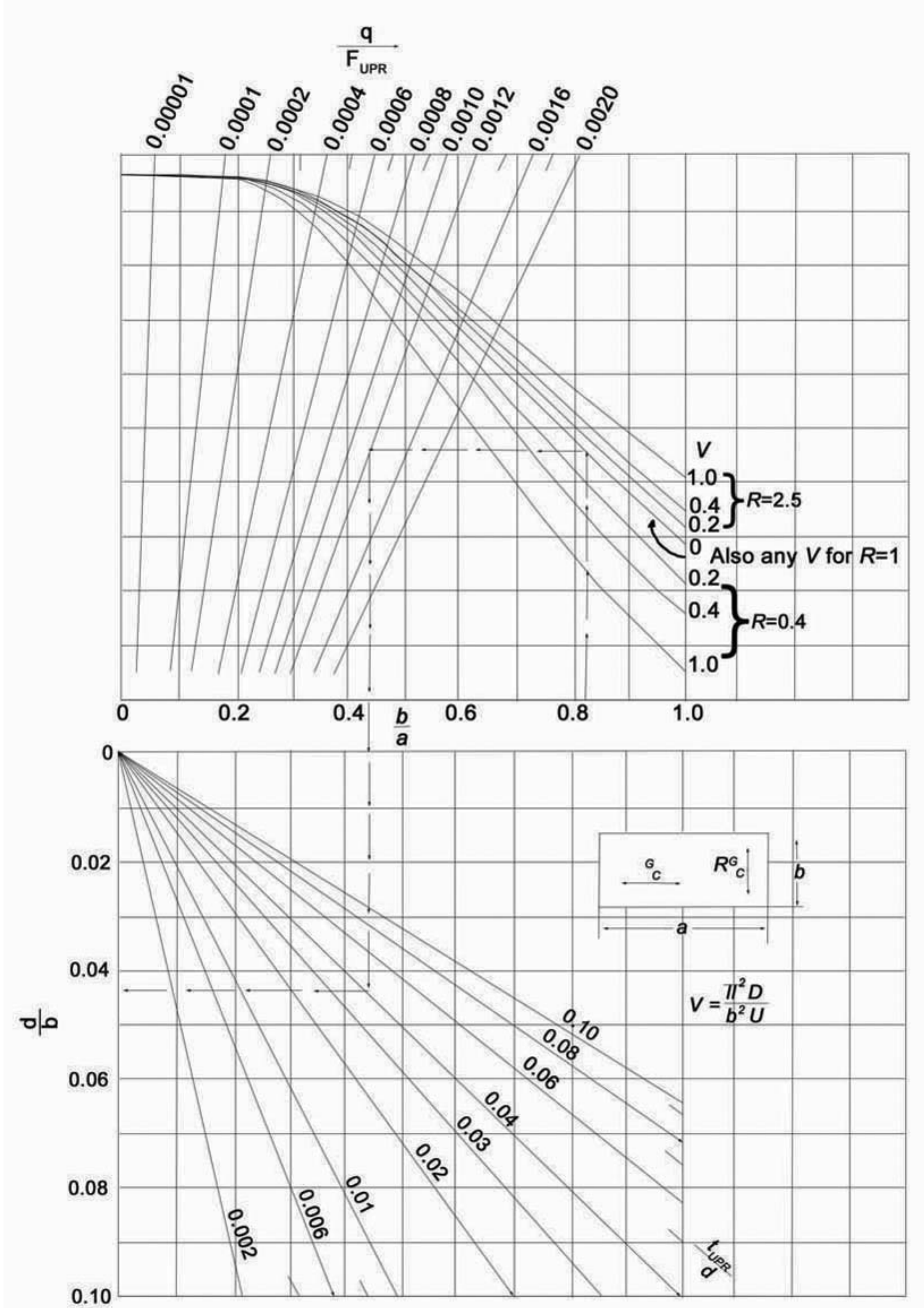

FIGURE 4.7.2.1(a) *Chart for determining $\frac{d}{b}$ ratio for flat rectangular sandwich panel, with isotropic face sheets and isotropic core ($R = 1$) or orthotropic core ($R = 0.4$ or 2.5), under uniformly distributed normal load, q, so that face sheet stress will be F_{UPR} or F_{LWR}.*

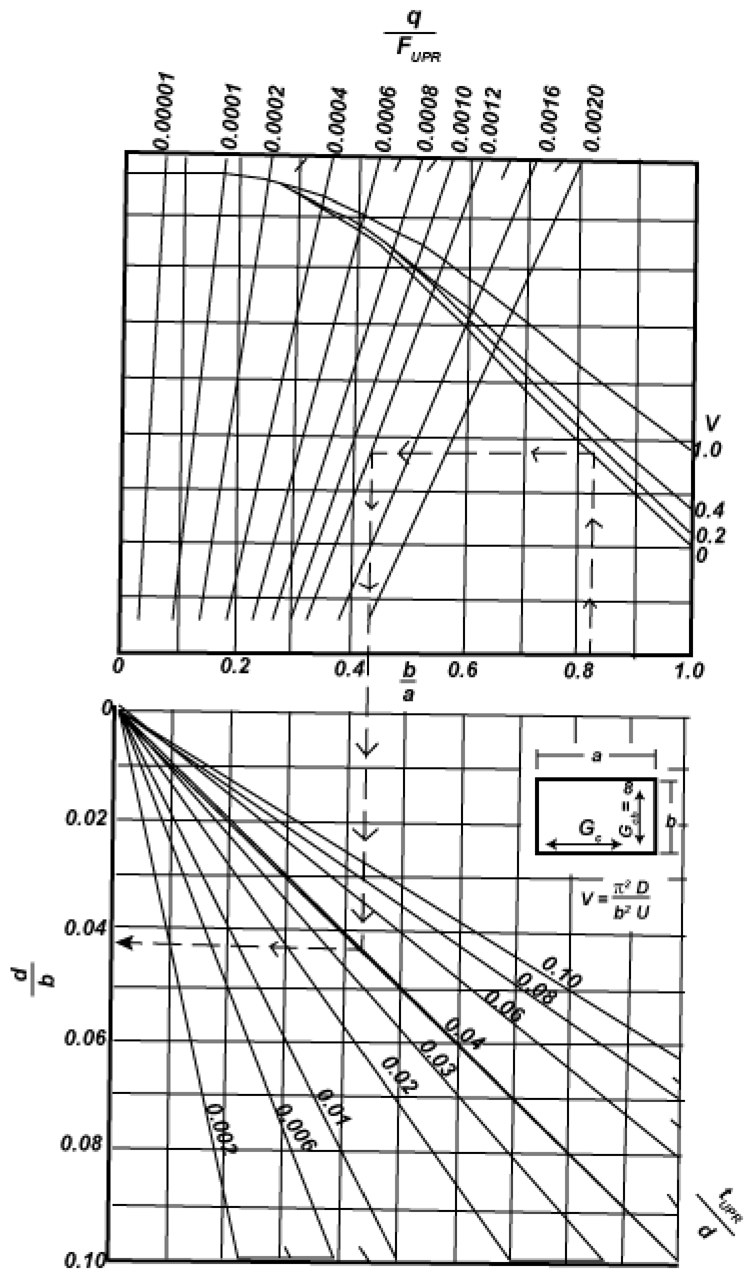

FIGURE 4.7.2.1(b) *Chart for determining h/b ratio for flat rectangular sandwich panel, with isotropic face sheets and corrugated core with flutes perpendicular to panel edge of length a ($G_{cb} = \infty$), under uniformly distributed normal load, q, so that face sheet stress will be F_{UPR} or F_{LWR}.*

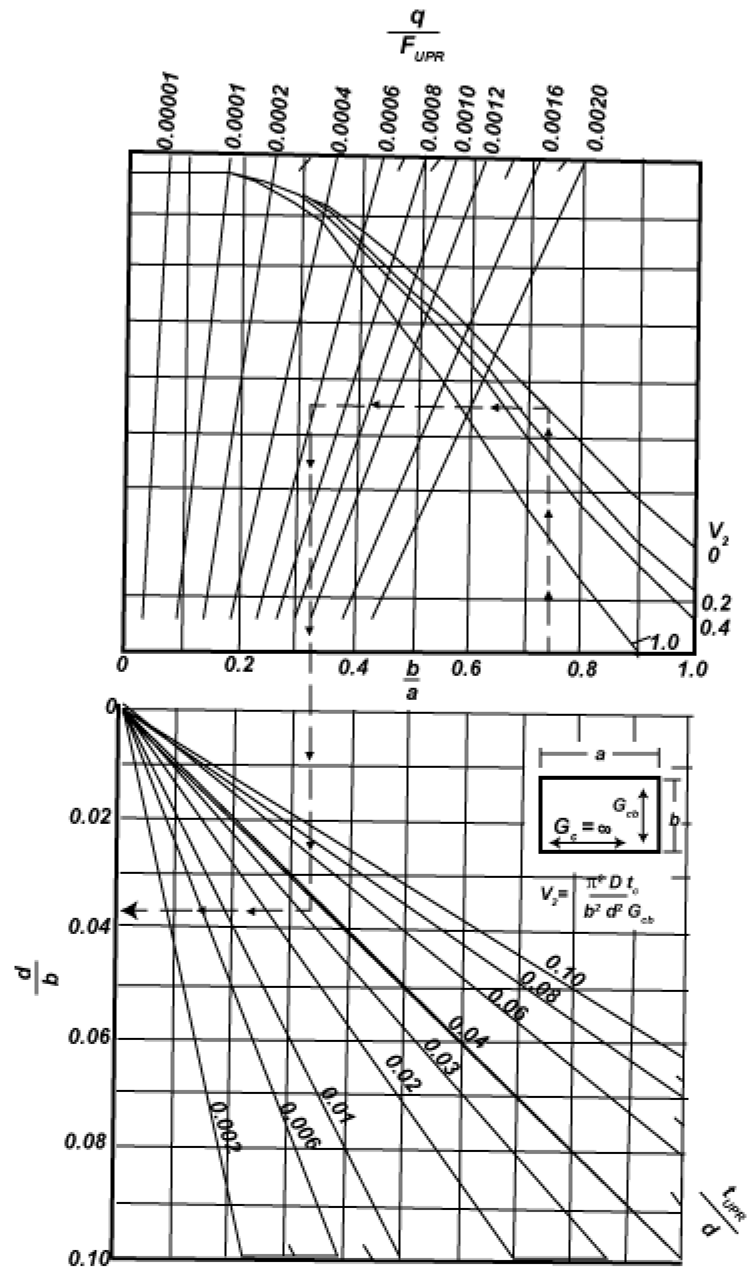

FIGURE 4.7.2.1(c) *Chart for determining $\frac{d}{b}$ ratio for flat rectangular sandwich panel, with isotropic face sheets and corrugated core with flutes parallel to panel edge of length a ($G_c = \infty$), under uniformly distributed normal load, q, so that face sheet stress will be F_{UPR} or F_{LWR}.*

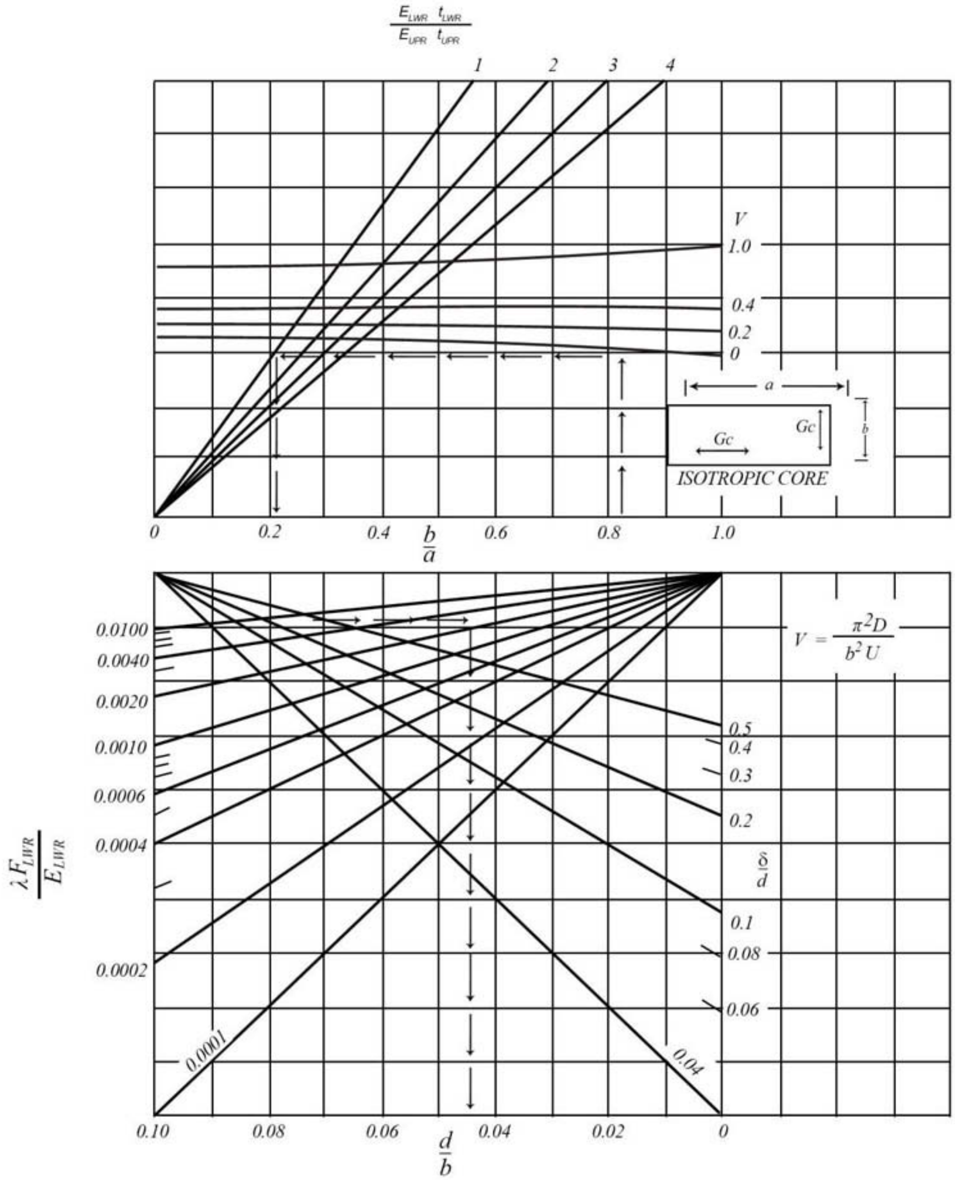

FIGURE 4.7.2.1(d) *Chart for determining $\frac{d}{b}$ ratio for flat rectangular sandwich panel, with isotropic face sheets and isotropic core, under uniformly distributed normal load, q, producing deflection ratio $\frac{\delta}{d}$.*

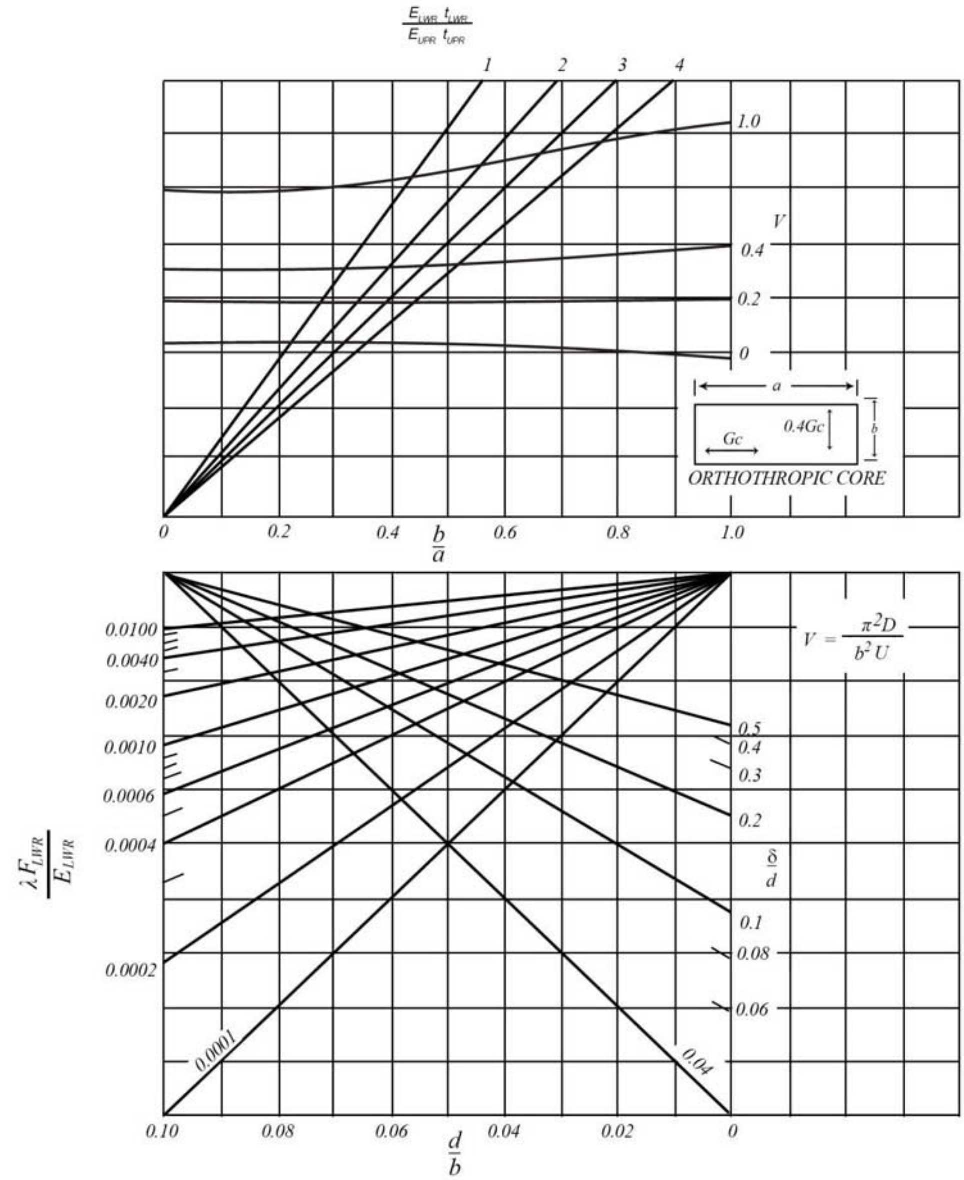

FIGURE 4.7.2.1(e) *Chart for determining $\frac{d}{b}$ ratio for flat rectangular sandwich panel, with isotropic face sheets and orthotropic core ($R = 0.4$), under uniformly distributed normal load, q, producing deflection ratio $\frac{\delta}{d}$.*

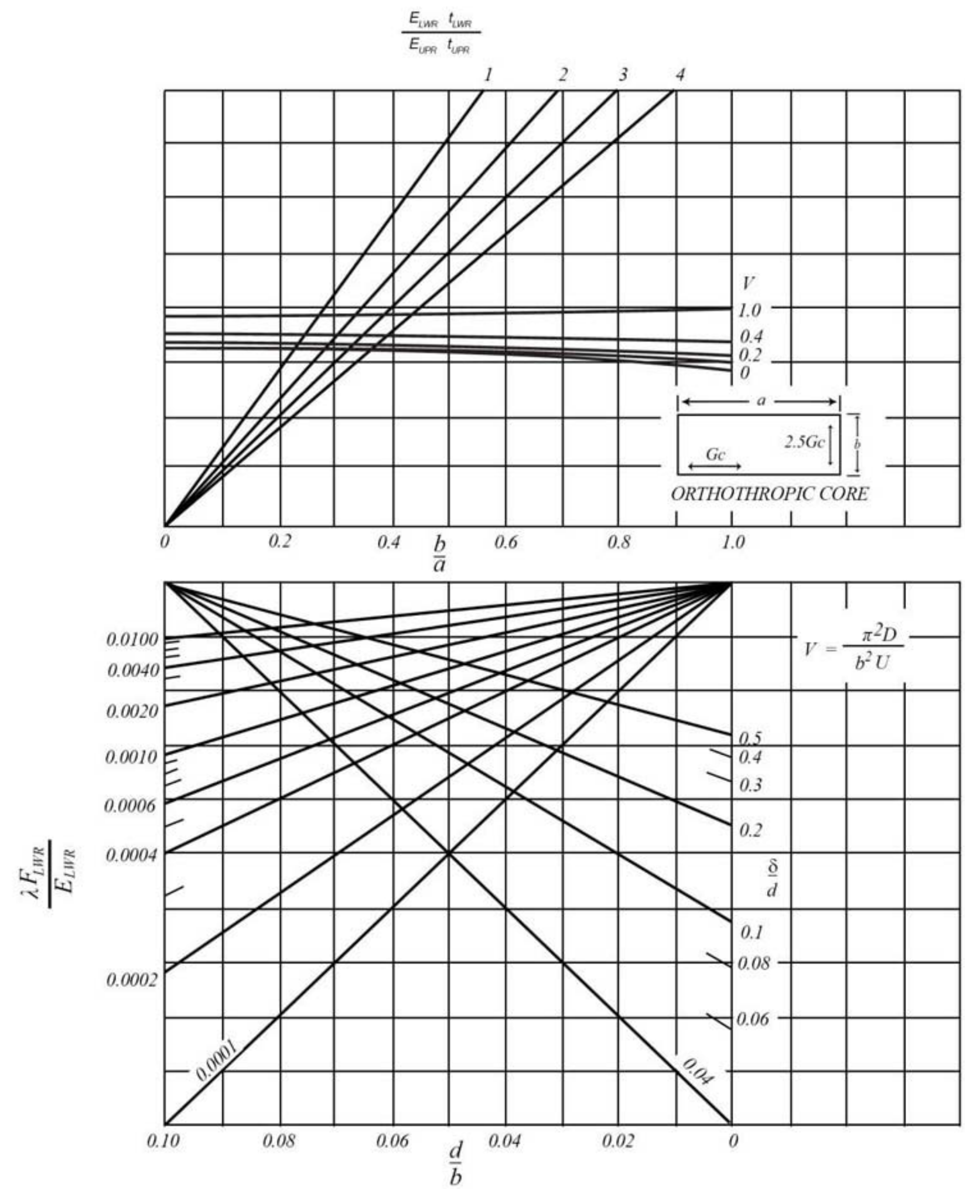

FIGURE 4.7.2.1(f) *Chart for determining $\frac{d}{b}$ ratio for flat rectangular sandwich panel, with isotropic face sheets and orthotropic core ($R = 2.5$), under uniformly distributed normal load, q, producing deflection ratio $\frac{\delta}{d}$.*

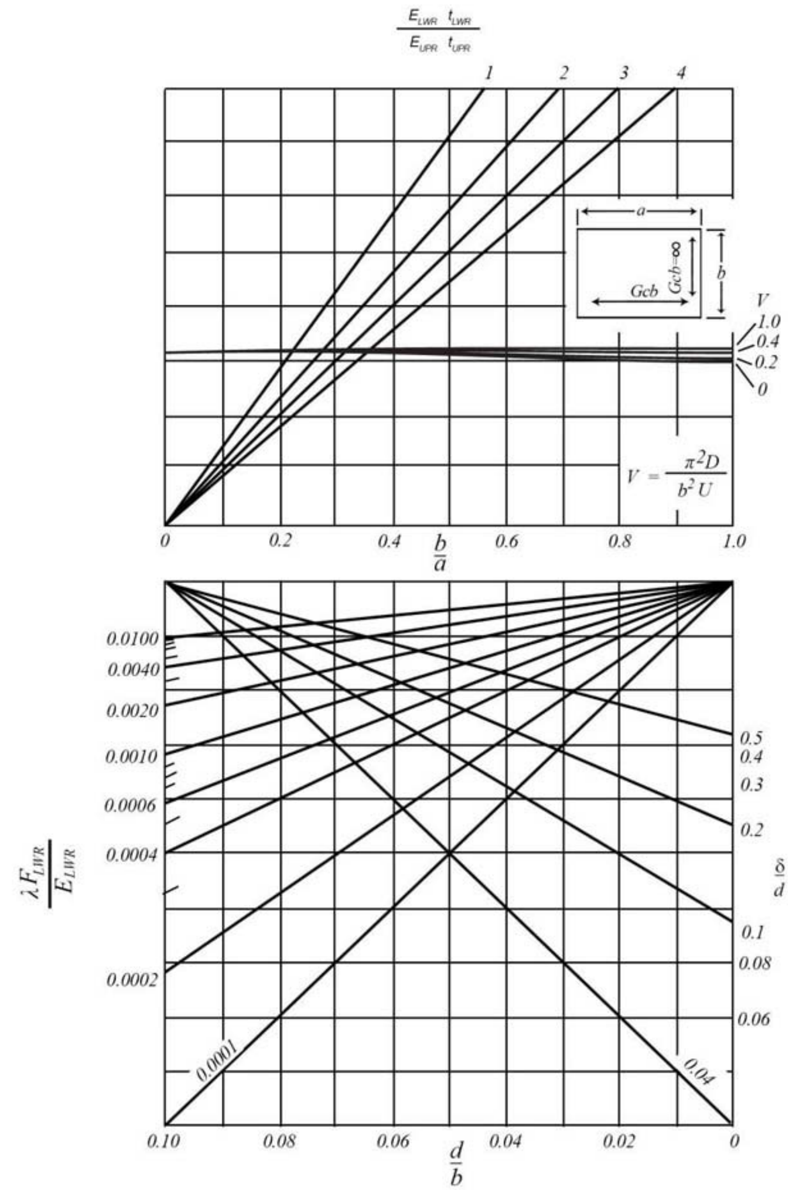

FIGURE 4.7.2.1(g) *Chart for determining $\frac{d}{b}$ ratio for flat rectangular sandwich panel, with isotropic face sheets and corrugated core with flutes perpendicular to panel edge of length a ($G_{cb} = \infty$), under uniformly distributed normal load, q, producing deflection ratio $\frac{\delta}{d}$.*

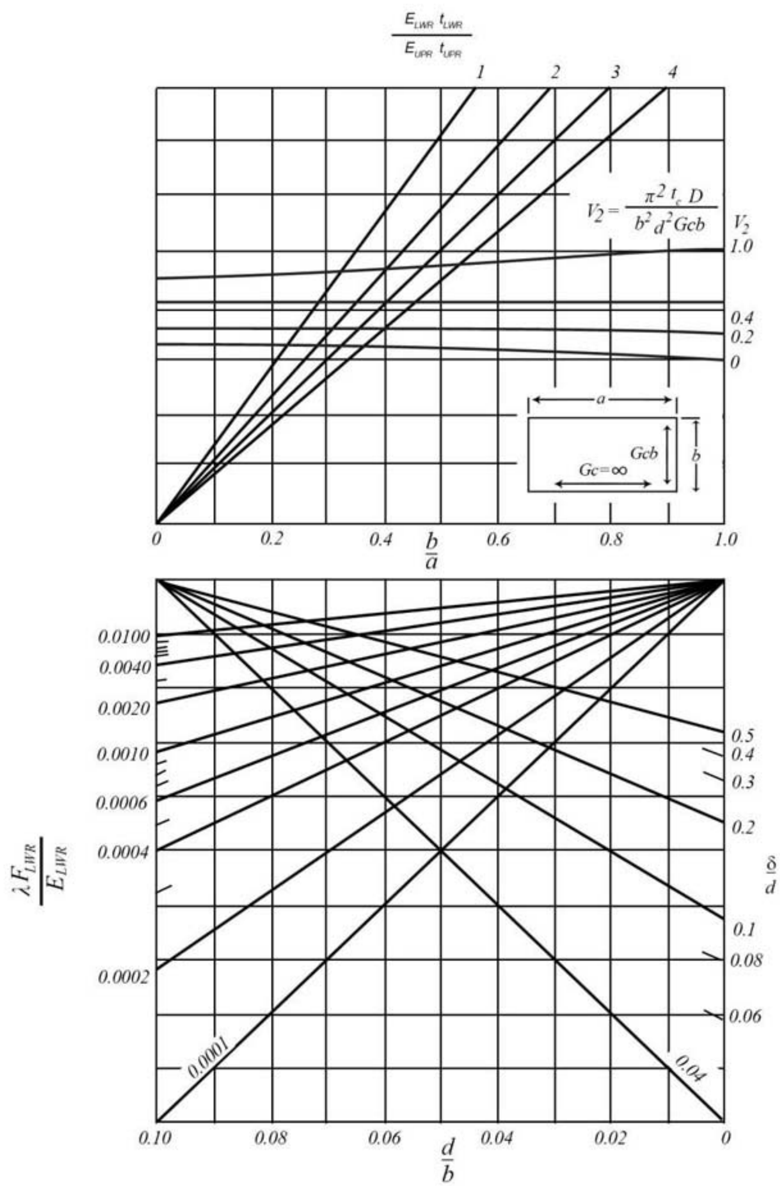

FIGURE 4.7.2.1(h) *Chart for determining $\frac{d}{b}$ ratio for flat rectangular sandwich panel, with isotropic face sheets and corrugated core with flutes parallel to panel edge of length a ($G_c = \infty$), under uniformly distributed normal load, q, producing deflection ratio $\frac{\delta}{d}$.*

4.7.2.1.1 Use of design charts

The sandwich must be designed by iterative procedures, and the charts enable rapid determination of the various quantities sought. The charts for panels with isotropic and orthotropic core were derived for isotropic face sheets with a Poisson's ratio of 0.3, and can be used with small error for face sheets having

other values of Poisson's ratio. The charts for panels with corrugated core were derived for isotropic face sheets with a Poisson's ratio of 0.25.

As a first approximation, it will be assumed that $V = 0$ or $V_2 = 0$. If the design is controlled by face sheet stress criteria, as may be determined, this assumption will lead to an exact value of d if the core is isotropic; to a minimum value of d if the core is orthotropic with a greater core shear modulus across the panel width than lengthwise ($R > 1$); and too large a value of d if the core is orthotropic with a smaller core shear modulus across the panel width than lengthwise ($R < 1$).

If the design is determined by deflection requirements, the assumption that $V = 0$ will produce a minimum value of d. The value of d is minimum because $V = 0$ if the core shear modulus is infinite. For any actual core, the shear modulus is not infinite; hence a thicker core must be used to keep deflection to the required value.

The following procedure is suggested:

1. Enter Figures 4.7.2.1(a), (b), or (c), as appropriate for the core to be used, with the desired value for the parameters b/a and q/F_{UPR} using the curve for V or $V_2 = 0$. Assume a value for t_{UPR}/d and determine $\frac{d}{b}$. Compute d and t_{UPR}. Modify the ratio t_{UPR}/d, if necessary, to determine more suitable values for d and t_{UPR}. Using this value of $\frac{d}{b}$, and the desired values of b/a and q/F_{LWR}, find the corresponding value for t_{LWR}/d and compute t_{LWR}. Check the stress in the a direction as described in Footnote 1 in Section 4.7.2.1.

2. Enter Figures 4.7.2.1(d), (e), (f), (g), or (h) with desired values for the parameters b/a, $E_{LWR}t_{LWR}/E_{UPR}t_{UPR}$ and $\lambda F_{LWR}/E_{LWR}$, and assume $V = 0$ or $V_2 = 0$. Assume a value for $\frac{\delta}{d}$ and determine $\frac{d}{b}$. Compute d and δ. Modify the ratio $\frac{\delta}{d}$, if necessary, and determine more suitable values for d and δ.

3. Repeat steps 1 and 2, using lower chosen design face sheet stresses, until d determined by step 2 is equal to, or a bit less than, d determined by step 1.

4. Compute core thickness t_c using equations.

$$t_c = d - \frac{t_{UPR} + t_{LWR}}{2} \qquad 4.7.2.1.1$$

$$t_c = d - t \quad \text{(for equal face sheets)}$$

This first approximation was based on a core with an infinite shear modulus. Since actual core shear modulus values are not very large, a somewhat larger value of t_c must be used. Successive approximations can be made by entering Figure 4.7.2.1(a) through 4.7.2.1(h) with values of V or V_2 as computed by Equations 4.7.2.1(b) and 4.7.2.1(c).

NOTE: For honeycomb cores with core ribbon parallel to panel length a, $G_c = G_{TL}$ and the shear modulus parallel to panel width b is G_{TW}. For honeycomb cores with core ribbons parallel to panel dimension b, $G_c = G_{Tw}$, and the shear modulus parallel to panel width b is G_{TL}. If core ribbons are at an angle θ to the panel length a, $G_c = \frac{G_{TL}G_{TW}}{\left(G_{TL}\sin^2\theta + G_{TW}\cos^2\theta\right)}$. (See Section 4.6.2.)

In using Figures 4.7.2.1(a) through 4.7.2.1(h) for V or $V_2 \neq 0$, it is necessary to iterate because V and V_2 are directly proportional to the core thickness t_c. As an aid to determining final values of t_c and G_c, Figure 4.7.2.1.1 presents a number of lines representing V or V_2 for various values of G_c, with V or V_2 ranging from 0.01 to 2 and G_c ranging from 1,000 to 1,000,000 pounds per square inch (psi). The following procedure is suggested:

1. Determine core thickness a using a value of 0.01 for V or V_2.

2. Compute the constant relating V or V_2 to G_c or G_{cb}.

$$VG_c \text{ or } V_2G_{cb} = \left[\frac{\pi^2\, t_c\, E'_{UPR}\, t_{UPR}\, E'_{LWR}\, t_{LWR}}{\left(E'_{UPR}\, t_{UPR} + E'_{LWR}\, t_{LWR}\right)\lambda\, b^2}\right]$$

$$VG_c \text{ or } V_2G_{cb} = \left[\frac{\pi^2 t_c E' t}{2\lambda b^2}\right] \quad (\text{for equal facings})$$

3. Find the line in Figure 4.7.2.1.1 with this constant, and determine necessary G_c or G_{cb} from the chart.

4. If the shear modulus is outside the range of values for materials available, slide up the appropriate line of Figure 4.7.2.1.1 and pick a new value of V or V_2, for a reasonable value of core shear modulus.

5. Reenter Figures 4.7.2.1(a) through 4.7.2.1(h) with the new value of V or V_2 and repeat all the previous steps.

4.7.2.1.2 Determining core shear stress

This section gives procedures for determining the maximum core shear stress of a simply supported flat rectangular panel under uniformly distributed normal load. The core shear stress is maximum at the panel edges, at midlength of each edge. The maximum shear stress, F_{sc}, is given by the formula

$$F_{sc} = K_3\, q\, \frac{b}{d} \qquad 4.7.2.1.2$$

where K_3 is a theoretical coefficient dependent upon panel aspect ratio and the parameter V. If the core is isotropic, values of V do not affect the core shear stress.

The charts of Figures 4.7.2.1.2(a) through 4.7.2.1.2(c) present a graphical solution of Equation 4.7.2.1.2. Figure 4.7.2.1.2(a) includes curves for sandwich with isotropic core ($R = 1$), and certain orthotropic cores ($R = 0.4$ or 2.5), where R is the ratio of core shear stiffness in the b direction to that in the a direction ($G_{cb} = R\, G_c$). Figures 4.7.2.1.2(b) and (c) apply to corrugated core with the core flutes perpendicular to the panel edge of length a ($G_{cb} = \infty$) and parallel to the edge of length a ($G_c = \infty$), respectively. The appropriate chart should be entered with values of thicknesses and other parameters previously determined.

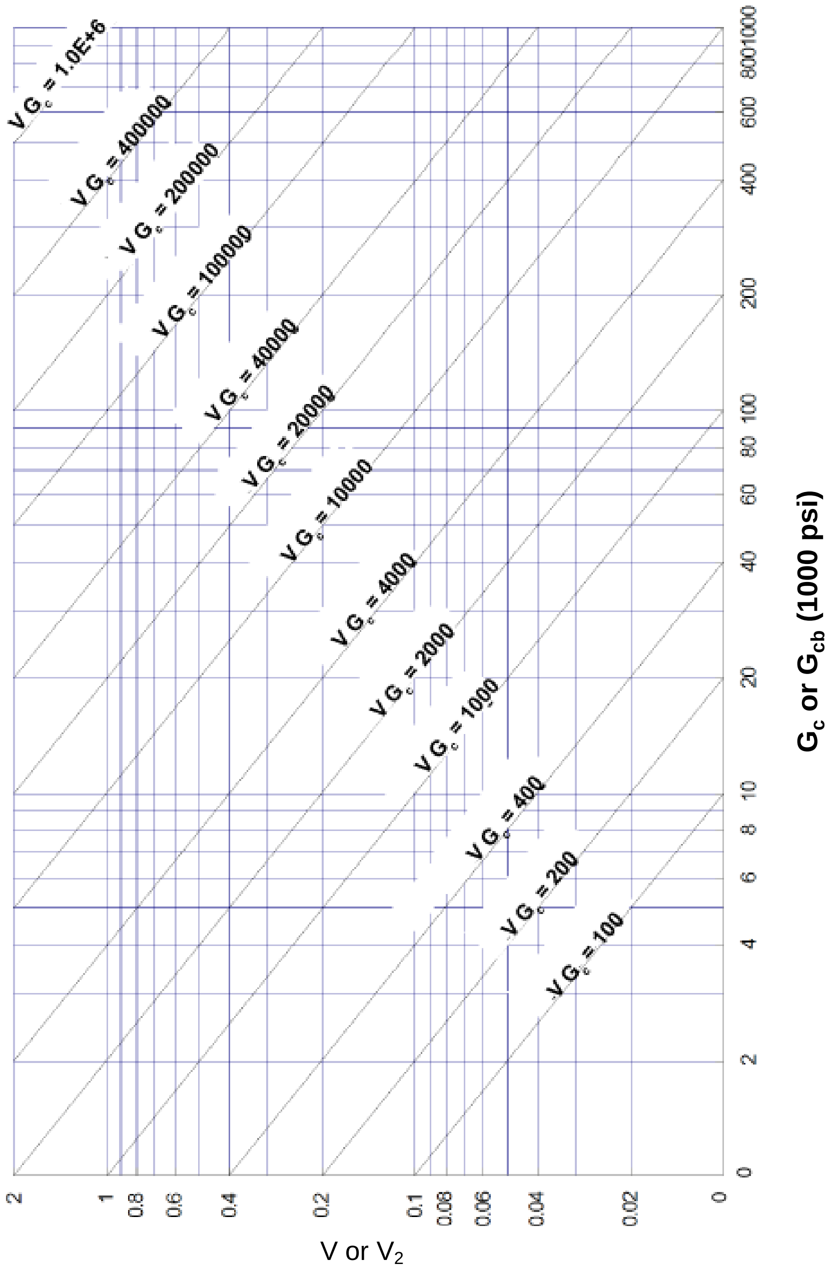

FIGURE 4.7.2.1.1 *Chart for determining V or V_2 and G_c or G_{cb} for sandwich under uniformly distributed normal load.*

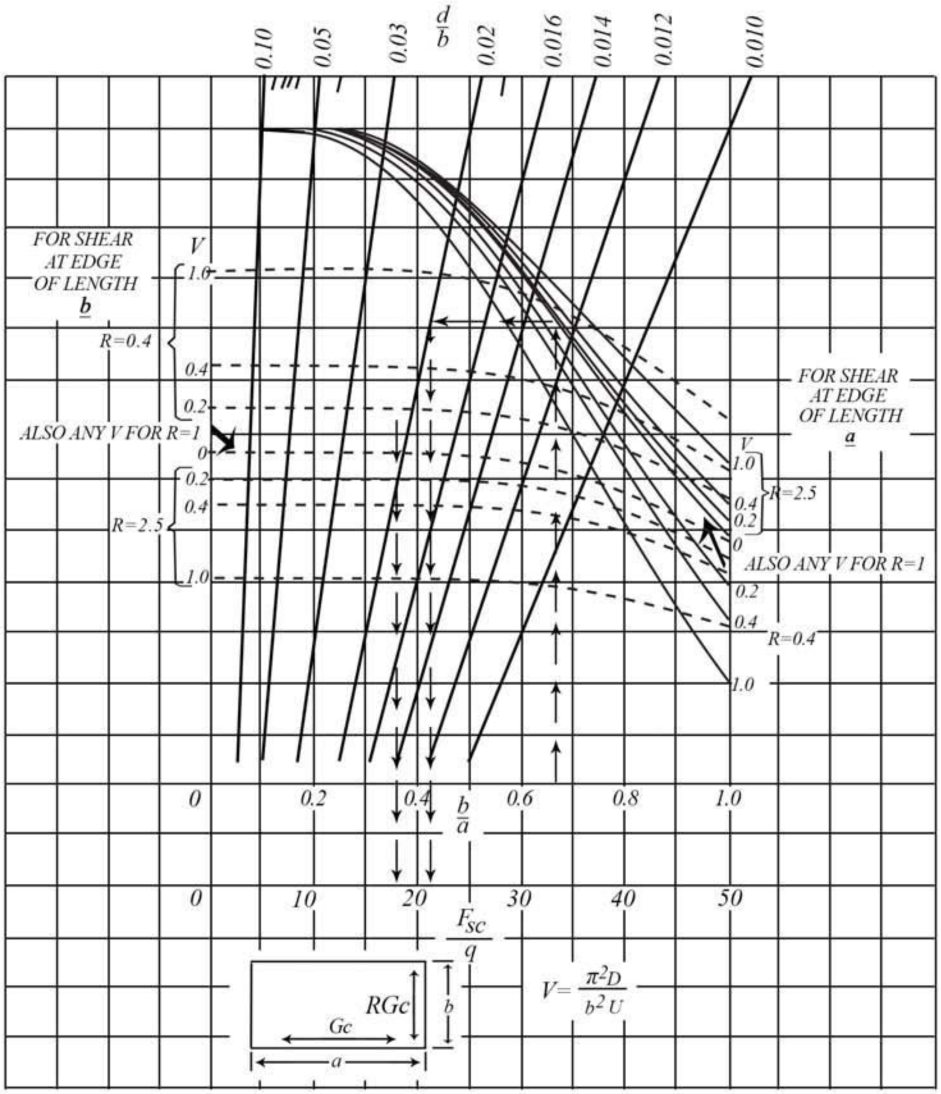

FIGURE 4.7.2.1.2(a) *Chart for determining core shear stress ratio F_{sc}/q for flat rectangular sandwich panel, with isotropic face sheets and isotropic core ($R = 1$) or orthotropic core ($R = 0.4$ or 2.5), under uniformly distributed normal load, q.*

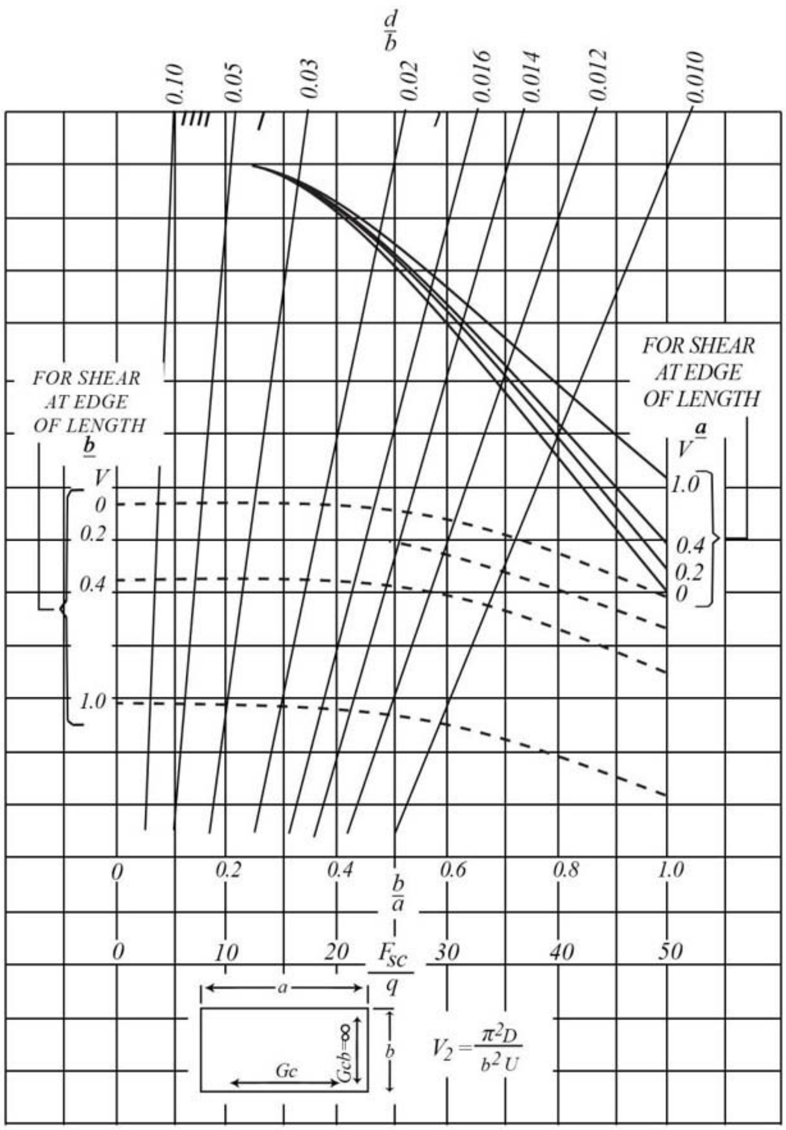

FIGURE 4.7.2.1.2(b) *Chart for determining core shear stress ratio F_{sc}/p for flat rectangular sandwich panel, with isotropic face sheets and corrugated core with flutes perpendicular to panel edge of length a ($G_{cb} = \infty$), under uniformly distributed normal load, q.*

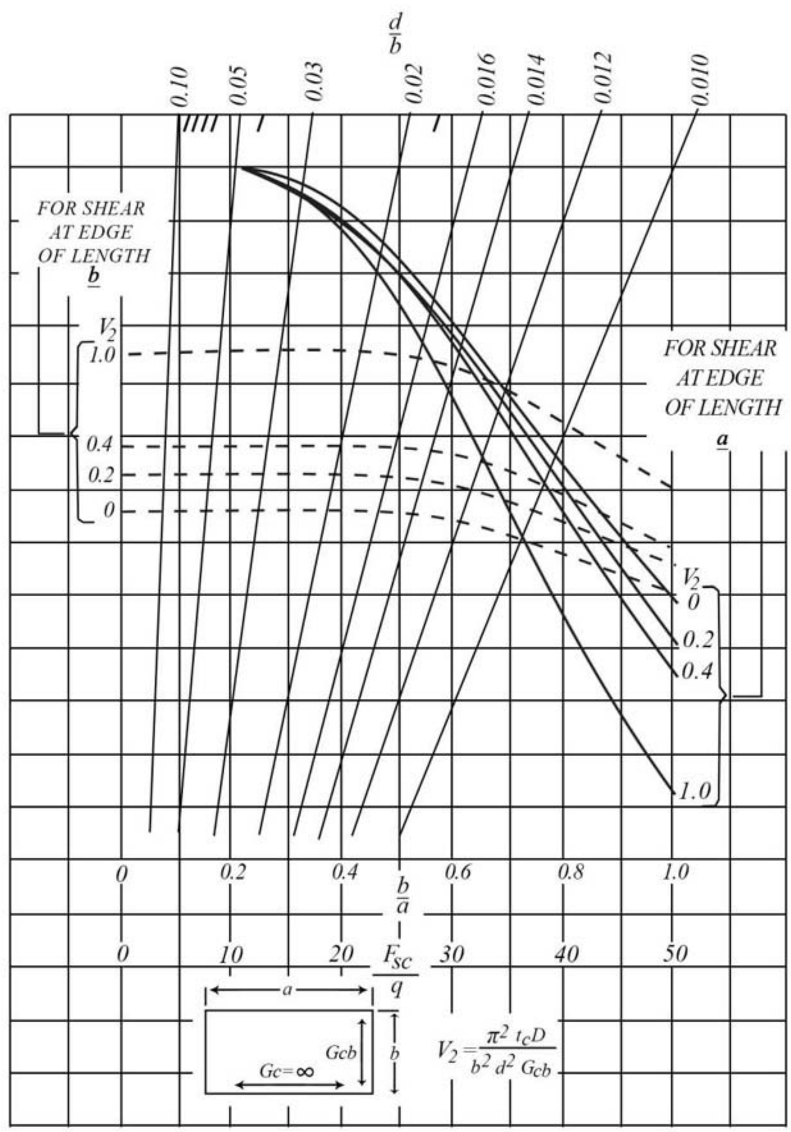

FIGURE 4.7.2.1.2(c) *Chart for determining core shear stress ratio F_{sc}/q for flat rectangular sandwich panel, with isotropic face sheets and corrugated core with flutes parallel to panel edge of length a ($G_c = \infty$), under uniformly distributed normal load, q.*

4.7.2.1.3 Checking procedures

The design may be checked by using the graphs of Figure 4.7.2.1.3(a) through 4.7.2.1.3(l) to determine theoretical coefficients K_2, K'_2, K_1, and K_3 to compute face sheet stresses, deflection, and core shear stresses. If the graphs do not apply to honeycomb core because ratios of core shear moduli are far different from those given in the graphs, or it is desired to check by a more accurate analysis, the equations given in Reference 4.7.2.1.3(a) may be used. The graphs for panels having corrugated core apply to panels where the ratio of bending stiffnesses, (D_a/D_b), is equal to 1. If the core corrugations contribute significantly to the panel bending stiffness, ($D_a/D_b \neq 1$), the graphs given in Reference 4.7.2.1.3(b) should be used.

Figures 4.7.2.1.3 (a) through (c) are for parameter K_2 (see Equation 4.7.2.1(a)). Figures (d) through (f) are for parameter K'_2, Figures (g) through (i) are for K_1 (see Equation 4.7.2.1(e)). Figures (j) through (l) are for K_3 (see Equation 4.7.2.1.2). Figures 4.7.2.1.3 (a), (d), (g), and (j) are for isotropic core ($R = 1$) or orthotropic core ($R = 0.4$ or 2.5). Figures (b), (e), (h), and (k) are for corrugated core with the flutes perpendicular to the panel edge of length a ($G_{cb} = \infty$), and Figures (c), (f), (i), and (l) are for corrugated core with the flutes parallel to the panel edge of length a ($G_c = \infty$).

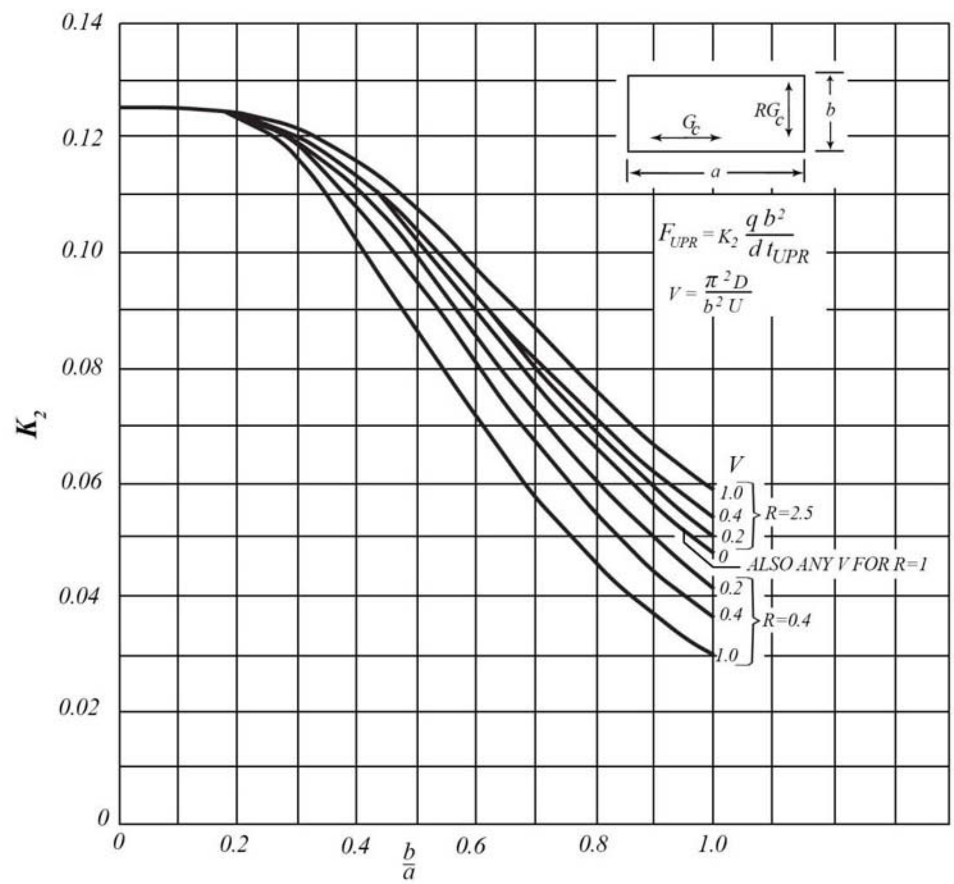

FIGURE 4.7.2.1.3(a) *K_2 for determining face sheet stress, F, in b direction of flat rectangular sandwich panel, with isotropic face sheets and isotropic core ($R = 1$) or orthotropic core ($R = 0.4$ or 2.5) core, under uniformly distributed normal load, q.*

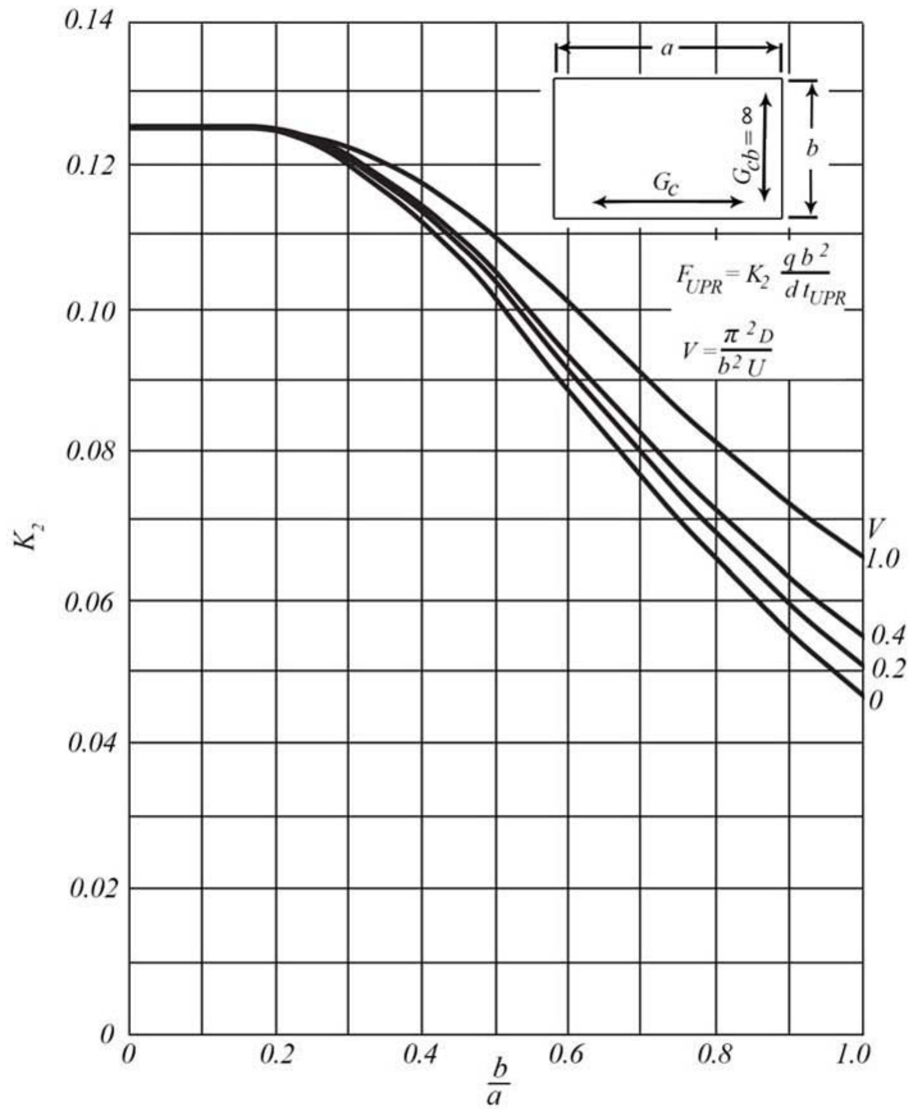

FIGURE 4.7.2.1.3(b) *K_2 for determining face sheet stress, F, in b direction of flat rectangular sandwich panel, with isotropic face sheets and corrugated core with the flutes perpendicular to panel edge of length a ($G_{cb} = \infty$), under uniformly distributed normal load, q.*

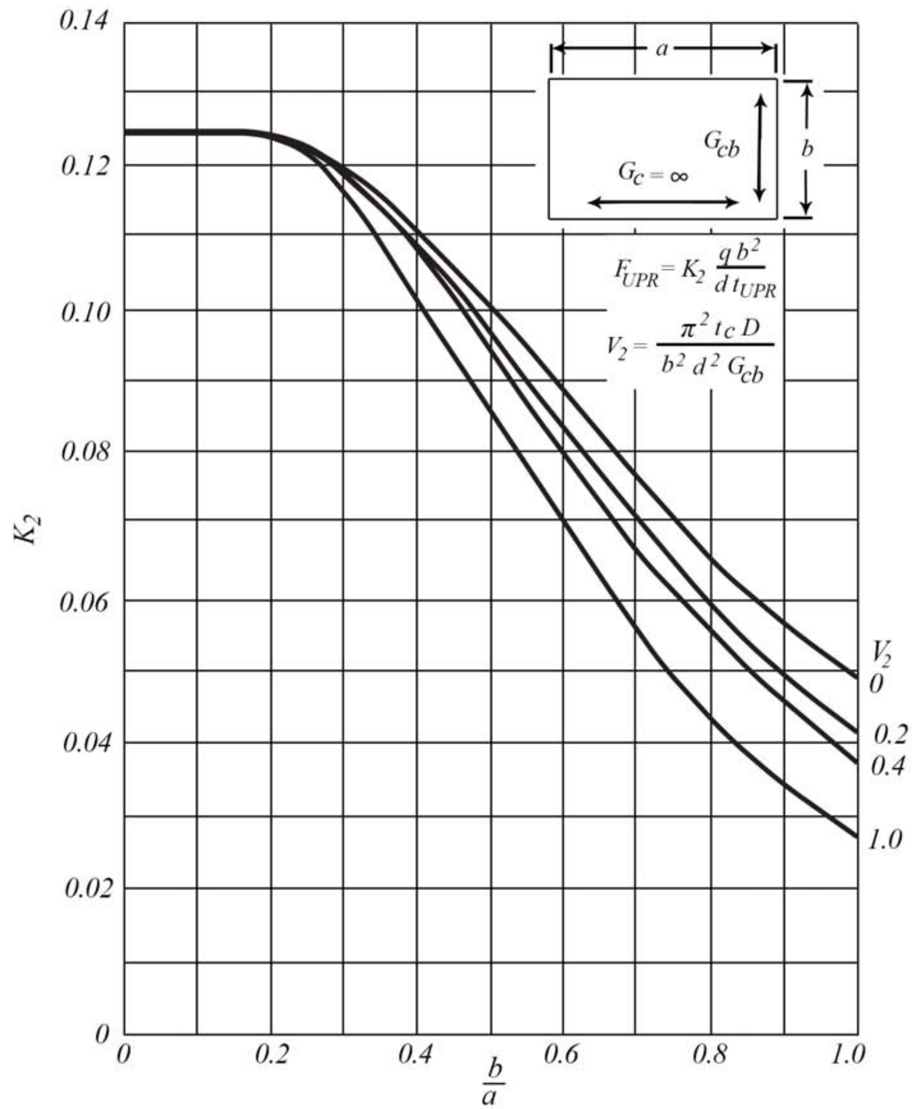

FIGURE 4.7.2.1.3(c) *K_2 for determining face sheet stress, F, in b direction of flat rectangular sandwich panel, with isotropic face sheets and corrugated core with the flutes parallel to panel edge of length a ($G_c = \infty$), under uniformly distributed normal load, q.*

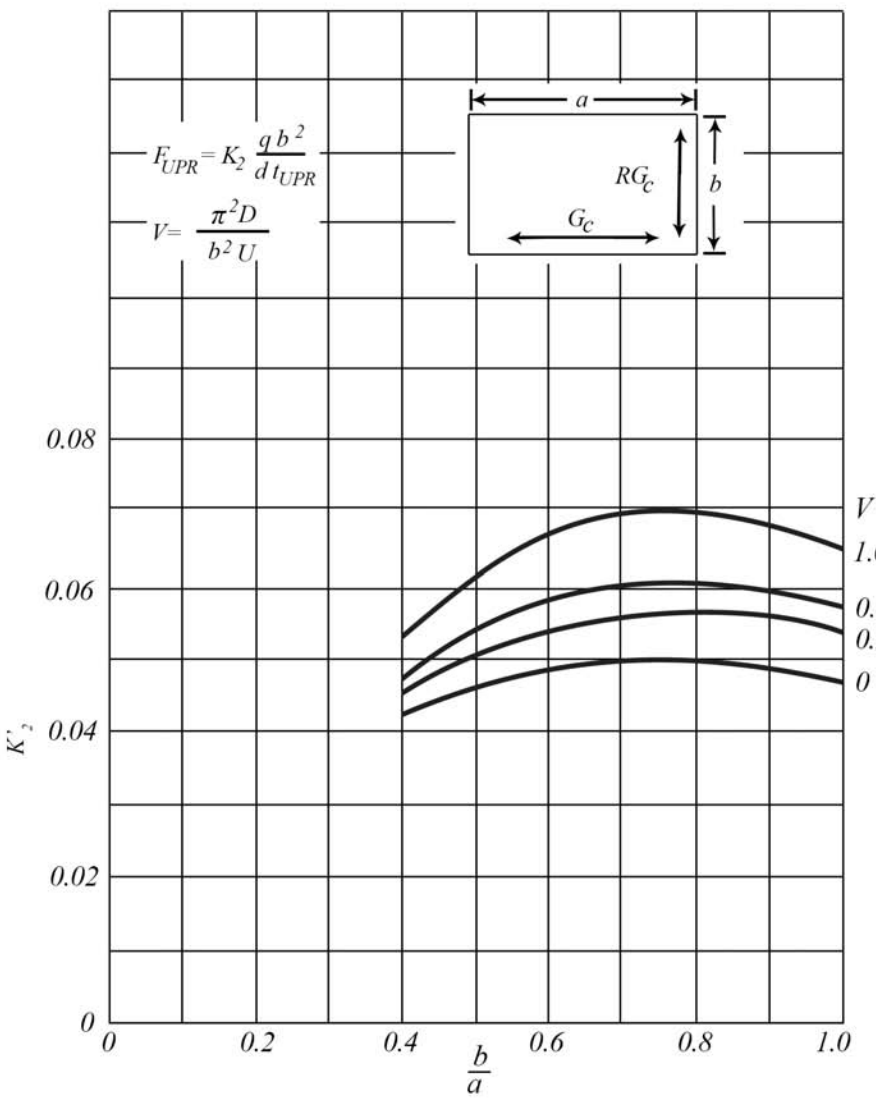

FIGURE 4.7.2.1.3(d) *K'_2 for determining face sheet stress, F, in a direction of flat rectangular sandwich panel, with isotropic face sheets and orthotropic core ($R = 0.4$), under uniformly distributed normal load, q. For $R = 1$ and $R = 2.5$, the maximum stress is given by K_2 of Figure 4.7.2.1.3(a).*

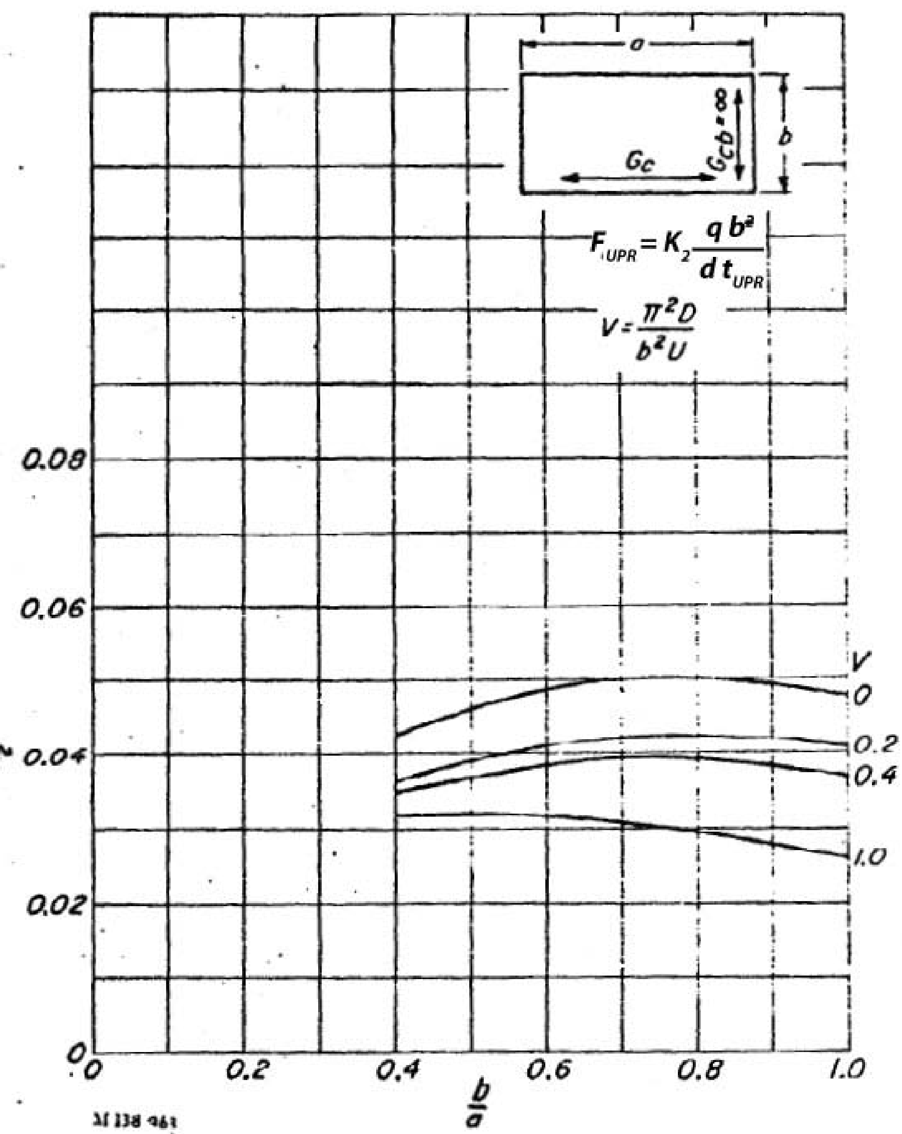

FIGURE 4.7.2.1.3(e) *K'_2 for determining face sheet stress, F, in a direction of flat rectangular sandwich panel, with isotropic face sheets and corrugated core with the flutes perpendicular to panel edge of length a ($G_{cb} = \infty$), under uniformly distributed load, q.*

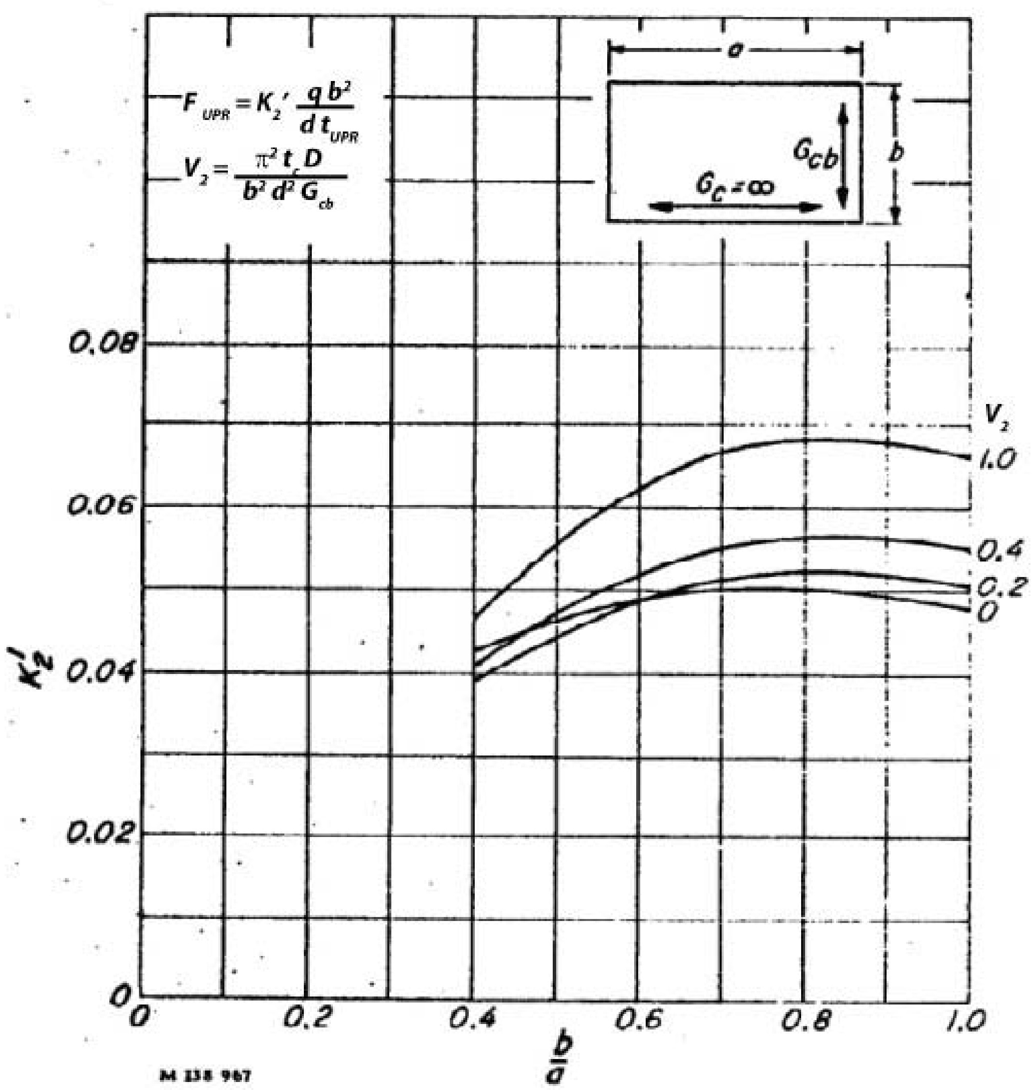

FIGURE 4.7.2.1.3(f) *K'_2 for determining face sheet stress, F, in a direction of flat rectangular sandwich panel, with isotropic face sheets and corrugated core with flutes parallel to panel edge of length a ($G_c = \infty$), under uniformly distributed load, q.*

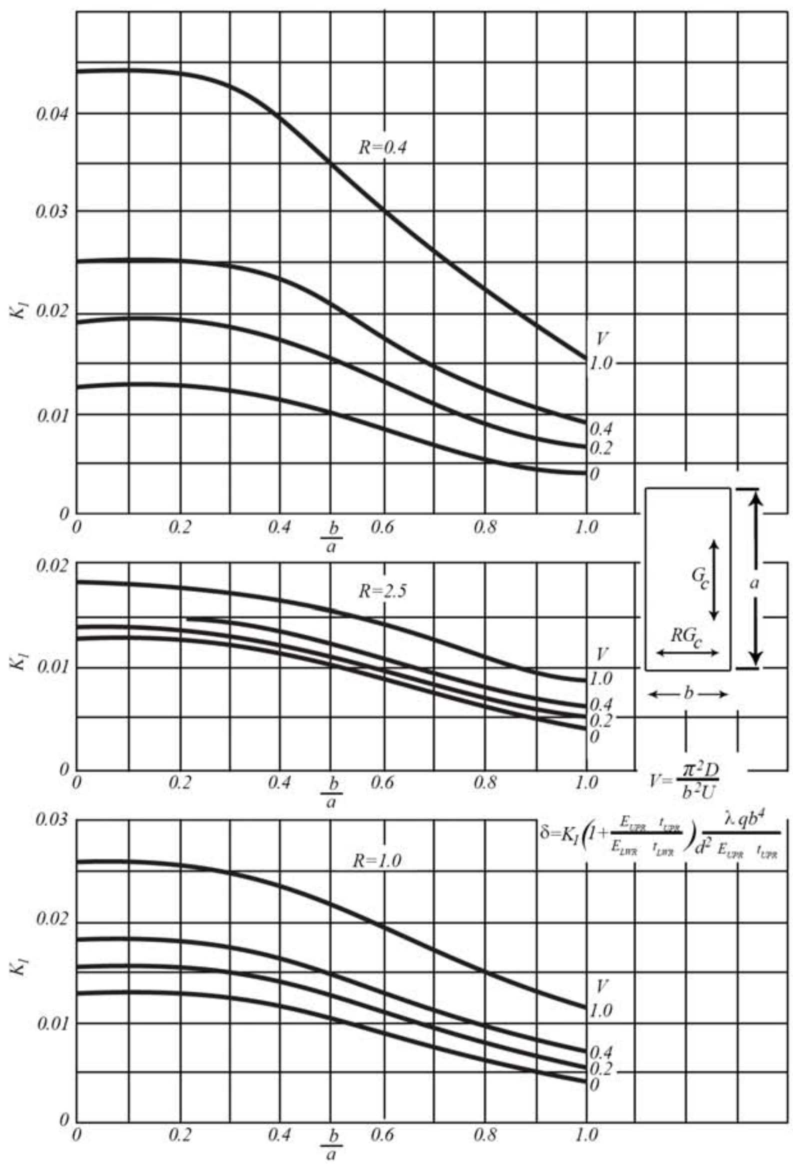

FIGURE 4.7.2.1.3(g) *K_1 for determining maximum deflection, δ, of flat rectangular sandwich panel, with isotropic face sheets and isotropic core (R = 1) or orthotropic core ($R = 0.4$ or 2.5), under uniformly distributed load, q.*

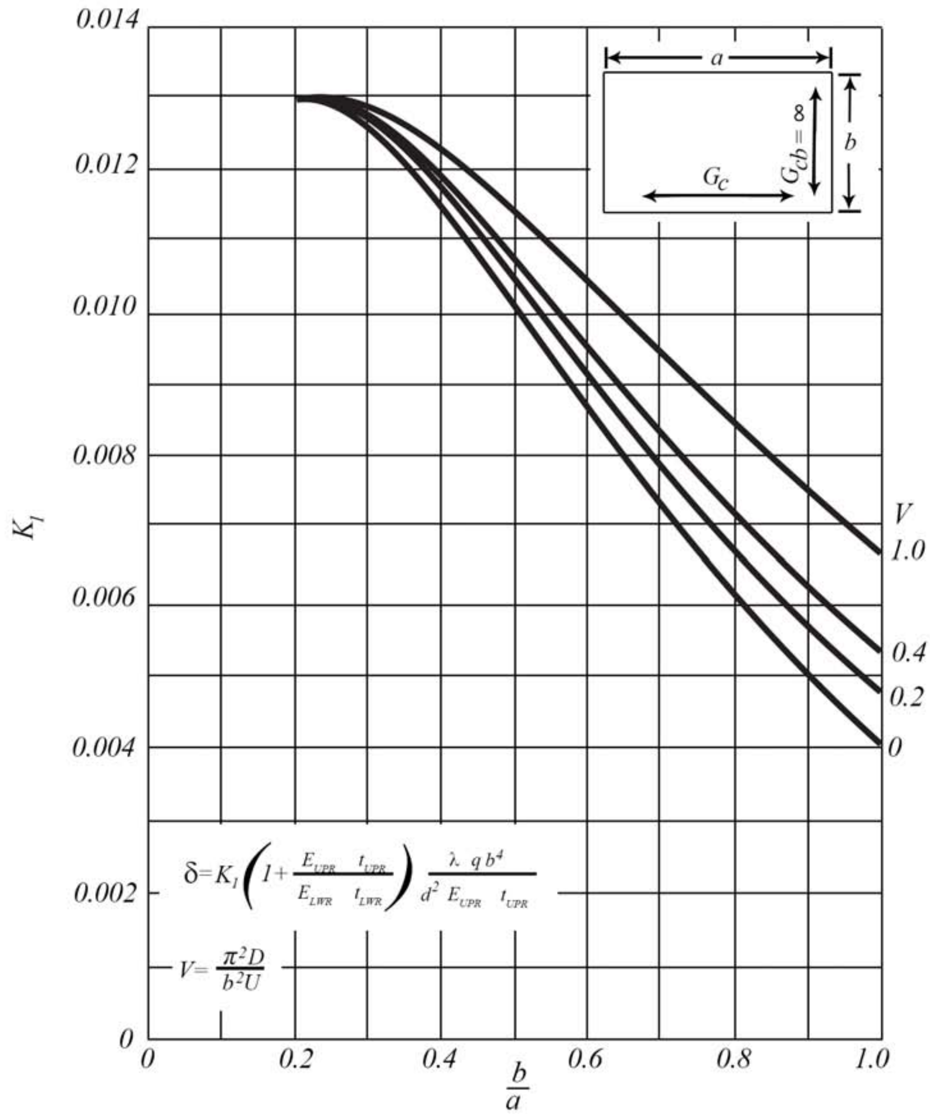

FIGURE 4.7.2.1.3(h) *K_1 for determining maximum deflection, δ, of flat rectangular sandwich panel, with isotropic face sheets and corrugated core with the flutes perpendicular to panel edge of length a ($G_{cb} = \infty$), under uniformly distributed load, q.*

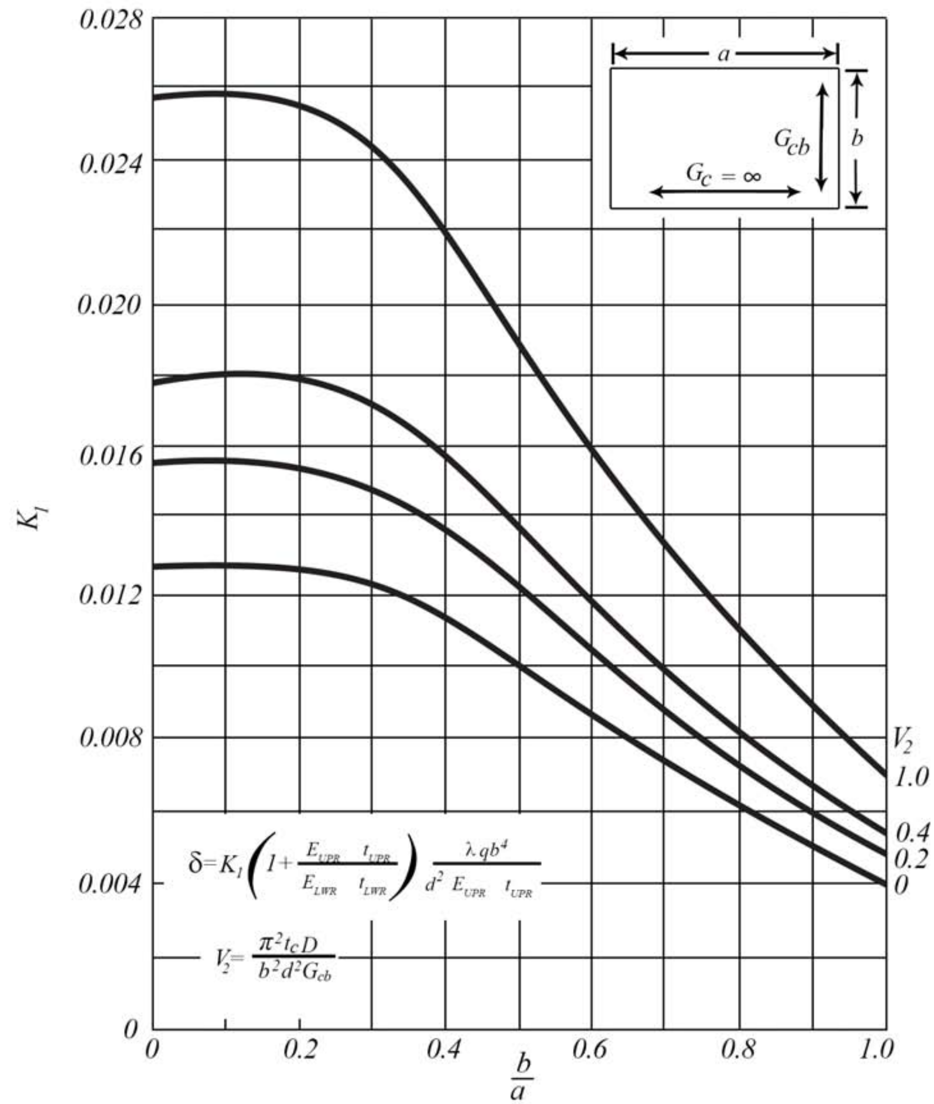

FIGURE 4.7.2.1.3(i) *K_1 for determining maximum deflection, δ, of flat rectangular sandwich panel, with isotropic face sheets and corrugated core with the flutes parallel to panel edge of length a ($G_c = \infty$), under uniformly distributed load, q.*

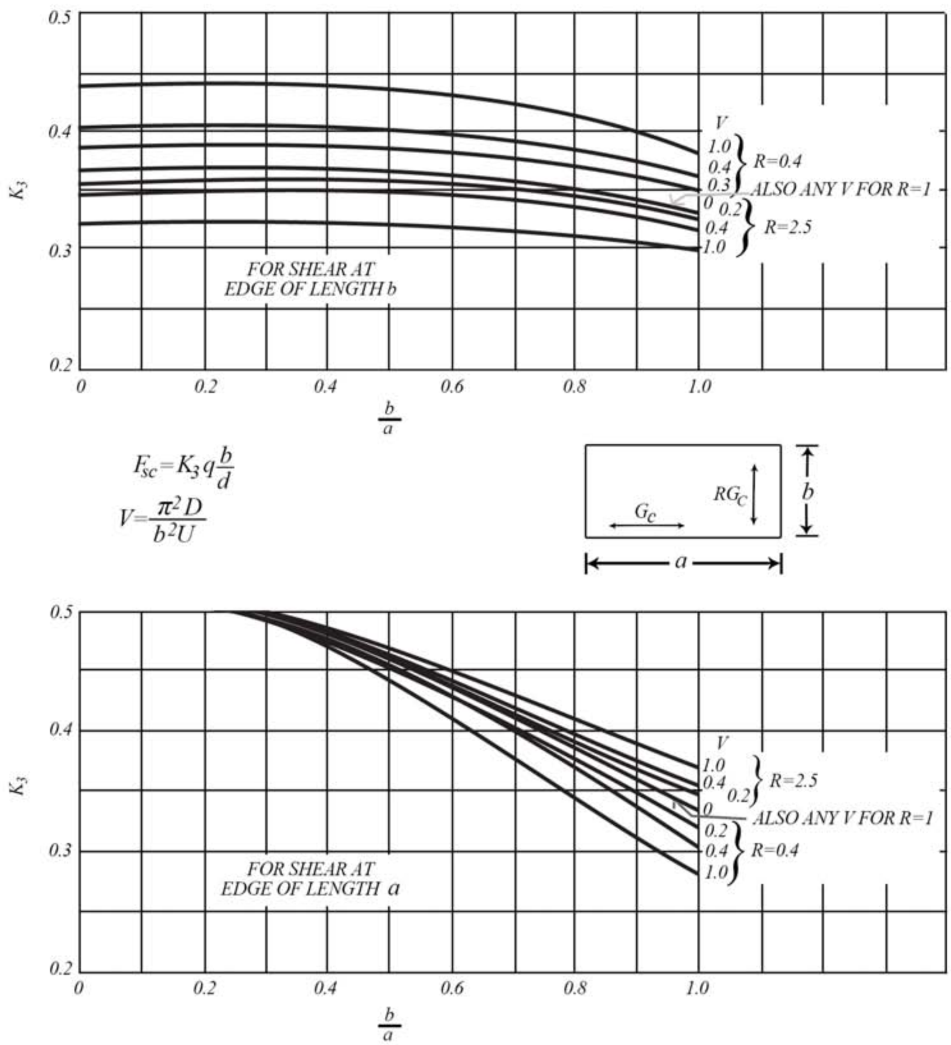

FIGURE 4.7.2.1.3(j) *K_3 for determining maximum core shear stress, F_{sc}, for flat rectangular sandwich panel, with isotropic face sheets and isotropic core ($R = 1$) or orthotropic core ($R = 0.4$ or 2.5), under uniformly distributed normal load, q.*

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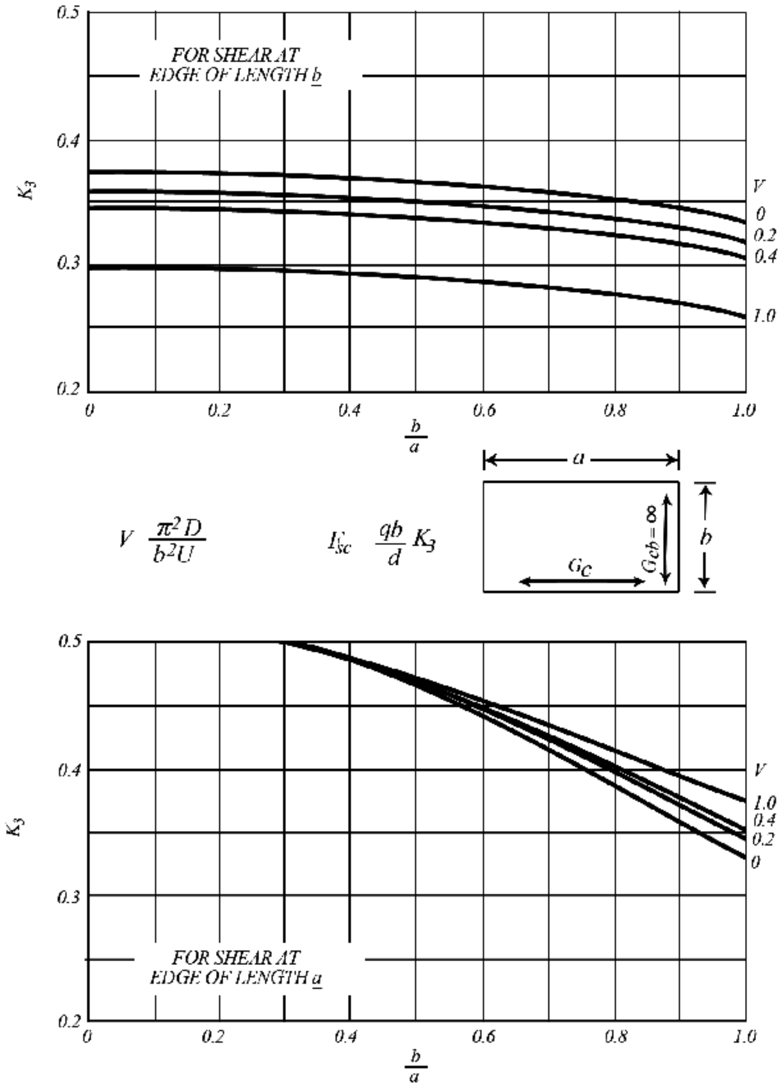

FIGURE 4.7.2.1.3(k) *K_3 for determining maximum core shear stress, F_{sc}, for flat rectangular sandwich panel, with isotropic face sheets and corrugated core with the flutes perpendicular to panel edge of length a ($G_{cb} = \infty$), under uniformly distributed normal load, q.*

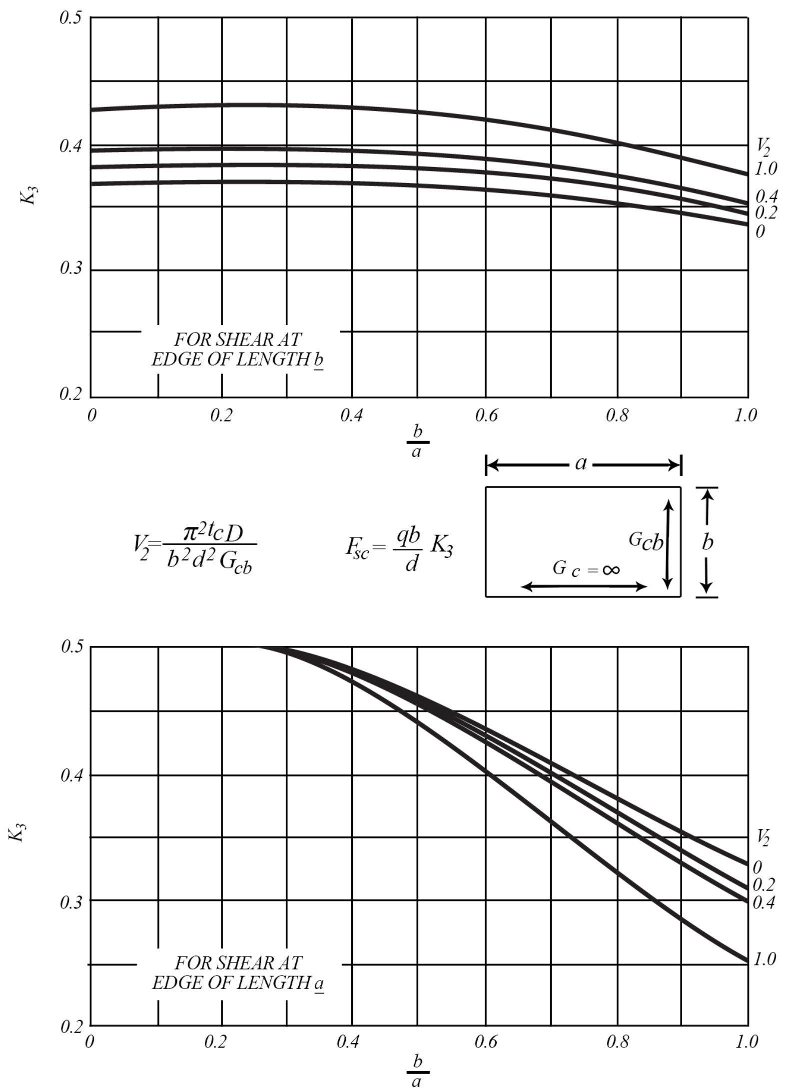

FIGURE 4.7.2.1.3(l) *K_3 for determining maximum core shear stress, F_{sc}, for flat rectangular sandwich panel, with isotropic face sheets and corrugated core with the flutes parallel to panel edge of length a ($G_c = \infty$), under uniformly distributed normal load, q.*

4.7.2.2 Determining face sheet thickness, core thickness, and core shear modulus for simply supported flat circular panels under uniform load

This section gives procedures for determining face sheet and core thicknesses and core shear modulus so that chosen design face sheet stresses and allowable panel deflections will not be exceeded under a given uniformly distributed normal load (Reference 4.7.2.2). The face sheet stresses, produced by bending moment, are a maximum at the center of a simply supported circular panel under uniformly distributed pressure load. If restraint exists at the panel edges, a redistribution of stresses may cause higher stresses near panel edges.

The procedures given apply only to panels with simply supported edges, isotropic face sheets, and isotropic cores. A solution is presented in the form of charts with which, by iterative process, the face sheets and core thicknesses and core shear modulus can be determined.

The average face sheet stress, F (stress at face sheet centroid), is given by the equation:

$$F_{UPR} = \left(\frac{3+\nu}{16}\right)\left(\frac{q\,r^2}{t_{UPR}\,d}\right)$$
$$F_{LWR} = \left(\frac{3+\nu}{16}\right)\left(\frac{q\,r^2}{t_{LWR}\,d}\right) \qquad 4.7.2.2(a)$$

$$F = \left(\frac{3+\nu}{16}\right)\left(\frac{q\,r^2}{t\,d}\right) \text{ (for equal face sheets)}$$

where ν is Poisson's ratio of the face sheets (in this equation it is assumed that ν is the same for both face sheets), r is the radius of the circular panel, and other quantities are as previously defined.

Solving Equation 4.7.2.2(a) for $\frac{d}{r}$ gives

$$\frac{d}{r} = \frac{\sqrt{3+\nu}\sqrt{\frac{q}{F_{UPR}}}}{4\sqrt{\frac{t_{UPR}}{d}}}$$
$$\frac{d}{r} = \frac{\sqrt{3+\nu}\sqrt{\frac{q}{F_{LWR}}}}{4\sqrt{\frac{t_{LWR}}{d}}} \qquad 4.7.2.2(b)$$
$$\frac{d}{r} = \frac{\sqrt{3+\nu}\sqrt{\frac{q}{F}}}{4\sqrt{\frac{t}{d}}} \text{ (for equal face sheets)}$$

A chart for solving Equation 4.7.2.2(b) graphically is given in Figure 4.7.2.2(a). The equation and chart include the ratio $\frac{t}{d}$, which is usually unknown. But by iteration satisfactory ratios of $\frac{t}{d}$ and $\frac{d}{r}$ can be found.

The deflection, δ, of the center of the panel is given by the equation:

$$\delta = K_4\left(1+\frac{E_{UPR}\,t_{UPR}}{E_{LWR}\,t_{LWR}}\right)\frac{\lambda F_{UPR}}{E_{UPR}}\frac{r^2}{d}$$
$$\delta = K_4\left(1+\frac{E_{LWR}\,t_{LWR}}{E_{UPR}\,t_{UPR}}\right)\frac{\lambda F_{LWR}}{E_{LWR}}\frac{r^2}{d} \qquad 4.7.2.2(c)$$

$$\delta = 2K_4 \frac{\lambda F}{E} \frac{r^2}{d} \quad \text{(for equal face sheets)}$$

where K_4 depends on sandwich bending and shear rigidities as incorporated in the parameter V, which relates the bending and shear stiffnesses of the sandwich panel. When written in coordinates for a circular plate, $V = \frac{\pi^2 D}{(2r)^2 U}$ becomes

$$V = \frac{\pi^2 t_c E_{UPR} t_{UPR} E_{LWR} t_{LWR}}{4\lambda r^2 G_c (E_{UPR} t_{UPR} + E_{LWR} t_{LWR})} \qquad 4.7.2.2(d)$$

$$V = \frac{\pi^2 t_c E t}{8\lambda r^2 G_c} \quad \text{(for equal face sheets)}$$

where r is panel radius and all other terms are as previously defined.

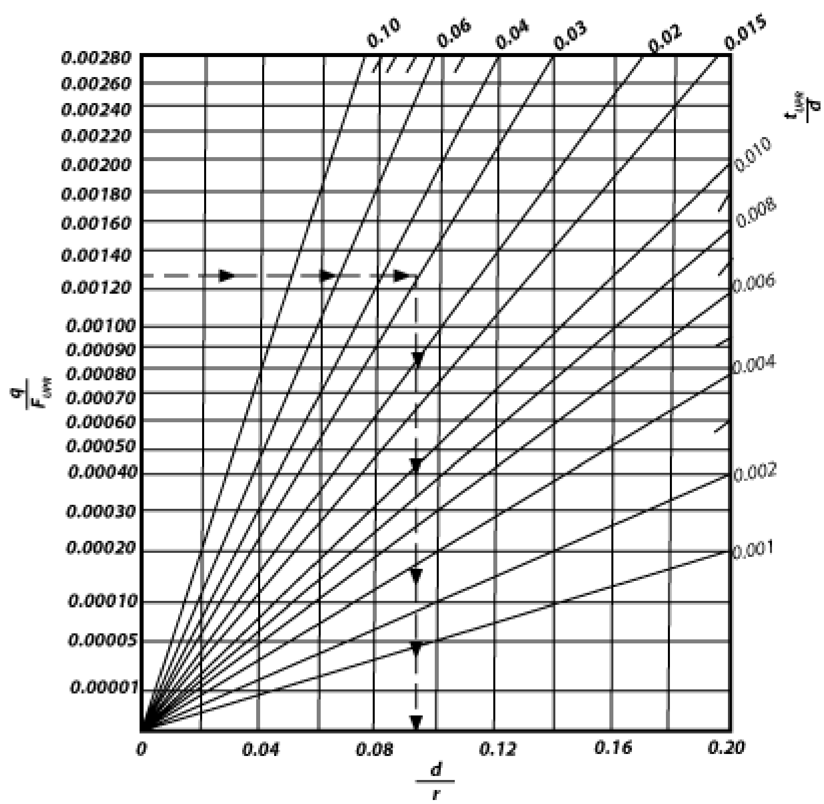

FIGURE 4.7.2.2(a) *Chart for determining $\frac{d}{r}$ ratio for flat circular sandwich panel, with isotropic face sheets and core, under uniformly distributed normal load, q, so that face sheet stress will be F_{UPR}, F_{LWR} ; $\nu = 0.3$.*

Solving Equation 4.7.2.2(c) for $\frac{d}{r}$, gives

$$\frac{d}{r} = \frac{\sqrt{K_4}\sqrt{\frac{\lambda F_{UPR}}{E_{UPR}}}\sqrt{\left(1+\frac{E_{UPR}\, t_{UPR}}{E_{LWR}\, t_{LWR}}\right)}}{\sqrt{\frac{\delta}{d}}}$$

4.7.2.2(e)

$$\frac{d}{r} = \frac{\sqrt{K_4}\sqrt{\frac{\lambda F_{LWR}}{E_{LWR}}}\sqrt{\left(1+\frac{E_{LWR}\, t_{LWR}}{E_{UPR}\, t_{UPR}}\right)}}{\sqrt{\frac{\delta}{d}}}$$

$$\frac{d}{r} = \frac{\sqrt{2K_4}\sqrt{\frac{\lambda F}{E}}}{\sqrt{\frac{\delta}{d}}} \quad \text{(for equal face sheets)}$$

A chart for solving Equation 4.7.2.2(e) is given in Figure 4.7.2.2(b). Use of equations and charts beyond $\frac{\delta}{d} = 0.5$ is not recommended.

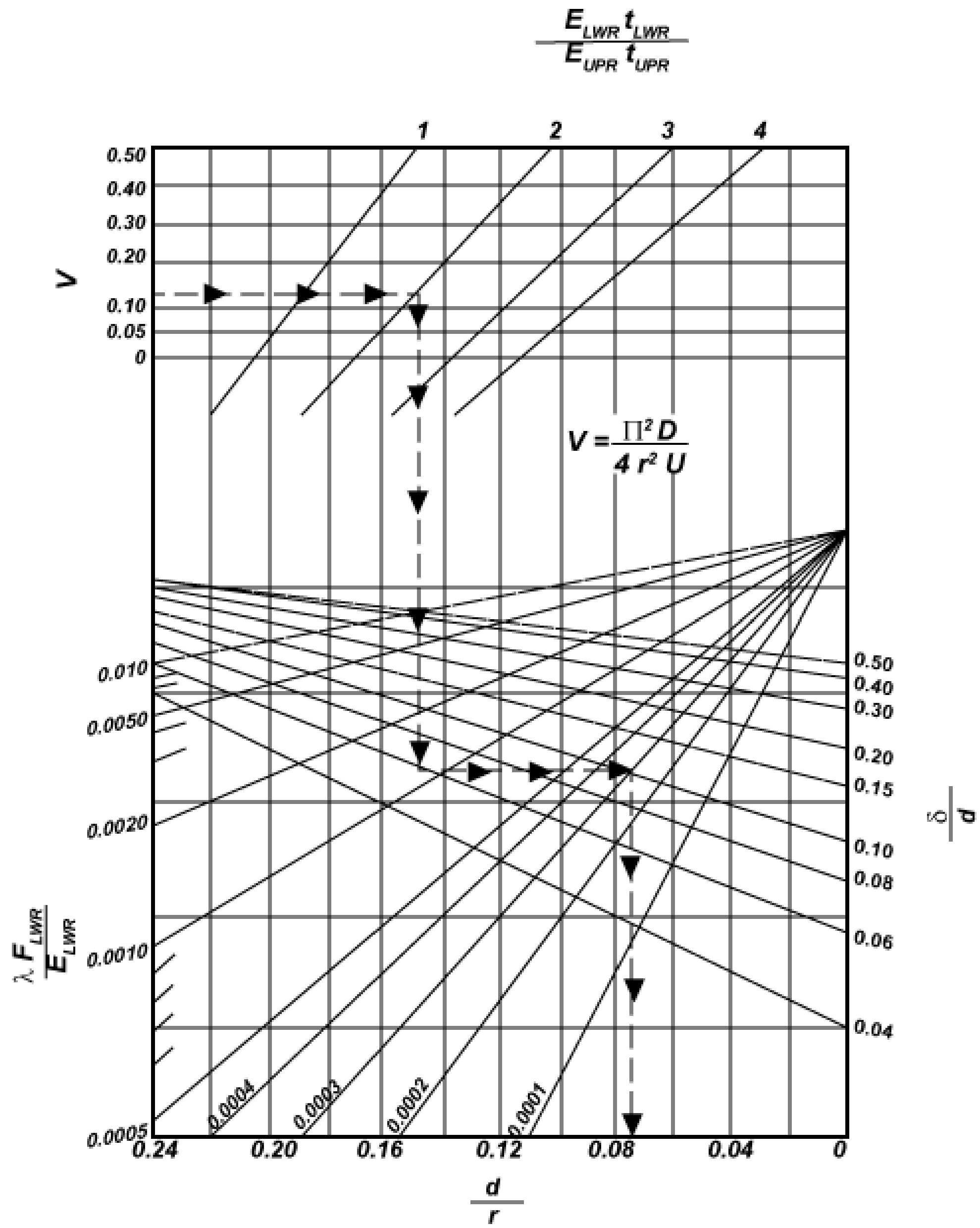

FIGURE 4.7.2.2(b) *Chart for determining $\frac{d}{r}$ ratio for flat circular sandwich panel, with isotropic face sheets and core, under uniformly distributed normal load, q, producing center deflection ratio $\frac{\delta}{d}$.*

4.7.2.2.1 Use of design charts

The sandwich must be designed by iterative procedures and the charts enable rapid determination of the various quantities sought. The charts were derived for a Poisson's ratio of 0.3, and can be used with small error for face sheets having other values of Poisson's ratio.

As a first approximation, it will be assumed that $V = 0$. If the design is controlled by face sheet stress criteria, as may be determined, this assumption will lead to an exact value of d. If the design is determined by deflection requirements, the assumption that $V = 0$ will produce a minimum value of d. The value of d is minimum because $V = 0$ if the core shear modulus is infinite. For any actual core, the shear modulus is not infinite; hence a thicker core must be used.

The following procedure is suggested:

1. Enter Figure 4.7.2.2(a) with the desired value for the parameter q/F_{UPR}. Assume a value for t_{UPR}/d and determine $\frac{d}{r}$. Compute d and t_{UPR}. Modify ratio t_{UPR}/d, if necessary, and determine more suitable values for d and t_{UPR}. Repeat this process with F_{LWR} and t_{LWR}.

2. Enter Figure 4.7.2.2(b) with the desired value for the parameters $E_{LWR}t_{LWR}/E_{UPR}t_{UPR}$ and $\lambda F_{LWR}/E_{LWR}$ and assume $V = 0$. Assume a value for $\frac{\delta}{d}$ and determine $\frac{d}{r}$. Compute d and δ. Modify ratio $\frac{\delta}{d}$, if necessary, and determine more suitable values for d and δ.

3. Repeat Steps 1 and 2, using lower chosen design face sheet stresses, until d determined by Step 2 is equal to, or a bit less than, d determined by Step 1.

4. Compute core thickness t_c using equations

$$t_c = d - \left(\frac{t_{UPR} + t_{LWR}}{2}\right)$$

$$t_c = d - t \quad \text{(for equal face sheets)}$$

This first approximation was based on a core with an infinite shear modulus. Since actual core shear modulus values are not very large, a somewhat larger value of t_c must be used. Successive approximations can be made by entering Figure 4.7.2.2(b) with values of V as computed by Equation 4.7.2.2(d).

In using Figure 4.7.2.2(b) for $V \neq 0$, it is necessary to iterate because V is directly proportional to core thickness t_c. As an aid to finally determining t_c and G_c, Figure 4.7.2.1.1 can again be used. The constant relating V to G_c may be computed from the equation

$$VG_c = \left[\frac{\pi^2 t_c E_{UPR} t_{UPR} E_{LWR} t_{LWR}}{4\lambda r^2 \left(E_{UPR} t_{UPR} + E_{LWR} t_{LWR}\right)}\right] \quad \text{or} \quad \left[\frac{\pi^2 t_c E t}{8\lambda r^2}\right] \quad \text{(for equal face sheets)}$$

With this constant, Figure 4.7.2.1.1 may be entered. Use of the figure is described in Section 4.7.2.1.

4.7.2.2.2 Determining core shear stress

This section gives procedures for determining the maximum core shear stress of a simply supported flat circular sandwich panel under uniformly distributed normal load. The core shear stress is maximum near the panel edge. The maximum shear stress, F_{asc}, is given by the formula

$$F_{sc} = \frac{q r}{2 d} \qquad 4.7.2.2.2$$

4.7.2.2.3 Checking procedure

The design may be checked by computing the face sheet stresses by Equation 4.7.2.2(a) and the deflection by Equation 4.7.2.2(c). The value of K_4 to be used in Equation 4.7.2.2(c) is given by

$$K_4 = \frac{16}{\pi^2 (3+\nu)} \left[\frac{(5+\nu)\pi^2}{64(1+\nu)} + V \right] \qquad 4.7.2.2.3(a)$$

which reduces to $K_4 = 0.309 + 0.491V$ when $\nu = 0.3$. Values of V may be computed by Equation 4.7.2.2(d).

An alternate method for computing the deflection at a panel center is given by the formula

$$\delta = K_5 \left(1 + \frac{E_{UPR}\, t_{UPR}}{E_{LWR}\, t_{LWR}} \right) \frac{\lambda q r^4}{\pi^2 E_{UPR}\, t_{UPR}\, d^2}$$
$$\delta = K_5 \left(1 + \frac{E_{LWR}\, t_{LWR}}{E_{UPR}\, t_{UPR}} \right) \frac{\lambda q r^4}{\pi^2 E_{LWR}\, t_{LWR}\, d^2} \qquad 4.7.2.2.3(b)$$
$$\delta = 2 K_5 \frac{\lambda q r^4}{\pi^2 E t d^2}$$

where

$$K_5 = \frac{(5+\nu)\pi^2}{64(1+\nu)} + V$$

which reduces to $K_5 = 0.629 + V$ when $\nu = 0.3$.

The core selected for the panel should be checked to be sure that it has a core shear modulus, G_c, at least as high as that assumed in computing the deflection in Equation 4.7.2.2(c), and that the core shear strength is sufficient to withstand the maximum core shear stress calculated from Equation 4.7.2.2.2.

If it is desired to check by a more accurate analysis, the equations given in Reference 4.7.2.2 may be used.

4.8 CURVED SANDWICH PANEL INTERNAL LOADS AND STRESSES

Curved sandwich panels under loading have deformation and stress fields that differ from flat panels. Curved panels may be stronger than flat panels in certain applications, but weaker in others. The differences appear in the equilibrium and kinematic equations for curved panels due to inclusion of the radii of curvature R_x and R_y.

4.8.1 General equations and analysis method

The equilibrium equations given here for a curved sandwich panel under pressure loading are based on the first-order shear deformation theory for a thick, anisotropic curved plate. In terms of loads, N_{ij}, and bending moments, M_{ij}, the coupled differential equations for a general shell can be written

$$\begin{aligned}
&\frac{\partial(\alpha_2 N_{11})}{\partial\xi_1} - \frac{\partial(\alpha_1 N_{21})}{\partial\xi_2} + N_{12}\frac{\partial\alpha_1}{\partial\xi_2} - N_{22}\frac{\partial\alpha_2}{\partial\xi_1} + \frac{\alpha_1\alpha_2 V_1}{R_1} = 0 \\
&\frac{\partial(\alpha_2 N_{12})}{\partial\xi_1} - \frac{\partial(\alpha_1 N_{22})}{\partial\xi_2} + N_{21}\frac{\partial\alpha_2}{\partial\xi_1} - N_{11}\frac{\partial\alpha_1}{\partial\xi_2} + \frac{\alpha_1\alpha_2 V_2}{R_2} = 0 \\
&\frac{\partial(\alpha_2 V_1)}{\partial\xi_1} + \frac{\partial(\alpha_1 V_2)}{\partial\xi_2} - \alpha_1\alpha_2\left(\frac{N_{11}}{R_1} + \frac{N_{22}}{R_2}\right) = \alpha_1\alpha_2 q(x,y) \\
&\alpha_1\alpha_2 V_1 = \frac{\partial(\alpha_2 M_{11})}{\partial\xi_1} - \frac{\partial(\alpha_1 M_{21})}{\partial\xi_2} + M_{12}\frac{\partial\alpha_1}{\partial\xi_2} - M_{22}\frac{\partial\alpha_2}{\partial\xi_1} \\
&\alpha_1\alpha_2 V_2 = \frac{\partial(\alpha_2 M_{12})}{\partial\xi_1} - \frac{\partial(\alpha_1 M_{22})}{\partial\xi_2} + M_{21}\frac{\partial\alpha_2}{\partial\xi_1} - M_{11}\frac{\partial\alpha_1}{\partial\xi_2} \\
&N_{12} + \frac{M_{12}}{R_1} = N_{21} + \frac{M_{21}}{R_2}
\end{aligned} \qquad \text{4.8.1(a)}$$

where $q(x,y)$ is the pressure load. The variables ξ_1 and ξ_2 are the shell reference surface coordinates, R_1 and R_2 are the principal radii of curvature of the middle surface of the shell, and α_1 and α_2 are scale factors relating the curvilinear surface coordinates to the Cartesian coordinates and are equal to the values of the Lamé coefficients at the reference surface, as shown in Figure 4.8.1. Equations for the calculation of these quantities can be found in many texts on differential geometry, such as References 4.8.1(a) and (b). The shear loads and bending moments are defined as in Section 4.5.2, and the transverse shear strains, γ_{13} and γ_{23}, and curvatures, κ_{11}, κ_{22}, and κ_{12}, in those definitions are written in terms of the out-of-plane displacement, w, and rotations about the midplane, ψ_x and ψ_θ, that are independent of the slope of w:

$$\begin{aligned}
&\gamma_{13} = \psi_1 + \frac{1}{\alpha_1}\frac{\partial w_1}{\partial\xi_1} - \frac{u_1}{R_1} \\
&\gamma_{23} = \psi_2 + \frac{1}{\alpha_2}\frac{\partial w_2}{\partial\xi_2} - \frac{u_2}{R_2} \\
&\kappa_{11} = \frac{1}{\alpha_1}\frac{\partial\psi_1}{\partial\xi_1} + \frac{1}{\alpha_1\alpha_2}\frac{\partial\alpha_1}{\partial\xi_2}\psi_2 \\
&\kappa_{22} = \frac{1}{\alpha_2}\frac{\partial\psi_2}{\partial\xi_2} + \frac{1}{\alpha_1\alpha_2}\frac{\partial\alpha_2}{\partial\xi_1}\psi_1 \\
&\kappa_{12} = \frac{1}{\alpha_2}\frac{\partial\psi_1}{\partial\xi_2} - \frac{1}{\alpha_1\alpha_2}\frac{\partial\alpha_2}{\partial\xi_1}\psi_2 + \frac{1}{\alpha_1}\frac{\partial\psi_2}{\partial\xi_1} - \frac{1}{\alpha_1\alpha_2}\frac{\partial\alpha_1}{\partial\xi_2}\psi_1
\end{aligned} \qquad \text{4.8.1(b)}$$

Note that for shallow shells, those for which the radii of curvature are large compared to the shell thickness, the last non-differential equation in Equation 4.8.1(a) becomes $N_{12} \approx N_{21}$. So, for a shallow cylindrical shell with constant radius, Equation 4.8.1(a) becomes:

$$
\begin{aligned}
&\frac{\partial N_x}{\partial x} - \frac{1}{R}\frac{\partial N_{x\theta}}{\partial \theta} = 0 \\
&\frac{\partial N_{x\theta}}{\partial x} - \frac{1}{R}\frac{\partial N_{\theta}}{\partial \theta} + \frac{V_\theta}{R} = 0 \\
&\frac{\partial V_x}{\partial x} + \frac{1}{R}\frac{\partial V_{\theta}}{\partial \theta} - \frac{N_\theta}{R} = q(x,y) \\
&V_x = \frac{\partial M_x}{\partial x} - \frac{1}{R}\frac{\partial M_{x\theta}}{\partial \theta} \\
&V_\theta = \frac{\partial M_{x\theta}}{\partial x} - \frac{1}{R}\frac{\partial M_{\theta}}{\partial \theta}
\end{aligned}
\qquad 4.8.1(c)
$$

These equations may be solved similarly to the flat panel equations by using an energy method in which the displacement distributions are assumed functions, which satisfy the required displacement boundary conditions. For example

$$
w(x,\theta) = \sum_{m=0}^{m\,\max} \sum_{n=0}^{n\,\max} c_{mn} \varphi_m(x) \varphi_n(\theta) \qquad 4.8.1(d)
$$

where c_{mn} are unknown coefficients to be determined through minimization of potential energy, and $\phi_m(\theta)$ and $\phi_n(\theta)$ are displacement functions.

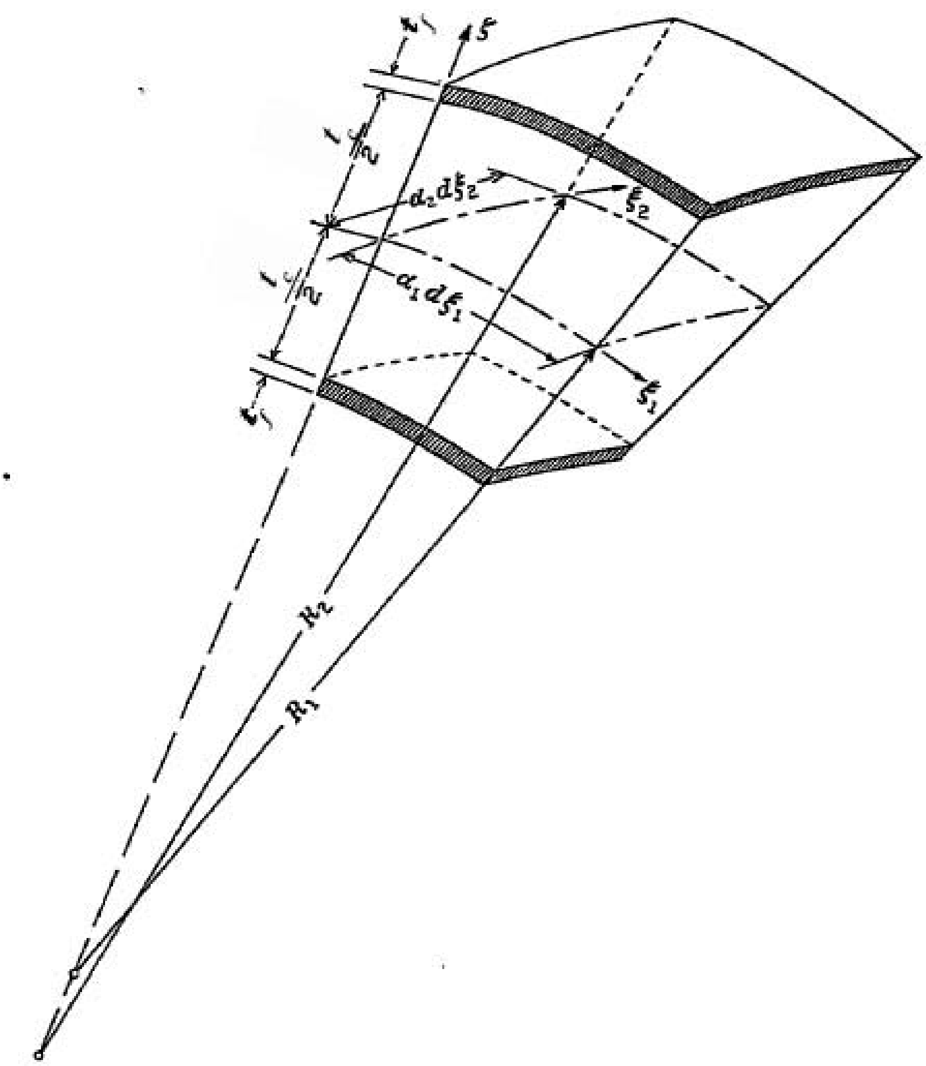

FIGURE 4.8.1 *Curved sandwich shell element.*

4.9 FLAT PANEL STABILITY ANALYSIS METHODS

This section provides equations and graphs for predicting the load at onset of global buckling for a flat sandwich column or flat sandwich panel, and for selecting the core material and the thickness of core and face sheets that will avoid buckling. This section addresses global buckling; the other failure modes listed in Section 4.4 should be checked separately.

For large or complex structure, general (global) stability solutions are best obtained from a computer analysis, using either a closed form buckling solution or a finite element analysis. The analysis must include the influence of the core transverse shear flexibility. In most cases of sandwich panels with thin, orthotropic face sheets, the panel stiffness matrix terms A_{16}, A_{26}, B_{ij}, $D_{16,}$ and D_{26} are usually small or zero, and thus the analysis method does not have to include the capability for including the effects of these terms. Most panel buckling codes, and all finite element codes, can analyze combinations of in-plane compression (axial or bi-axial) and shear loads.

Some computerized buckling analysis tools will predict the shear crimping instability mode. This mode occurs when the core shear stiffness is low. This mode can be detected from the analysis output by either a large number of half waves or a very short wavelength of the critical stability mode shape. See Section 4.6.7 for further information on this failure mode.

4.9.1 Buckling of flat rectangular sandwich columns

Assuming that a design begins with chosen design stresses and a given load to transmit, a flat rectangular column of sandwich construction shall be designed to comply with the basic design principles summarized in Section 4.2.1. These conditions must be met. This section addresses global buckling. The other failure modes listed in Section 4.4 should be checked separately.

Overall buckling of the sandwich, or local instabilities such as dimpling or wrinkling of the face sheets, may lead to total collapse of the column. Detailed procedures giving theoretical equations and graphs for determining dimensions of the face sheets and core, as well as necessary core properties, are given in the following paragraphs. Face sheet modulus of elasticity, E', and stress values, F_c, shall be compression values at the conditions of use; that is, if the application is at elevated temperature, then face sheet properties at elevated temperature shall be used in design. The face sheet modulus of elasticity is the effective value at the face sheet stress. If this stress is beyond the proportional limit value, an appropriate tangent, reduced, or modified compression modulus of elasticity shall be used (Reference 4.6.6.1(c)).

In the buckling analysis of sandwich columns, transverse shear deformations must be accounted for. This decreases the buckling load compared with Eulerian buckling calculations. The critical buckling load can be approximately written

$$\frac{1}{P_{cr}} = \frac{1}{P_b} + \frac{1}{P_s} \qquad 4.9.1(a)$$

where P_b is the buckling load in pure bending and P_s in pure shear.

When the face sheets are thin, the buckling loads are given by

$$P_b = \frac{n^2\pi^2 D}{(\beta L)^2} \quad \text{and} \quad P_s = S \qquad 4.9.1(b)$$

where L is the length of the column, D and S are the bending and shear stiffness of the panel, as defined in Section 4.5.1, n is the number of half-waves in the buckling mode, and the factor β depends on the boundary conditions, so that βL is the effective length of the column. Using the expressions for P_b and P_s in Equation 4.9.1(a), the buckling load can be written

$$P_{cr} = \frac{n^2\pi^2 D/(\beta L)^2}{1 + n^2\pi^2 D/S(\beta L)^2} \qquad 4.9.1(c)$$

When accounting for thick face sheets, this equation becomes

$$P_{cr} = \frac{\dfrac{2n^4\pi^4 D_f D_o}{S(\beta L)} + \dfrac{n^2\pi^2 D_o}{(\beta L)^2}}{1 + \dfrac{n^2\pi^2 D_o}{S(\beta L)^2}} \qquad 4.9.1(d)$$

where S, D_o and D_f are calculated as in Section 4.5.1.

4.9.2 Design of flat rectangular sandwich panels under edgewise compression load

Assuming that a design begins with chosen design stresses and a given load to transmit, a flat rectangular panel of sandwich construction under edgewise compression load shall be designed to comply with the basic design principles summarized in Section 4.2.1. These conditions must be met. This section addresses global buckling. The other failure modes listed in Section 4.4 should be checked separately.

Overall buckling of the sandwich, or local instabilities such as dimpling or wrinkling of the face sheets, may lead to total collapse of the panel. Detailed procedures giving theoretical equations and graphs for determining dimensions of the face sheets and core, as well as necessary core properties, are given in the following paragraphs. Double equations are given, one equation for sandwich with face sheets of different materials and thicknesses and another equation for sandwich with each face sheet of the same material and thickness. Face sheet modulus of elasticity, E', and stress values, F_c, shall be compression values at the conditions of use; that is, if application is at elevated temperature, then face sheet properties at elevated temperature shall be used in design. The face sheet modulus of elasticity is the effective value at the face sheet stress. If this stress is beyond the proportional limit value, an appropriate tangent, reduced, or modified compression modulus of elasticity shall be used (Reference 4.6.6.1(c)).

4.9.2.1 Determining face sheet thickness

Face sheet stresses are related to the edge load by the equations:

$$t_{UPR} F_{cUPR} + t_{LWR} F_{cLWR} = N \qquad 4.9.2.1$$

$$t = \frac{N}{2F_c} \quad \text{(for equal face sheets)}$$

where t is face sheet thickness, F_c is chosen design face sheet compressive stress, N is design compression load per unit length of panel edge, and UPR, LWR are subscripts denoting upper and lower face sheets.

In determining thicknesses of face sheets for sandwich with face sheets of different materials, this equation must be satisfied, but also the stresses $F_{C\ UPR}$ and $F_{C\ LWR}$ must be chosen so that $F_{C\ UPR}/E_{S\ UPR} = F_{C\ LWR}/E_{S\ LWR}$ (where E_S is face sheet secant modulus of elasticity), to achieve equal strains in the two face sheets and thus avoid overstressing of either face sheet. For example, if the upper face sheet is of a material such that the ratio $F_{C\ UPR}/E_{S\ UPR} = 0.005$, and the lower face sheet is of a material such that the ratio $F_{C\ LWR}/E_{S\ LWR} = 0.002$, then the design must be based on a ratio of 0.002, otherwise the lower face sheet will be overstressed. In order to accomplish this, the chosen design stress for the upper face sheet must be lowered. For many combinations of face sheet materials it will be found advantageous to choose thicknesses such that $E_{UPR}\, t_{UPR} = E_{LWR}\, t_{LWR}$.

If the core can support edge load, N should be replaced by the quantity $(N - F_{c\ core}\, t_c)$, where $F_{c\ core}$ is the compressive stress carried by the core, and t_c is the core thickness.

4.9.2.2 Determining core thickness and core shear modulus

This section gives procedures for determining core thickness and core shear modulus so that overall buckling of the sandwich panel will not occur (References 4.9.2.2(a) through 4.9.2.2(c)). The load per unit panel width at which buckling of a sandwich panel will occur is given by the theoretical formula:

$$N_{cr} = K \frac{\pi^2}{b^2} D$$

where D is sandwich bending stiffness. This formula, solved for the critical buckling face sheet stress, F_{cr}, becomes;

$$F_{cr\,UPR} = \frac{\pi^2 K E'_{UPR}}{\lambda} \frac{E'_{UPR} t_{UPR} E'_{LWR} t_{LWR}}{\left(E'_{UPR} t_{UPR} + E'_{LWR} t_{LWR}\right)^2} \left(\frac{d}{b}\right)^2$$

$$F_{cr\,LWR} = \frac{\pi^2 K E'_{LWR}}{\lambda} \frac{E'_{UPR} t_{UPR} E'_{LWR} t_{LWR}}{\left(E'_{UPR} t_{UPR} + E'_{LWR} t_{LWR}\right)^2} \left(\frac{d}{b}\right)^2 \qquad 4.9.2.2(a)$$

$$F_{cr} = \frac{\pi^2 K E'}{4\lambda} \left(\frac{d}{b}\right)^2 \text{ (for equal face sheets)}$$

where E' is effective compressive modulus of elasticity of face sheet at stress F_C (for orthotropic face sheets $E' = \sqrt{E'_a E'_b}$), $\lambda = 1 - \nu^2$, ν is Poisson's ratio of face sheets (in Equation 4.9.2.2(a), it is assumed that the two face sheets have the same Poisson ratio, $\nu = \nu_{UPR} = \nu_{LWR}$), d is distance between face sheet centroids, b is length of loaded panel edge, $K = K_F + K_M$, K_F is a theoretical coefficient dependent on face sheet stiffness and panel aspect ratio, and K_M is a theoretical coefficient dependent on sandwich bending and shear rigidities and panel aspect ratio. Information on calculating K_F and K_M is given in Section 4.9.2.3.

Solving for $\frac{d}{b}$ gives:

$$\frac{d}{b} = \frac{1}{\pi\sqrt{K}} \sqrt{\frac{\lambda F_{cr\,UPR}}{E'_{UPR}} \left(\frac{E'_{UPR} t_{UPR} + E'_{LWR} t_{LWR}}{\sqrt{E'_{UPR} t_{UPR} E'_{LWR} t_{LWR}}} \right)}$$

$$\frac{d}{b} = \frac{1}{\pi\sqrt{K}} \sqrt{\frac{\lambda F_{cr\,LWR}}{E'_{LWR}} \left(\frac{E'_{UPR} t_{UPR} + E'_{LWR} t_{LWR}}{\sqrt{E'_{UPR} t_{UPR} E'_{LWR} t_{LWR}}} \right)} \qquad 4.9.2.2(b)$$

$$\frac{d}{b} = \frac{2}{\pi\sqrt{K}} \sqrt{\frac{\lambda F_{cr}}{E'}} \text{ (for equal face sheets)}$$

Therefore, if K is known, Equation 4.9.2.2(b) can be solved directly to obtain d because all other quantities are known. After d is obtained, the core thickness, t_C is computed from the equations

$$t_c = d - \frac{t_{UPR} + t_{LWR}}{2} \qquad 4.9.2.2(c)$$

$$t_c = d - t \text{ (for equal face sheets)}$$

As a first approximation, it will be assumed that $K_F = 0$, hence $K = K_M$. Values of K_M depend upon the bending and shear rigidities of the sandwich as incorporated in the parameter

$$V = \frac{\pi^2 D}{b^2 U}$$

which can be written as:

$$V = \frac{\pi^2 t_c E'_{UPR} t_{UPR} E'_{LWR} t_{LWR}}{\left(E'_{UPR} t_{UPR} + E'_{LWR} t_{LWR}\right)\lambda b^2 G_{ca}} \qquad 4.9.2.2(d)$$

$$V = \frac{\pi^2 t_c E' t}{2\lambda b^2 G_{ca}} \quad \text{(for equal face sheets)}$$

where U is sandwich shear stiffness, G_{ca} is the core shear modulus associated with the axes parallel to direction of loading (also parallel to panel side of length a) and perpendicular to the plane of the panel. As values of core shear modulus decrease, values of V increase and values of K_M gradually decrease.

For sandwich with corrugated core having corrugation flutes parallel to the direction of loading, the parameter V is replaced by the parameter

$$V_2 = \frac{\pi^2 t_c E'_{UPR} t_{UPR} E'_{LWR} t_{LWR}}{\left(E'_{UPR} t_{UPR} + E'_{LWR} t_{LWR}\right)\lambda b^2 G_{cb}} \qquad 4.9.2.2(e)$$

$$V_2 = \frac{\pi^2 t_c E' t}{2\lambda b^2 G_{cb}} \quad \text{(for equal face sheets)}$$

where G_{cb} is the core shear modulus associated with the axes perpendicular to the direction of loading (parallel to panel side of length b) and perpendicular to the plane of the panel.

4.9.2.2.1 Determining minimum value of d

A minimum value of d required will be determined by assuming $V = 0$ or $V_2 = 0$ for a first approximation. The value of d is minimum because $V = 0$ or $V_2 = 0$ only if the core shear modulus is infinite; for any actual core the shear modulus is not infinite, hence a thicker core must be used. The chart of Figure 4.9.2.2.1(a) gives minimum values of d for sandwich panels with isotropic or orthotropic face sheets and core and for various edge conditions. Panels with clamped edges are included in the chart of Figure 4.9.2.2.1(a), although truly clamped edges are not actually attainable. Panels with edges supported by beams are discussed at the end of Section 4.9.2.3.

The charts of Figure 4.9.2.2.1(a) and figures in Sections 4.9.2.2.2 and 4.9.2.3 are applicable to sandwich with isotropic face sheets for which $\alpha = 1.0$, $\beta = 1.0$, $\gamma = 0.375$, and to sandwich with orthotropic face sheets, such as glass fabric laminates[1], for which $\alpha = 1.0$, $\beta = 0.6$, $\gamma = 0.2$.

The constants α, β, and γ depend upon elastic properties of the face sheets as follows:

$$\alpha = \sqrt{\frac{E'_b}{E'_a}}; \quad \beta = \alpha\nu_{ab} + 2\gamma; \quad \gamma = \frac{\lambda G'_{ba}}{\sqrt{E'_a E'_b}} \qquad 4.9.2.2.1(a)$$

where E'_a and E'_b are the moduli of elasticity parallel and perpendicular to the direction of loading, G'_{ba} is the face sheet shear modulus associated with those directions, ν_{ab} is the Poisson's ratio of the contraction in the b direction to extension in the a direction due to a tensile stress in the a direction, ν_{ba} is similarly defined, and $\lambda = 1 - \nu_{ab}\nu_{ba}$. For isotropic face sheets, the figures were generated assuming $\nu = 0.25$. For orthotropic face sheets it was assumed that $\nu_{ab} = \nu_{ba} = 0.2$, $E'_a = E'_b$, and $G_{ab} = 0.21\ E_a$.

---

[1] Laminates giving these values of α, β, and γ were of polyester and epoxy laminates with glass fabrics 112, 116 ,120, 128, 162, 164, 181, 182, 183 and 184 (Reference 4.9.2.2(d)).

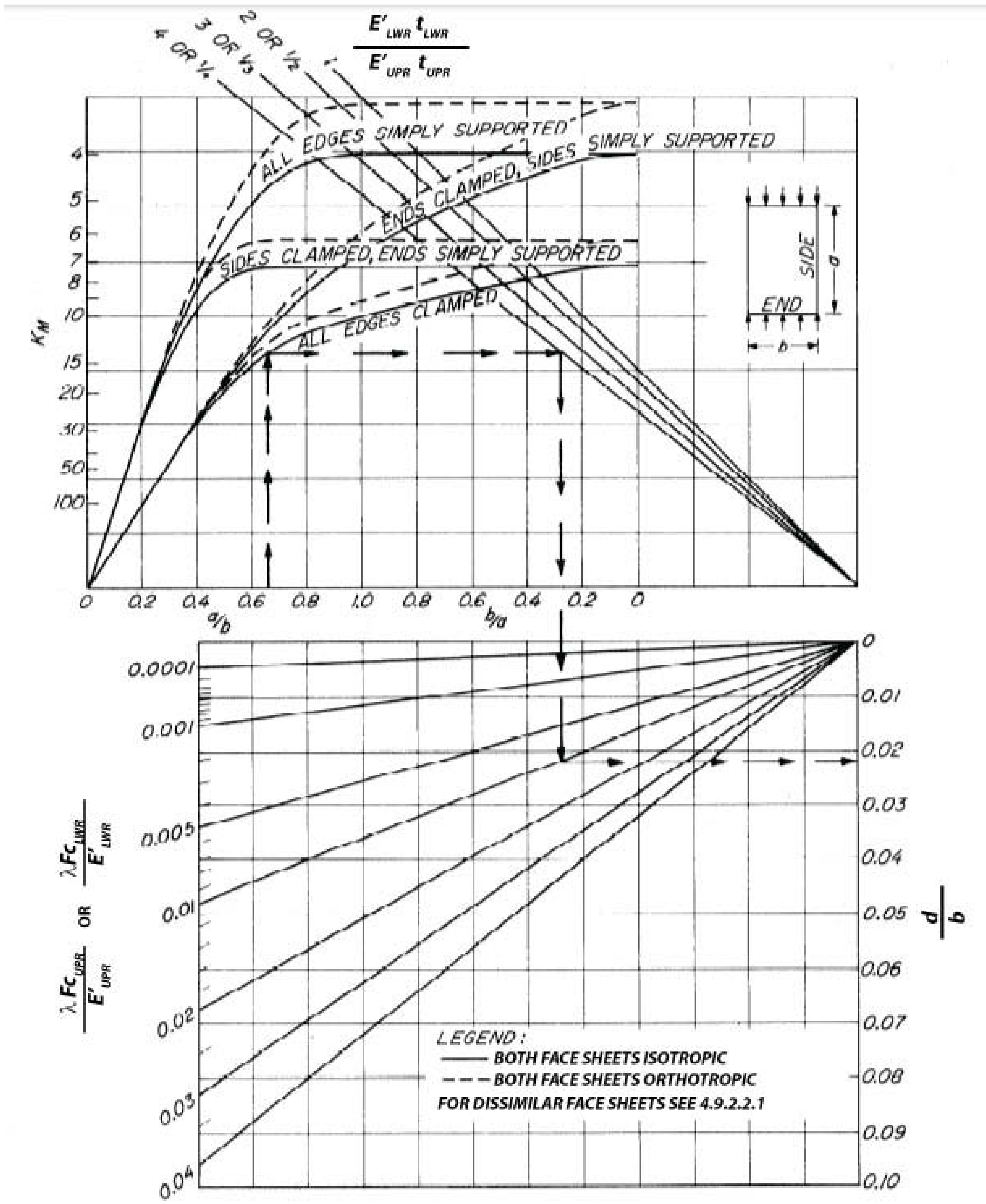

FIGURE 4.9.2.2.1(a) *Chart for determining $\frac{d}{b}$ ratio ($V = 0$) such that a sandwich panel will not buckle under edgewise compression load.*

Parameters needed for use of the chart of Figure 4.9.2.2.1(a) are:

1. Panel aspect ratio $\frac{a}{b}$ or $\frac{b}{a}$

2. Face sheet properties $\frac{\lambda F_{c\,UPR}}{E'_{UPR}}$ and $\frac{\lambda F_{c\,LWR}}{E'_{LWR}}$

3. Ratio of $E'_{LWR} t_{LWR} / E'_{UPR} t_{UPR}$

The charts of Figure 4.9.2.2.1(a) and figures in Sections 4.9.2.2.2 and 4.9.2.3 are also applicable to sandwich with dissimilar face sheets wherein the upper face sheet is isotropic (α_{UPR} = 1.0, β_{UPR} = 1.0, and γ_{UPR} = 0.375) and the lower face sheet is orthotropic (α_{LWR} = 1.0, β_{LWR} = 0.6, and γ_{LWR} = 0.2). For such a sandwich, linear interpolation is made between curves for sandwich with both face sheets isotropic and curves for sandwich with both face sheets orthotropic by means of the parameter

$$Q = \frac{1}{1 + \left(\frac{\lambda_{UPR}}{\lambda_{LWR}}\right)\left(\frac{t_{LWR}}{t_{UPR}}\right)\sqrt{\frac{E'_{aLWR}\, E'_{bLWR}}{E'_{aUPR}\, E'_{bUPR}}}}$$

With the above assumptions that $\alpha_{UPR} = \alpha_{LWR} = 1.0$ and $\lambda_{UPR} = \lambda_{LWR}$

$$Q = \frac{1}{1 + E'_{LWR} t_{LWR} / E'_{UPR} t_{UPR}}$$

Values of Q = 0 correspond to sandwich with both face sheets isotropic and Q = 1 to sandwich with both face sheets orthotropic, although in both cases it is not necessary that the two face sheets have the same modulus and thickness. This is demonstrated by substitution of these Q values in the general expression for K_M (see definition of K_M in Section 4.9.2.3 and discussion in Reference 4.9.2.2(c)). Thus, for example, if Q = 1/4, interpolation is at 1/4 of the distance from the curve for both face sheets isotropic toward the curve for both face sheets orthotropic.

4.9.2.2.2 Determining actual value of d

Figure 4.9.2.2.1(a) was generated assuming that the core shear modulus is infinite. Since actual core shear modulus values are not very large, a value of d somewhat greater than given on Figure 4.9.2.2.1(a) must be used.

Charts for determining d for sandwich with all edges simply supported are shown in Figures 4.9.2.2.2(a) through 4.9.2.2.2(e). These figures are entered with values of the panel aspect ratio and values of V as computed by Equation 4.9.2.2(d). Figure 4.9.2.2.2(a) applies to sandwich with isotropic cores, for which the core shear modulus perpendicular to the direction of loading is equal to the core shear modulus parallel to the direction of loading. Figure 4.9.2.2.2(b) applies to sandwich with cores for which the core shear modulus perpendicular to the direction of loading is equal to 0.40 times the core shear modulus parallel to the direction of loading. Figure 4.9.2.2.2(c) applies to sandwich with honeycomb cores for which the core shear modulus perpendicular to the direction of loading is 2.50 times the core shear modulus parallel to the direction of loading.

NOTE: For honeycomb cores with core ribbons parallel to direction of loading, $G_c = G_{TL}$ and the shear modulus perpendicular to loading is G_{TW}. For honeycomb cores with core ribbons perpendicular to direc-

tion of loading $G_c = G_{Tw}$ and the shear modulus perpendicular to loading is G_{TL}. If core ribbons are at an angle θ to the panel length a, $G_c = \dfrac{G_{TL}G_{TW}}{\left(G_{TL}\sin^2\theta + G_{TW}\cos^2\theta\right)}$.

Figure 4.9.2.2.2(d) applies to sandwich with corrugated core having the core flutes perpendicular to the direction of loading. Figure 4.9.2.2.2(e) applies to sandwich with corrugated core having the core flutes parallel to the direction of loading and requires values of the parameter V_2 given by Equation 4.9.2.2(e) instead of values of V.

In using Figures 4.9.2.2.2(a) through 4.9.2.2.2(e), it is necessary to iterate because V is directly proportional to the core thickness t_c. As an aid to determining t_c and G_c, Figure 4.9.2.2.2(f) presents a number of lines representing V for various values of G_c with V ranging from 0.01 to 2 and G_c ranging from 1,000 to 1,000,000 pounds per square inch. The following procedure is suggested:

1. Determine a core thickness t_c from Figures 4.9.2.2.2(a), (b), (c), (d), or (e) using a value of 0.01 for V or V_2.

2. Compute the constant relating V or V_2 to G_c.

$$\left[\frac{\pi^2 t_c E'_{UPR} t_{UPR} E'_{LWR} t_{LWR}}{\left(E'_{UPR} t_{UPR} + E'_{LWR} t_{LWR}\right)\lambda b^2}\right] \text{ or } \left[\frac{\pi^2 t_c E' t}{2\lambda b^2}\right]\left(\text{for equal face sheets}\right) = VG_c \text{ or } V_2 G_c$$

3. With this constant enter Figure 4.9.2.2.2(f) and determine necessary G_c.

4. If the shear modulus is outside the range of values for materials available, slide up the appropriate line of Figure 4.9.2.2.2(f) and pick a new value of V or V_2, for a reasonable value of core shear modulus.

5. Reenter Figure 4.9.2.2.2(a), (b), (c), (d), or (e) with the new value of V or V_2 and repeat previous steps 1, 2, and 3.

Charts of the type used in Figures 4.9.2.2.2(a), (b), (c), (d), or (e) have not been prepared for panels with ends or sides clamped. True clamping at panel edges is never attained, particularly for sandwich constructions. It is suggested that each panel be designed as simply supported on all edges and then enter Figure 4.9.2.2.1(a) to estimate any possible reduction that can be made in core thickness due to edge clamping.

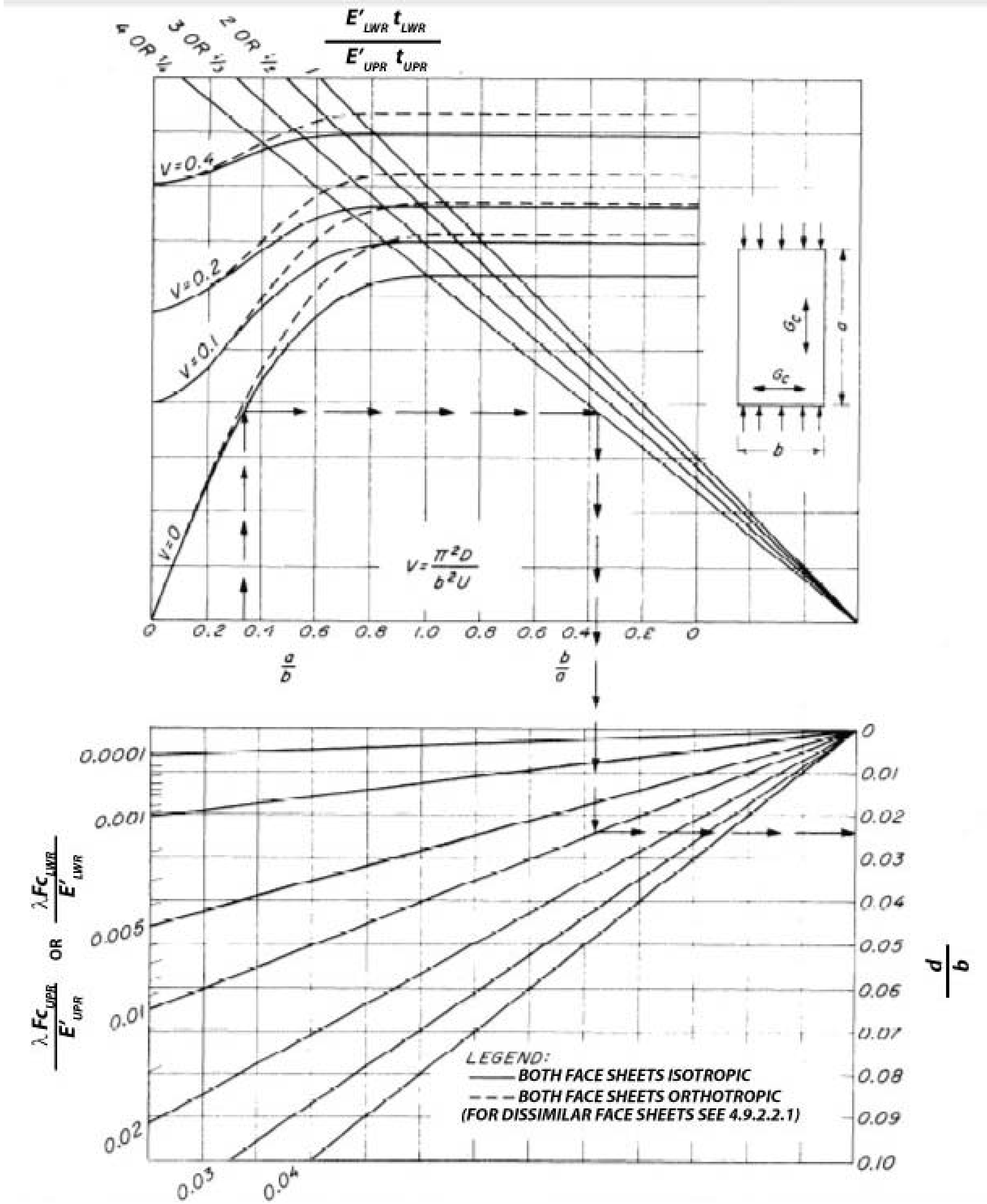

FIGURE 4.9.2.2.2(a) *Chart for determining $\frac{d}{b}$ ratio such that a simply-supported sandwich panel with isotropic core ($G_{cb} = G_{ca}$) will not buckle under edgewise compression load.*

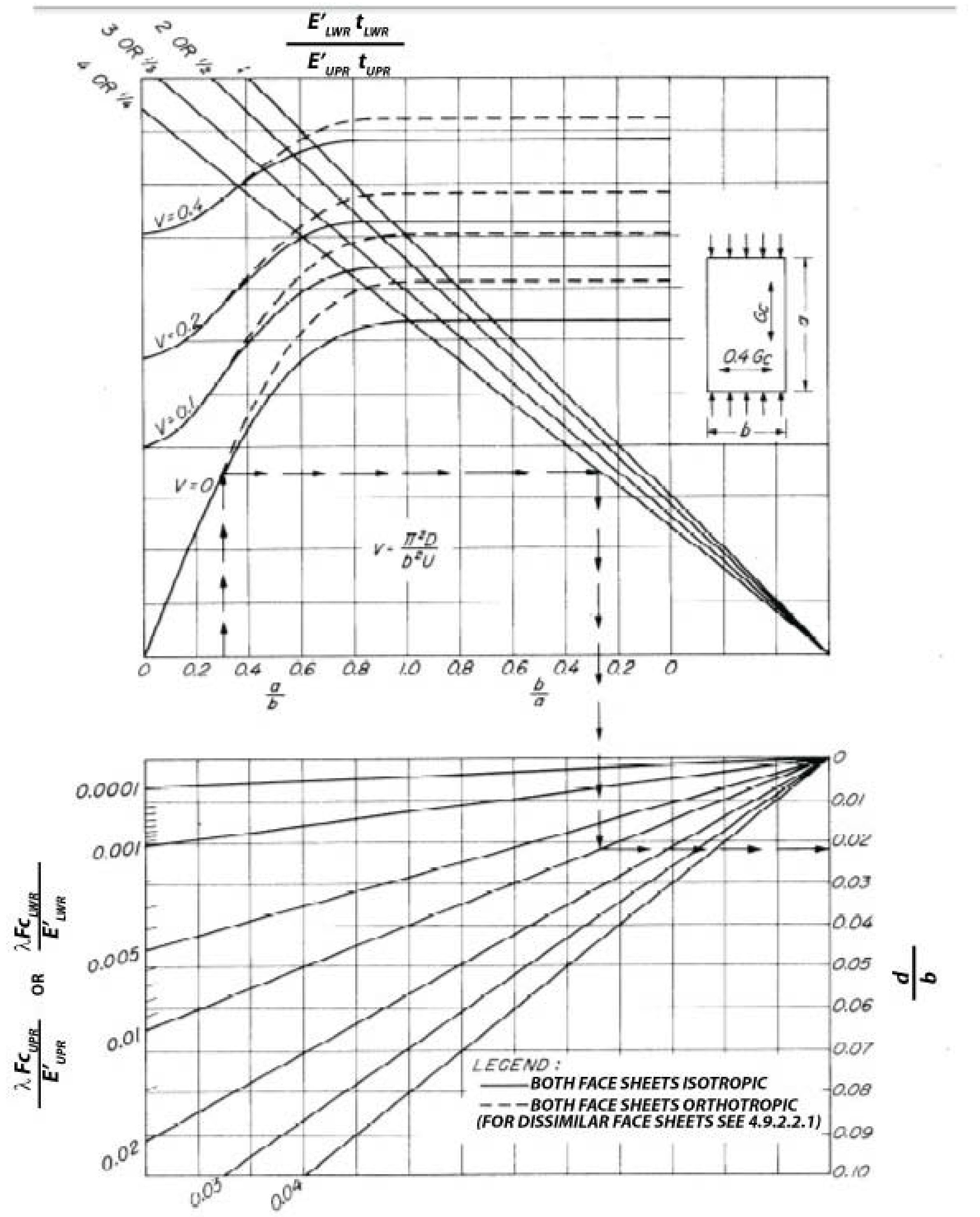

FIGURE 4.9.2.2.2(b) *Chart for determining $\frac{d}{b}$ ratio such that a simply-supported sandwich panel with orthotropic core ($G_{cb} = 0.4G_{ca}$) will not buckle under edgewise compression load.*

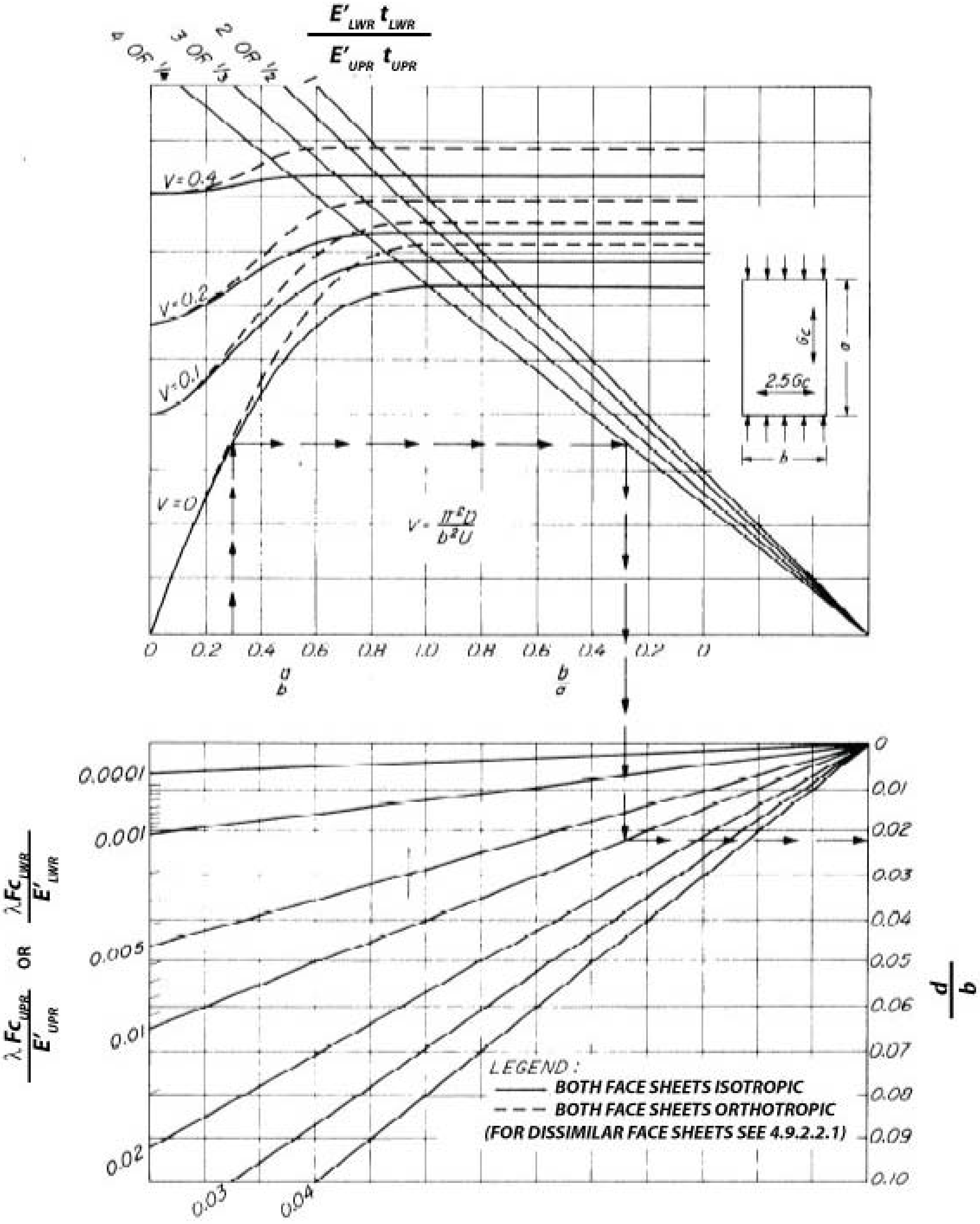

FIGURE 4.9.2.2.2(c) *Chart for determining $\frac{d}{b}$ ratio such that a simply-supported sandwich panel with orthotropic core ($G_{cb} = 2.5G_{ca}$) will not buckle under edgewise compression load.*

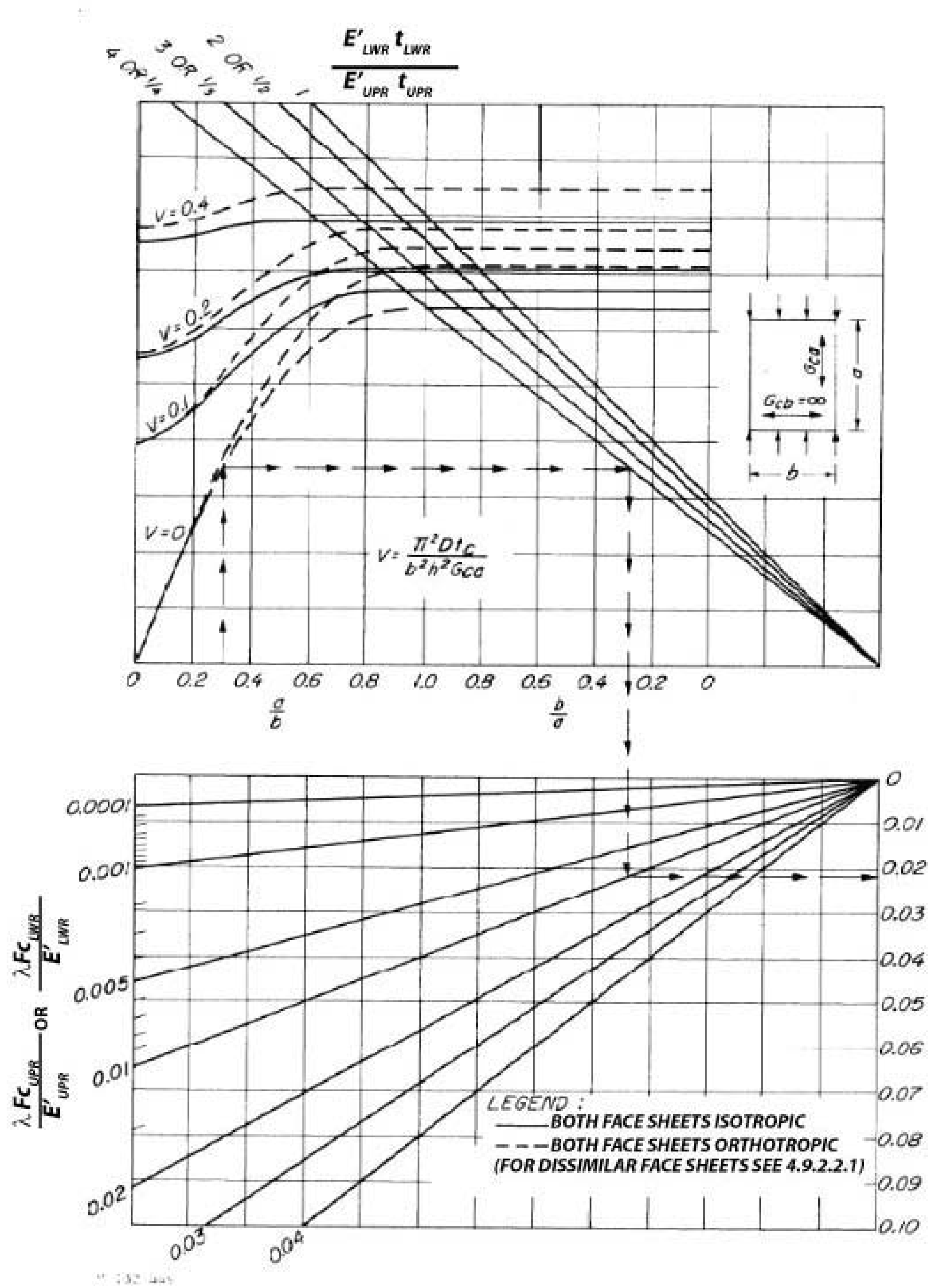

FIGURE 4.9.9.2.2(d) *Chart for determining $\frac{d}{b}$ ratio such that a simply-supported sandwich panel with corrugated core will not buckle under edgewise compression load; core corrugation flutes perpendicular to load direction.*

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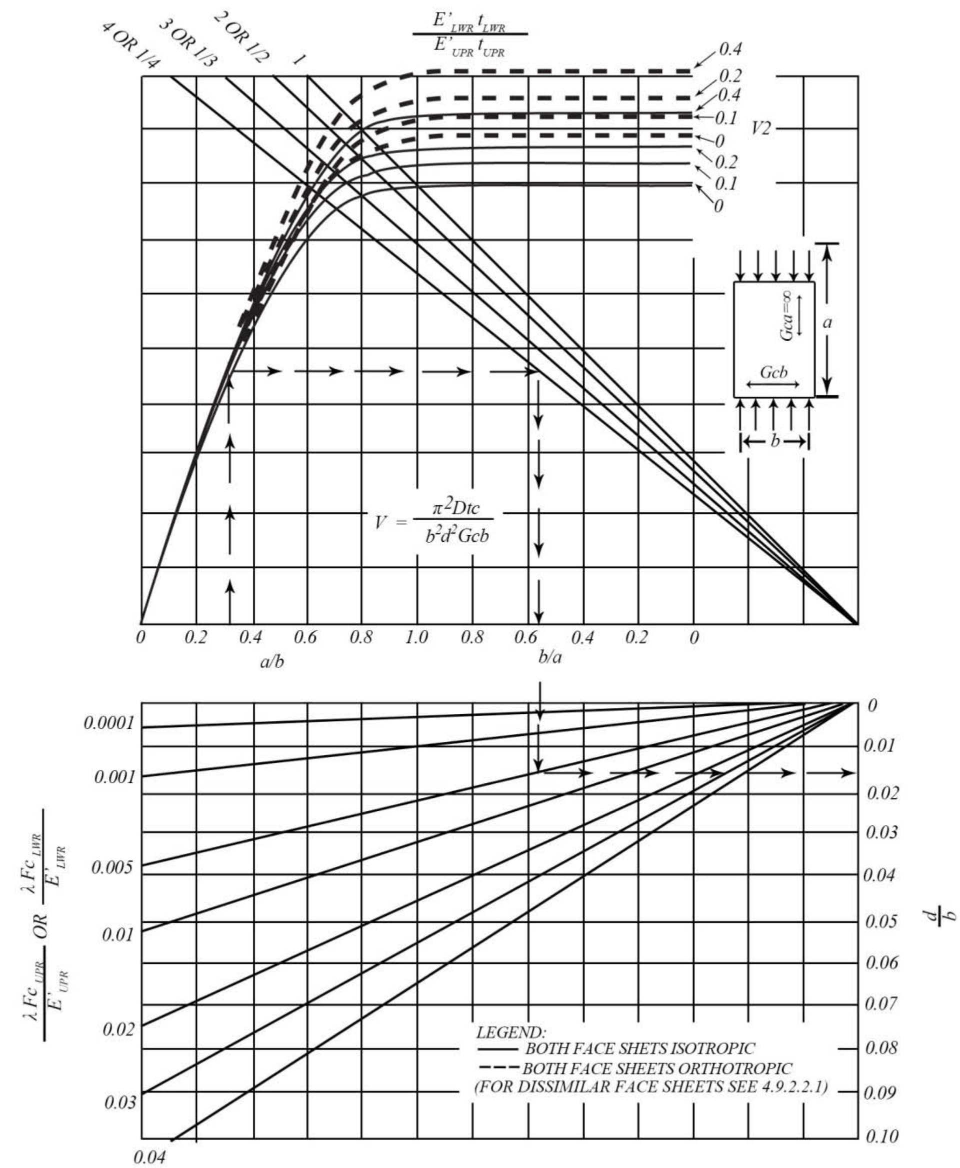

FIGURE 4.9.2.2.2(e) *Chart for determining $\frac{d}{b}$ ratio such that a simply-supported sandwich panel with corrugated core will not buckle under edgewise compression load; core corrugation flutes parallel to load direction.*

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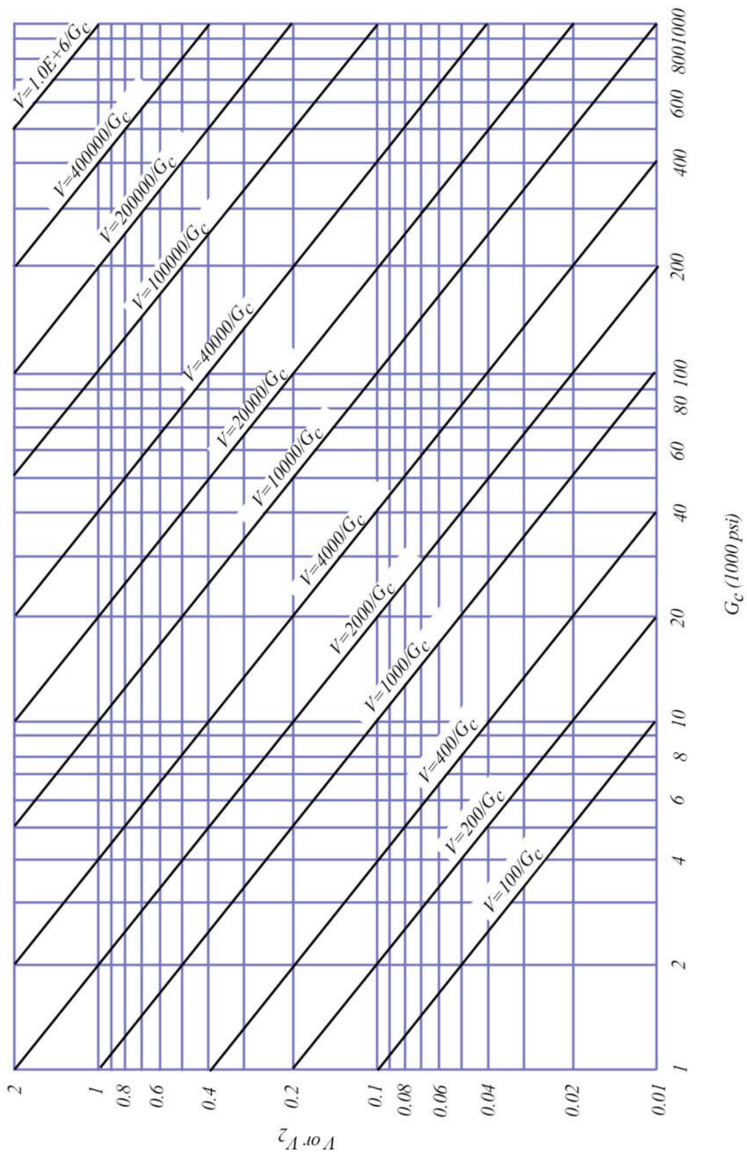

FIGURE 4.9.2.2.2(f) *Chart for determining V or V_2 and G_c for sandwich in edgewise compression.*

4.9.2.3 Checking procedure for determining buckling stress, F_{cr}

The design shall be checked by using the graphs of Figures 4.9.2.3(a) through 4.9.2.3(o) to determine values of K_M for use in evaluating $K = K_F + K_M$ to substitute into Equation 4.9.2.2(a) to compute actual buckling stress, F_{cr}.

The figures apply to sandwich panels with edges simply supported and clamped, and to sandwich with isotropic or certain orthotropic face sheets cores (see Section 4.9.2.2.1).

For each value of the parameter V, there is a cusped curve giving values of K_M for various values of the ratios $\frac{a}{b}$ or $\frac{b}{a}$. These cusps are indicated by dotted lines for the top curve in each figure. The cusps show the sandwich panel buckling coefficients calculated for different values of n, the number of half waves into which the panel buckles. Only the portions of each cusped curve for which K_M is a minimum are shown. Envelope curves indicate values of K_M for use in design.

Values of K_F shall be determined by the equation

$$K_F = \frac{\left(E'_{UPR}\, t^3_{UPR} + E'_{LWR}\, t^3_{LWR}\right)\left(E'_{UPR}\, t_{UPR} + E'_{LWR}\, t_{LWR}\right)}{12\, E'_{UPR}\, t_{UPR}\, E'_{LWR}\, t_{LWR}\, d^2} K_{MO} \qquad 4.9.2.3(a)$$

$$K_F = \frac{t^2}{3d^2} K_{MO} \text{ for equal face sheets}$$

where K_{MO} is determined from the chart of Figure 4.9.2.3(o). ($K_{MO} = K_M$ when $V = 0$.) For panels with $\frac{a}{b}$ ratios larger than shown on Figure 4.9.2.3(o), it can be assumed that $K_F = 0$. Then K shall be computed as $K = K_F + K_M$ and Equation 4.9.2.2(a) solved for F_{cr}. It should be understood that if the desired F_{cr} is above proportional limit values, the value of E' shall be an effective value, used in computing V and F_{cr}.

If the charts do not apply because ratios of core shear moduli are far different from what is given on the charts, or it is desired to check by a more accurate analysis, the equations given in the following shall be used (References 4.9.2.2(a) through (c)):

$$K_M = \frac{\psi_1 K_{LWR} + \left(1 + \frac{R}{c_4}\right) B_{LWR}\, V}{\psi_2 + \psi_3\, \Phi_{LWR}\, V + \frac{R}{c_4} B_{LWR}\, V^2} \qquad 4.9.2.3(b)$$

$$K_i = \alpha_i c_1 + 2\beta_i c_2 + \frac{c_3}{\alpha_i} \qquad 4.9.2.3(c)$$

where

$$\psi_1 = Q + (1-Q)\, \frac{K_{UPR}}{K_{LWR}}\, \frac{B_{LWR}}{B_{UPR}} \qquad 4.9.2.3(d)$$

$$\psi_2 = Q^2 + 2Q(1-Q)\frac{B_{UL}}{B_{UPR}} + (1-Q)^2\, \frac{B_{LWR}}{B_{UPR}} \qquad 4.9.2.3(e)$$

$$\psi_3 = Q + (1-Q)\frac{\Phi_{UPR}}{\Phi_{LWR}}\, \frac{B_{LWR}}{B_{UPR}} \qquad 4.9.2.3(f)$$

$$B_i = c_1\, c_3 - \beta_i^2\, c_2^2 + \gamma_i\, c_2\, K_i \qquad 4.9.2.3(g)$$

$$B_{UL} = \left(\frac{\alpha^2_{UPR} + \alpha^2_{LWR}}{2\,\alpha_{UPR}\,\alpha_{LWR}}\right) c_1\, c_3 - \beta_{UPR}\, \beta_{LWR}\, c_2^2 + \frac{c_2}{2}\left(\gamma_{UPR} K_{LWR} + \gamma_{LWR} K_{UPR}\right) \qquad 4.9.2.3(h)$$

$$\Phi_i = \alpha_i c_1 \frac{R}{c_4} + \left(1 + \frac{R}{c_4}\right)\gamma_i c_2 + \frac{c_3}{\alpha_i} \tag{4.9.2.3(i)}$$

The parameters of these equations are given by the following expressions:

$$Q = \frac{A_{UPR}}{A_{UPR} + A_{LWR}} \tag{4.9.2.3(j)}$$

$$V = \frac{A_{UPR}\, A_{LWR}}{A_{UPR} + A_{LWR}} \frac{\pi^2\, t_c}{b^2\, G_{ca}} \tag{4.9.2.3(k)}$$

$$R = \frac{G_{ca}}{G_{cb}} \tag{4.9.2.3(l)}$$

$$A_i = \frac{t_i}{\lambda_i}\sqrt{E'_{ai}\, E'_{bi}} \tag{4.9.2.3(m)}$$

where G_{cb} and G_{ca} are the moduli of transverse rigidity of the core associated with the directions of the loaded and unloaded edges of the panel, and the subscript i may be replaced by UPR or LWR to refer to the upper or lower face sheet, respectively. The parameters α, β, and γ are defined in Equation 4.9.2.2.1(a).

The values of c_1, c_2, c_3, and c_4 depend upon the panel aspect ratio, $\frac{b}{a}$, the integral number of longitudinal half waves, n, into which the panel buckles, and the panel edge conditions. Values of n are chosen to produce minimum values of the compression load per unit of panel edge, N.

For a panel with all edges simply supported:

$$c_1 = c_4 = \frac{a^2}{n^2 b^2}, \quad c_2 = 1, \quad \text{and } c_3 = \frac{n^2 b^2}{a^2} \tag{4.9.2.3(n)}$$

For a panel with loaded edges simply supported and other edges clamped:

$$c_1 = \frac{16a^2}{3n^2 b^2}, \quad c_2 = \frac{4}{3}, \quad c_3 = \frac{n^2 b^2}{a^2}, \quad \text{and } c_4 = \frac{4a^2}{3n^2 b^2} \tag{4.9.2.3(o)}$$

For a panel with loaded edges clamped and other edges simply supported:

$$\text{For } n = 1 \quad c_1 = c_4 = \frac{3a^2}{4b^2}, \quad c_2 = 1, \quad \text{and } c_3 = \frac{4b^2}{a^2} \tag{4.9.2.3(p)}$$

$$\text{For } n \geq 2 \quad c_1 = c_4 = \frac{a^2}{\left(n^2+1\right)b^2}, \quad c_2 = 1, \quad c_3 = \left(\frac{n^4 + 6n^2 + 1}{n^2+1}\right)\left(\frac{b^2}{a^2}\right) \tag{4.9.2.3(q)}$$

For a panel with all edges clamped:

$$\text{For } n = 1 \quad c_1 = 4c_4 = \frac{4a^2}{b^2}, \quad c_2 = \frac{4}{3}, \quad \text{and } c_3 = \frac{4b^2}{a^2} \qquad 4.9.2.3(r)$$

$$\text{For } n \geq 2 \quad c_1 = 4c_4 = \frac{16a^2}{3\left(n^2+1\right)b^2}, \quad c_2 = \frac{4}{3}, \quad c_3 = \left(\frac{n^4+6n^2+1}{n^2+1}\right)\left(\frac{b^2}{a^2}\right) \qquad 4.9.2.3(s)$$

Adaptations of these equations to sandwich with corrugated core are made by considering core shear modulus infinite in the direction of the corrugation flutes (details in Reference 4.9.2.2(b)). If the corrugation flutes are parallel to the direction of loading, they can carry load in proportion to their area and elastic modulus.

For panels with edges not simply supported but supported by beams with low torsional rigidity but finite bending stiffness, the buckling coefficient may be much lower than for panels with simply supported edges (Reference 4.9.2.3). The buckling coefficient for such a panel is dependent upon the parameters ζ and ϕ in addition to the usual parameters, where ζ and ϕ depend upon bending stiffness and cross sectional area of the beam supports. Charts showing effects of beam support stiffness and area on buckling coefficients are given in Figures 4.9.2.3(p) and (q).

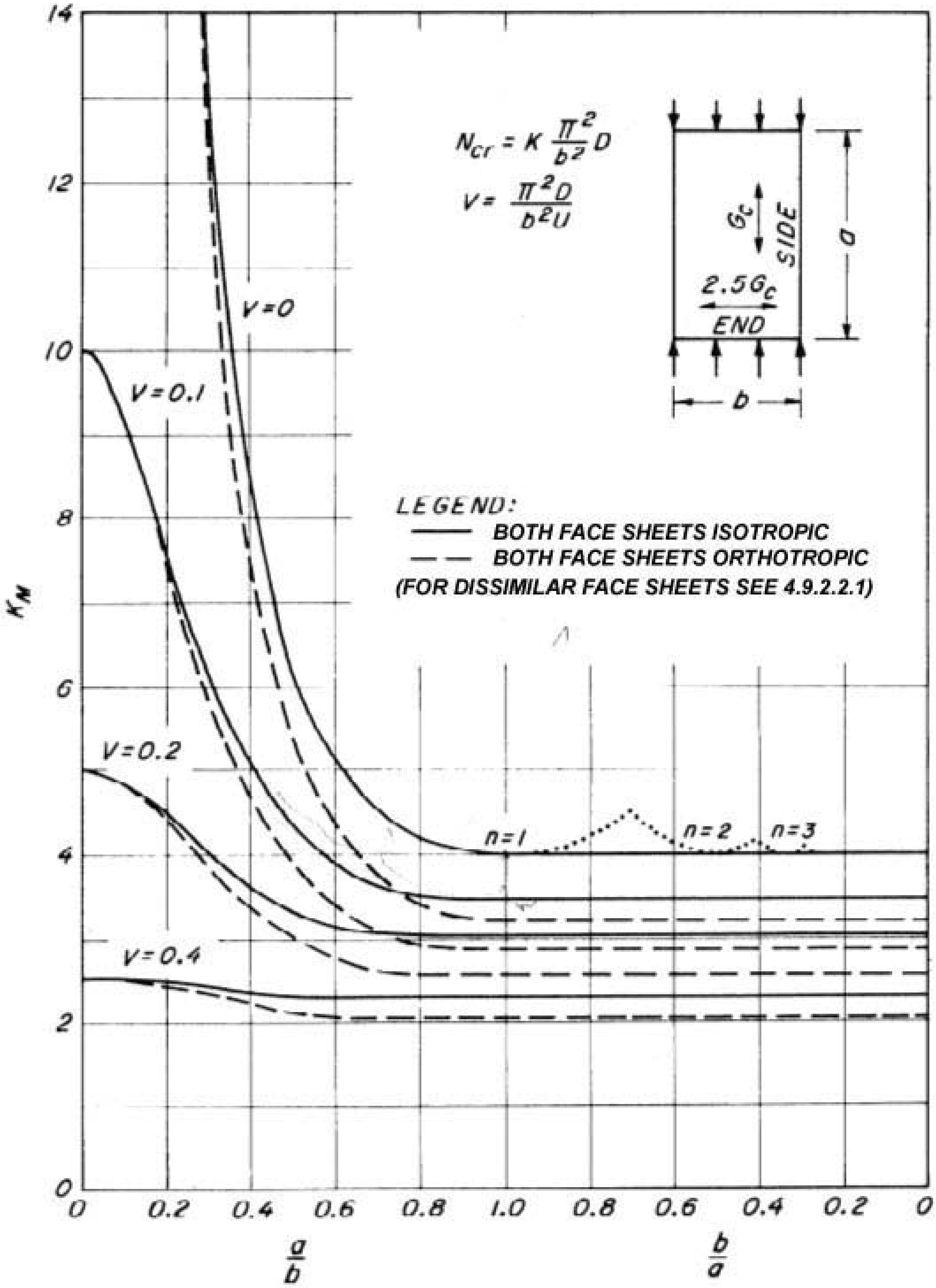

FIGURE 4.9.2.3(a) *K_M for sandwich panel with ends and sides simply supported and orthotropic core ($G_{cb} = 2.5G_{ca}$).*

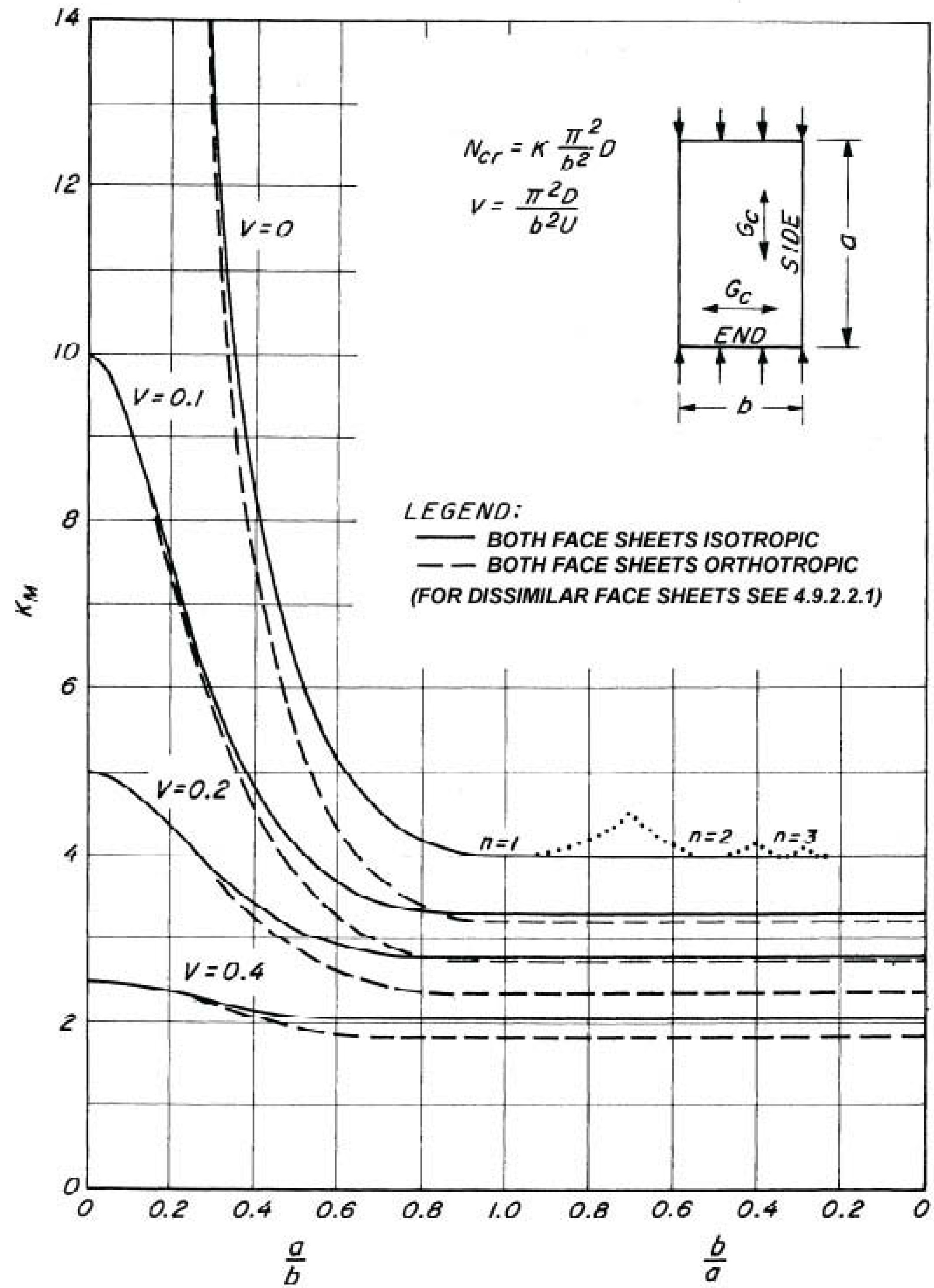

FIGURE 4.9.2.3(b) *K_M for sandwich panel with ends and sides simply supported and isotropic core ($G_{cb} = G_{ca}$).*

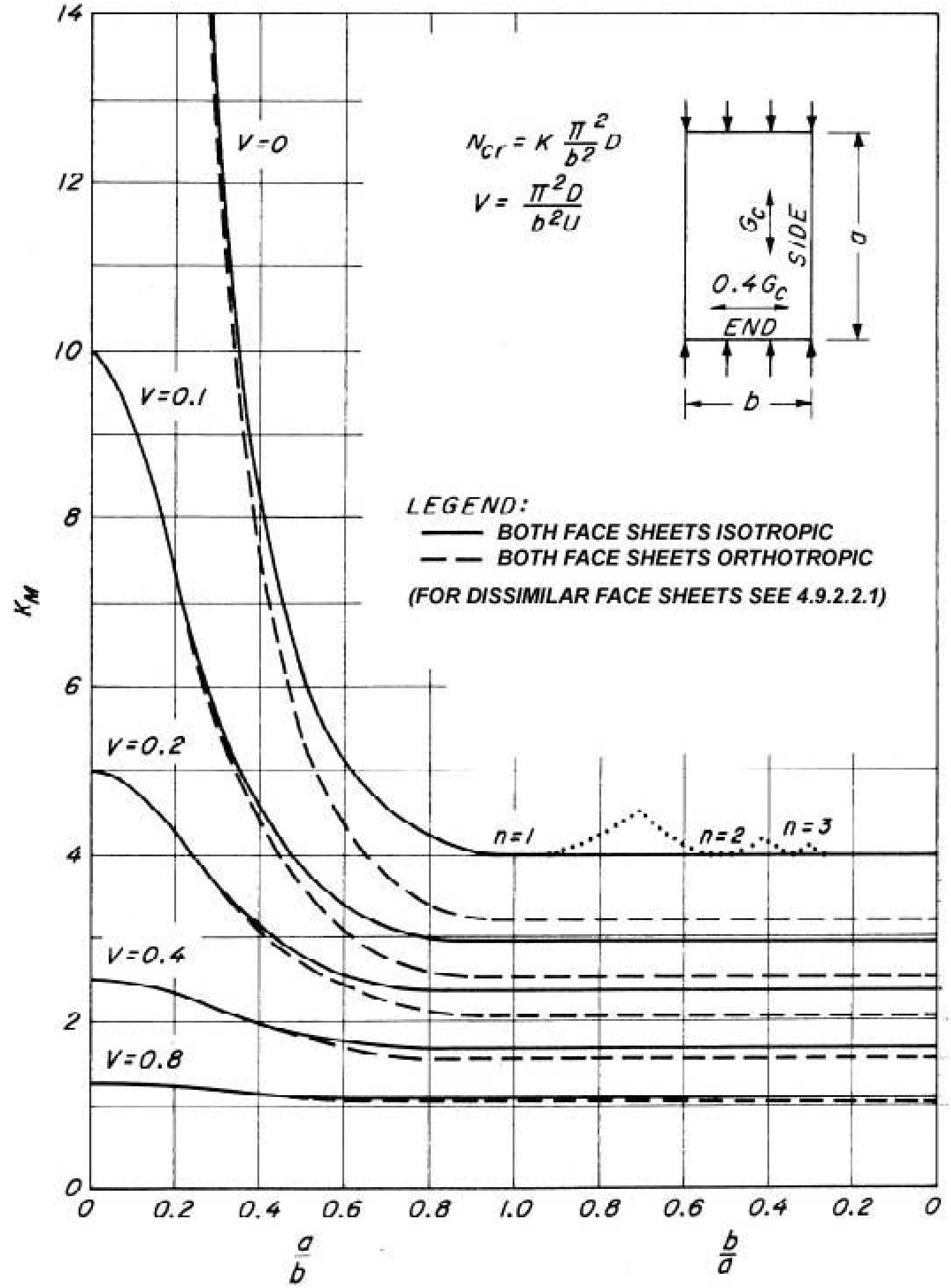

FIGURE 4.9.2.3(c) *K_M for sandwich panel with ends and sides simply supported and orthotropic core ($G_{cb} = 0.4G_{ca}$).*

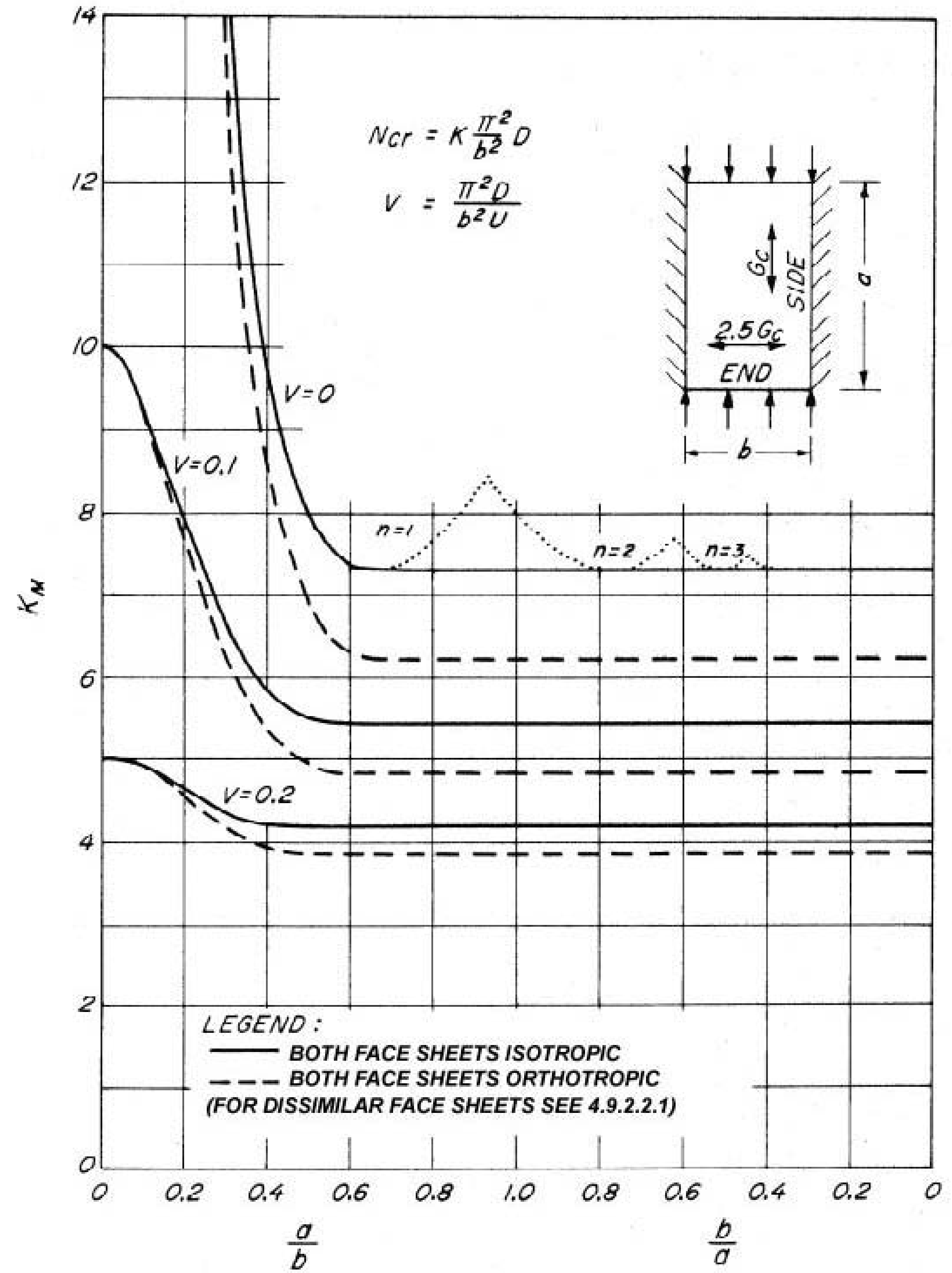

FIGURE 4.9.2.3(d) *K_M for sandwich panel with ends simply supported and sides clamped and orthotropic core ($G_{cb} = 2.5G_{ca}$).*

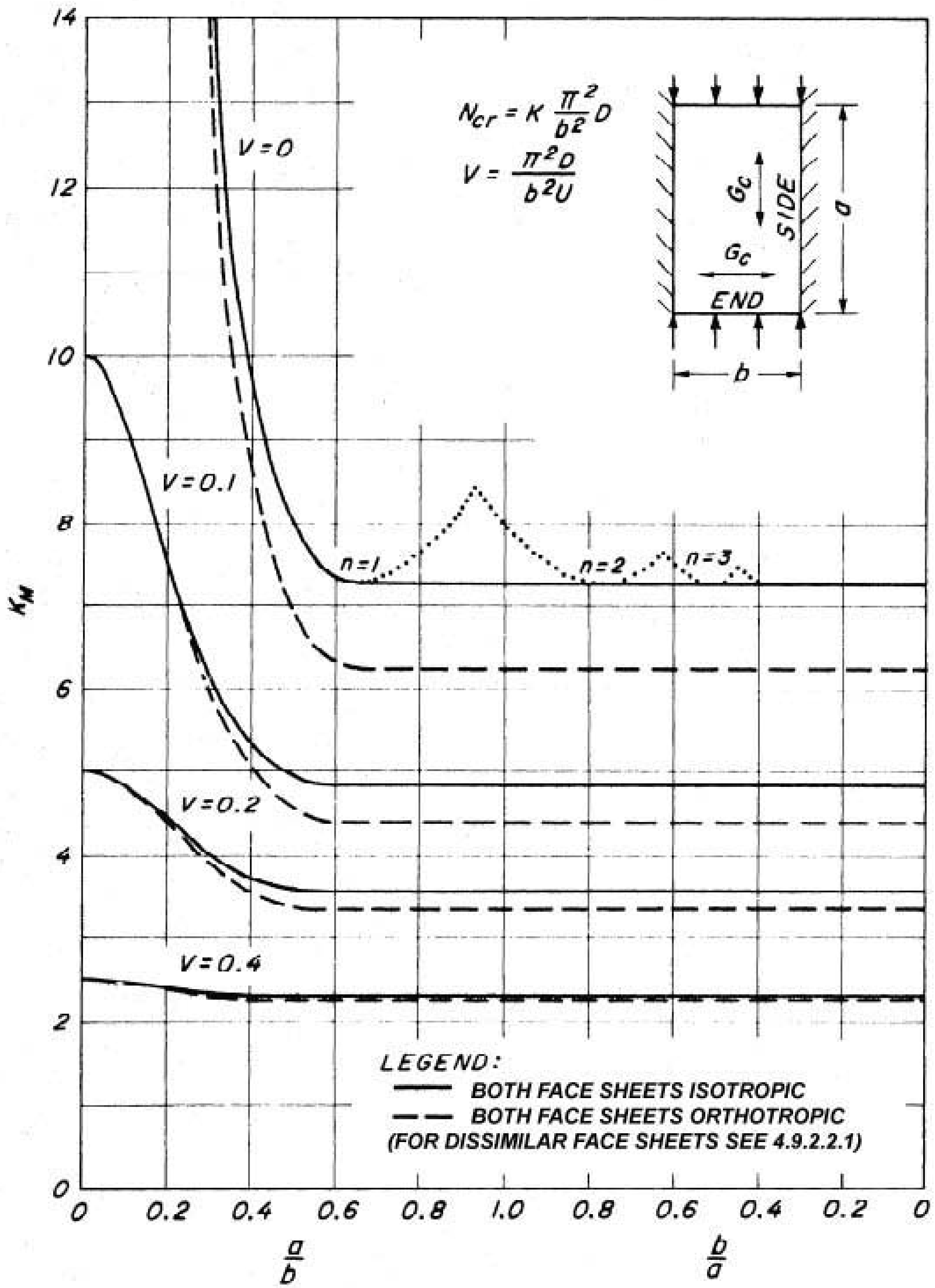

FIGURE 4.9.2.3(e) *K_M for sandwich panel with ends simply supported sides clamped and isotropic core ($G_{cb} = G_{ca}$).*

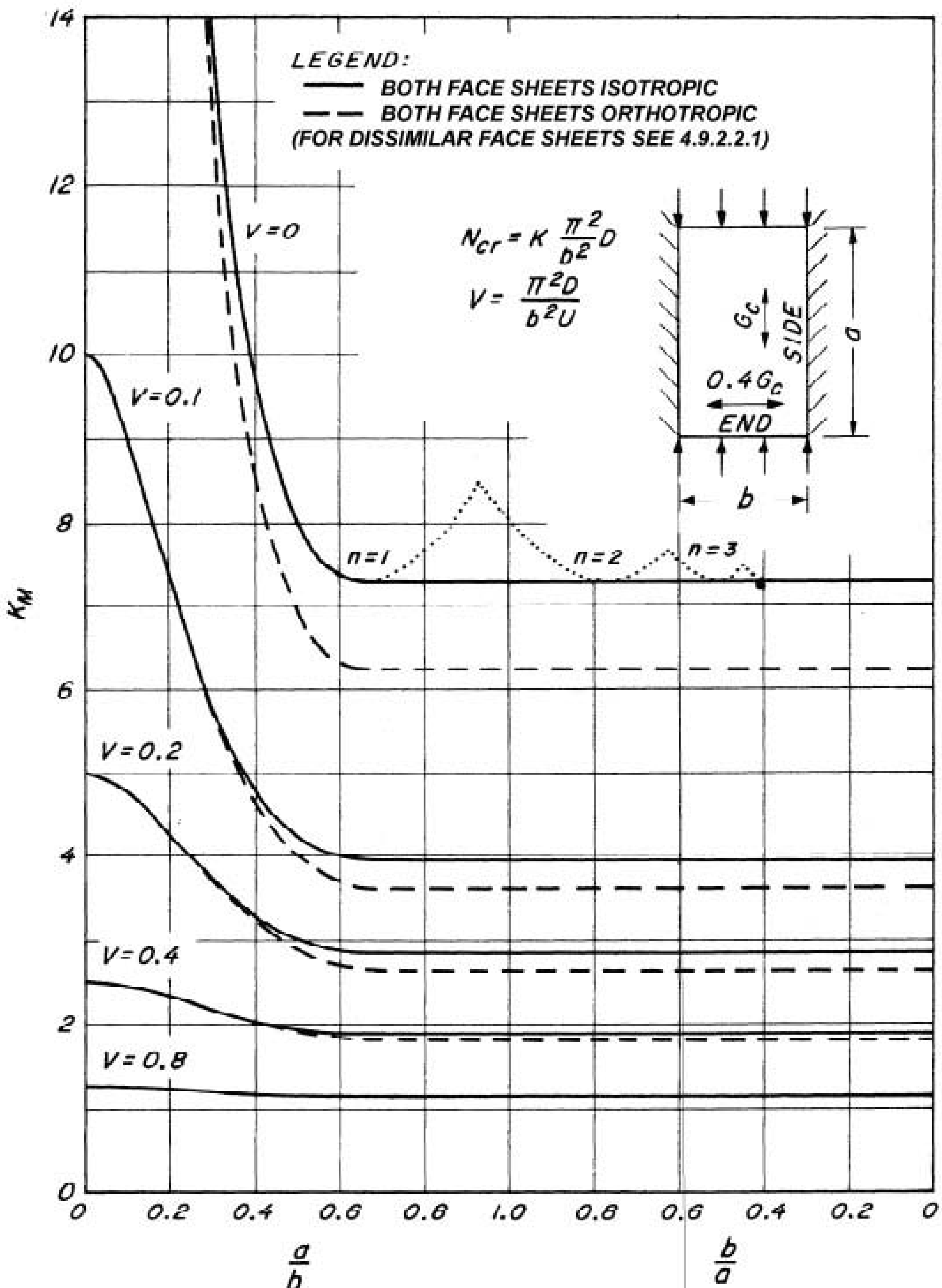

FIGURE 4.9.2.3(f) *K_M for sandwich panel with ends simply supported and sides clamped and orthotropic core ($G_{cb} = 0.4G_{ca}$).*

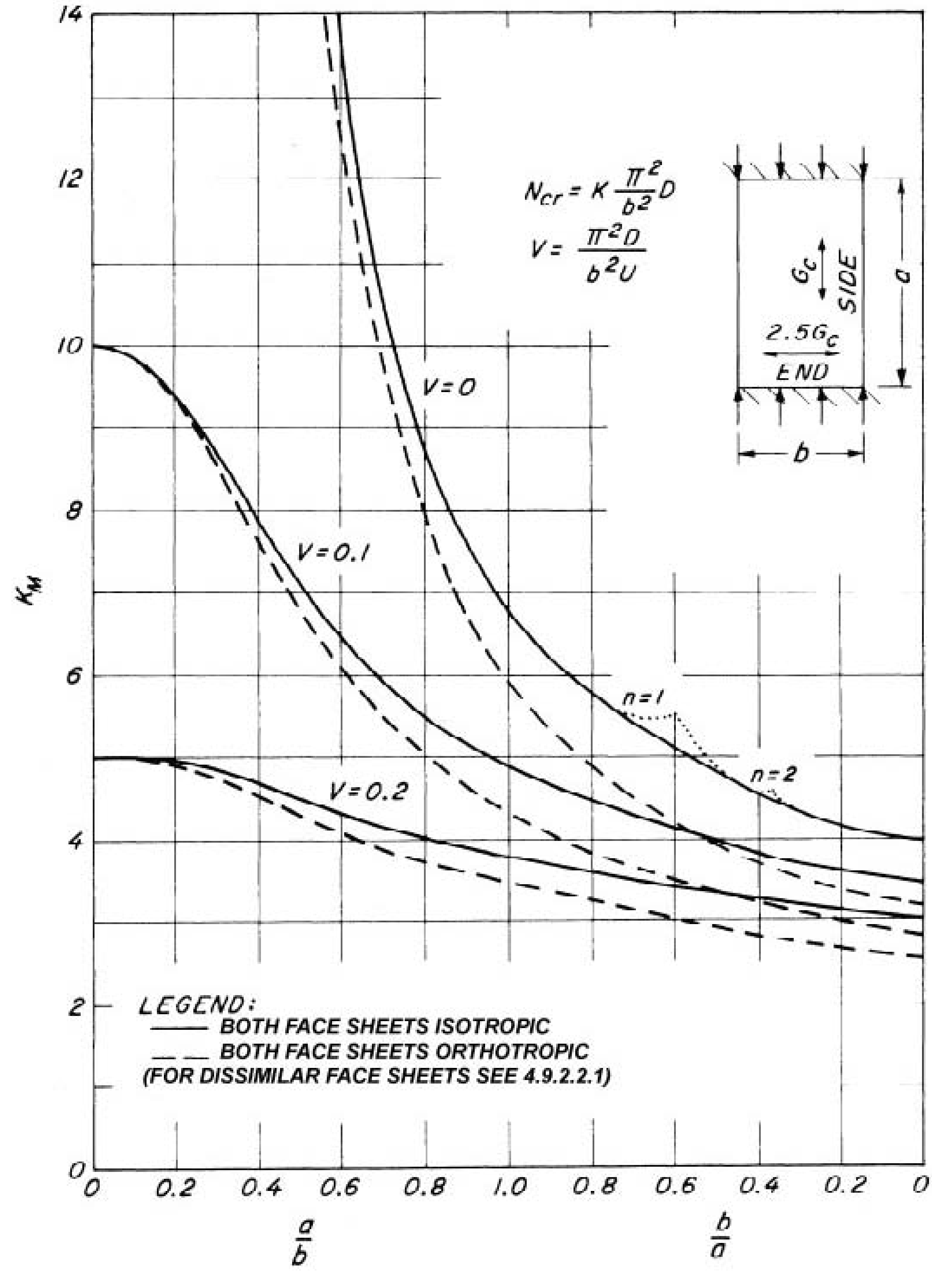

FIGURE 4.9.2.3(g) *K_M for sandwich panel with ends clamped and sides simply supported and orthotropic core ($G_{cb} = 2.5G_{ca}$).*

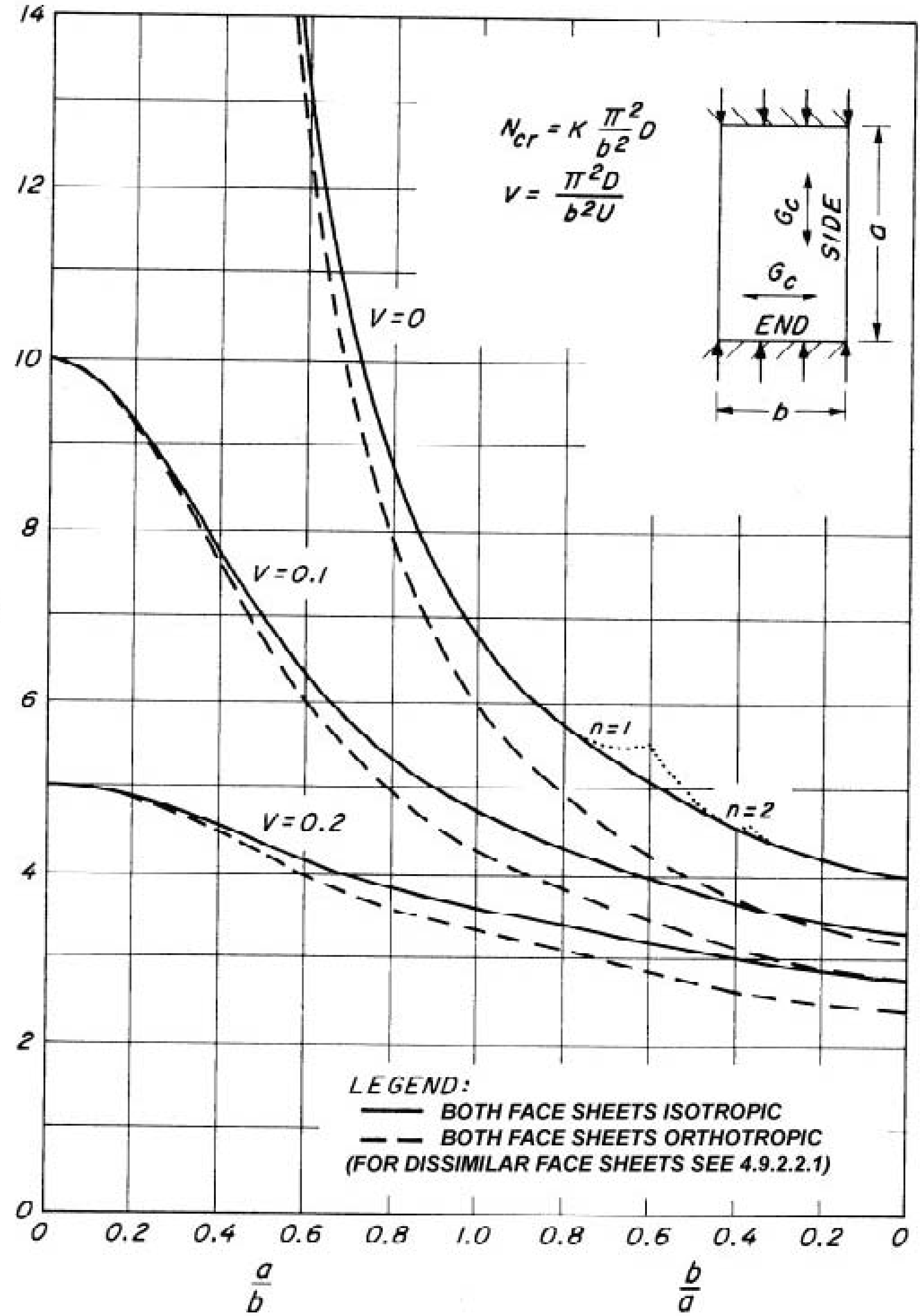

FIGURE 4.9.2.3(h) *K_M for sandwich panel with ends clamped and sides simply supported and isotropic core ($G_{cb} = G_{ca}$).*

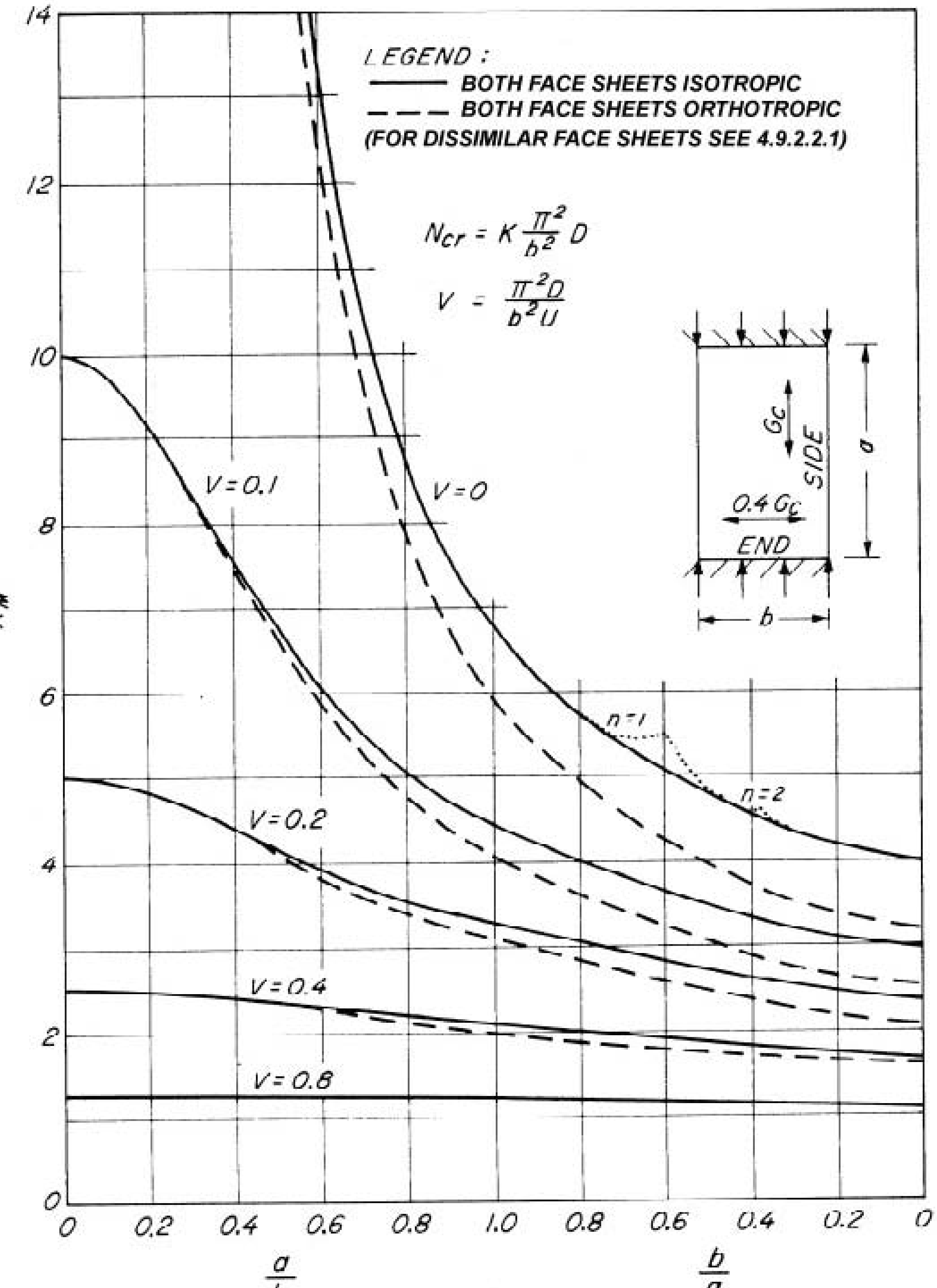

FIGURE 4.9.2.3(i) *K_M for sandwich panel with ends clamped and sides simply supported and orthotropic core ($G_{cb} = 0.4G_{ca}$).*

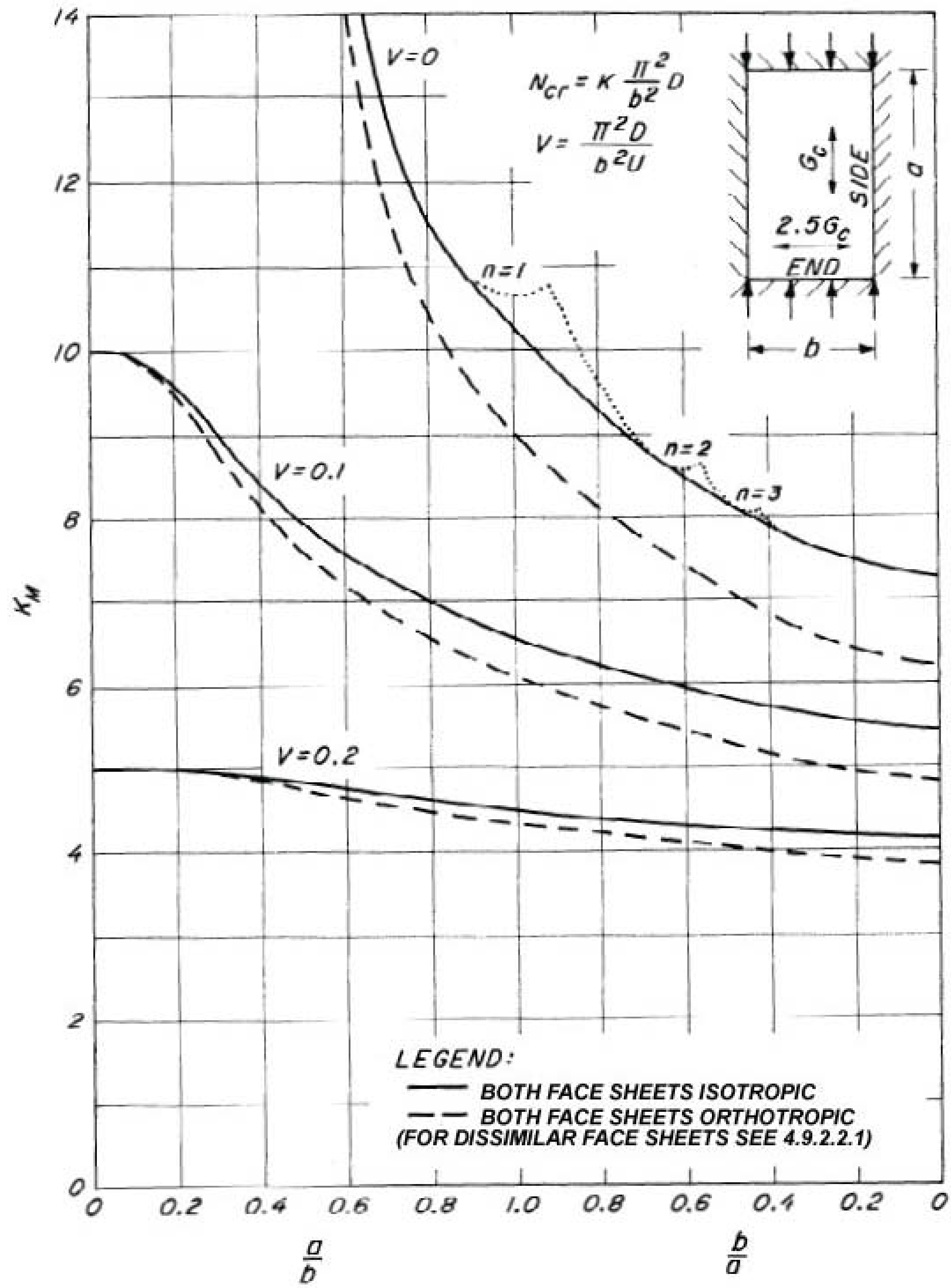

FIGURE 4.9.2.3(j) *K_M for sandwich panel with ends and sides clamped and orthotropic core ($G_{cb} = 2.5G_{ca}$).*

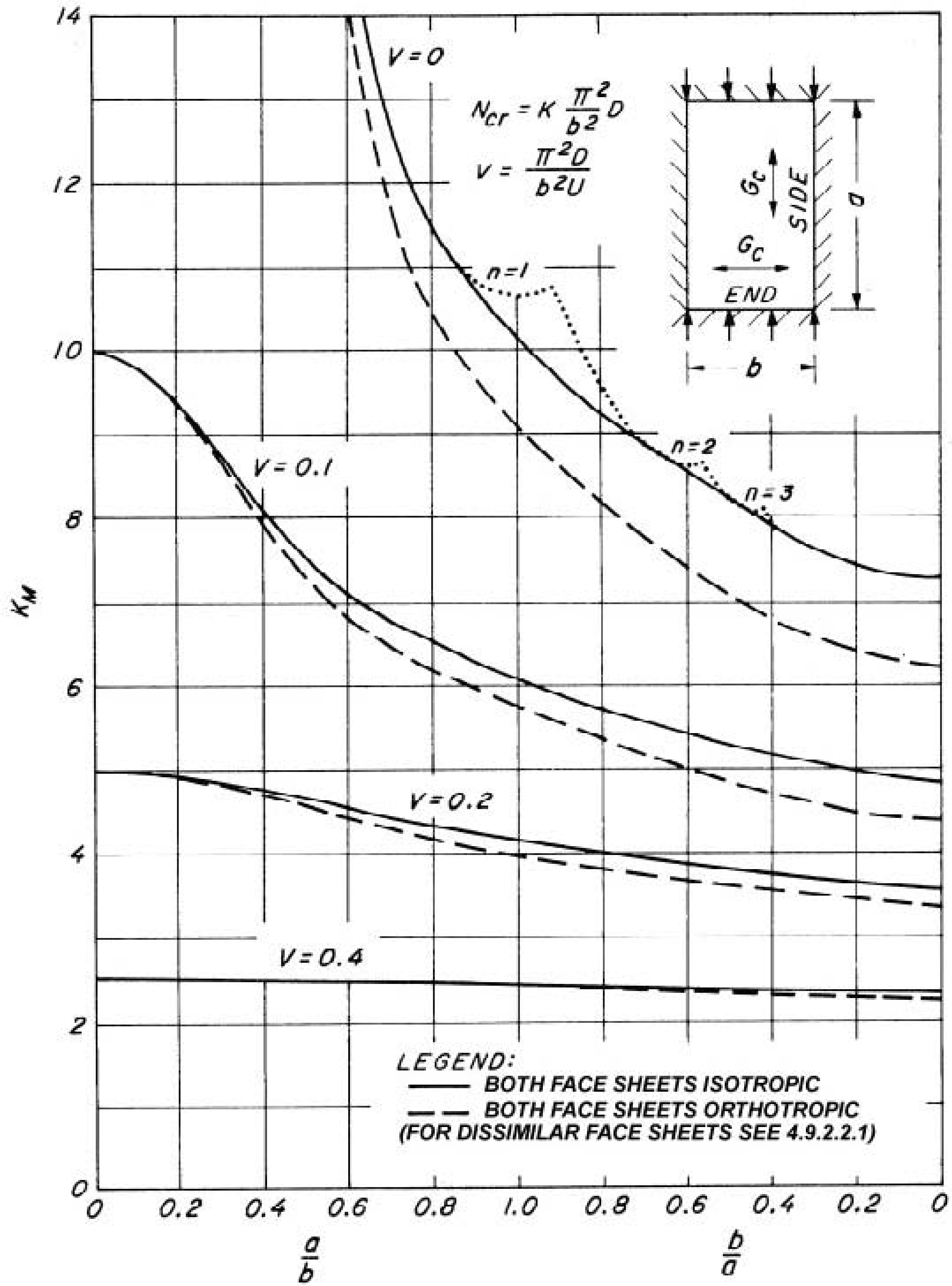

FIGURE 4.9.2.3(k) *K_M for sandwich panel with ends and sides clamped and isotropic core ($G_{cb} = G_{ca}$).*

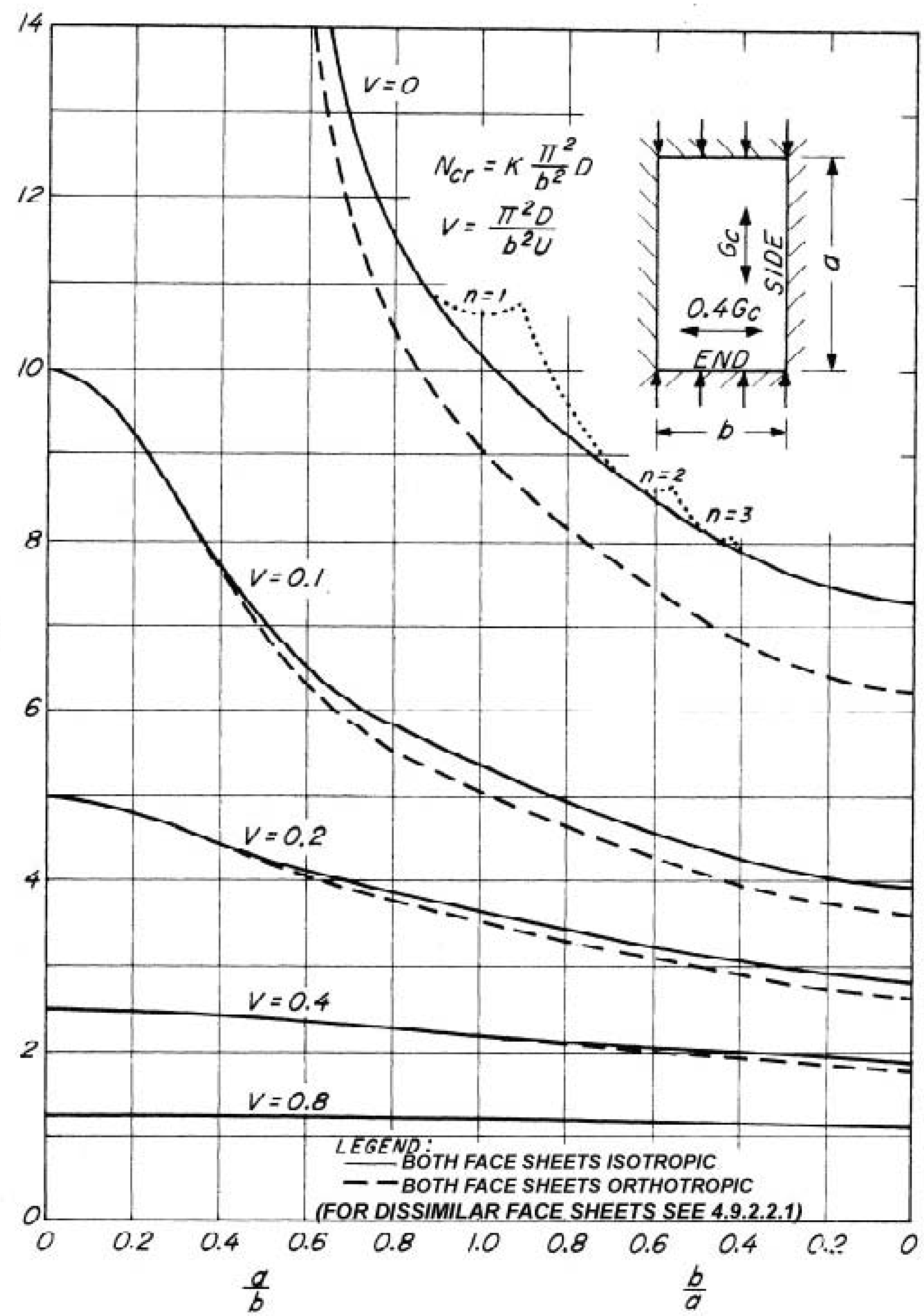

FIGURE 4.9.2.3(l) *K_M for sandwich panel with ends and sides clamped and orthotropic core ($G_{cb} = 0.4G_{ca}$).*

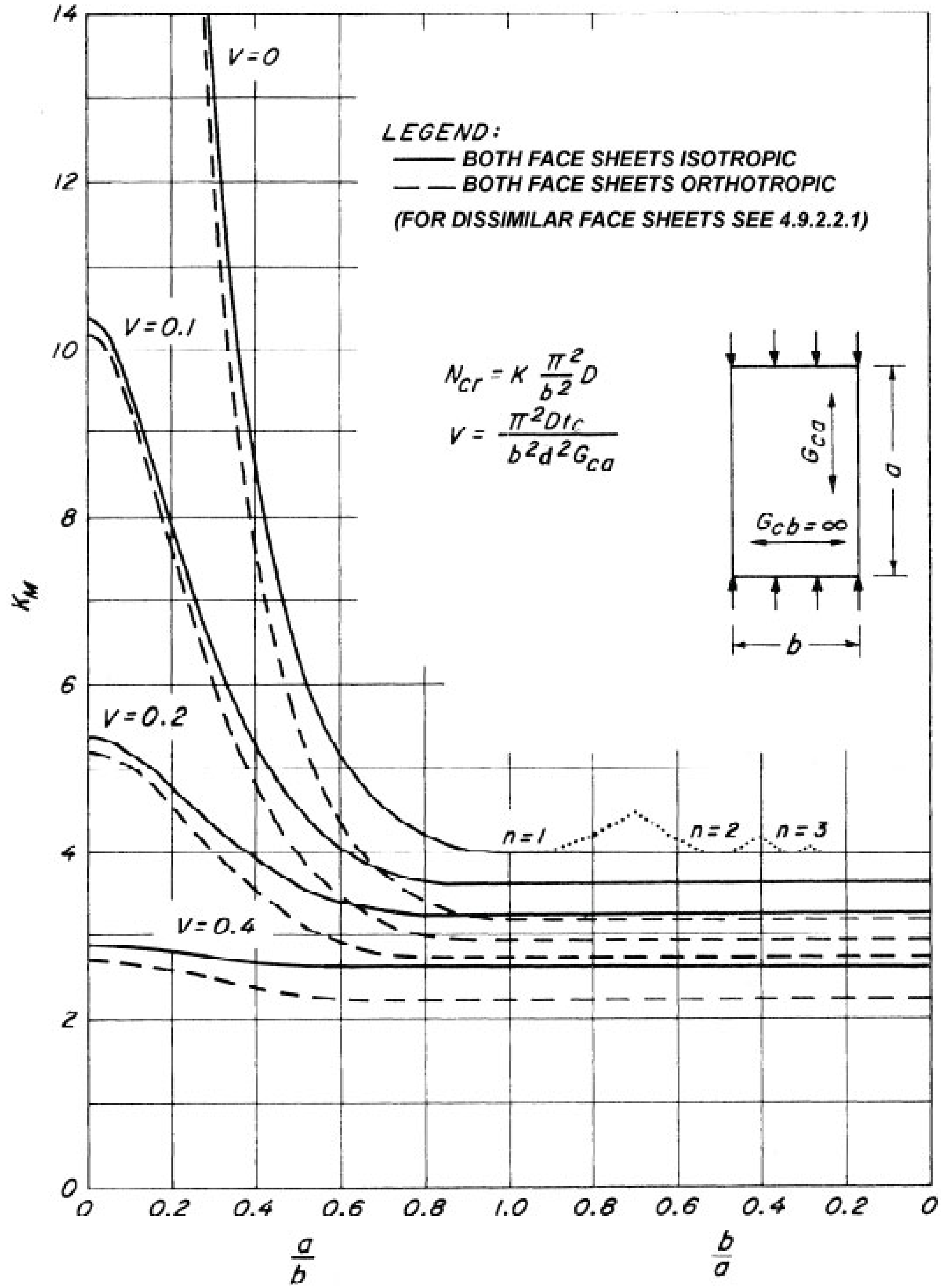

FIGURE 4.9.2.3(m) *K_M for sandwich panel having a corrugated core. Core corrugation flutes perpendicular to load direction.*

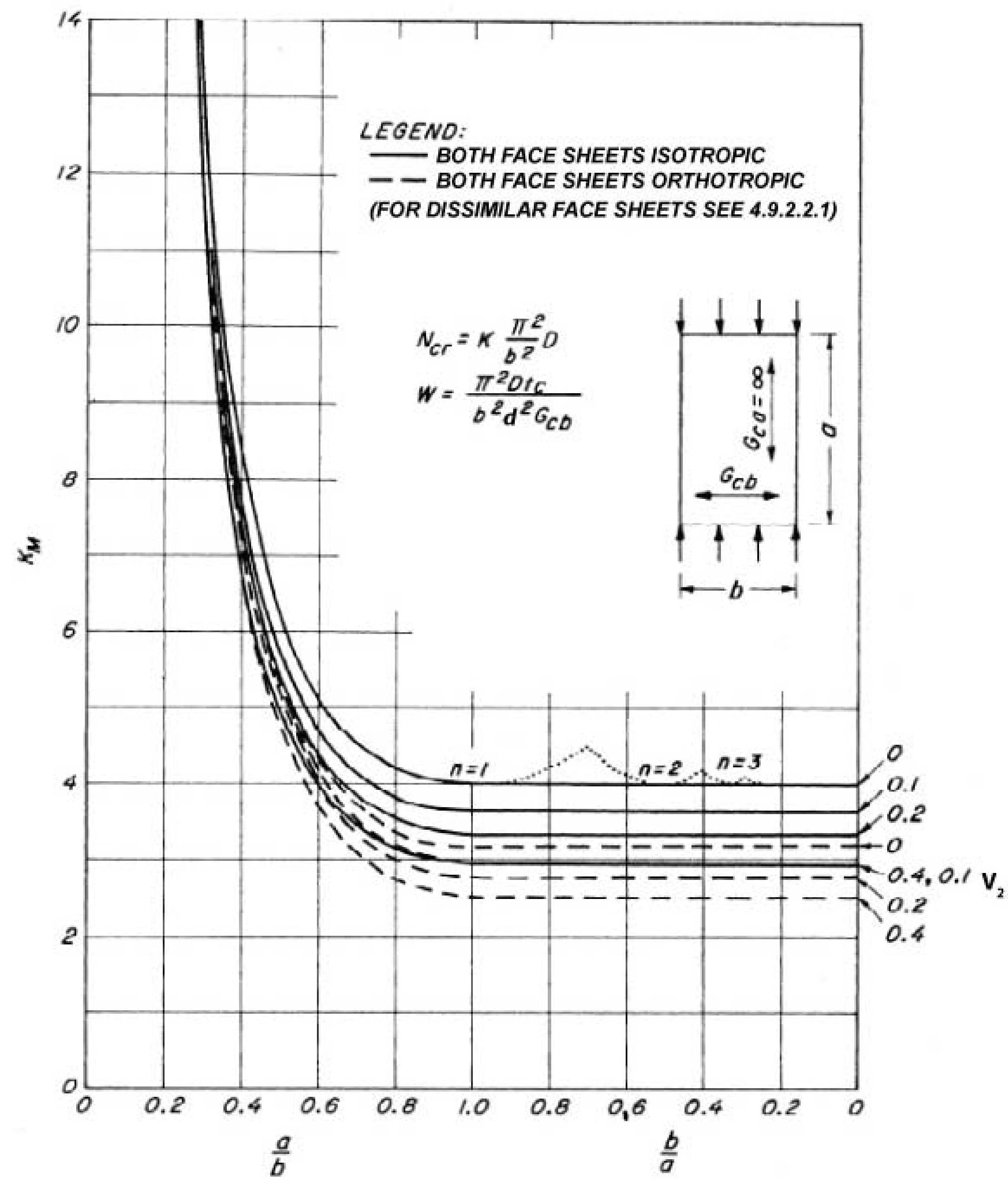

FIGURE 4.9.2.3(n) *K_M for simply supported sandwich panel having a corrugated core. Core corrugation flutes parallel to load direction.*

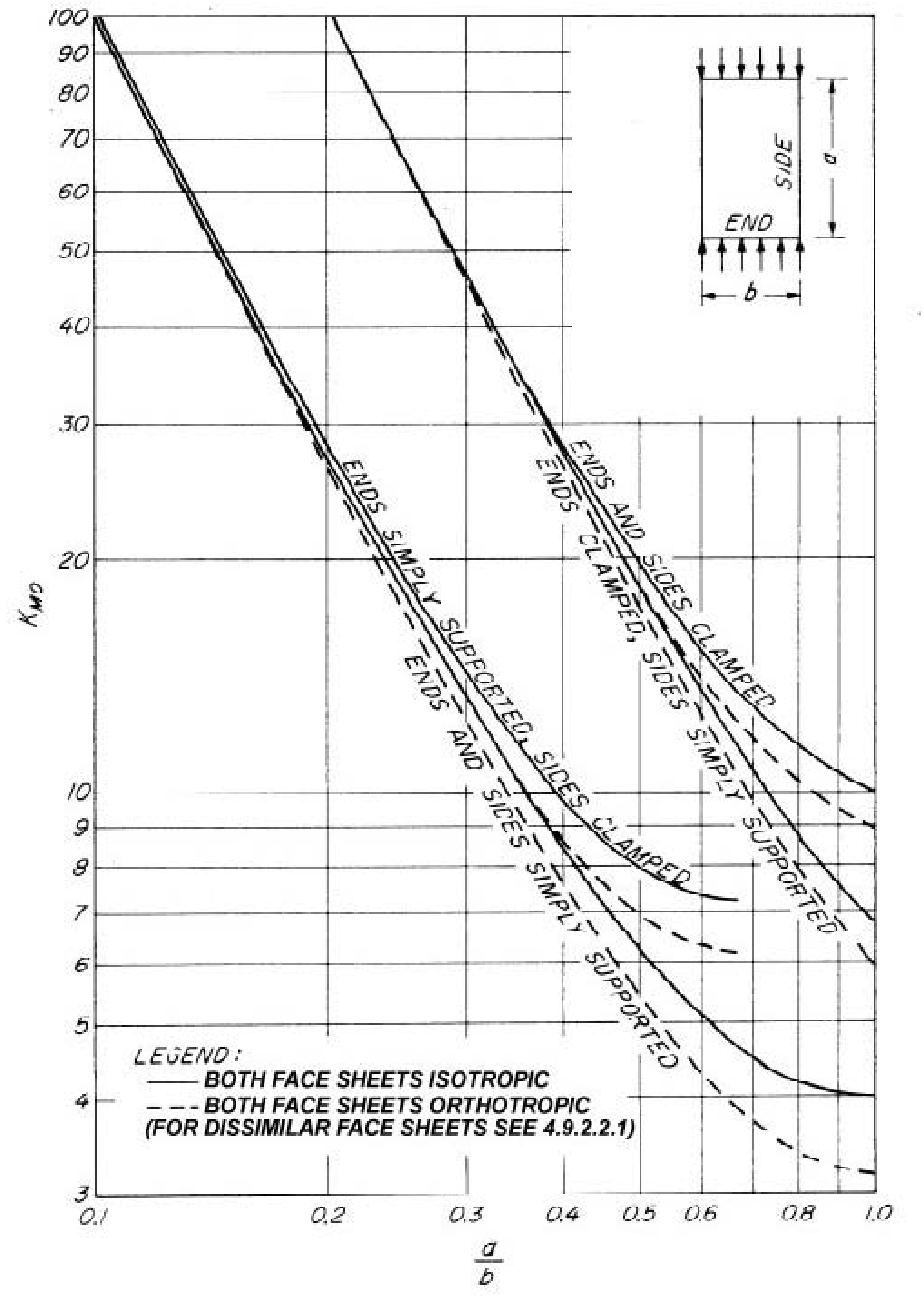

FIGURE 4.9.2.3(o) *Values of K_{MO} for sandwich panels in edgewise compression.*

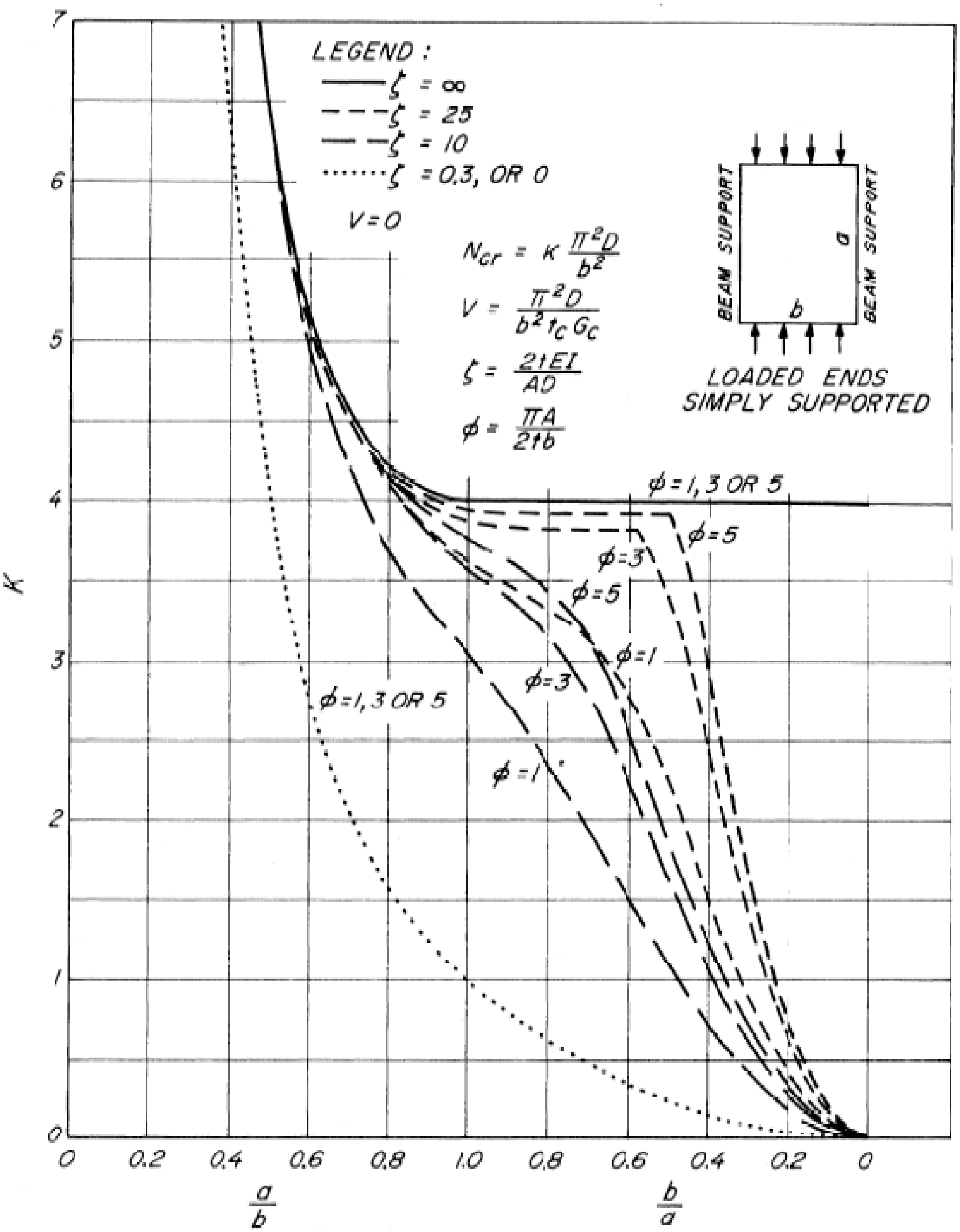

FIGURE 4.9.2.3(p) *Edgewise compressive buckling coefficient, K, for flat, isotropic, sandwich panels with loaded ends simply supported and sides supported by beams; ν = 0.3, V = 0.*

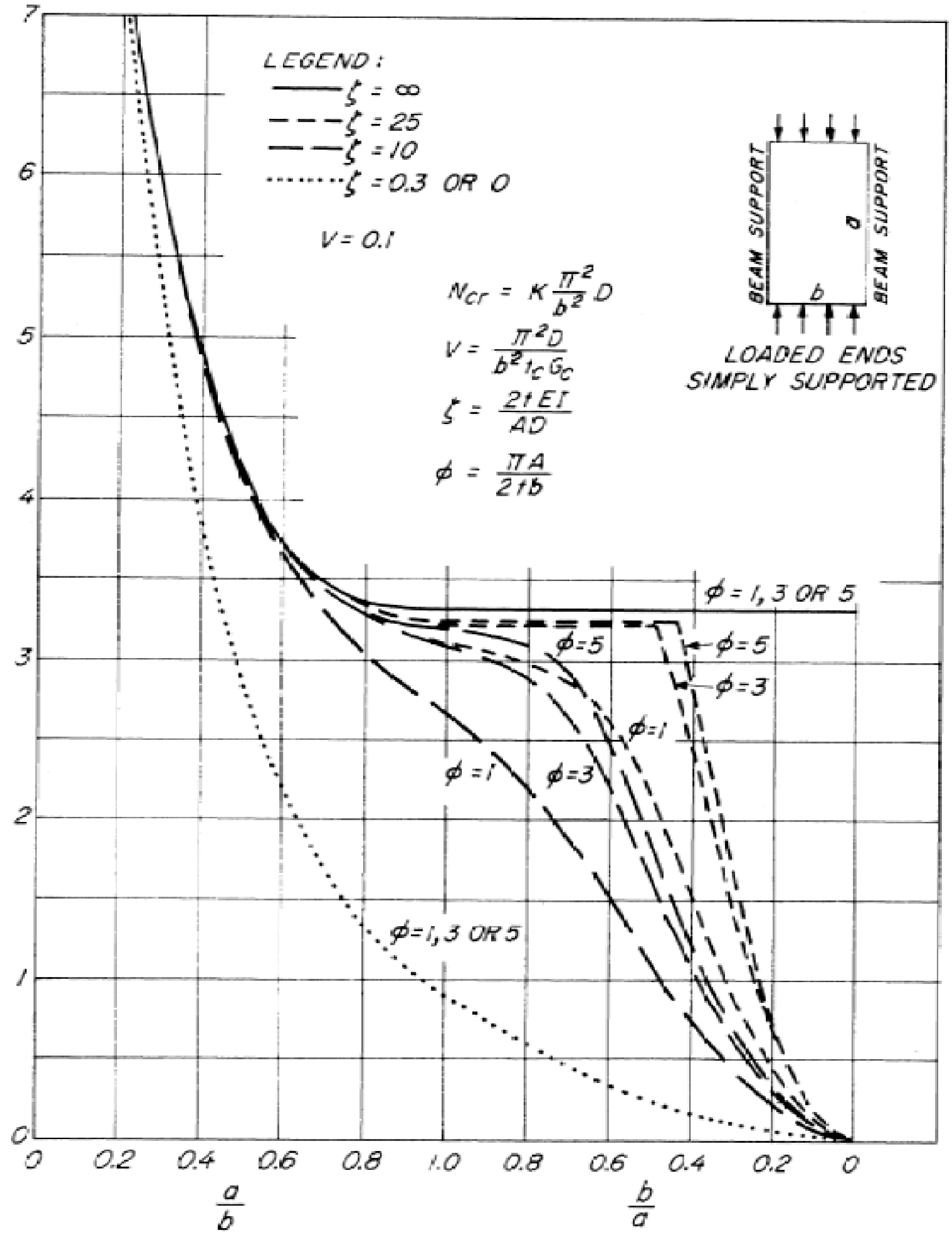

FIGURE 4.9.2.3(q) *Edgewise compressive buckling coefficient, K, for flat, isotropic, sandwich panels with loaded ends simply supported and sides supported by beams; ν = 0.3, V = 0.1.*

4.9.3 Design of flat rectangular sandwich panels under edgewise shear load

Assuming that a design begins with chosen design stresses and a given load to transmit, a flat rectangular panel of sandwich construction under edgewise shear load shall be designed to comply with the basic design principles summarized in Section 4.2.1. These conditions must be met. This section addresses global buckling. The other failure modes listed in Section 4.4 should be checked separately.

Overall buckling of the sandwich, or local instabilities such as dimpling or wrinkling of the face sheets, may lead to total collapse of the panel. Detailed procedures giving theoretical equations and graphs for determining dimensions of the face sheets and core, as well as necessary core properties, are given in the following paragraphs. Double equations are given, one equation for sandwich with face sheets of different materials and thicknesses and another equation for sandwich with each face sheet of the same material and thickness. Face sheet modulus of elasticity, E', shear modulus, G', and stress values, F_s, shall be shear loading values at the conditions of use; that is, if application is at elevated temperature, then face sheet properties at elevated temperature shall be used in design. The face sheet shear modulus or modulus of elasticity is the effective value at the face sheet stress. If this stress is beyond the proportional limit value, an appropriate tangent, reduced, or modified shear modulus of elasticity shall be used (Reference 4.6.6.1(c)).

4.9.3.1 Determining face sheet thickness

$$t_{UPR}\,F_{sUPR} + t_{LWR}\,F_{sLWR} = N_s \qquad 4.9.3.1$$

$$t = \frac{N_s}{2F_s}\text{ (for equal face sheets)}$$

where t is face sheet thickness, F_s is chosen design face sheet compressive stress, N_s is design shear load per unit length of panel edge, and UPR, LWR are subscripts denoting upper and lower face sheets.

In determining thicknesses of face sheets for sandwich with face sheets of different materials, Equation 4.9.3.1 must be satisfied, but also the stresses $F_{s\,UPR}$ and $F_{s\,LWR}$ must be chosen so that $F_{s\,UPR}/G_{S\,UPR} = F_{sLWR}/G_{S\,LWR}$ (where G_S is face sheet secant shear modulus), thus avoiding overstressing of either face sheet. For example, if the upper face sheet is of a material such that the ratio $F_{s\,UPR}/G_{S\,UPR}$ = 0.005 and the lower face sheet is of a material such that the ratio $F_{s\,LWR}/G_{S\,LWR}$ = 0.002, the design must be based on a ratio of 0.002, otherwise the lower face sheet will be overstressed. In order to accomplish this, the chosen design stress for the upper face sheet must be lowered. For many combinations of face sheet materials, it will be found advantageous to choose thicknesses such that $G_{UPR}\,t_{UPR} = G_{LWR}\,t_{LWR}$ or $E_{UPR}\,t_{UPR} = E_{LWR}\,t_{LWR}$.

4.9.3.2 Determining core thickness and core shear modulus

This section gives procedures for determining core thickness and core shear modulus so that overall buckling of the sandwich panel will not occur (References 4.9.3.2(a) and (b)). The load per unit panel width at which buckling of a sandwich panel will occur is given by the theoretical formula:

$$N_{s\,cr} = K_s \frac{\pi^2}{b^2} D \qquad 4.9.3.2(a)$$

where D is sandwich bending stiffness. This formula, solved for the face sheet stress, becomes

$$F_{sUPR} = \frac{\pi^2 K_s E'_{UPR}}{\lambda} \frac{E'_{UPR} t_{UPR} E'_{LWR} t_{LWR}}{\left(E'_{UPR} t_{UPR} + E'_{LWR} t_{LWR}\right)^2} \left(\frac{d}{b}\right)^2 \qquad 4.9.3.2(b)$$

$$F_{sLWR} = \frac{\pi^2 K_s E'_{LWR}}{\lambda} \frac{E'_{UPR} t_{UPR} E'_{LWR} t_{LWR}}{\left(E'_{UPR} t_{UPR} + E'_{LWR} t_{LWR}\right)^2} \left(\frac{d}{b}\right)^2$$

$$F_s = \frac{\pi^2 K_s E'}{4\lambda} \left(\frac{d}{b}\right)^2 \text{ (for equal face sheets)}$$

where E' is effective compressive modulus of elasticity of face sheet at stress F_s, $\lambda = 1 - \nu^2$, ν is Poisson's ratio of face sheets (in Equation 4.9.3.2(b) it is assumed that $\nu = \nu_{UPR} = \nu_{LWR}$), d is distance between face sheet centroids, b is length of loaded panel edge, $K_s = K_F + K_M$, K_F is a theoretical coefficient dependent on face sheet stiffness and panel aspect ratio, and K_M is a theoretical coefficient dependent on sandwich bending and shear rigidities and panel aspect ratio. Information on calculating K_F and K_M is given in Section 4.9.3.3.

Solving for $\frac{d}{b}$ gives:

$$\frac{d}{b} = \frac{1}{\pi\sqrt{K_s}} \sqrt{\frac{\lambda F_{sUPR}}{E'_{UPR}}} \left(\frac{E'_{UPR} t_{UPR} + E'_{LWR} t_{LWR}}{\sqrt{E'_{UPR} t_{UPR} E'_{LWR} t_{LWR}}}\right) \qquad 4.9.3.2(c)$$

$$\frac{d}{b} = \frac{1}{\pi\sqrt{K_s}} \sqrt{\frac{\lambda F_{sLWR}}{E'_{LWR}}} \left(\frac{E'_{UPR} t_{UPR} + E'_{LWR} t_{LWR}}{\sqrt{E'_{UPR} t_{UPR} E'_{LWR} t_{LWR}}}\right)$$

$$\frac{d}{b} = \frac{2}{\pi\sqrt{K_s}} \sqrt{\frac{\lambda F_s}{E'}} \text{ (for equal face sheets)}$$

Therefore, if K_s is known, Equation 4.9.3.2(c) can be solved directly to obtain d because all other quantities are known. After d is obtained, the core thickness, t_C, is computed from the equations

$$t_c = d - \frac{t_{UPR} + t_{LWR}}{2} \qquad 4.9.3.2(d)$$

$$t_c = d - t \text{ (for equal face sheets)}$$

As a first approximation, it will be assumed that $K_F = 0$, hence $K_s = K_M$. Values of K_M depend upon the bending and shear rigidities of the sandwich as incorporated in the parameter

$$V = \frac{\pi^2 D}{b^2 U} \qquad 4.9.3.2(e)$$

which can be written as:

$$V = \frac{\pi^2 t_c E'_{UPR} t_{UPR} E'_{LWR} t_{LWR}}{\left(E'_{UPR} t_{UPR} + E'_{LWR} t_{LWR}\right)\lambda b^2 G_c} \qquad 4.9.3.2(f)$$

$$V = \frac{\pi^2 t_c E' t}{2\lambda b^2 G_c} \text{ (for equal face sheets)}$$

where U is sandwich shear stiffness, G_C is the core shear modulus associated with the axes parallel to panel side of length a, and perpendicular to the plane of the panel. As values of core shear modulus decrease, values of V increase and values of K_M gradually decrease.

For sandwich with corrugated core having corrugation flutes parallel to direction of loading, the parameter V is replaced by the parameter, V_2

$$V_2 = \frac{\pi^2 t_c D}{b^2 d^2 G_{cb}} \qquad 4.9.3.2(g)$$

which can be written as:

$$V_2 = \frac{\pi^2 t_c E'_{UPR} t_{UPR} E'_{LWR} t_{LWR}}{\left(E'_{UPR} t_{UPR} + E'_{LWR} t_{LWR}\right)\lambda b^2 G_{cb}} \qquad 4.9.3.2(h)$$

$$V_2 = \frac{\pi^2 t_c E' t}{2\lambda b^2 G_{cb}} \quad \text{(for equal face sheets)}$$

where G_{cb} is the core shear modulus associated with the axes parallel to panel side of length b and perpendicular to the plane of the panel (Reference 4.9.3.2(a)).

4.9.3.2.1 Determining minimum value of d

A minimum value of d required will be determined by assuming $V = 0$ or $V_2 = 0$ for a first approximation. The value of d is minimum because $V = 0$ or $V_2 = 0$ only if the core shear modulus is infinite; for any actual core the shear modulus is not infinite, hence a thicker core must be used. The curves for $V = 0$ and $V_2 = 0$ in the charts of Figures 4.9.3.2.1(a) through (e) give minimum values of d for sandwich panels with isotropic, orthotropic, or corrugated cores with simply supported edges. Panels with clamped edges are not included in the design charts because truly clamped edges are not actually attainable. Approximate curves for checking clamped sandwich are included and discussed in Section 4.9.3.3.

The charts of Figures 4.9.3.2.1(a) through (c) are applicable to simply supported sandwich with isotropic face sheets for which $\alpha = 1.0$, $\beta = 1.0$, $\gamma = 0.375$ and to sandwich with orthotropic face sheets such as glass fabric laminates for which $\alpha = 1.0$, $\beta = 0.6$, $\gamma = 0.2$[1].

The constants α, β, and γ depend upon elastic properties of the face sheets as follows:

$$\alpha = \sqrt{\frac{E'_b}{E'_a}}; \quad \beta = \alpha\nu_{ab} + 2\gamma; \quad \gamma = \frac{\lambda G'_{ba}}{\sqrt{E'_a E'_b}} \qquad 4.9.3.2.1(a)$$

where E'_a and E'_b are the moduli of elasticity parallel to sides a and b, respectively, G_{ba}' is the face sheet shear modulus associated with those directions, ν_{ab} is the Poisson's ratio of the contraction in the b direction to extension in the a direction due to a tensile stress in the a direction, ν_{ba} is similarly defined, and $\lambda = 1 - \nu_{ab}\nu_{ba}$. For isotropic face sheets, it was assumed that $\nu = 0.25$. For orthotropic face sheets, it was assumed that $\nu_{ab} = \nu_{ba} = 0.2$, $E'_a = E'_b$, and $G_{ab}' = 0.21\, E_a$.

Parameters needed for use of the charts of Figure 4.9.3.2.1(a) through 4.9.3.2.1(c) are:

[1] Laminates giving these values of α, β, and γ were of polyester and epoxy laminates with glass fabrics 112, 116 ,120, 128, 162, 164, 181, 182, 183 and 184 (Reference 4.9.3.2.1).

1. Panel aspect ratio $\frac{a}{b}$ or $\frac{b}{a}$

2. Face sheet properties $\frac{\lambda F_{s\,UPR}}{E'_{UPR}}$ and $\frac{\lambda F_{s\,LWR}}{E'_{LWR}}$

3. Ratio of $E'_{LWR} t_{LWR} / E'_{UPR} t_{UPR}$

The charts of Figures 4.9.3.2.1(a) to 4.9.3.2.1(c) are also applicable to sandwich with dissimilar face sheets wherein the upper face sheet is isotropic (α_{UPR} = 1.0, β_{UPR} = 1.0, and γ_{UPR} = 0.375) and the lower face sheet is orthotropic (α_{LWR} = 1.0, β_{LWR} = 0.6, and γ_{LWR} = 0.2). For such a sandwich, linear interpolation is made between curves for sandwich with both face sheets isotropic and curves for sandwich with both face sheets orthotropic by means of the parameter

$$Q = \frac{1}{1 + \left(\frac{\lambda_{UPR}}{\lambda_{LWR}}\right)\left(\frac{t_{LWR}}{t_{UPR}}\right)\sqrt{\frac{E'_{a\,LWR}\; E'_{b\,LWR}}{E'_{a\,UPR}\; E'_{b\,UPR}}}} \qquad \text{4.9.3.2.1(b)}$$

With the above assumptions that $\alpha_{UPR} = \alpha_{LWR}$ = 1.0 and $\lambda_{UPR} = \lambda_{LWR}$

$$Q = \frac{1}{1 + E'_{UPR} t_{UPR} / E'_{LWR} t_{LWR}} \qquad \text{4.9.3.2.1(c)}$$

Values of Q = 0 correspond to sandwich with both face sheets isotropic and Q = 1 to sandwich with both face sheets orthotropic, although in both cases it is not necessary that the two face sheets have the same modulus and thickness. This is demonstrated by substitution of these Q values in the general expression for K_M (see definition of K_M in Section 4.9.3.3 and discussion in Reference 4.9.3.2(b)). Thus, for example, if Q = 1/4, interpolation is at 1/4 of the distance from the curve for both face sheets isotropic toward the curve for both face sheets orthotropic.

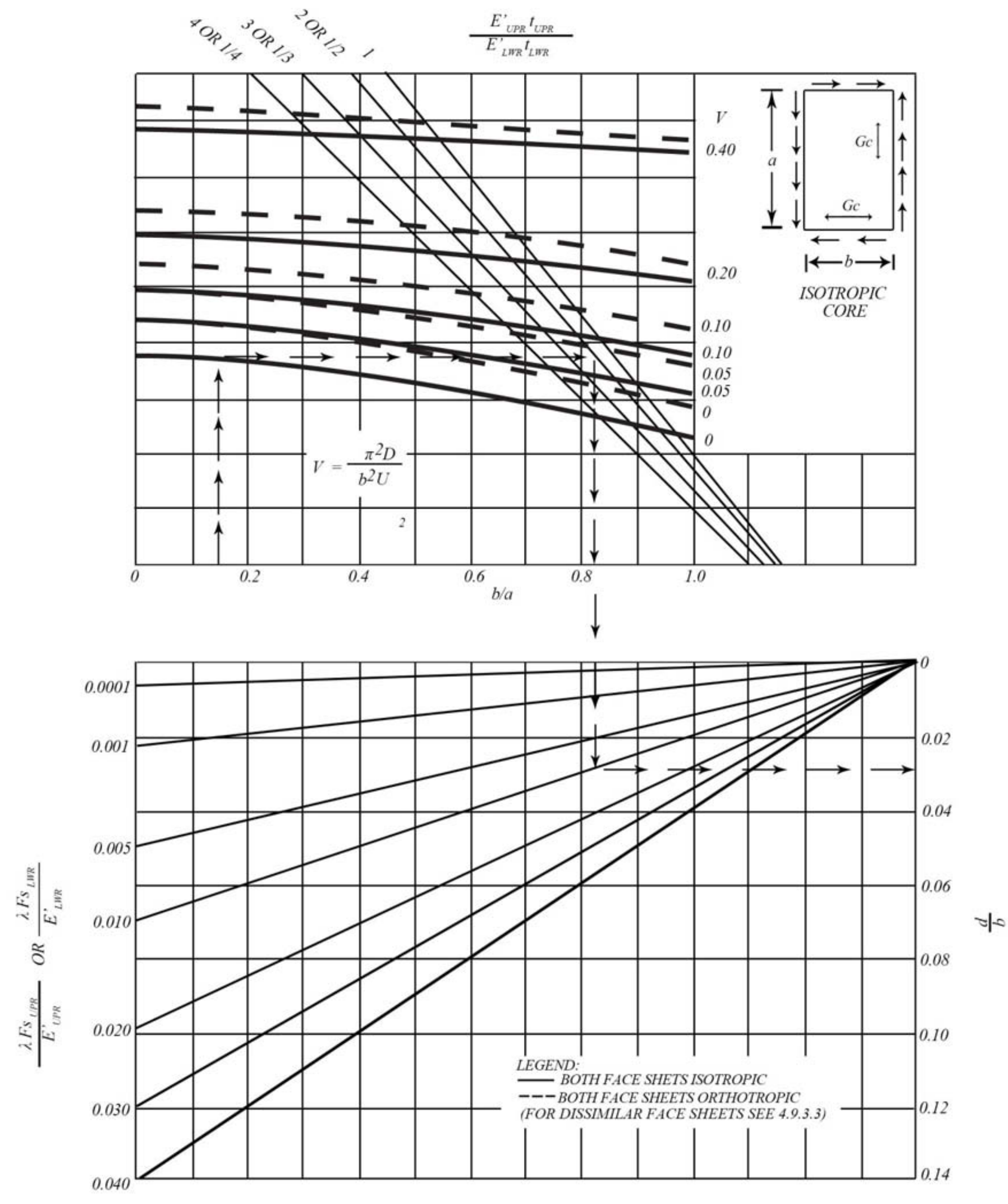

FIGURE 4.9.3.2.1(a) *Chart for determining $\frac{d}{b}$ ratio such that a simply-supported sandwich panel with isotropic core ($G_{cb} = G_{ca}$) will not buckle under edgewise shear load.*

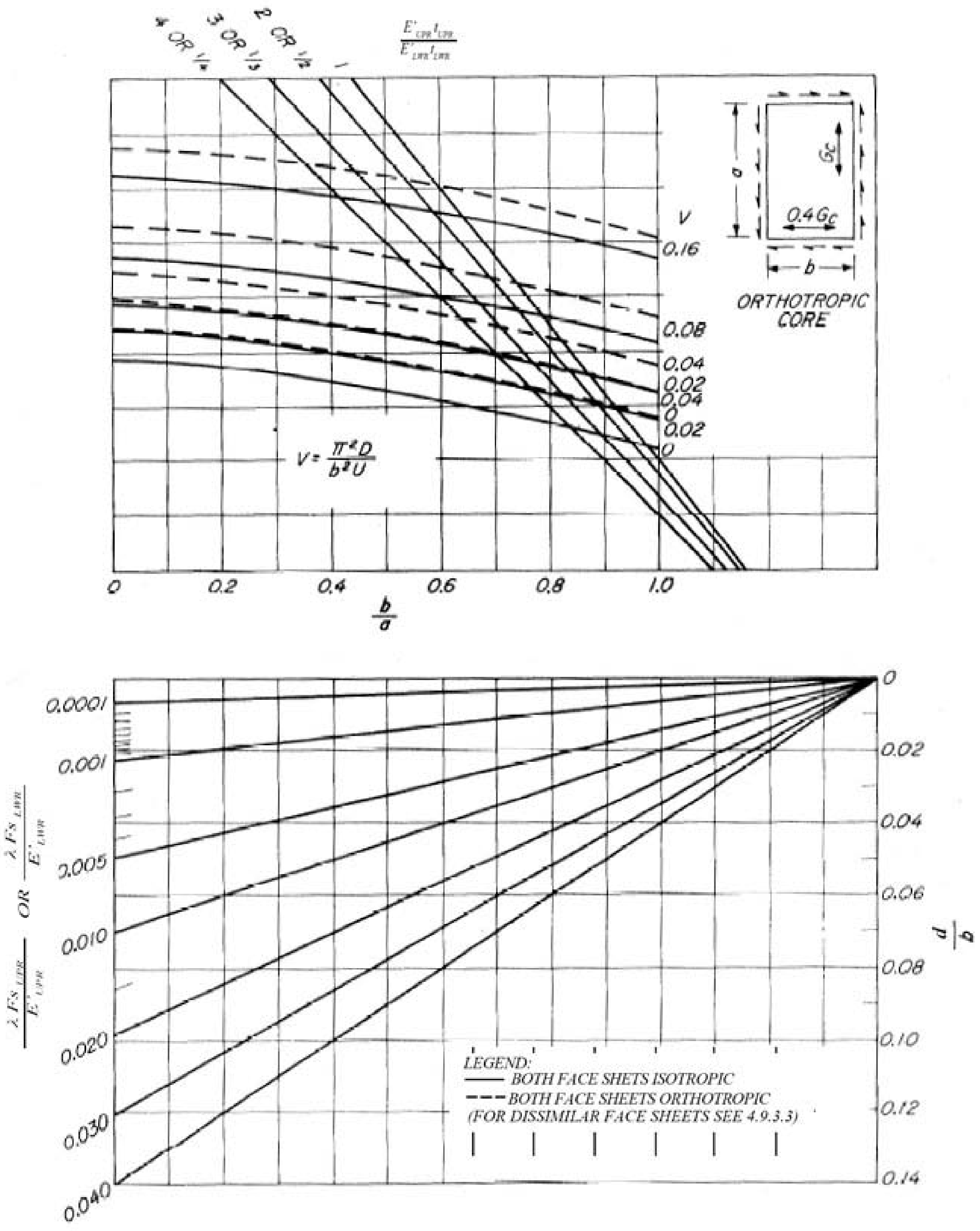

FIGURE 4.9.3.2.1(b) *Chart for determining $\frac{d}{b}$ ratio such that a simply-supported sandwich panel with orthotropic core ($G_{cb} = 0.4G_{ca}$) will not buckle under edgewise shear load.*

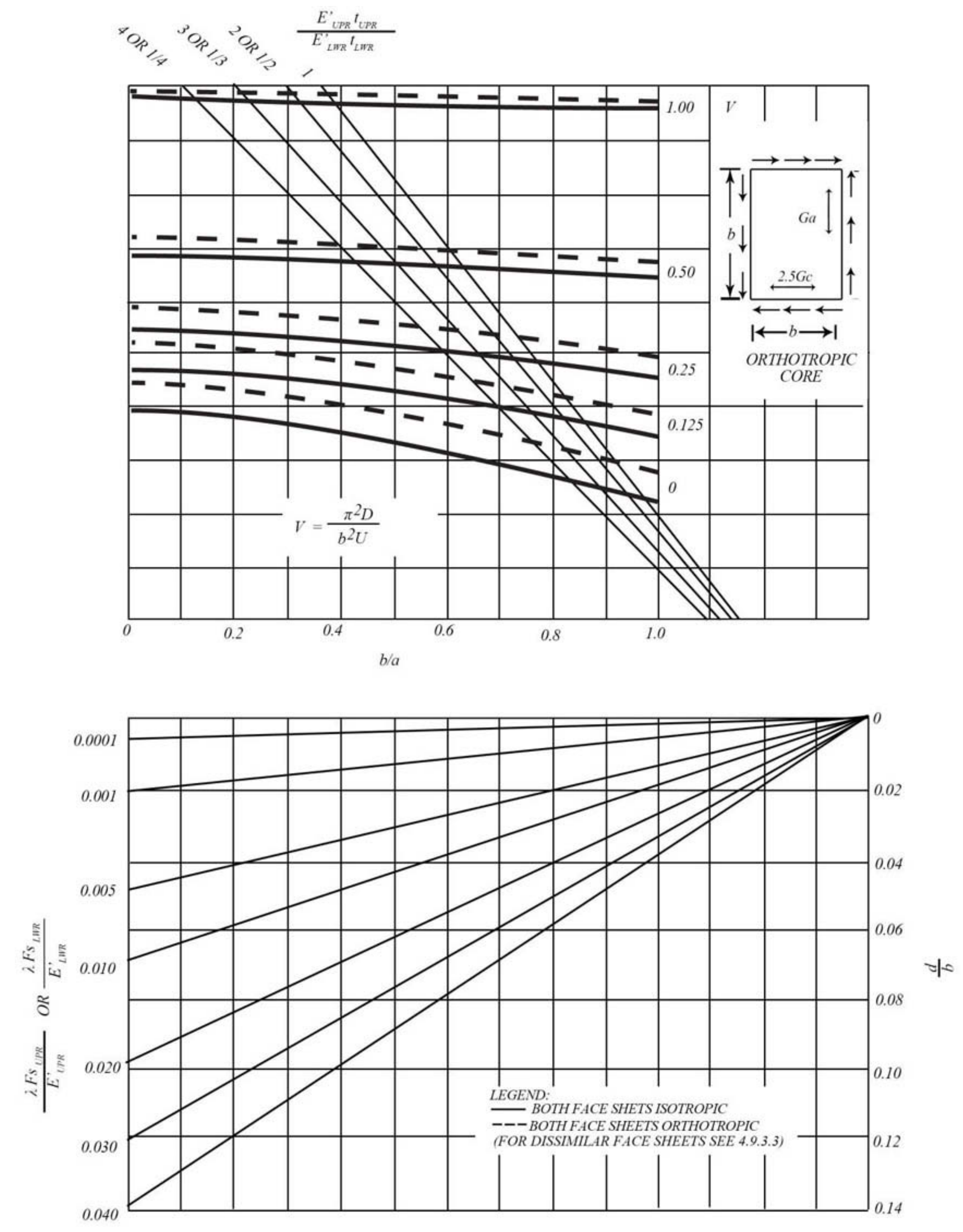

FIGURE 4.9.3.2.1(c) *Chart for determining* $\frac{d}{b}$ *ratio such that a simply-supported sandwich panel with orthotropic core* ($G_{cb} = 2.5G_{ca}$) *will not buckle under edgewise shear load.*

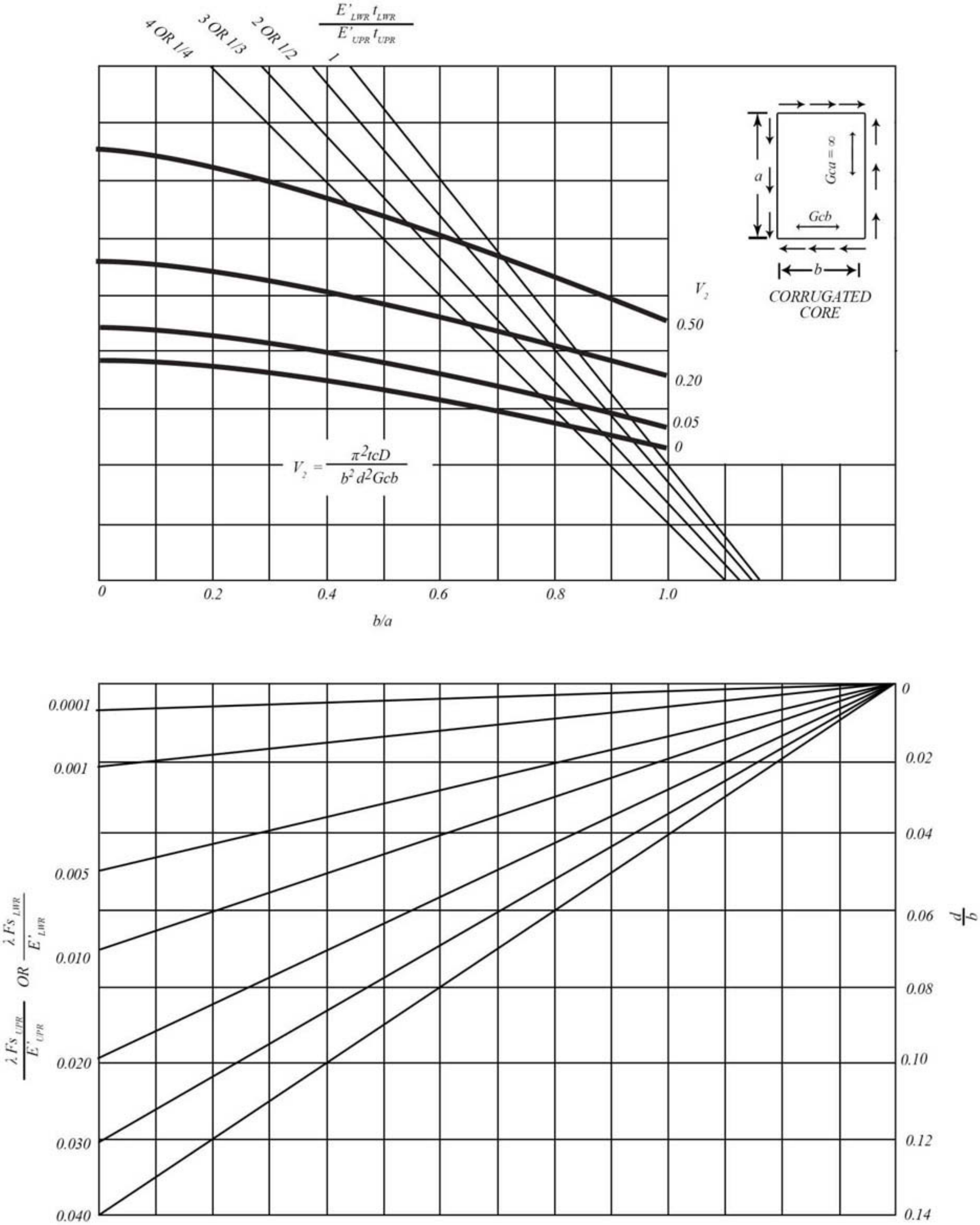

FIGURE 4.9.3.2.1(d) *Chart for determining $\frac{d}{b}$ ratio such that a simply-supported sandwich panel with isotropic face sheets and a corrugated core will not buckle under edgewise shear load; core corrugation flutes parallel to edge a.*

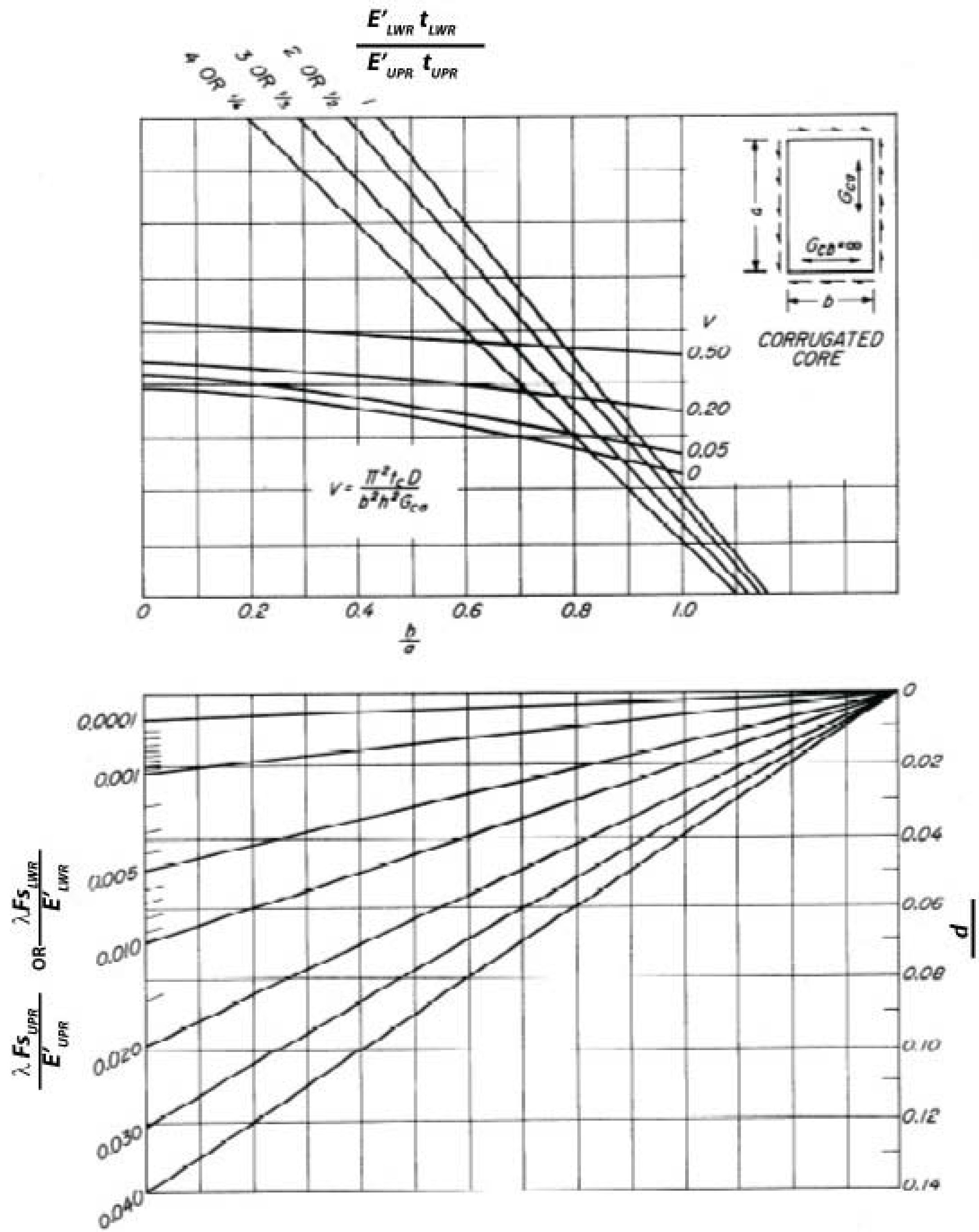

FIGURE 4.9.3.2.1(e) *Chart for determining $\frac{d}{b}$ ratio such that a simply-supported sandwich panel with isotropic face sheets and a corrugated core will not buckle under edgewise shear load; core corrugation flutes parallel to edge b.*

4.9.3.2.2 Determining actual value of d

Since actual core shear modulus values are not very large, a value of d somewhat greater than given by the curves for V = 0 and V_2 = 0 in Figures 4.9.3.2.1(a) through (e) must be used. These figures are entered with values of the panel aspect ratio and values of V or V_2 as computed by Equation 4.9.3.2(f) or 4.9.3.2(h). Figure 4.9.3.2.1(a) applies to sandwich with isotropic cores for which the core shear modulus perpendicular to the panel length is equal to the core shear modulus parallel to the panel length. Figure 4.9.3.2.1(b) applies to sandwich with cores for which the core shear modulus perpendicular to the panel length is equal to 0.40 times the core shear modulus parallel to the panel length. Figure 4.9.3.2.1(c) applies to sandwich with cores for which the core shear modulus perpendicular to the panel length is 2.50 times the core shear modulus parallel to the panel length.

NOTE: For honeycomb cores with core ribbons parallel to direction of loading, $G_c = G_{TL}$ and the shear modulus perpendicular to loading is G_{TW}. For honeycomb cores with core ribbons perpendicular to direction of loading, $G_c = G_{Tw}$ and the shear modulus perpendicular to loading is G_{TL}. If core ribbons are at an angle θ to the panel length a, $G_c = \dfrac{G_{TL}G_{TW}}{\left(G_{TL}\sin^2\theta + G_{TW}\cos^2\theta\right)}$.

Figure 4.9.3.2.1(d) applies to sandwich having isotropic face sheets and a corrugated core having the core flutes parallel to the edge length a. The parameter V_2, given in Equation 4.9.3.2(h), is used instead of V. Figure 4.9.3.2.1(e) applies to sandwich having isotropic face sheets and a corrugated core having the core flutes parallel to the edge of length b. Solution of the charts gives the ratio $\dfrac{d}{b}$.

In using the curves of Figures 4.9.3.2.1(a) through (e) for values of V or V_2 other than zero, it is necessary to iterate because V is directly proportional to the core thickness t_c. As an aid to determining t_c and G_c, Figure 4.9.3.2.2 presents a number of lines representing V for various values of G_c with V ranging from 0.01 to 2 and G_c ranging from 1,000 to 1,000,000 pounds per square inch. The following procedure is suggested:

1. Determine core thickness t_c from Figures 4.9.3.2.1(a) through (e) using a value of 0.01 for V or V_2.

2. Compute the constant relating V or V_2 to G_c.

$$\left[\frac{\pi^2 t_c E'_{UPR} t_{UPR} E'_{LWR} t_{LWR}}{\left(E'_{UPR} t_{UPR} + E'_{LWR} t_{LWR}\right)\lambda b^2}\right] \text{ or } \left[\frac{\pi^2 t_c E' t}{2\lambda b^2}\right]\left(\text{for equal face sheets}\right) = VG_c \text{ or } V_2G_c$$

3. With this constant enter Figure 4.9.3.2.2 and determine necessary G_c.

4. If the shear modulus is outside the range of values for materials available, slide up the appropriate line of Figure 4.9.3.2.2 and pick a new value of V or V_2, for a reasonable value of core shear modulus.

5. Reenter Figures 4.9.3.2.1(a) through (e) with the new value of V or V_2 and repeat previous steps 1, 2, and 3.

Charts of the type used in Figures 4.9.3.2.1(a) through (e) have not been prepared for panels with ends or sides clamped. True clamping at panel edges is never attained, particularly for sandwich constructions. It is suggested that each panel be designed as simply supported on all edges and consult Section 4.9.3.3 to estimate any possible reduction that can be made in core thickness due to edge clamping.

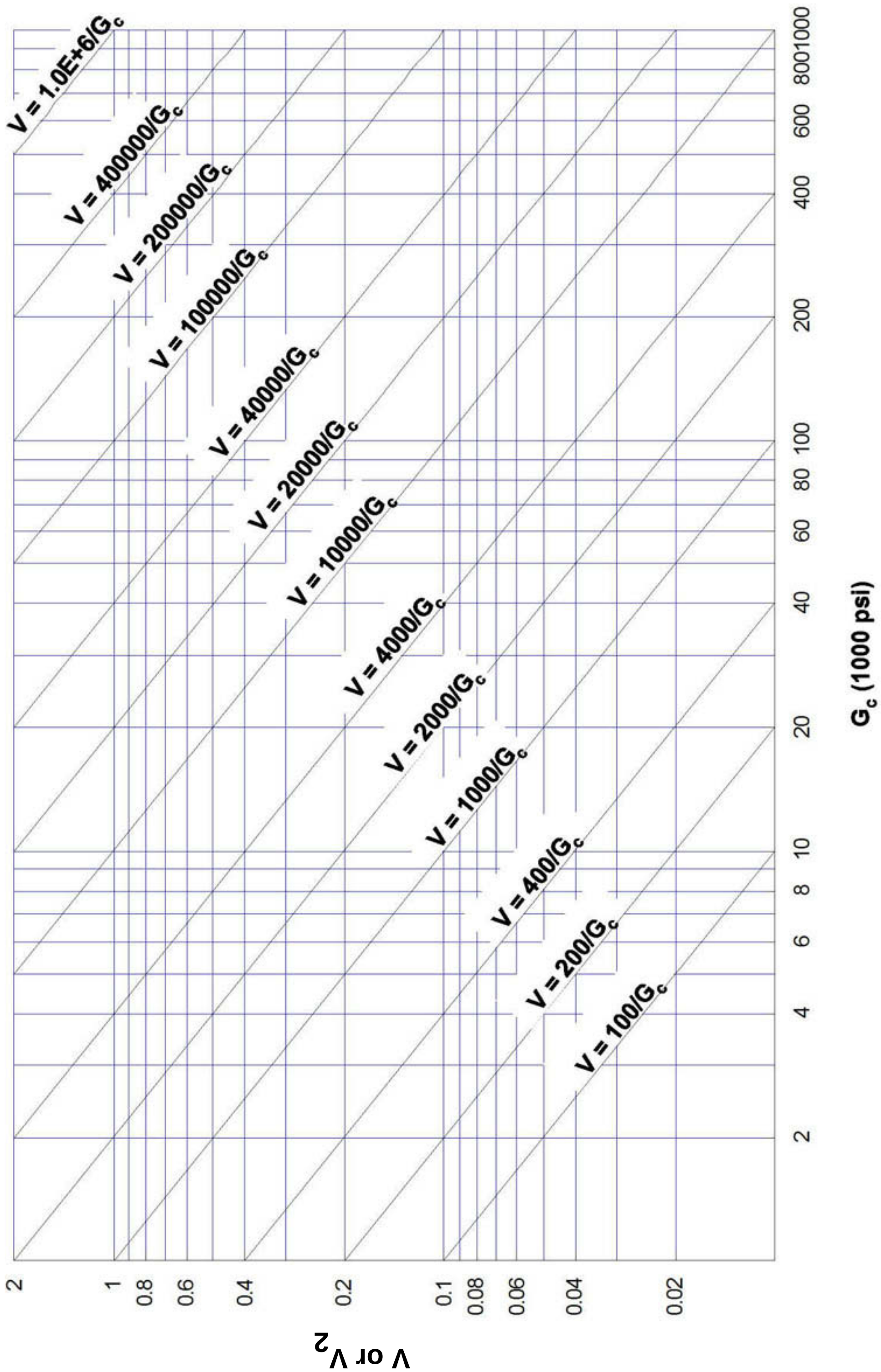

FIGURE 4.9.3.2.2 *Chart for determining V or V_2 and G_c for sandwich in edgewise shear.*

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4.9.3.3 Checking procedure for determining buckling stress, F_{cr}

The design shall be checked by using the graphs of Figures 4.9.3.3(a) through (e) to determine values of K_M for use in evaluating $K = K_F + K_M$ to substitute into Equation 4.9.3.2(b) to compute actual buckling stress, F_{cr}. The figures apply to sandwich panels with edges simply supported and isotropic or certain orthotropic face sheet cores. Curves in Figures 4.9.3.3(a) through (c) for isotropic face sheets and $V \neq 0$ were derived on the assumption the $\nu = 0.25$, and in Figures 4.9.3.3(d) and (e) that $\nu = 0.3$.

Values of K_F shall be determined by the equation

$$K_F = \frac{\left(E'_{UPR} t^3_{UPR} + E'_{LWR} t^3_{LWR}\right)\left(E'_{UPR} t_{UPR} + E'_{LWR} t_{LWR}\right) K_{MO}}{12 E'_{UPR} t_{UPR} E'_{LWR} t_{LWR} d^2} \qquad 4.9.3.3$$

$$K_F = \frac{t^2}{3d^2} K_{MO} \qquad \text{(for equal face sheets)}$$

where K_{MO} is determined from the curve for $V = 0$ or $V_2 = 0$ of Figure 4.9.3.3(a) through (e).

It should be understood that if the desired F_{cr} is above proportional limit values, the value of E' shall be an effective value, used in computing V or V_2 and F_{cr}.

If the charts do not apply because ratios of core shear moduli are far different from what is given on the charts, or it is desired to check by a more accurate analysis, the equations given in References 4.9.3.2(a) and (b) shall be used.

Graphs of K_M for sandwich panels having isotropic face sheets and isotropic or orthotropic core with all edges clamped are presented in Figures 4.9.3.3(f) through (h). Curves for clamped sandwich panels with orthotropic core are approximate because they were obtained by multiplying buckling coefficients for simply supported orthotropic sandwich by the ratio of clamped to simply supported buckling coefficients for isotropic sandwich. Values of K_M from these figures may be used to compute face sheet stress F_s from Equation 4.9.3.2(b) or to solve Equation 4.9.3.2(c) for $\frac{d}{b}$. The values of $\frac{d}{b}$ so obtained may then be compared with the values obtained for simply supported panels given by the design charts of Figures 4.9.3.2.1(a) to 4.9.3.2.1(e) to determine possible reductions in core thickness due to edge clamping.

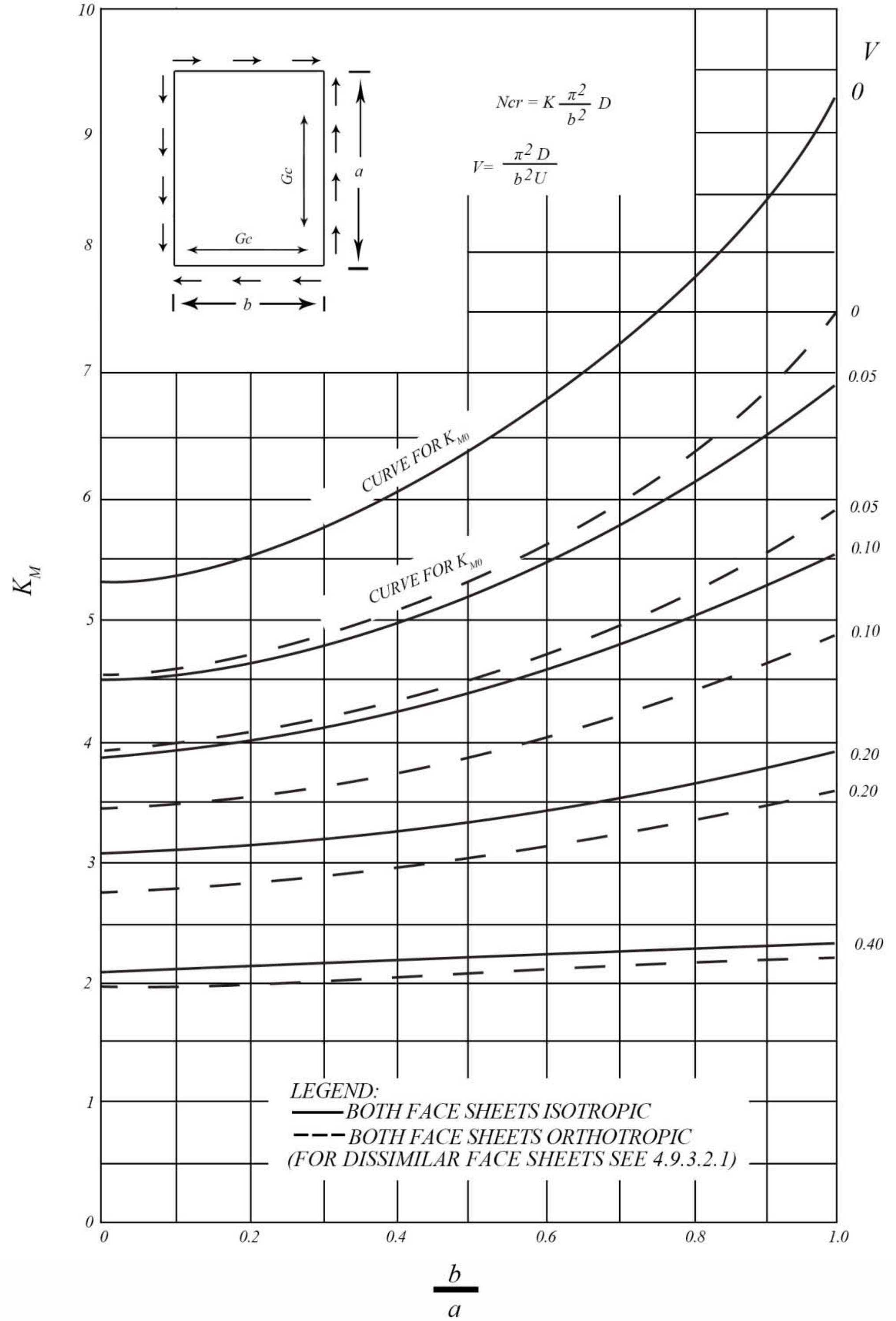

FIGURE 4.9.3.3(a) *Buckling coefficient K_M for sandwich panel with all edges simply supported and isotropic core, under edgewise shear load.*

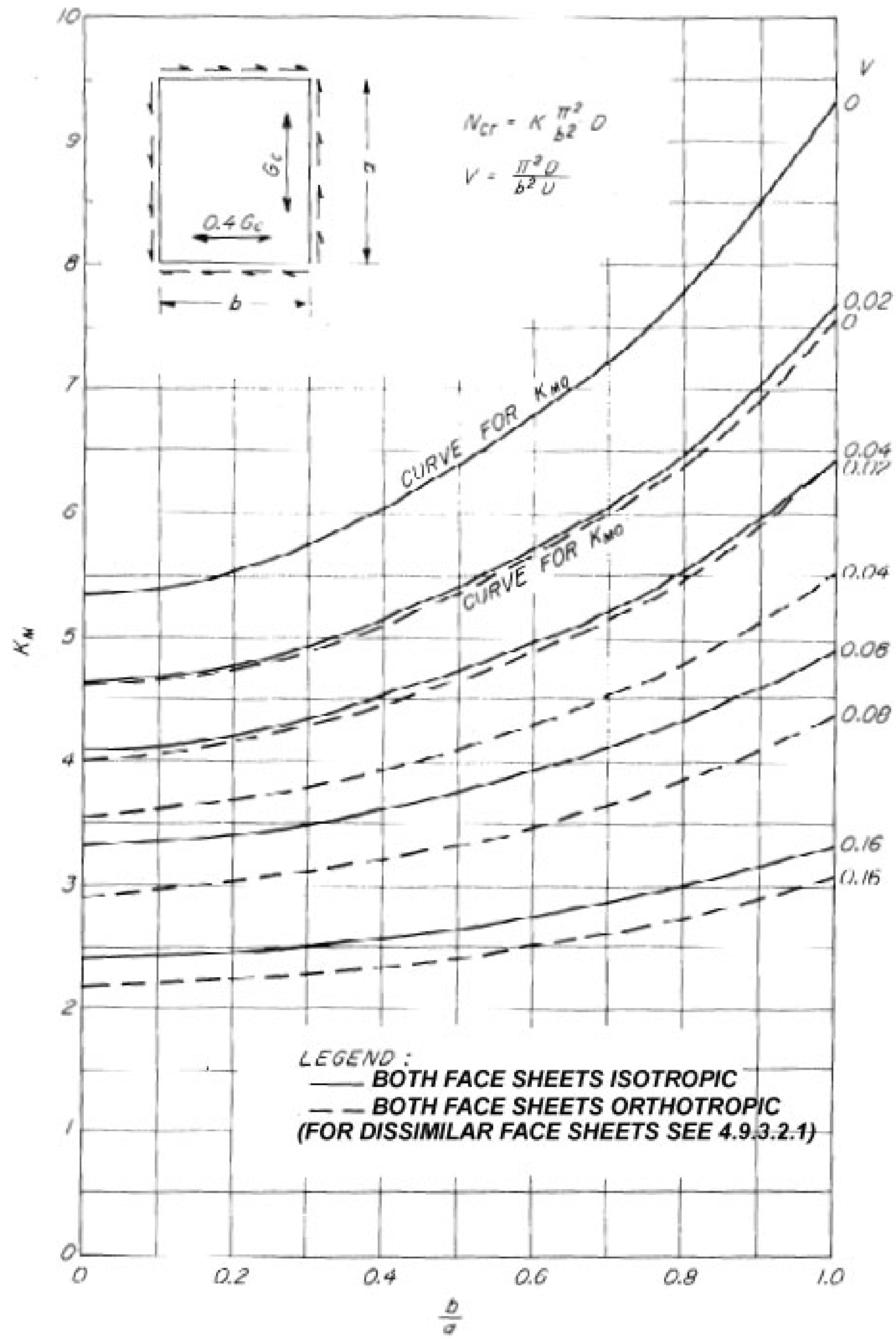

FIGURE 4.9.3.3(b) *Buckling coefficient K_M for sandwich panel with all edges simply supported and orthotropic core ($G_{cb} = 0.4G_{ca}$) , under edgewise shear load.*

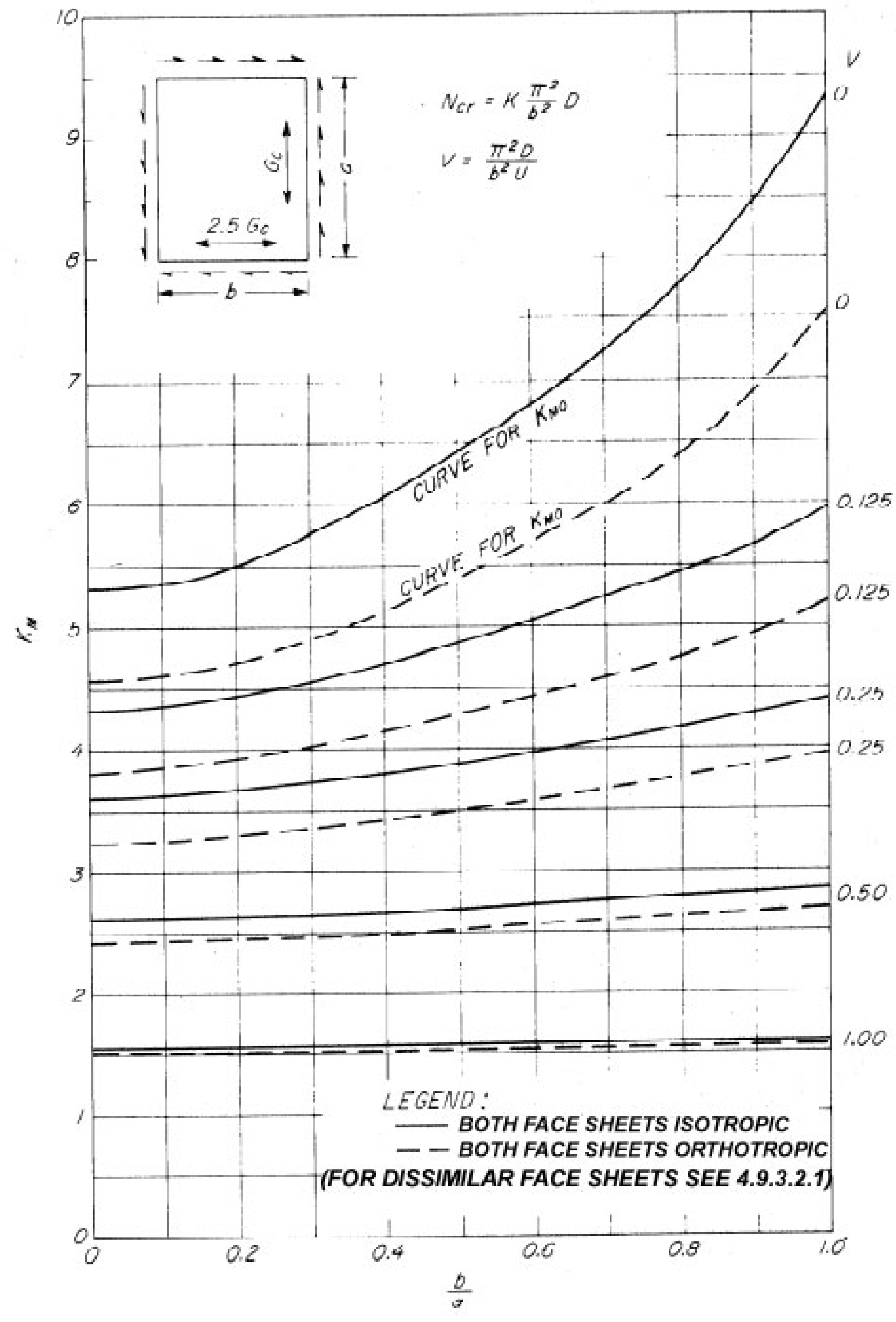

FIGURE 4.9.3.3(c) *Buckling coefficient K_M for sandwich panel with all edges simply supported and orthotropic core ($G_{cb} = 2.5G_{ca}$), under edgewise shear load.*

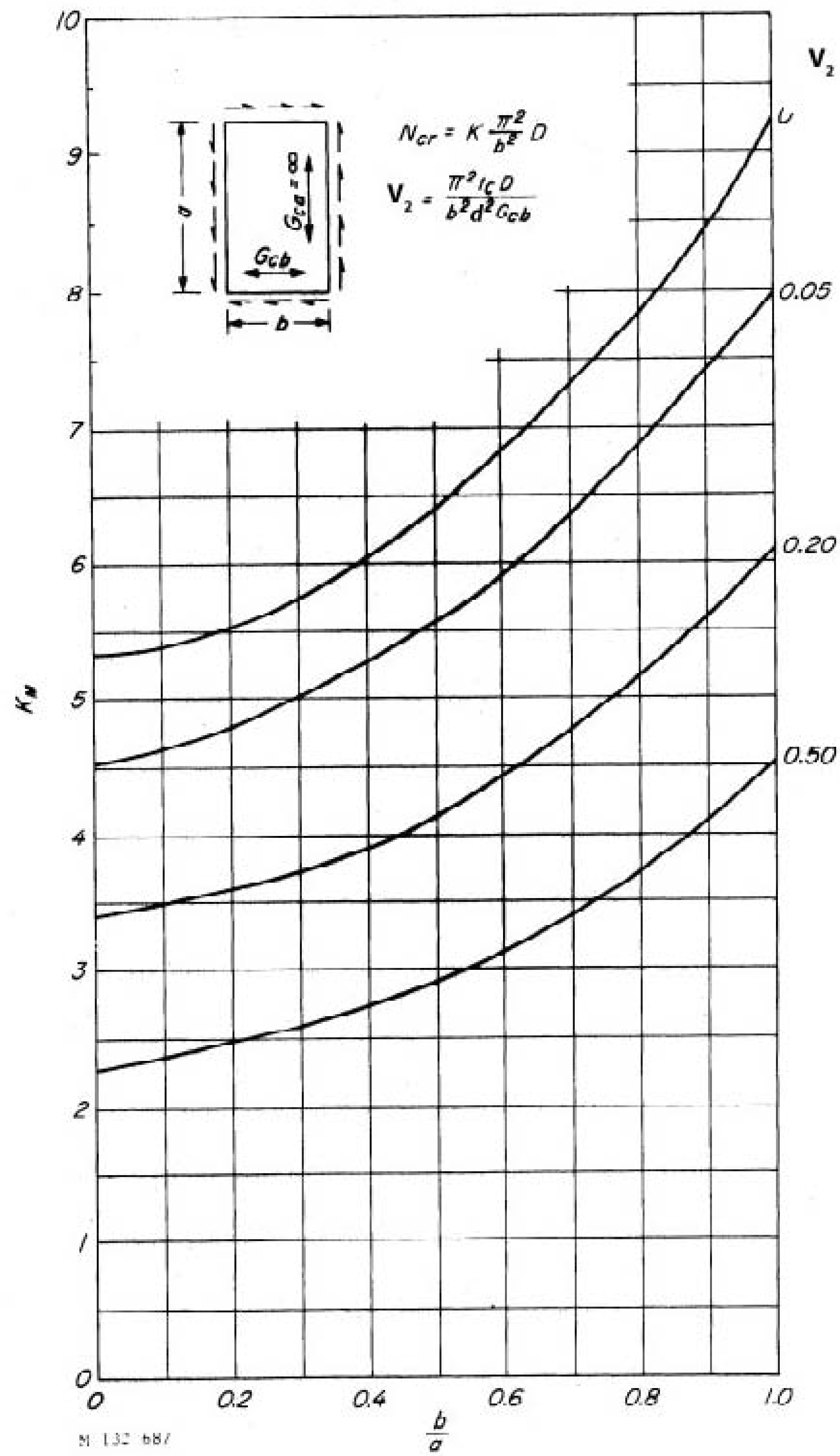

FIGURE 4.9.3.3(d) *Buckling coefficient K_M for sandwich panel with all edges simply supported, isotropic face sheets and corrugated core, under edgewise shear load. Core corrugation flutes parallel to edge a.*

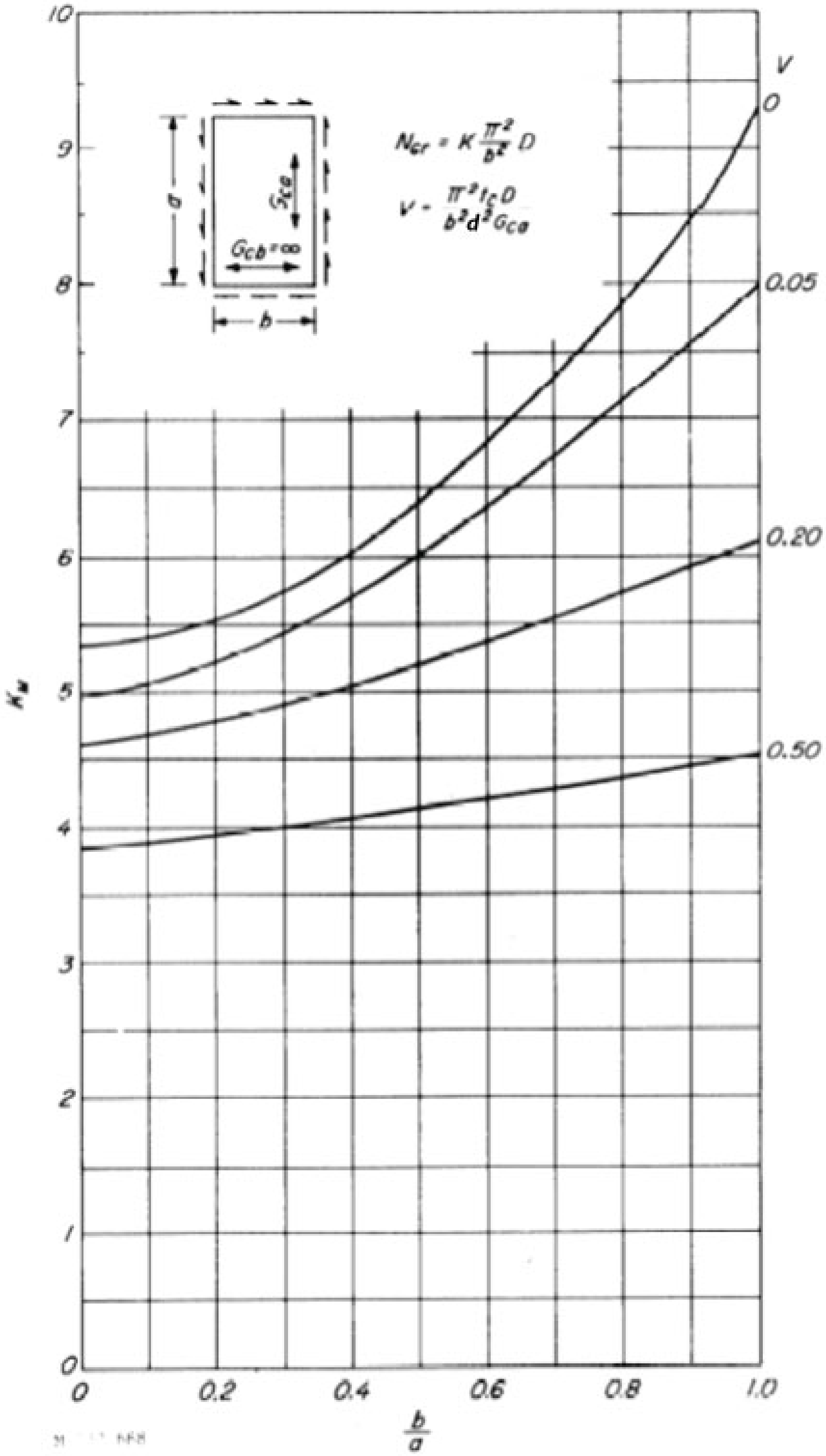

FIGURE 4.9.3.3(e) *Buckling coefficient K_M for sandwich panel with all edges simply supported, isotropic face sheets and corrugated core, under edgewise shear load. Core corrugation flutes parallel to edge b.*

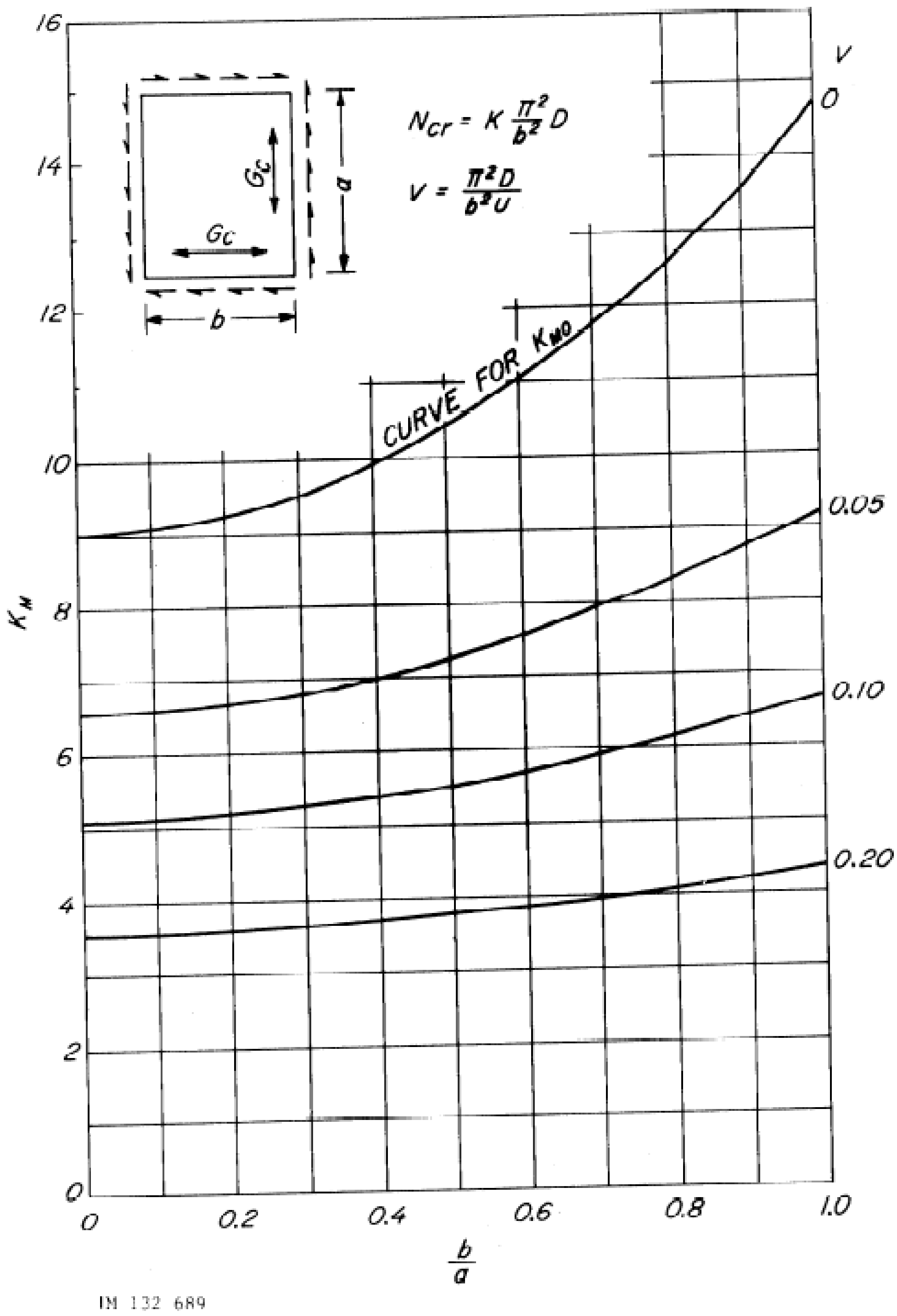

FIGURE 4.9.3.3(f) *Buckling coefficient K_M for sandwich panel with all edges clamped, isotropic face sheets and isotropic core, under edgewise shear load.*

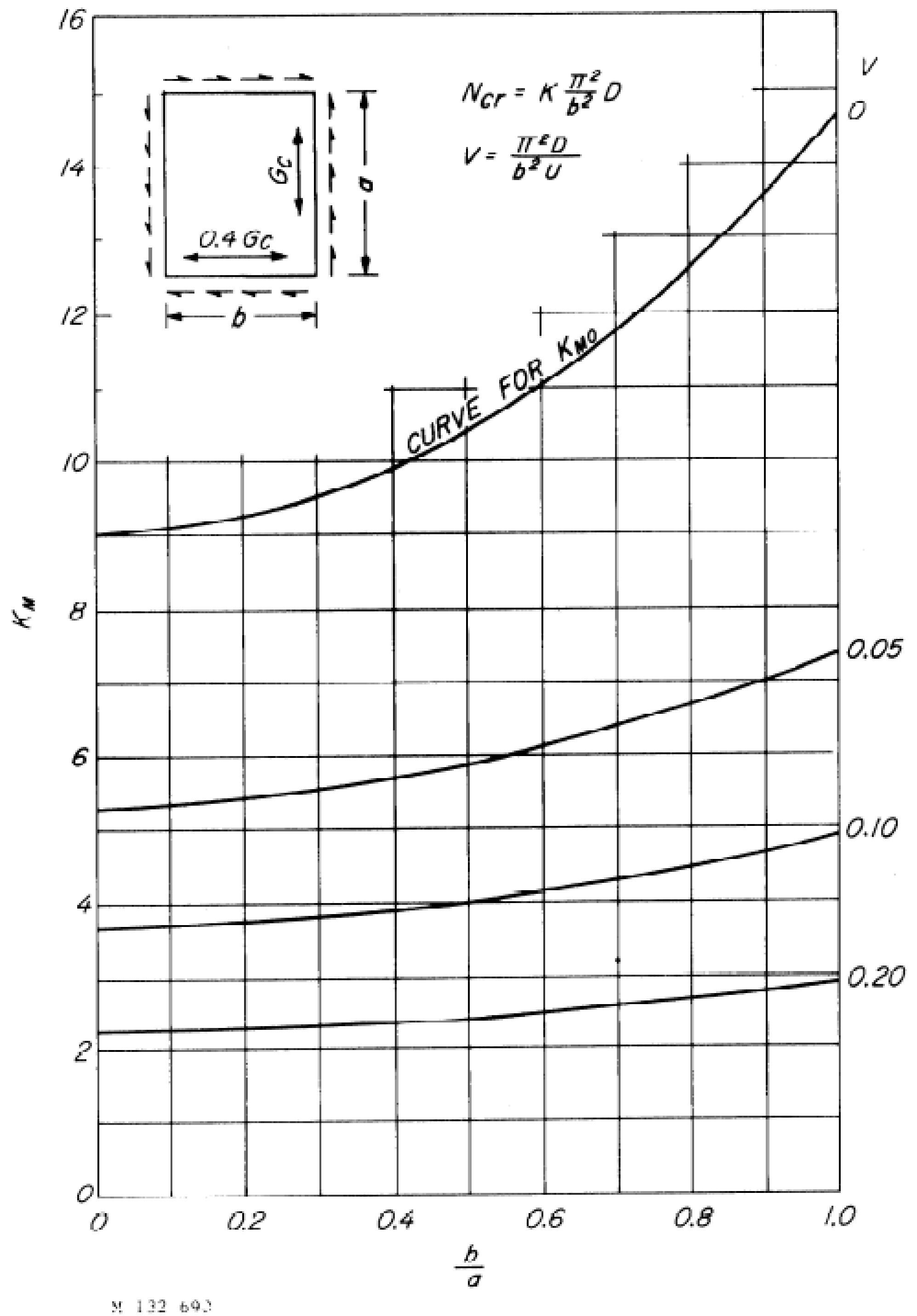

FIGURE 4.9.3.3(g) *Buckling coefficient K_M for sandwich panel with all edges clamped, isotropic face sheets and orthotropic core ($G_{cb} = 0.4G_{ca}$), under edgewise shear load.*

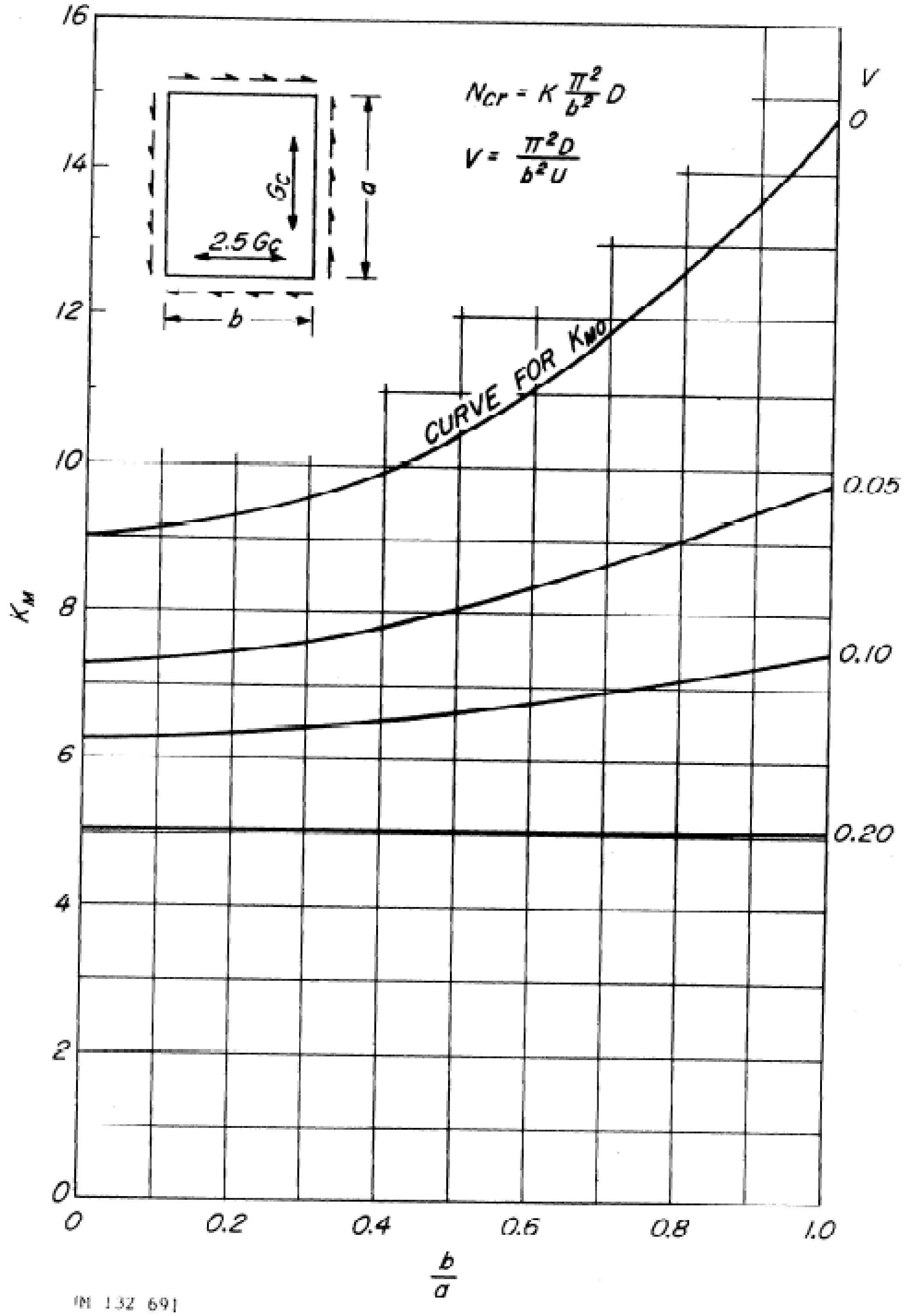

FIGURE 4.9.3.3(h) *Buckling K_M for sandwich panel with all edges clamped, isotropic face sheets and orthotropic core ($G_{cb} = 2.5G_{ca}$), under edgewise shear load.*

4.9.4 Design of sandwich strips under torsion load

Assuming that a design begins with chosen design stresses and a given load to transmit, a sandwich strip under torsion load shall be designed to comply with the basic design principles summarized in Section 4.2.1. The conditions must be met. This section addresses global buckling. The other failure modes listed in Section 4.4 should be checked separately.

The design of sandwich strips under torsion load is based primarily upon limitations on the amount of twist rather than limitations upon torque-produced stresses in sandwich face sheets.

Design information is presented for sandwich strips of trapezoidal (including rectangular) and triangular cross section. Strips of rectangular cross section are included as a limiting case of the trapezoidal cross section. The information presented applies to strips having thin isotropic face sheets of equal thickness.

Design procedures for sandwich strips are arranged in a manner similar to the design of other sandwich components, wherein face sheet and core thicknesses and properties can be determined for sandwich having a fixed width, length, and torsional rigidity. The shape of the cross section may be determined by nonstructural design features such as airfoil characteristics that require a specified angle between sandwich face sheets and a definite width of strip as for a control surface. Checking procedures are also presented.

A useful design hint for sandwich strips of any shape of cross section is that torque is directly proportional to face sheet thickness for a given twist, face sheet stress, or core stress. The following procedures are restricted to linear elastic behavior.

Overall buckling of the sandwich, or local instabilities such as dimpling or wrinkling of the face sheets, may lead to total collapse of the panel. Detailed procedures giving theoretical equations and graphs for determining dimensions of face sheets and core, as well as necessary core properties, are given in the following paragraphs. Face sheet modulus of rigidity values, G, and stress values, F_s, shall be values at the conditions of use; that is, if application is at elevated temperature, then face sheet properties at elevated temperature shall be used in design.

4.9.4.1 Determining face sheet thickness, core thickness and core shear modulus for sandwich strips of trapezoidal and rectangular cross section

This section gives procedures for determining sandwich face sheet and core thicknesses and core shear modulus so that chosen design face sheet stresses and allowable sandwich twist will not be exceeded (Reference 4.9.4.1).

For a sandwich in which the two face sheets are of the same isotropic material and the same thickness, the angle of twist on one end of a trapezoidal sandwich strip of length L relative to the other end is given by the formula

$$\theta = \frac{k_1 TL}{2td^2bG} \qquad 4.9.4.1(a)$$

where θ is angle of twist (radians), k_1 is a coefficient dependent upon a value of V_t and Z, T is applied torque, b is width of the sandwich strip, d is distance between centroids of sandwich face sheets, t is thickness of sandwich face sheet (see Figure 4.9.4.1(a) for notation), G is modulus of rigidity of sandwich face sheet, and V_t and Z are given by the equations

$$V_t = \frac{tdG}{2b^2G_c} \qquad 4.9.4.1(b)$$

$$Z = \frac{b}{d}\tan(\alpha) \qquad 4.9.4.1(c)$$

where G_c is core shear modulus in the x-z plane, where x and y define the midplane of the core, and z is perpendicular to it, as shown in Figure 4.9.4.1(a); α is the angle of the face sheets relative to the mid-plane, as shown in Figure 4.9.4.1(a), and the remaining symbols are as defined previously. This analysis assumes $\cos(\alpha) \approx 1$, and therefore α should not exceed 20° (Reference 4.9.4.1).

The face sheet shear stress is maximum near the center of the strip width ($y = \frac{b}{2}$) and is given by the formula

$$F_s = \frac{k_2 T}{2tdb} \qquad 4.9.4.1(d)$$

where F_s is the face sheet shear stress, and k_2 is a coefficient dependent upon values of V_t and Z.

The core shear stress is maximum at the thicker edge of the strip and is given by the formula

$$F_{sc} = \frac{k_3 T}{2db}\sqrt{\frac{2G_c}{tdG}} \qquad 4.9.4.1(e)$$

where F_{sc} is core shear stress and k_3 is a coefficient dependent upon values of V_t and Z.

Combining Equations 4.9.4.1(a) and 4.9.4.1(d) and solving for d results in

$$d = \frac{k_1 F_s L}{k_2 G \theta} \qquad 4.9.4.1(f)$$

Graphs of k_1 and k_2 are given in Figures 4.9.4.1(b) through (e) wherein these coefficients are presented as functions of V_t and Z. Note that Figures 4.9.4.1(b) and 4.9.4.1(c) show different ranges of the same curves, as do Figures 4.9.4.1(d) and (e). Figures 4.9.4.1(b) and (d) show the curves for values of V_t = 0 to 0.1, and are therefore applicable to sandwiches with stiffer core than Figures 4.9.4.1(c) and (e), which show the curves for V_t = 0.1 to 2.0.

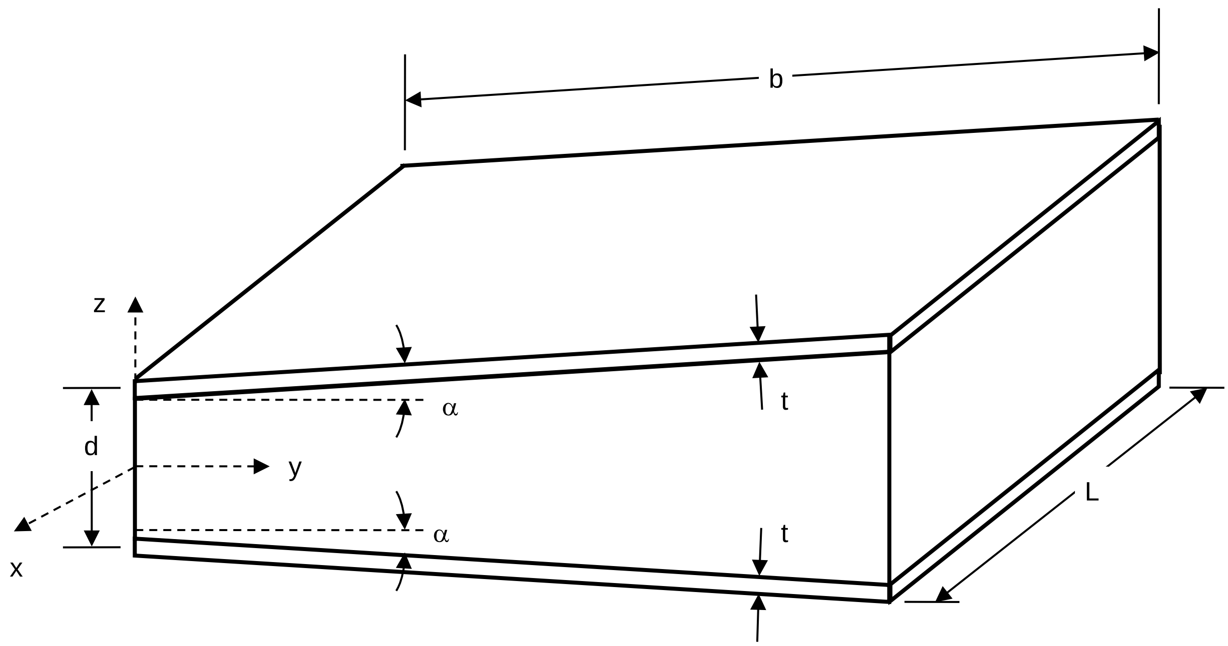

FIGURE 4.9.4.1(a) *Notation for sandwich of trapezoidal cross section in torsion.*

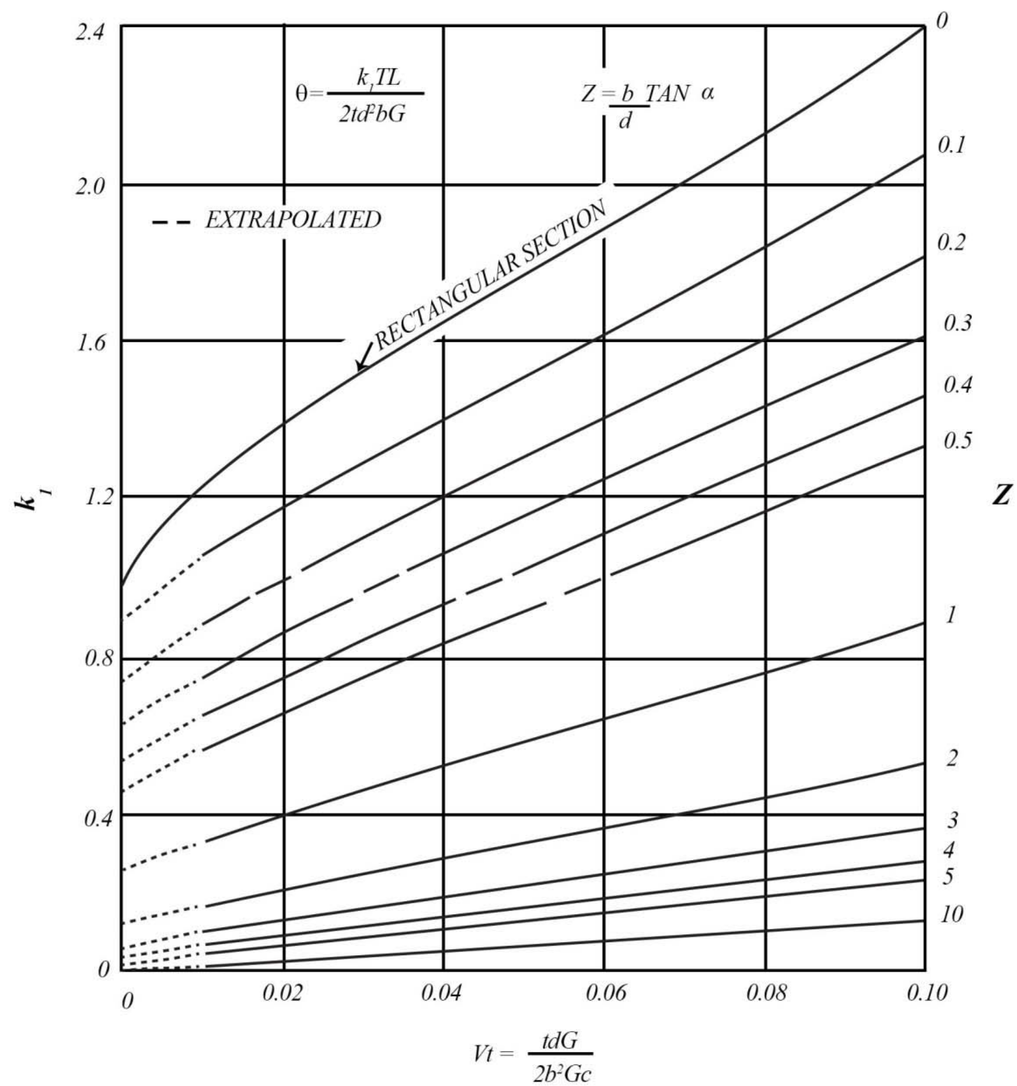

FIGURE 4.9.4.1(b) *Coefficient k_1 for designing sandwich of rectangular and trapezoidal cross section – stiff cores.*

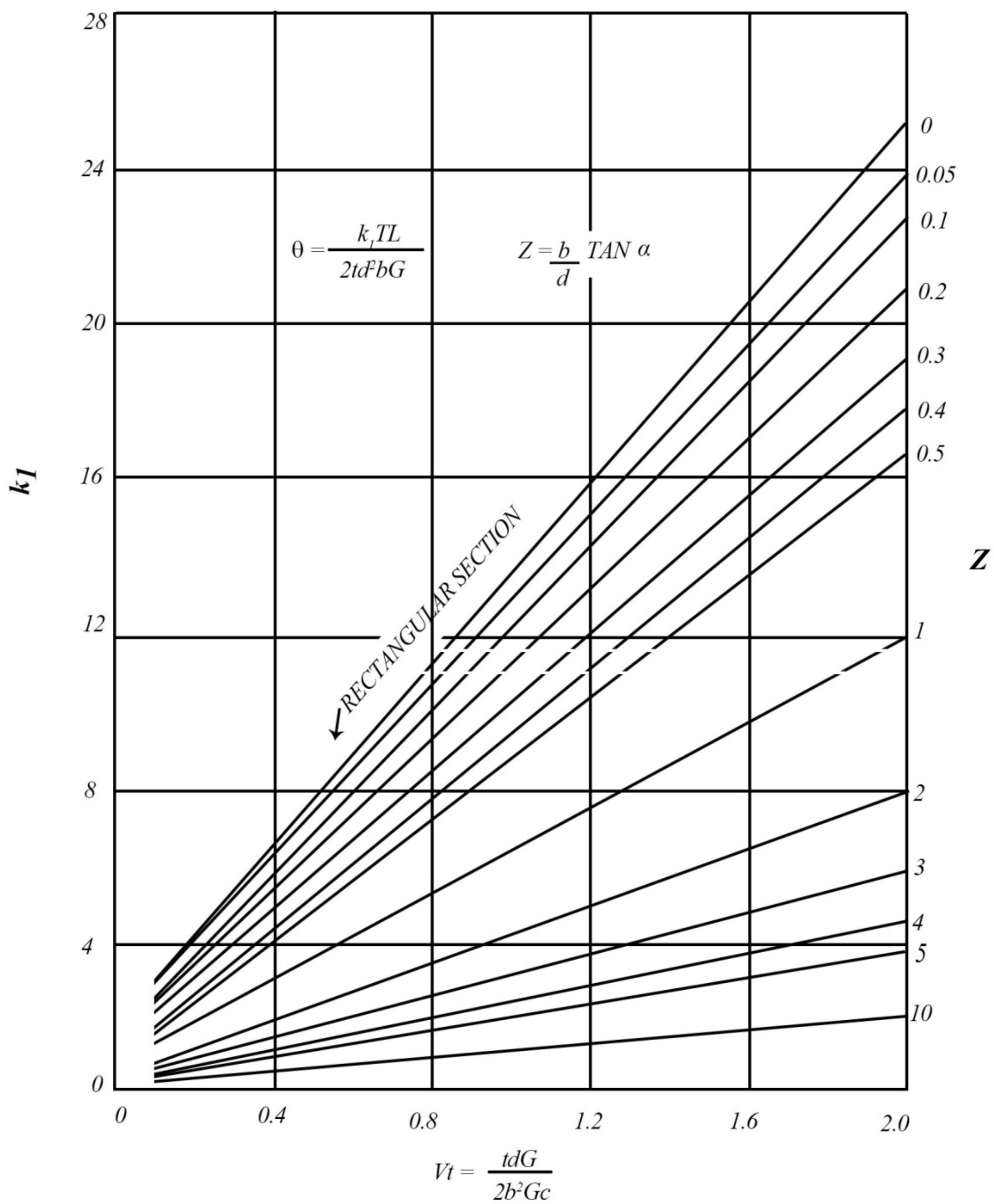

FIGURE 4.9.4.1(c) *Coefficient k_1 for designing sandwich of rectangular and trapezoidal cross section.*

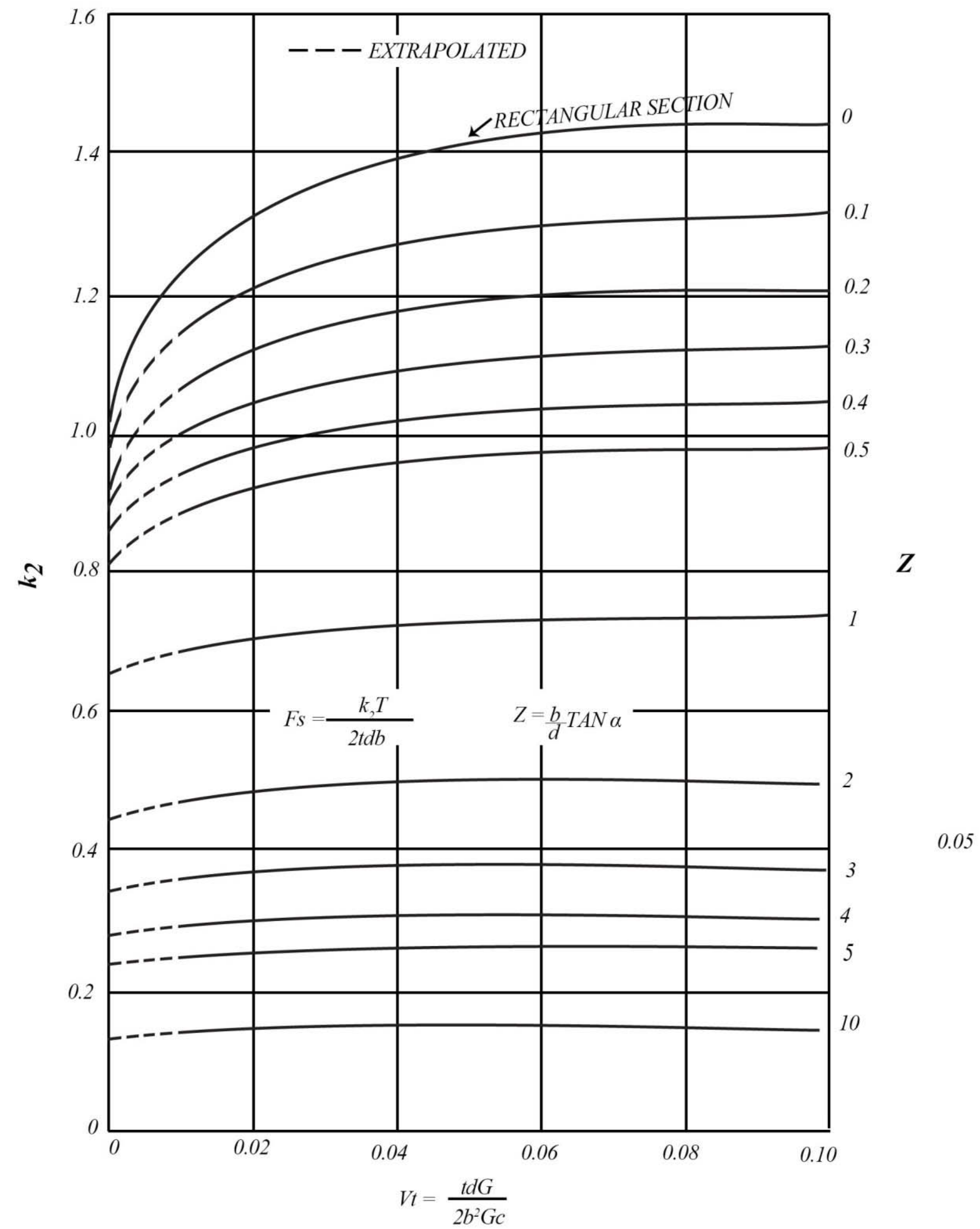

FIGURE 4.9.4.1(d) *Coefficient k_2 for designing sandwich of rectangular and trapezoidal cross section – stiff cores.*

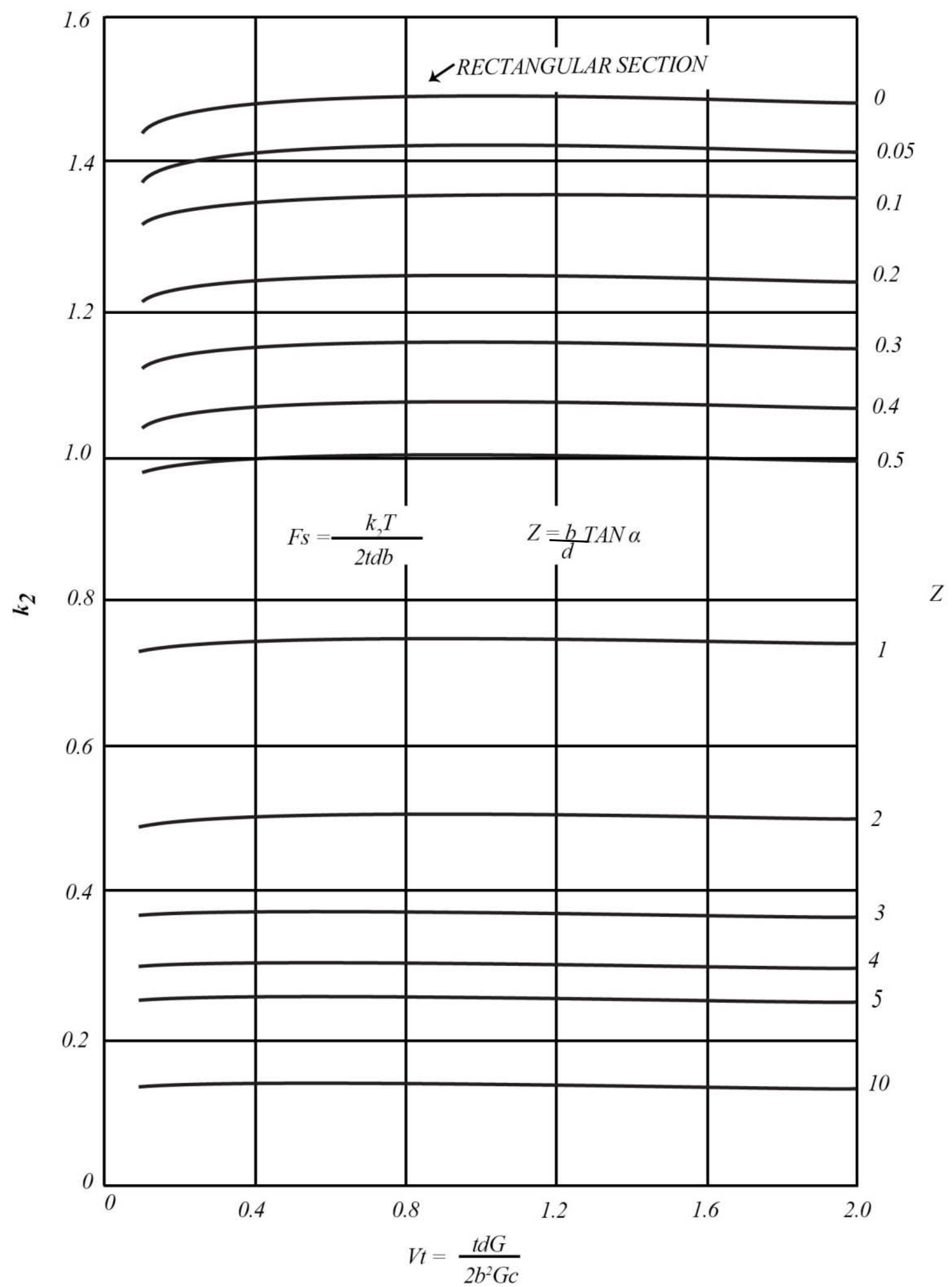

FIGURE 4.9.4.1(e) *Coefficient k_2 for designing sandwich of rectangular and trapezoidal cross section.*

4.9.4.1.1 Determining minimum values of d and t

Minimum values of d and t will be determined by assuming $V_t = 0$ for a first approximation. The values of d and t are minimum because $V_t = 0$ only if the core shear modulus is infinite; for any actual core the shear modulus is finite, hence thicker cores and face sheets must be used. Values of k_1 and k_2 for $V_t = 0$ are obtained from graphs of Figures 4.9.4.1(b) and (d) and substituted into Equation 4.9.4.1(f) to obtain a minimum value of d.

Substitution of this value of d into Equation 4.9.4.1(d) and solving for t results also in a minimum value for t.

If the resultant value of t is too small for a reasonable face sheet thickness, it will be necessary to lower the value of the face sheet stress F_s and begin the design again with Equation 4.9.4.1(f).

4.9.4.1.2 Determining actual values of d and t

Since actual core shear modulus values are not very large, values of d and t somewhat greater than that given by Equations 4.9.4.1(d) and 4.9.4.1(f) must be used. Actual values of d can be determined from Equation 4.9.4.1(f) with values of k_1 and k_2 read from graphs of Figures 4.9.4.1(b) through (e) for $V_t \neq 0$. In using these graphs, it is necessary to iterate because V_t is directly proportional to d and t, and Z is dependent upon d. As an aid to determining d and G_c, Figure 4.9.4.1.2 presents a number of lines representing V_t for various values of G_c with V_t ranging from 0.01 to 2 and G_c ranging from 1,000 to 1,000,000 psi. The following procedure is suggested:

1. Determine Z from Equation 4.9.4.1(c) using the minimum value of d determined in Section 4.9.4.1.2.

2. Determine k_1 and k_2 from Figures 4.9.4.1(b) and (d) using a value of 0.01 for V_t.

3. Compute d with Equation 4.9.4.1(f), and with this value of d compute a new value of Z using Equation 4.9.4.1(c).

4. Repeat steps 2 and 3 until value of d from Equation 4.9.4.1(f) agrees with the value used in Equation 4.9.4.1(c) to compute Z.

5. Compute t with Equation 4.9.4.1(d) solved for t

$$t = \frac{k_2 T}{2dbF_s}$$

6. Compute the constant relating V_t to G_c

$$\frac{tdG}{2b^2} = V_t G_c$$

7. With this constant enter Figure 4.9.4.1.2 and determine G_c

8. If the shear modulus is outside the range of values for materials available, slide up the appropriate line of Figure 4.9.4.1.2 and pick a new value for V_t, for a reasonable value of core shear modulus.

9. Reenter Figures 4.9.4.1(b) and (c) with new value of V_t and repeat previous steps.

4.9.4.1.3 Checking procedure for sandwich strips of trapezoidal and rectangular cross section

The design shall be checked by using the graphs of Figures 4.9.4.1(b) through (e), 4.9.4.1.2, and 4.9.4.1.3(a) and (b) to determine the k coefficients and Equations 4.9.4.1(a), 4.9.4.1(d), and 4.9.4.1(e) to determine theoretical performance.

For a rectangular cross section enclosed on all four sides (see inset in Figure 4.9.4.1.3(c)), the angle of twist can be estimated by elementary theory for torsion of rectangular cross sections. The coefficient k_1 for enclosed, thin-walled, rectangular cross sections is given in the graph of Figure 4.9.4.1.3(c).

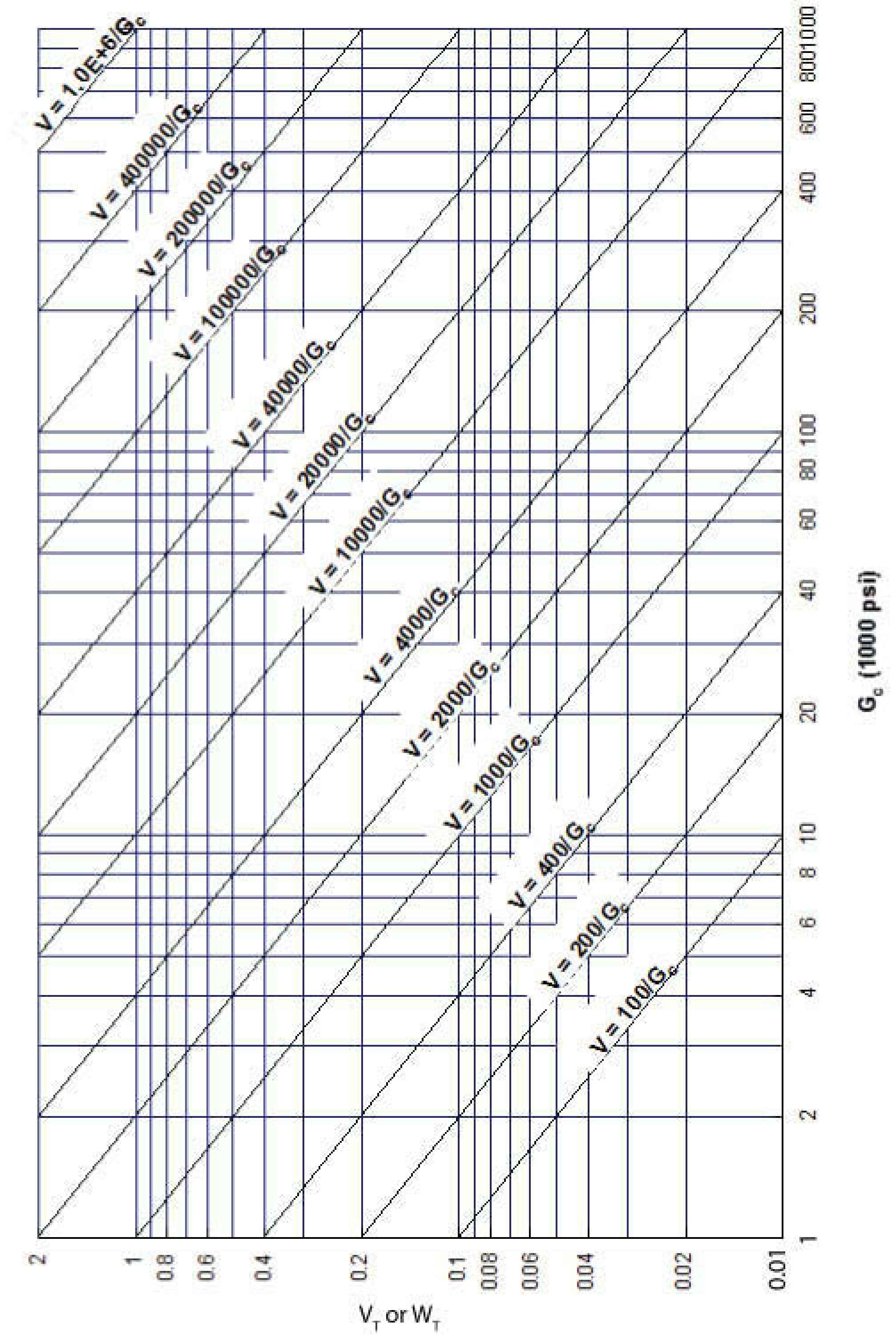

FIGURE 4.9.4.1.2 *Chart for determining V_t or W_t and G_c for sandwich strips in torsion.*

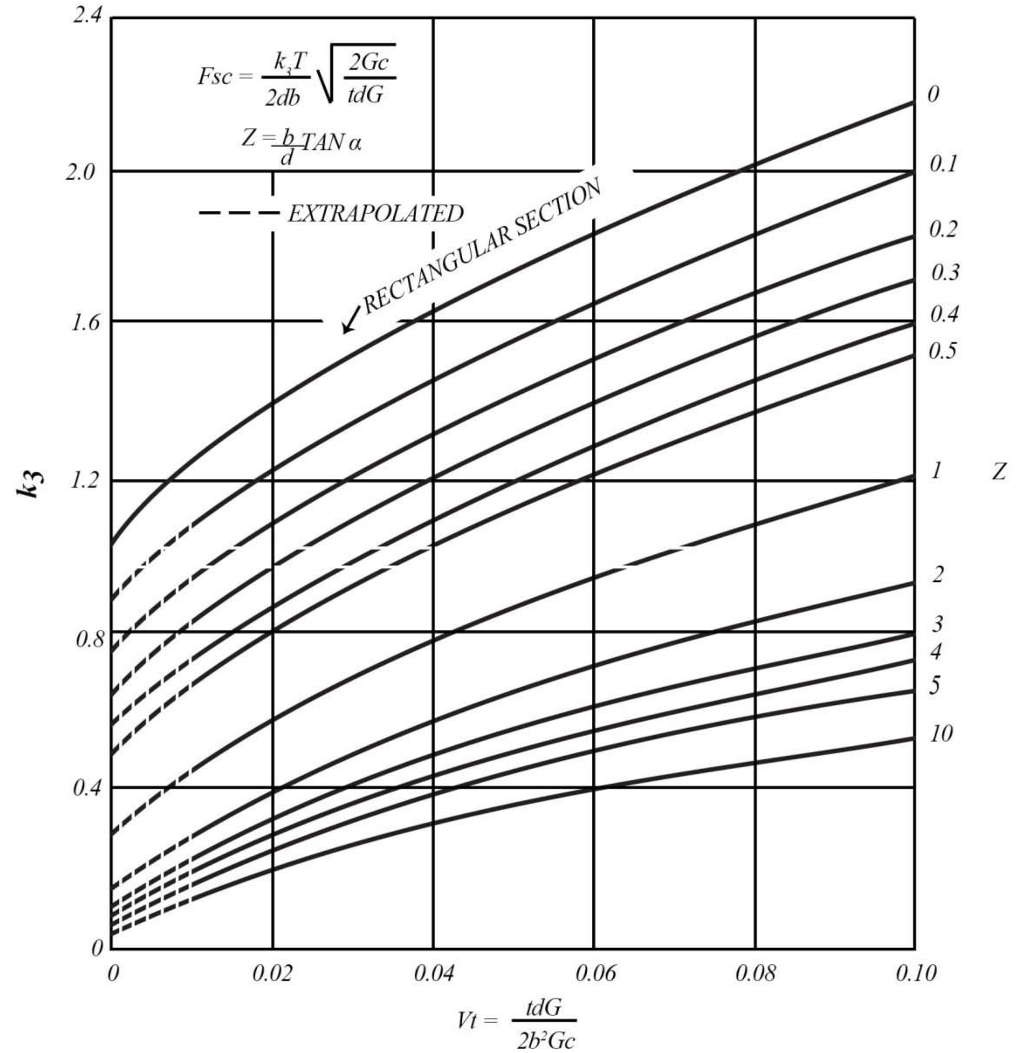

FIGURE 4.9.4.1.3(a) *Coefficient k_3 for designing sandwich of rectangular and trapezoidal cross section in torsion – stiff cores.*

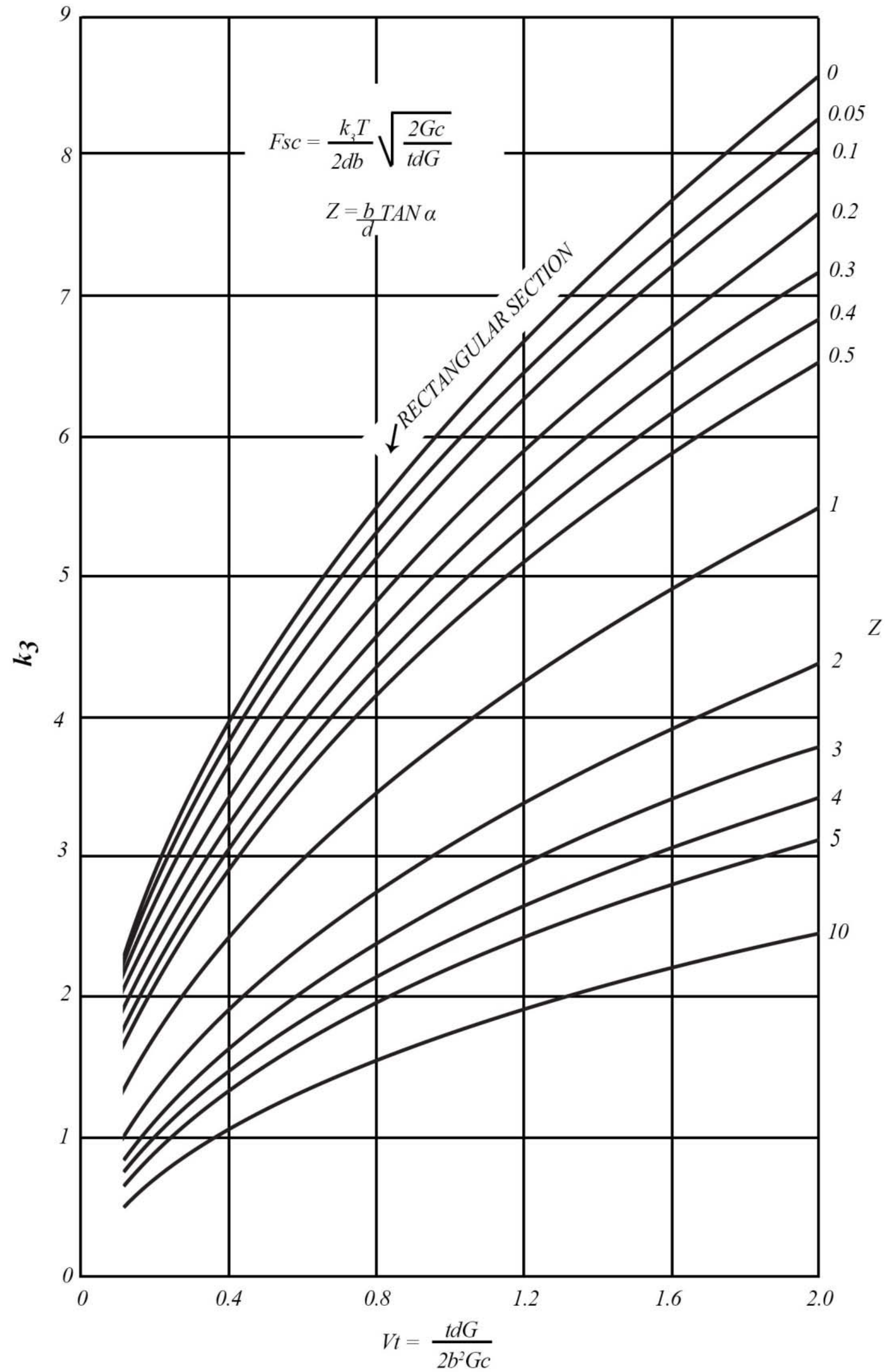

FIGURE 4.9.4.1.3(b) *Coefficient k_3 for designing sandwich of rectangular and trapezoidal cross section in torsion.*

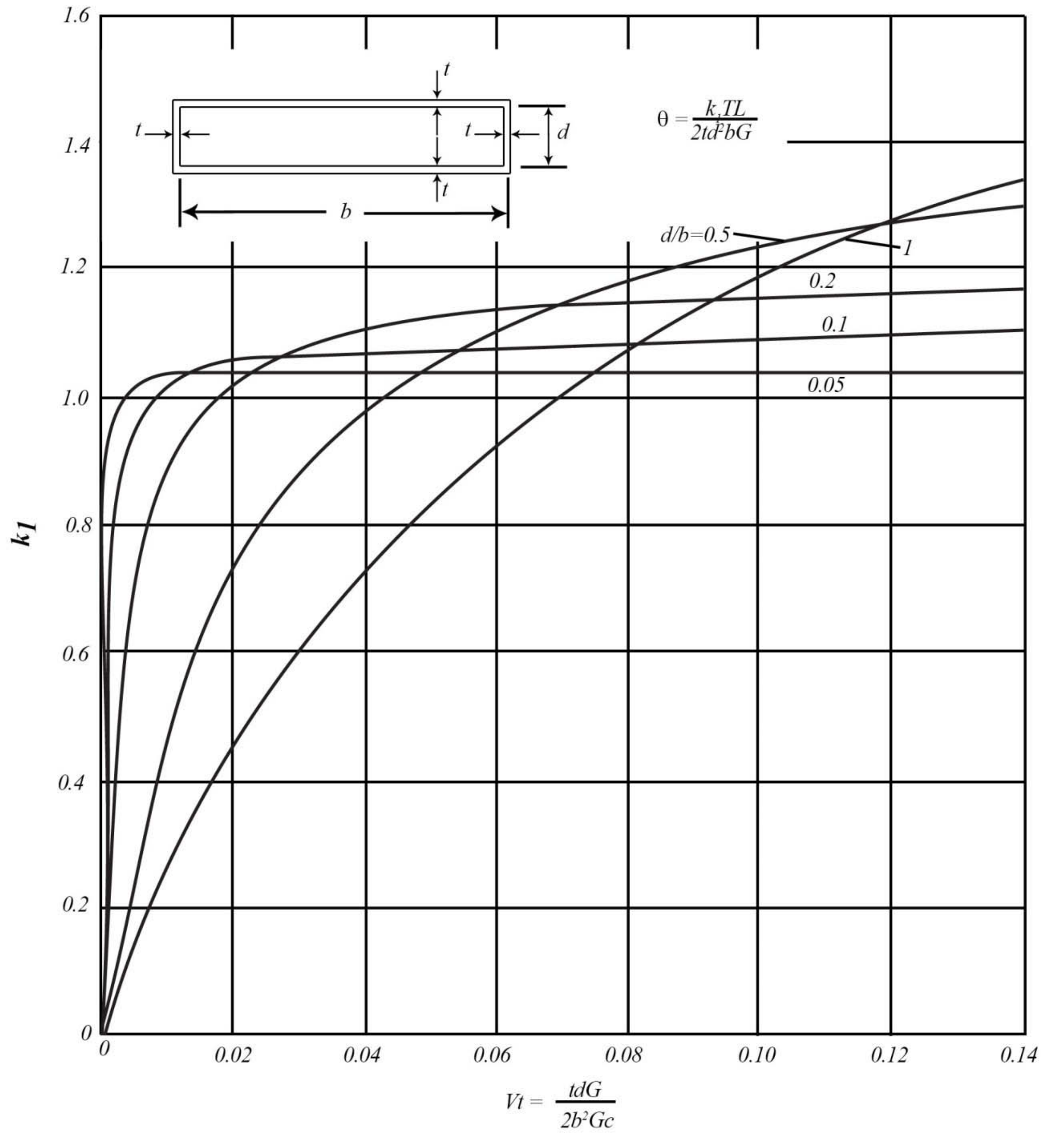

FIGURE 4.9.4.1.3(c) *Coefficient k_1 for designing sandwich strips of enclosed, thin-walled, rectangular cross section in torsion.*

4.9.4.2 Determining face sheet thickness and core shear modulus for sandwich strips of triangular cross section

This section gives procedures for determining sandwich face sheet thickness and core shear modulus, for sandwich of triangular cross section, so that chosen design face sheet stresses and allowable sandwich twist will not be exceeded (Reference 4.9.4.1).

The angle of twist on one end of a triangular sandwich strip of length L relative to the other end is given by the equation

$$\theta = \frac{k_{11} TL}{8tb^3 G} \qquad 4.9.4.2(a)$$

where θ is angle of twist (radians); k_{11} is a coefficient dependent upon a value of W_t and α (see Figure 4.9.4.2.1(a)); T is applied torque; b is width of the sandwich strip; t is thickness of sandwich face sheet; α is the angle of the face sheets relative to the midplane, as shown in the sketch of notation in Figure 4.9.4.2; G is modulus of rigidity of sandwich face sheet; and W_t is given by the formula

$$W_t = \frac{tG}{2bG_c} \qquad 4.9.4.2(b)$$

where G_c is core shear modulus in the x-z plane, where x and y define the midplane of the core, and z is perpendicular to it, as shown in Figure 4.9.4.2, and the remaining symbols are as defined previously.

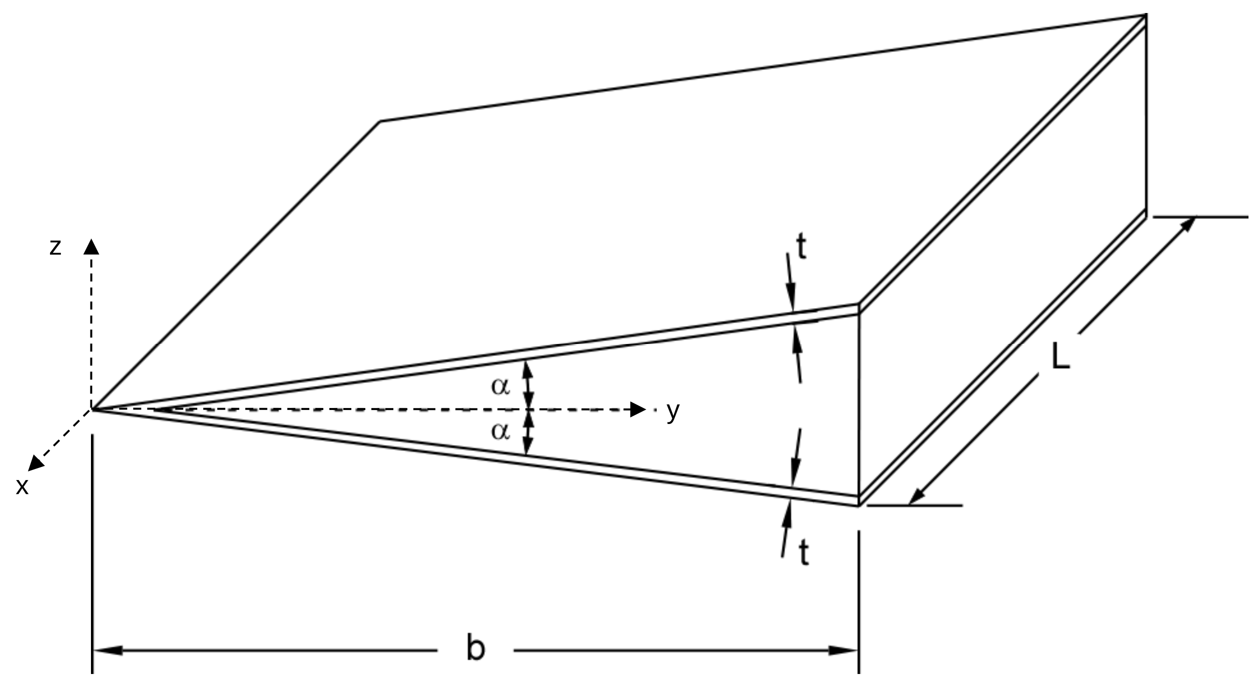

FIGURE 4.9.4.2 *Notation for sandwich of triangular cross section in torsion.*

The face sheet shear stress is maximum near the center of the strip width and is given by the formula

$$F_s = \frac{k_{22} T}{4tb^2} \qquad 4.9.4.2(c)$$

where F_s is the face sheet shear stress, and k_{22} is a coefficient dependent upon values of W_t and α (see Figure 4.9.4.2.1(b)).

The core shear stress is maximum at the thicker edge of the strip and is given by the formula

$$F_{sc} = \frac{k_{33} T}{4b^3} \qquad 4.9.4.2(d)$$

where F_{sc} is core shear stress and k_{33} is a coefficient dependent upon values of W_t and α (see Figure 4.9.4.2.3(a)).

Solution of Equations 4.9.4.2(a) and 4.9.4.2(c) for face sheet thickness, t, results in the following two expressions

$$t = \frac{k_{11} T L}{8 b^3 G \theta} \qquad 4.9.4.2(e)$$

and

$$t = \frac{k_{22} T}{4 b^2 F_s} \qquad 4.9.4.2(f)$$

and the larger value determined by Equation 4.9.4.2(e) or (f) must be used to prevent excessive twist or stress.

4.9.4.2.1 Determining minimum value of t

A minimum value of t will be determined by assuming $W_t = 0$ for a first approximation. The value of t is minimum because $W_t = 0$ only if the core shear modulus is infinite; for any actual core the shear modulus is finite, hence thicker cores and face sheets must be used. Values of k_{11} and k_{22} for $W_t = 0$ are obtained from graphs of Figures 4.9.4.2.1(a) and (b) and substituted into Equations 4.9.4.2(e) and (f) to obtain a minimum value of t. The larger value from Equation 4.9.4.2(e) and (f) will be the value of t for $W_t = 0$. If the resultant value of t is too small for a reasonable face sheet thickness, it will be necessary to lower the value of twist, θ, and the face sheet stress, F_s, and begin the design again with Equations 4.9.4.2(e) and (f).

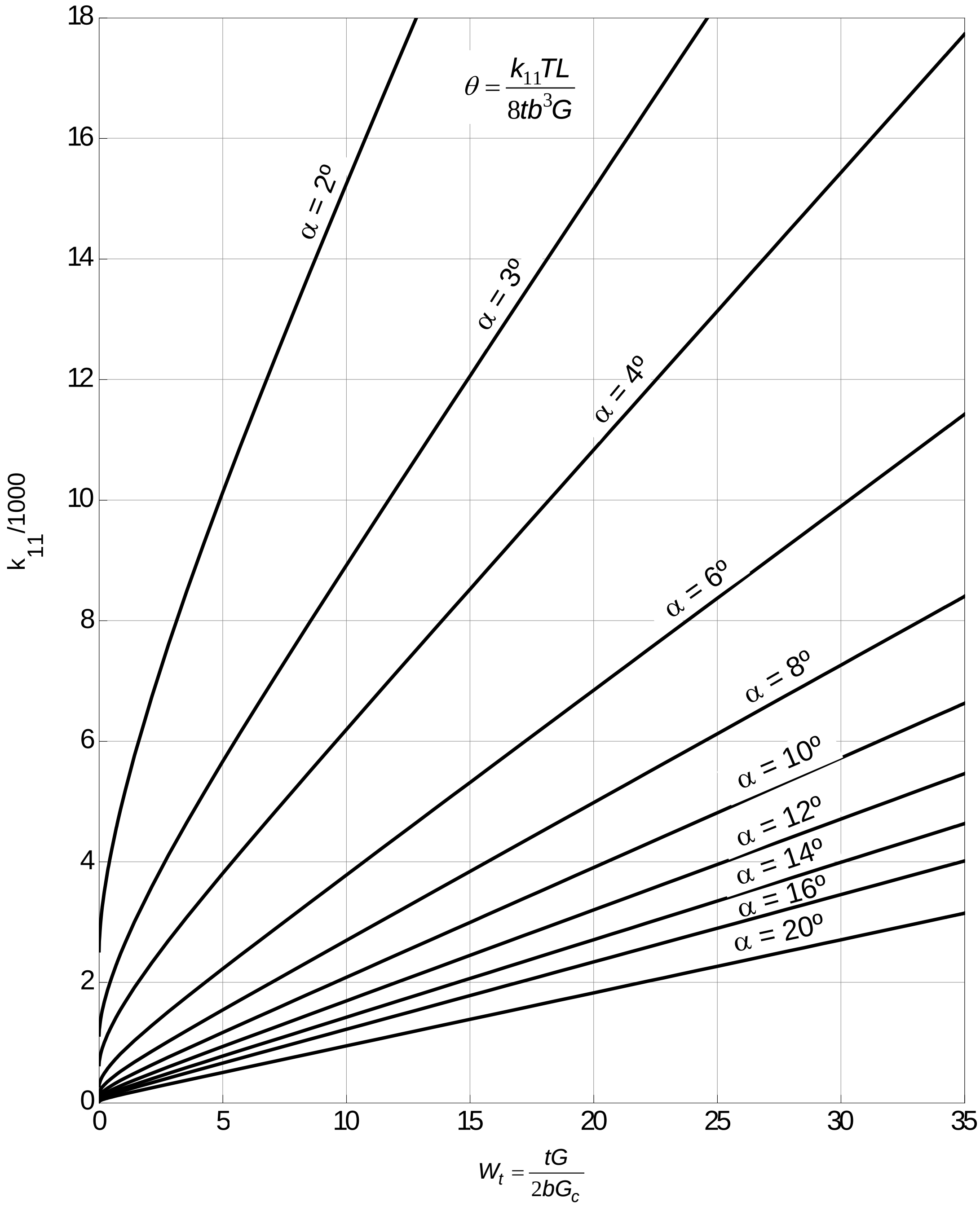

FIGURE 4.9.4.2.1(a) *Coefficient k_{11} for designing sandwich strips of triangular cross section in torsion.*

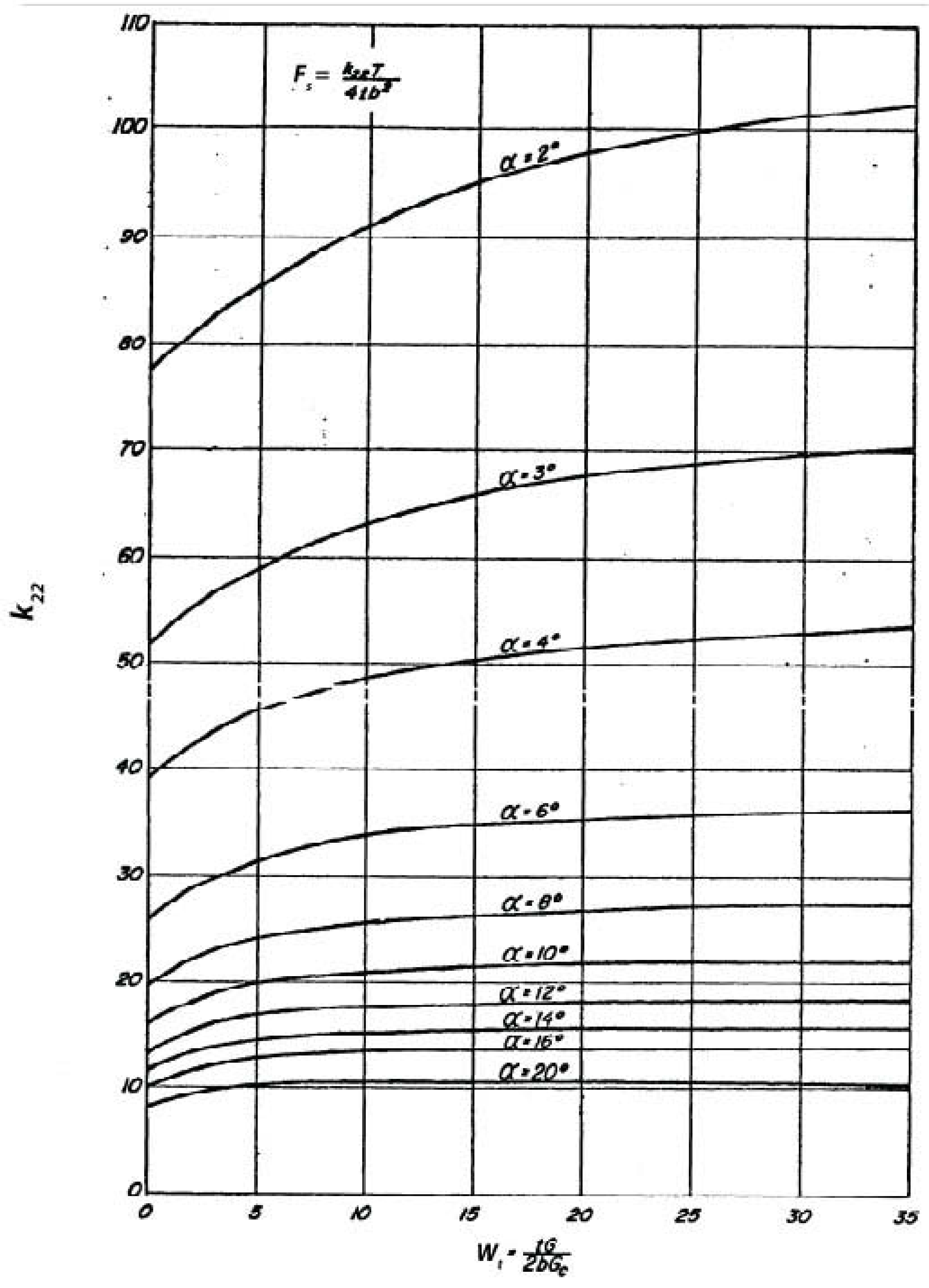

FIGURE 4.9.4.2.1(b) *Coefficient k_{22} for designing sandwich strips of triangular cross section in torsion.*

4.9.4.2.2 Determining actual value of t

Since actual core shear modulus values are not very large, a value of t somewhat greater than that given by Equations 4.9.4.2(e) and (f) must be used. Actual values of t can be determined from Equations 4.9.4.2(e) and (f) with values of k_{11} and k_{22} read from graphs of Figures 4.9.4.2.1(a) and (b) for $W_t \neq 0$. In using these graphs, it is necessary to iterate because W_t is directly proportional to t. As an aid to determining t and G_c, Figure 4.9.4.1.2 presents a number of lines representing V_t or W_t for various values of G_c with V_t or W_t ranging from 0.01 to 2 and G_c ranging from 1,000 to 1,000,000 psi. The following procedure is suggested:

1. Determine k_{11} and k_{22} from Figures 4.9.4.2.1(a) and (b) using a value of 1 for W_t.

2. Compute t as the larger value from Equations 4.9.4.2(e) and (f).

3. Compute the constant relating W_t to G_c

$$\frac{tG}{2b} = W_t G_c$$

4. With this constant enter Figure 4.9.4.1.2 and determine G_c.

5. If the shear modulus is outside the range of values for materials available, slide up the appropriate line of Figure 4.9.4.1.2 and pick a new value for W_t, for a reasonable value of core shear modulus.

6. Reenter Figures 4.9.4.2.1(a) and (b) with new value of W_t and repeat the previous steps.

4.9.4.2.3 Checking procedure for sandwich strips of triangular cross section

The design shall be checked by using the graphs of Figures 4.9.4.2.1(a) and (b) and 4.9.4.2.3(a) to determine the k coefficients and Equations 4.9.4.2(a), 4.9.4.2(c), and 4.9.4.2(d) to determine theoretical performance.

If the triangular cross section is enclosed, the angle of twist can be determined by theory of Reference 4.9.4.2.3. The coefficient k_{11} for enclosed, thin-walled, triangular cross sections is given in the graph of Figure 4.9.4.2.3(b).

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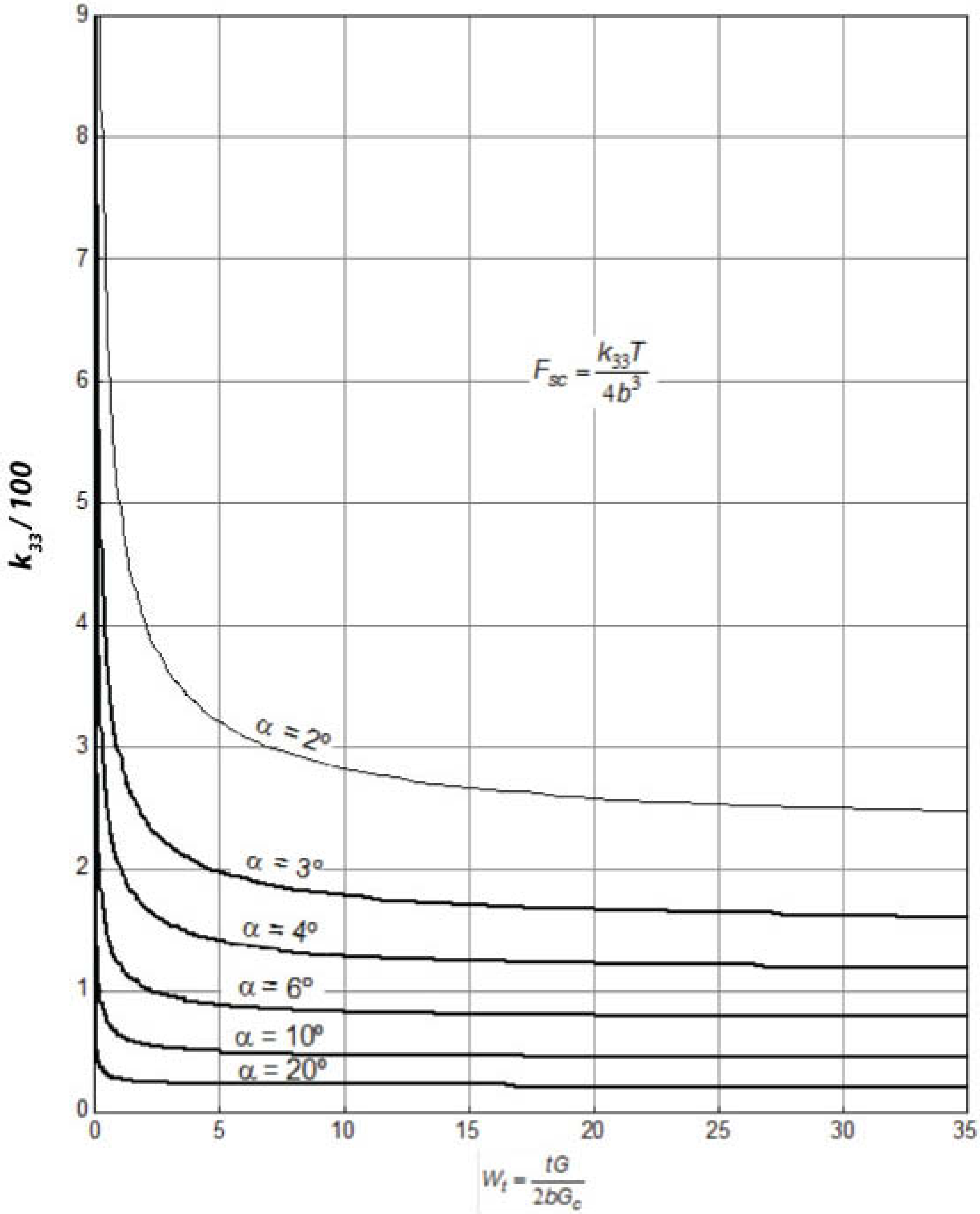

FIGURE 4.9.4.2.3(a) *Coefficient k_{33} for designing sandwich strips of triangular cross section in torsion.*

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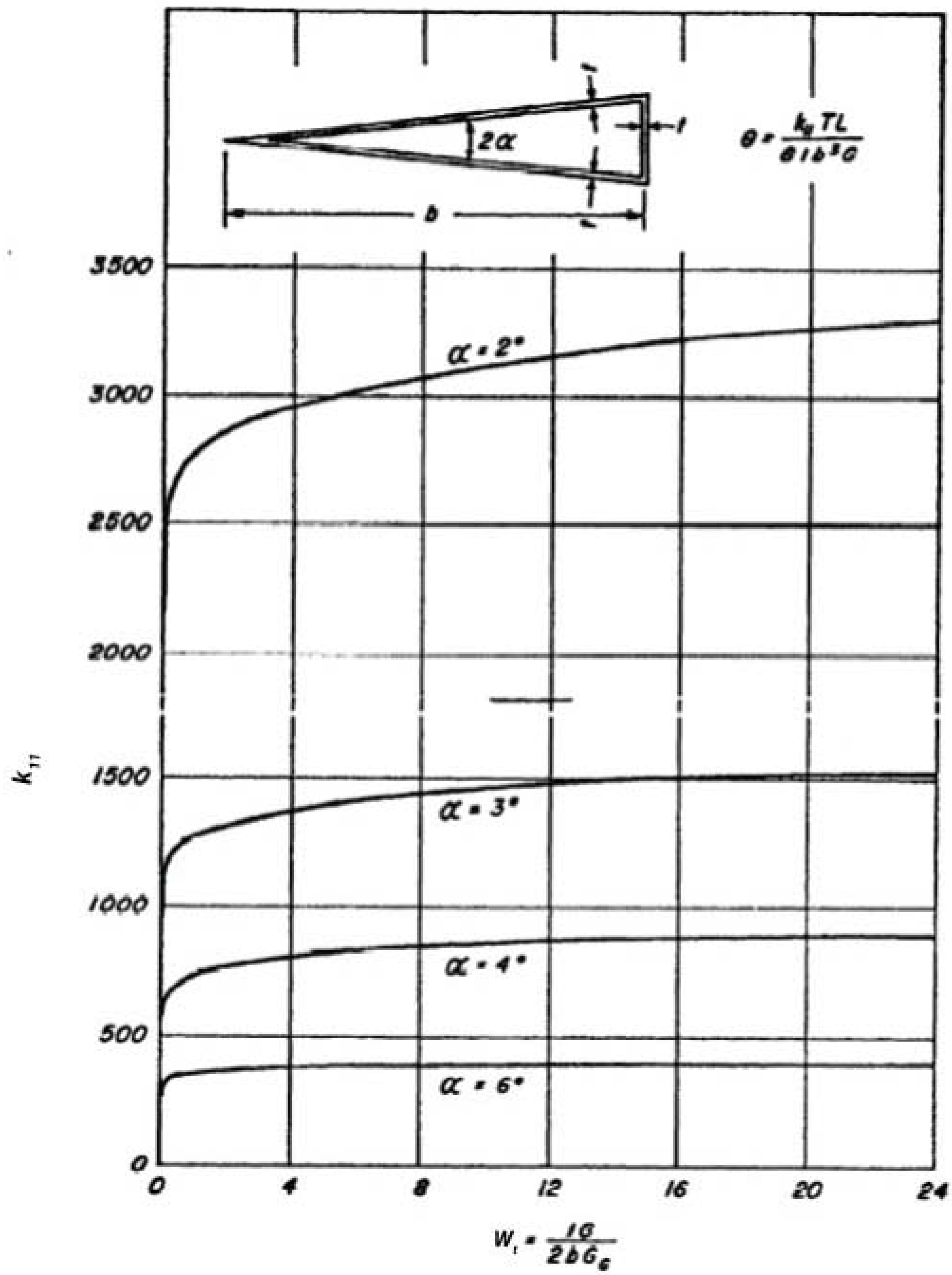


FIGURE 4.9.4.2.3(b) *Coefficient k_{11} for designing sandwich strips of enclosed, thin-walled triangular cross section in torsion.*

4.9.5 Design of flat rectangular sandwich panels under edgewise bending moment

Assuming that a design begins with chosen design stresses and a given load to transmit, a flat rectangular panel of sandwich construction under edgewise bending moment shall be designed to comply with the basic design principles summarized in Section 4.2.1. These conditions must be met. This sec-

tion addresses global buckling. The other failure modes listed in Section 4.4 should be checked separately.

Overall buckling of the sandwich, or local instabilities such as dimpling or wrinkling of the face sheets, may lead to total collapse of the panel. Detailed procedures giving theoretical equations and graphs for determining dimensions of the face sheets and core, as well as necessary core properties, are given in the paragraphs. Double equations are given, one equation for sandwich with face sheets of different materials and thicknesses and another equation for sandwich with each face sheet of the same material and thickness.

Because edgewise bending moment causes variations in face sheet stress across the panel width, extrapolation to buckling beyond the elastic range of face sheet stresses cannot be done by substituting an effective elastic modulus, such as the tangent modulus, in buckling equations. Proper extrapolation to stresses beyond the elastic range must consider the variation of effective elastic modulus across the panel width associated with the stress variation. The information given here is thus strictly applicable only to buckling at face sheet stresses within the elastic range. Face sheet modulus of elasticity, E, and stress values, F_c, shall be compression values at the conditions of use; that is, if application is at elevated temperature, then face sheet properties at elevated temperature shall be used in design.

4.9.5.1 Determining face sheet thickness

Edgewise bending moment applied to simply supported, flat rectangular sandwich panel produces the loading shown in the sketch in Figure 4.9.5.1. Half of the panel is in edgewise tension, which is a stable condition, but the other half is in edgewise compression. The edgewise compression load, varying from zero at the neutral axis to a maximum value, N, at the panel edge, can produce buckling.

The value of N at the panel edge is determined by the formula

$$N = \frac{6M}{b^2} \qquad 4.9.5.1(a)$$

where N is the load per unit edge width, M is the edgewise bending moment, and b is the panel width.

The equations for buckling in edgewise bending are similar to those for edgewise compression buckling, but the critical load, N_{cr}, is higher for edgewise bending.

Face sheet stresses are related to the edge load by the equations:

$$t_{UPR} F_{cUPR} + t_{LWR} F_{cLWR} = N \qquad 4.9.5.1(b)$$

$$t = \frac{N}{2F_c} \quad \text{(for equal face sheets)}$$

where t is face sheet thickness, F_c is chosen design face sheet compressive stress, N is design compression load per unit length of panel edge, and UPR, LWR are subscripts denoting upper and lower face sheets.

In determining thicknesses of face sheets for sandwich with face sheets of different materials, Equation 4.9.5.1(b) must be satisfied, but also the stresses $F_{c\,UPR}$ and $F_{c\,LWR}$ must be chosen so that $F_{c\,UPR}/E_{UPR} = F_{c\,LWR}/E_{LWR}$ (where E is face sheet modulus of elasticity), thus avoiding overstressing of either face sheet. For example, if the upper face sheet is of a material such that the ratio $F_{c\,UPR}/E_{UPR}$ = 0.005 and the lower face sheet is of a material such that the ratio $F_{c\,LWR}/E_{LWR}$ = 0.002, the design must be based on a ratio of 0.002, otherwise the lower face sheet will be overstressed. In order to accomplish this, the chosen design stress for the upper face sheet must be lowered. For many combinations of face sheet materials, it will be found advantageous to choose thicknesses such that $E_{UPR}\, t_{UPR} = E_{LWR}\, t_{LWR}$.

If the core can support edge load, N should be replaced by the quantity ($N - F_{c\ core}\ t_c$), where $F_{c\ core}$ is compressive stress in the core, and t_c is core thickness.

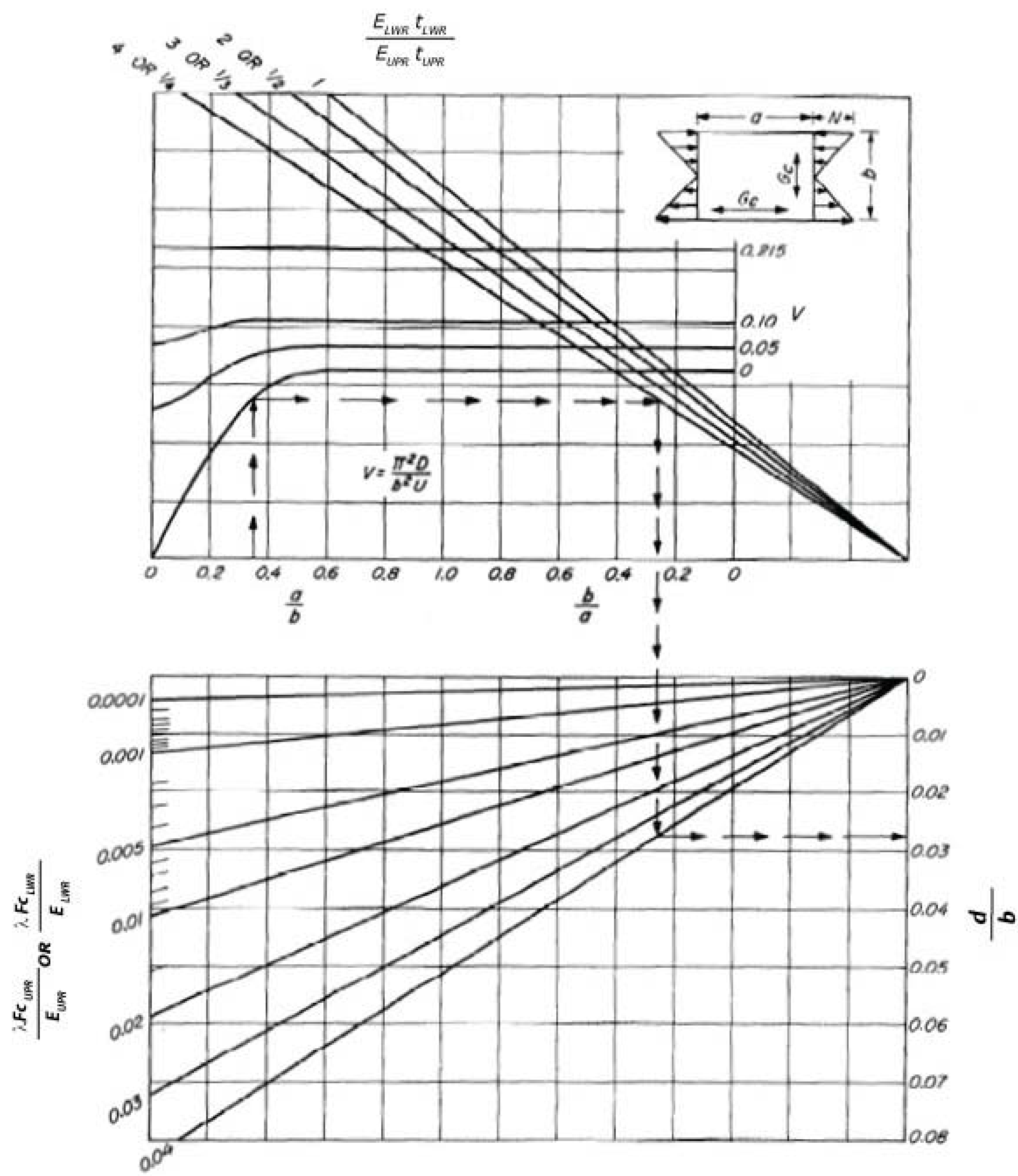


FIGURE 4.9.5.1 *Chart for determining $\frac{d}{b}$ ratio such that a simply-supported sandwich panel with isotropic face sheets and isotropic core ($G_{cb} = G_{ca}$) will not buckle under edgewise bending load.*

4.9.5.2 Determining core thickness and core shear modulus

This section gives procedures for determining core thickness and core shear modulus so that overall buckling of the sandwich panel will not occur (References 4.9.5.2(a) and (b)).

The load per unit panel width at which buckling of a sandwich panel will occur is given by the theoretical formula:

$$N_{cr} = K\frac{\pi^2}{b^2}D$$

where D is sandwich bending stiffness. This formula, solved for the face sheet stress, becomes;

$$F_{cUPR} = \frac{\pi^2 K E_{UPR}}{\lambda}\frac{E_{UPR}\,t_{UPR}\,E_{LWR}\,t_{LWR}}{\left(E_{UPR}\,t_{UPR}+E_{LWR}\,t_{LWR}\right)^2}\left(\frac{d}{b}\right)^2$$

$$F_{cLWR} = \frac{\pi^2 K E_{LWR}}{\lambda}\frac{E_{UPR}\,t_{UPR}\,E_{LWR}\,t_{LWR}}{\left(E_{UPR}\,t_{UPR}+E_{LWR}\,t_{LWR}\right)^2}\left(\frac{d}{b}\right)^2 \qquad \text{4.9.5.2(a)}$$

$$F_c = \frac{\pi^2 K E}{4\lambda}\left(\frac{d}{b}\right)^2 \text{ (for equal face sheets)}$$

where E is the modulus of elasticity of the face sheets, $\lambda = 1 - \nu^2$, ν is Poisson's ratio of the face sheets (in Equation 4.9.5.1(b) it is assumed that $\nu = \nu_{UPR} = \nu_{LWR}$), d is distance between face sheet centroids, b is length of loaded panel edge, $K = K_F + K_M$, K_F is a theoretical coefficient dependent on face sheet stiffness and panel aspect ratio, and K_M is a theoretical coefficient dependent on sandwich bending and shear rigidities and panel aspect ratio. Information on calculating K_F and K_M is given in Section 4.9.5.3.

Solving for $\frac{d}{b}$ gives:

$$\frac{d}{b} = \frac{1}{\pi\sqrt{K}}\sqrt{\frac{\lambda F_{cUPR}}{E_{UPR}}}\left(\frac{E_{UPR}t_{UPR}+E_{LWR}t_{LWR}}{\sqrt{E_{UPR}t_{UPR}E_{LWR}t_{LWR}}}\right)$$

$$\frac{d}{b} = \frac{1}{\pi\sqrt{K}}\sqrt{\frac{\lambda F_{cLWR}}{E_{LWR}}}\left(\frac{E_{UPR}t_{UPR}+E_{LWR}t_{LWR}}{\sqrt{E_{UPR}t_{UPR}E_{LWR}t_{LWR}}}\right) \qquad \text{4.9.5.2(b)}$$

$$\frac{d}{b} = \frac{2}{\pi\sqrt{K}}\sqrt{\frac{\lambda F_c}{E}} \text{ (for equal face sheets)}$$

Therefore, if K is known, Equation 4.9.5.2(b) can be solved directly to obtain d because all other quantities are known. After d is obtained, the core thickness, t_C is computed from the equations

$$t_c = d - \frac{t_{UPR}+t_{LWR}}{2} \qquad \text{4.9.5.2(c)}$$

$$t_c = d - t \text{ (for equal face sheets)}$$

As a first approximation, it will be assumed that $K_F = 0$, hence $K = K_M$. Values of K_M depend upon the bending and shear rigidities of the sandwich as incorporated in the parameter

$$V = \frac{\pi^2 D}{b^2 U}$$

which can be written as:

$$V = \frac{\pi^2 t_c E_{UPR} t_{UPR} E_{LWR} t_{LWR}}{\left(E_{UPR} t_{UPR} + E_{LWR} t_{LWR}\right) \lambda b^2 G_c} \qquad 4.9.5.2(d)$$

$$V = \frac{\pi^2 t_c E t}{2 \lambda b^2 G_c} \quad \text{(for equal face sheets)}$$

where U is sandwich shear stiffness, G_C is the core shear modulus associated with the axes parallel to direction of loading (also parallel to panel side of length a) and perpendicular to the plane of the panel. As values of core shear modulus decrease, values of V increase and values of K_M gradually decrease.

For sandwich with corrugated core having corrugation flutes parallel to direction of loading, the parameter V is replaced by the parameter

$$V_2 = \frac{\pi^2 t_c E_{UPR} t_{UPR} E_{LWR} t_{LWR}}{\left(E_{UPR} t_{UPR} + E_{LWR} t_{LWR}\right) \lambda b^2 G_{cb}} \qquad 4.9.5.2(e)$$

$$V_2 = \frac{\pi^2 t_c E t}{2 \lambda b^2 G_{cb}} \quad \text{(for equal face sheets)}$$

where G_{cb} is the core shear modulus associated with the axes perpendicular to direction of loading (parallel to panel side of length b) and perpendicular to the plane of the panel.

4.9.5.2.1 Determining minimum value of d

A minimum value of d required will be determined by assuming $V = 0$ or $V_2 = 0$ for a first approximation. This value of d is minimum because $V = 0$ or $V_2 = 0$ only if the core shear modulus is infinite; for any actual core the shear modulus is not infinite, hence a thicker core must be used. The minimum value of d may be found using $V = 0$ or $V_2 = 0$ in any of the charts in Figure 4.9.5.1 and Figures 4.9.5.2.1(a) through (c). These charts apply to simply supported sandwich panels having isotropic face sheets and isotropic, orthotropic, or corrugated cores.

Parameters needed for use of these charts are:

1. Panel aspect ratio $\frac{a}{b}$ or $\frac{b}{a}$
2. Face sheet properties $\frac{\lambda F_{cUPR}}{E_{UPR}}$ and $\frac{\lambda F_{cLWR}}{E_{LWR}}$
3. Ratio of $E_{LWR} t_{LWR} / E_{UPR} t_{UPR}$

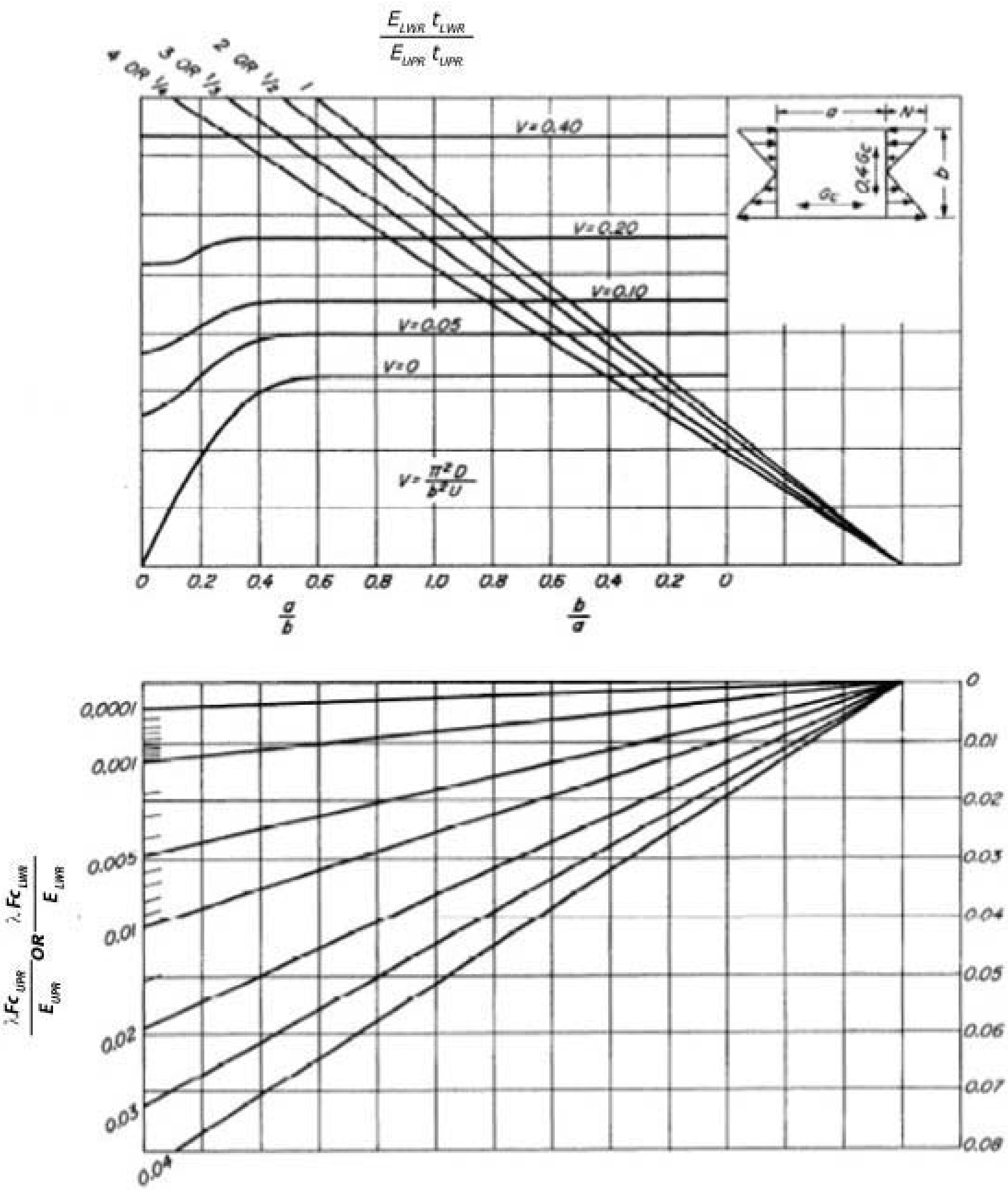


FIGURE 4.9.5.2.1(a) *Chart for determining $\frac{d}{b}$ ratio such that a simply-supported sandwich panel with isotropic face sheets and orthotropic core ($G_{cb} = 0.4G_{ca}$) will not buckle under edgewise bending load.*

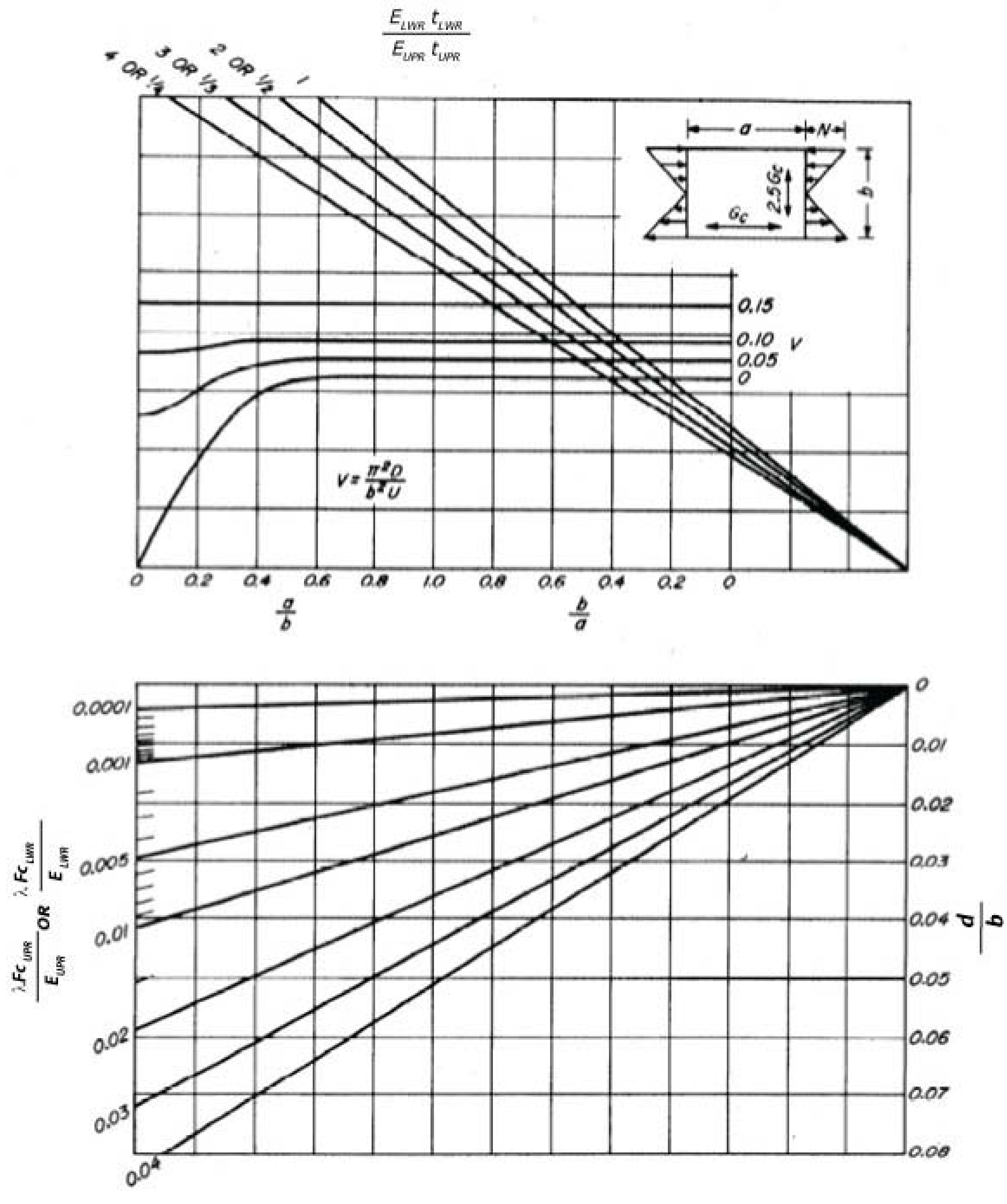


FIGURE 4.9.5.2.1(b) *Chart for determining $\frac{d}{b}$ ratio such that a simply-supported sandwich panel with isotropic face sheets and orthotropic core ($G_{cb} = 2.5G_{ca}$) will not buckle under edgewise bending load.*

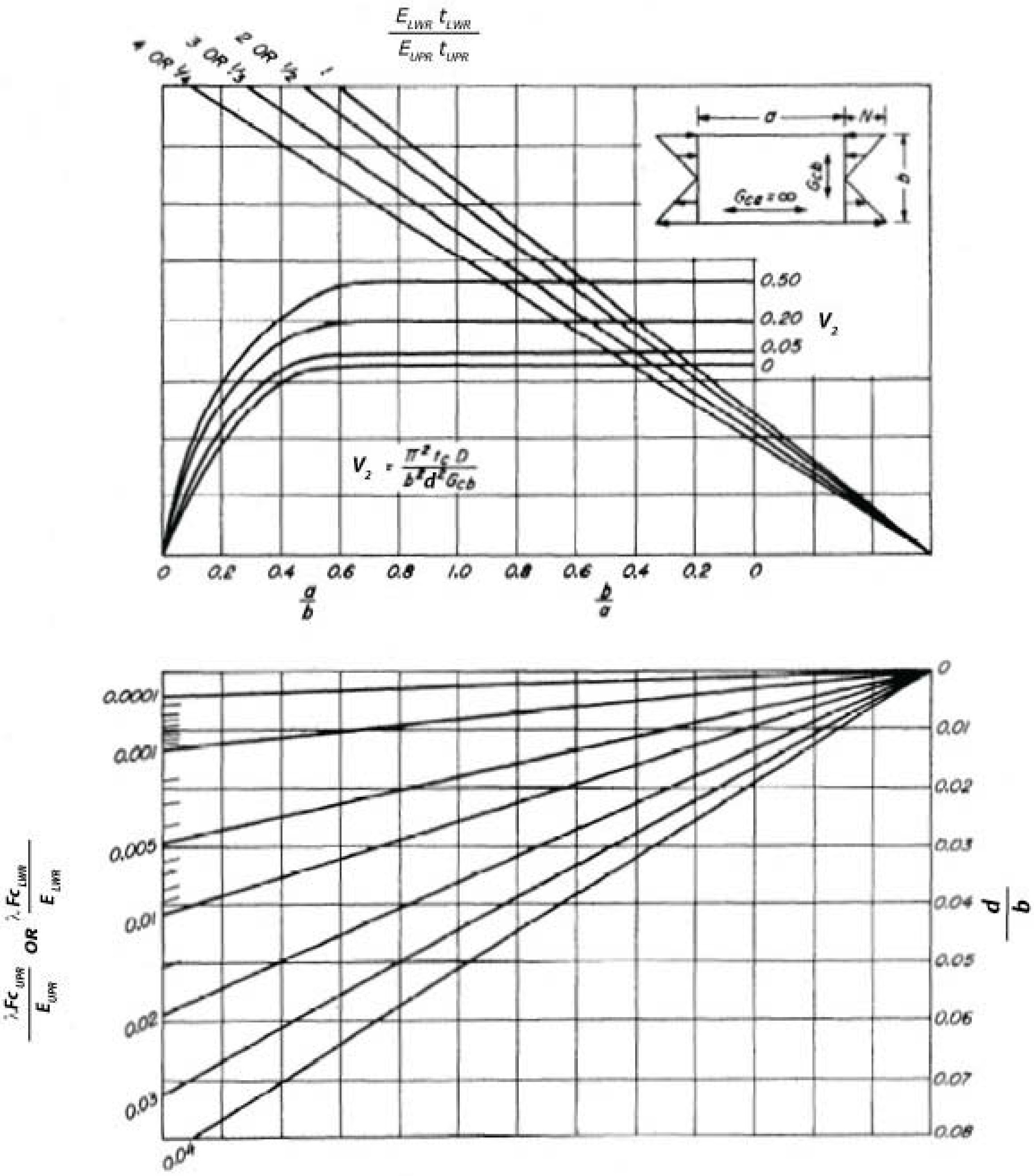


FIGURE 4.9.5.2.1(c) *Chart for determining $\frac{d}{b}$ ratio such that a simply-supported sandwich panel with isotropic face sheets and corrugated core will not buckle under edgewise bending load; core corrugation flutes parallel to side a.*

4.9.5.2.2 Determining actual value of d

Since actual core shear modulus values are not very large, a value of d somewhat greater than that determined by assuming $V = 0$ or $V_2 = 0$ must be used. Charts for determining d for sandwich with all edges simply supported are shown in Figure 4.9.5.1 and Figures 4.9.5.2.1(a) through (c). These figures are entered with values of the panel aspect ratio and values of V as computed by Equation 4.9.5.2(d), or values of V_2 as computed by Equation 4.9.5.2(e). Figure 4.9.5.1 applies to sandwich with isotropic cores, for which the core shear modulus perpendicular to the direction of loading is equal to the core shear modulus parallel to the direction of loading. Figure 4.9.5.2.1(a) applies to sandwich with honeycomb cores for which the core shear modulus perpendicular to the direction of loading is equal to 0.40 times the core shear modulus parallel to the direction of loading. Figure 4.9.5.2.1(b) applies to sandwich with honeycomb cores for which the core shear modulus perpendicular to the direction of loading is 2.50 times the core shear modulus parallel to the direction of loading.

NOTE: For honeycomb cores with core ribbons parallel to direction of loading, $G_c = G_{TL}$ and the shear modulus perpendicular to loading is G_{TW}. For honeycomb cores with core ribbons perpendicular to direction of loading, $G_c = G_{Tw}$ and the shear modulus perpendicular to loading is G_{TL}. If core ribbons are at an angle θ to the panel length a, $G_c = \dfrac{G_{TL} G_{TW}}{\left(G_{TL} \sin^2\theta + G_{TW} \cos^2\theta\right)}$.

Figure 4.9.5.2.1(c) applies to sandwich with corrugated core having the core flutes parallel to the direction of loading.

In using Figure 4.9.5.1 and Figures 4.9.5.2.1(a) through (c), it is necessary to iterate because V and V_2 are directly proportional to the core thickness t_c. As an aid to determining t_c and G_c, Figure 4.9.5.2.2 presents a number of lines representing V for various values of G_c with V ranging from 0.01 to 2 and G_c ranging from 1,000 to 1,000,000 pounds per square inch. The following procedure is suggested:

1. Determine core thickness t_c from Figure 4.9.5.1 and Figures 4.9.5.2.1(a) through (c) using a value of 0.01 for V or V_2.

2. Compute the constant relating V or V_2 to G_c.

$$\left[\frac{\pi^2\, t_c\, E_{UPR}\, t_{UPR}\, E_{LWR}\, t_{LWR}}{\left(E_{UPR}\, t_{UPR} + E_{LWR}\, t_{LWR}\right)\lambda\, b^2}\right] \text{ or } \left[\frac{\pi^2\, t_c\, E\, t}{2\lambda\, b^2}\right] \text{(for equal face sheets)} = V\,G_c \text{ or } V_2\,G_c$$

3. With this constant enter Figure 4.9.5.2.2 and determine necessary G_c.

4. If the shear modulus is outside the range of values for materials available, slide up the appropriate line of Figure 4.9.5.2.2 and pick a new value of V or V_2, for a reasonable value of core shear modulus.

5. Reenter Figure 4.9.5.1 and Figures 4.9.5.2.1(a) through (c) with the new value of V or V_2 and repeat previous steps 1, 2, and 3.

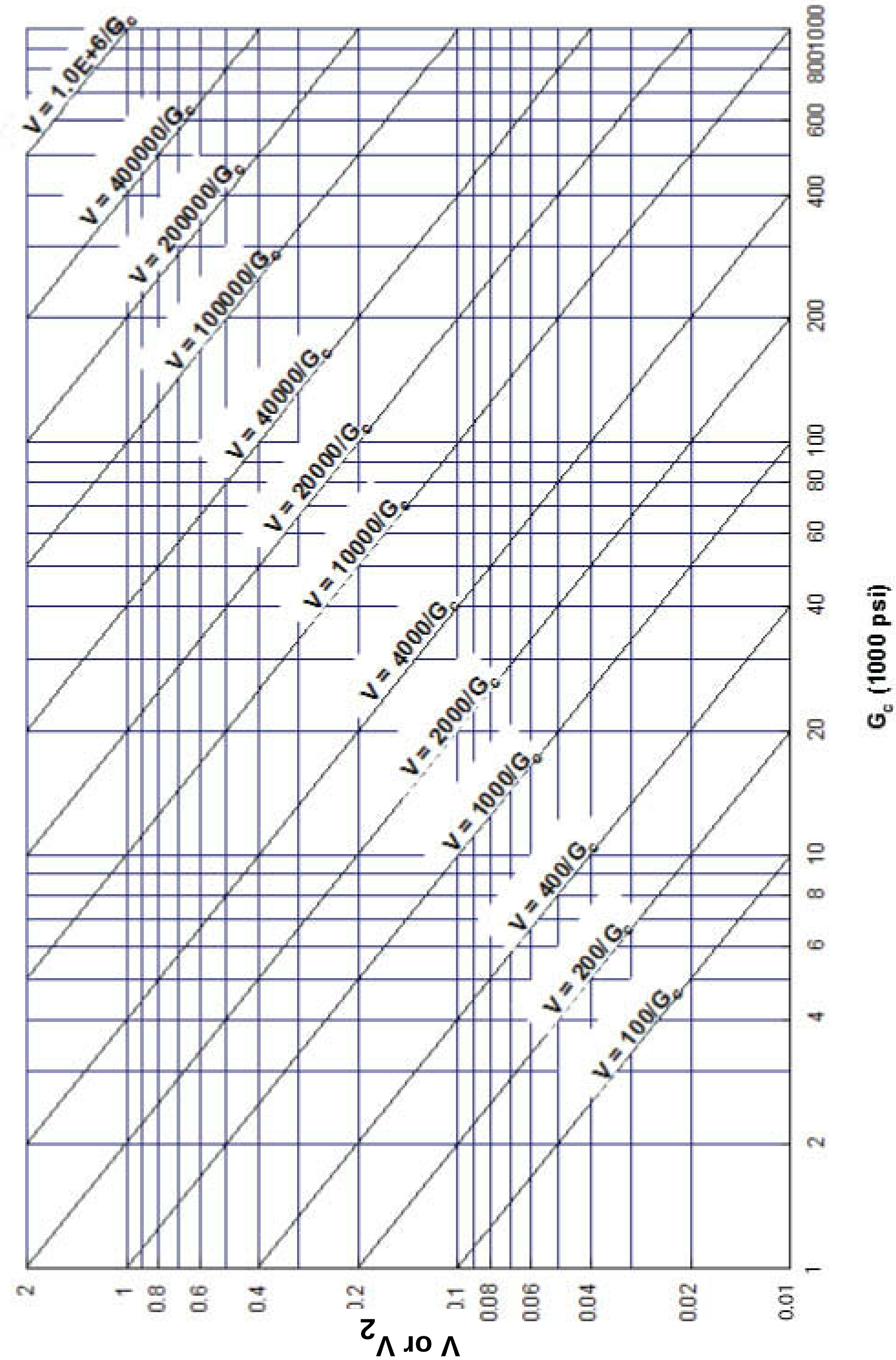


FIGURE 4.9.5.2.2 *Chart for determining V or V_2 and G_c for sandwich in edgewise bending load.*

4.9.5.3 Checking procedure for determining buckling stress, F_{cr}

The design shall be checked by using the graphs of Figures 4.9.5.3(a) through (d) to determine values of K_M for use in evaluating $K = K_F + K_M$ to substitute into Equation 4.9.5.2(a) to compute actual buckling stress, F_{cr}. The figures apply to sandwich panels with edges simply supported and with isotropic face sheets and isotropic or certain orthotropic cores (see Section 4.9.5.2).

For each value of the parameter V or V_2, there is a cusped curve giving values of K_M for various values of the ratios $\frac{a}{b}$ or $\frac{b}{a}$. These cusps are indicated by dotted lines for the top curve in each figure. The cusps show the sandwich panel buckling coefficients calculated for different values of n, the number of half waves into which the panel buckles. Only the portions of each cusped curve for which K_M is a minimum are shown. Envelope curves indicate values of K_M for use in design.

Values of K_F shall be determined by the equation

$$K_F = \frac{\left(E_{UPR}\, t_{UPR}^3 + E_{LWR}\, t_{LWR}^3\right)\left(E_{UPR}\, t_{UPR} + E_{LWR}\, t_{LWR}\right)K_{MO}}{12\, E_{UPR}\, t_{UPR}\, E_{LWR}\, t_{LWR}\, d^2} \qquad 4.9.5.3$$

$$K_F = \frac{t^2\, K_{MO}}{3d^2} \quad \text{(for equal face sheets)}$$

where $K_{MO} = K_M$ when $V = 0$ or $V_2 = 0$ and thus can be obtained the graphs of Figures 4.9.5.3(a) through (d). For panels with $\frac{a}{b}$ ratios ≥0.4, it can be assumed that $K_F = 0$. Then K shall be computed as $K = K_F + K_M = K_M$ and Equation 4.9.5.2(a) solved for F_{cr}.

If the charts do not apply because ratios of core shear moduli are far different from what is given on the charts, or it is desired to check by a more accurate analysis, the equations given in References 4.9.5.2(a) and (b) shall be used.

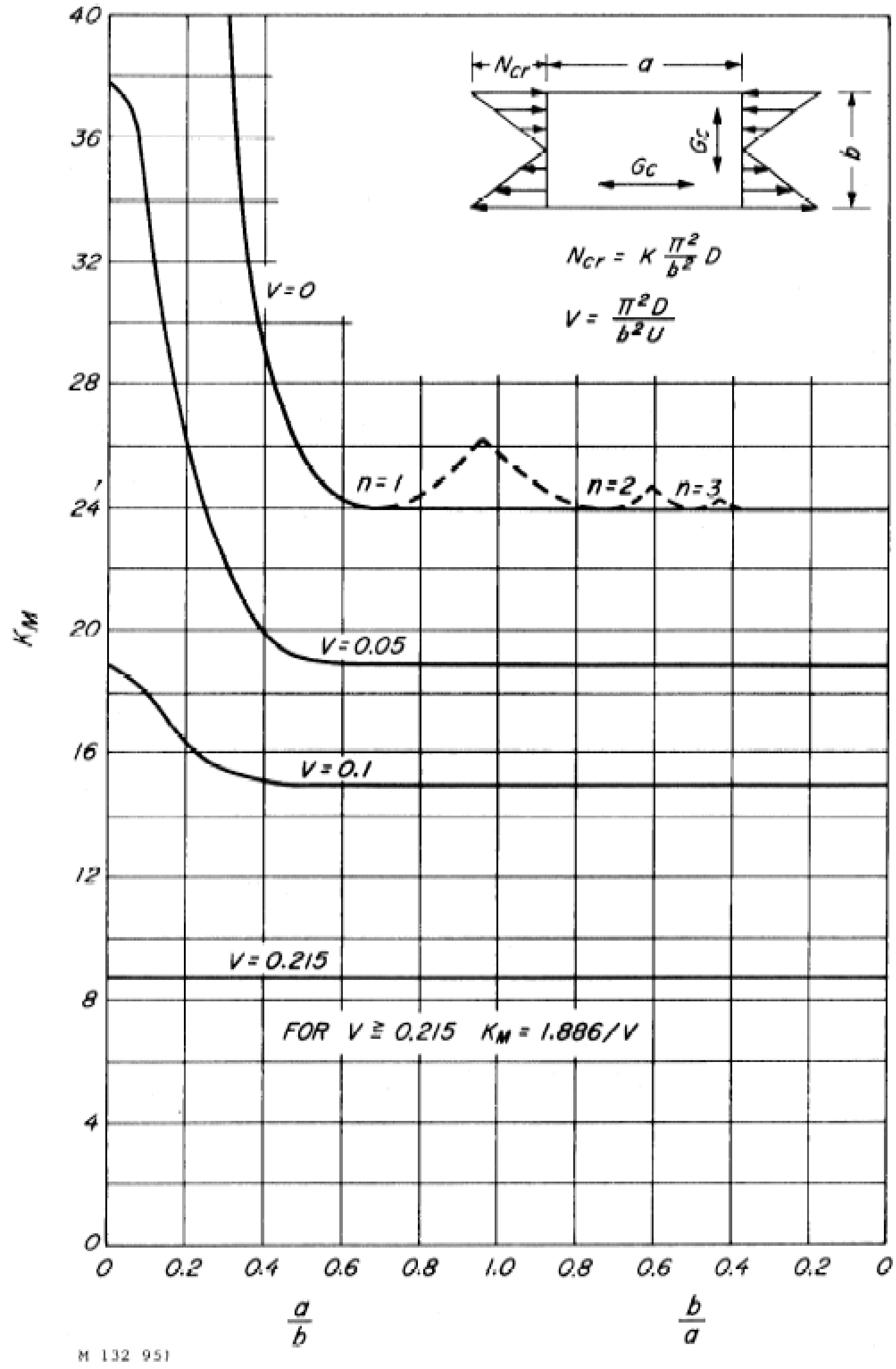


FIGURE 4.9.5.3(a) *Coefficient K_M for simply supported sandwich panels with isotropic core ($G_{cb} = G_{ca}$), under edgewise bending.*

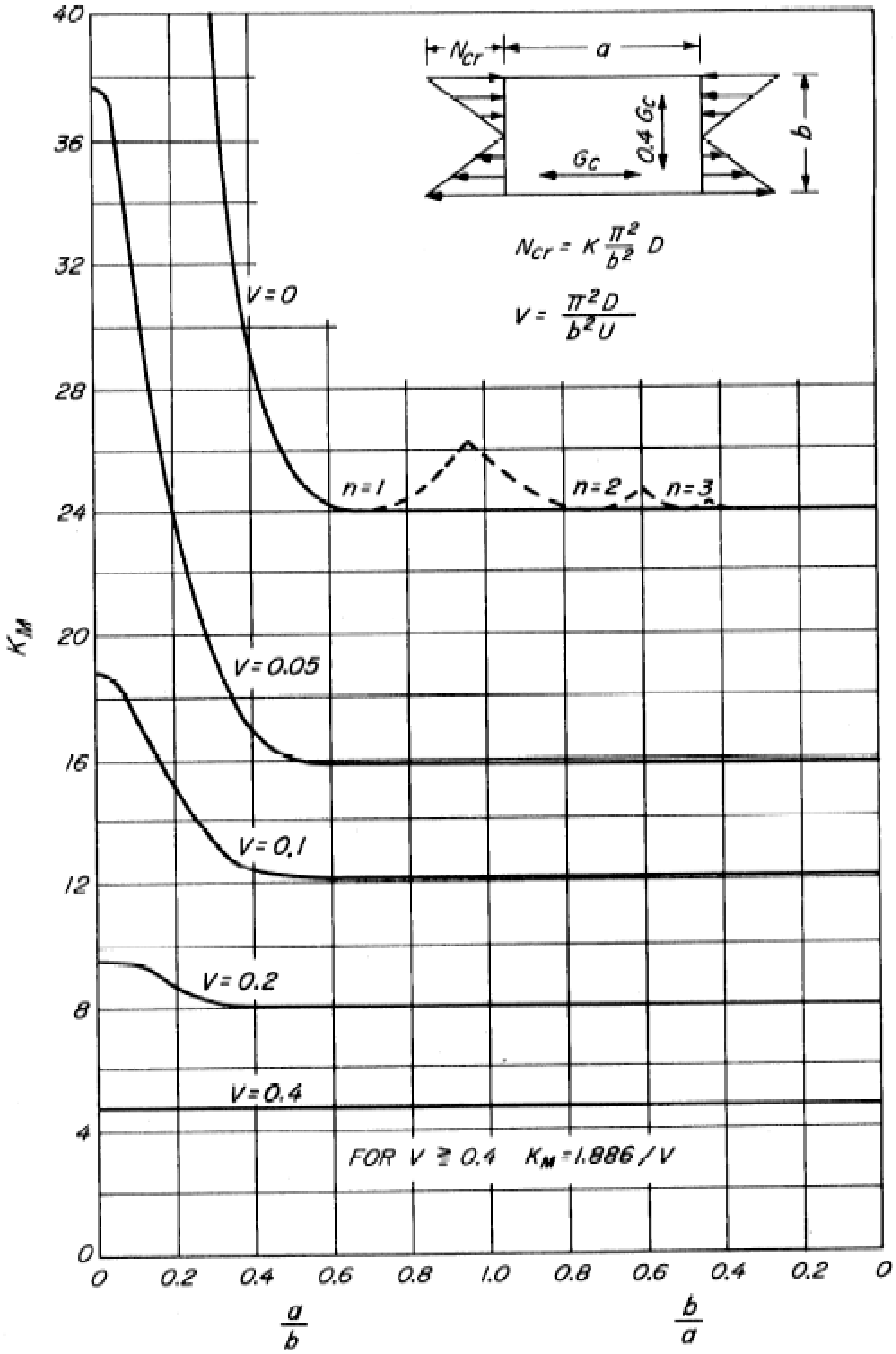


FIGURE 4.9.5.3(b) *Coefficient K_M for simply supported sandwich panels with orthotropic core ($G_{cb} = 0.4G_{ca}$) under edgewise bending.*

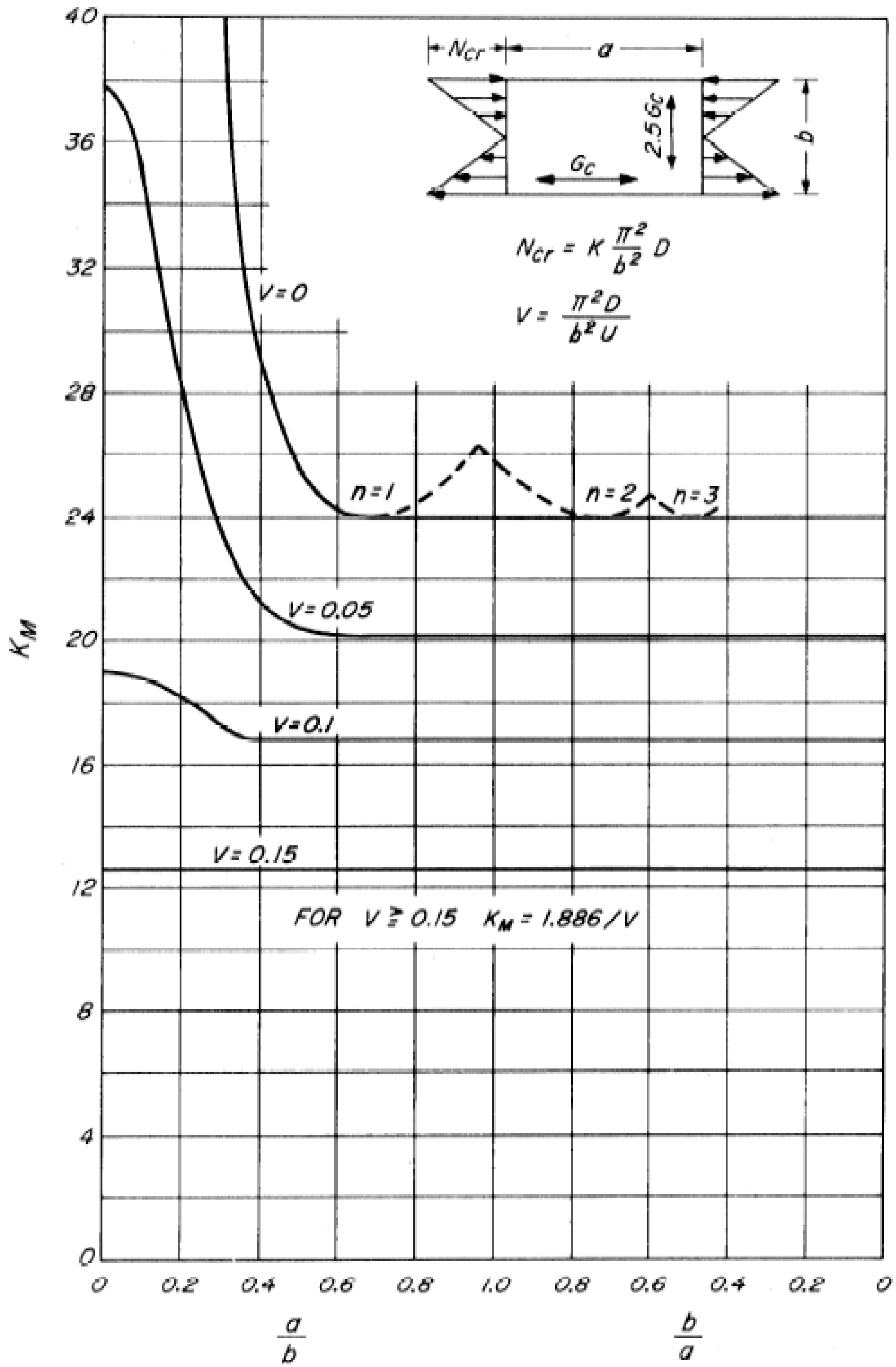


FIGURE 4.9.5.3(c) *Coefficient K_M for simply supported sandwich panels with orthotropic core ($G_{cb} = 2.5G_{ca}$), under edgewise bending.*

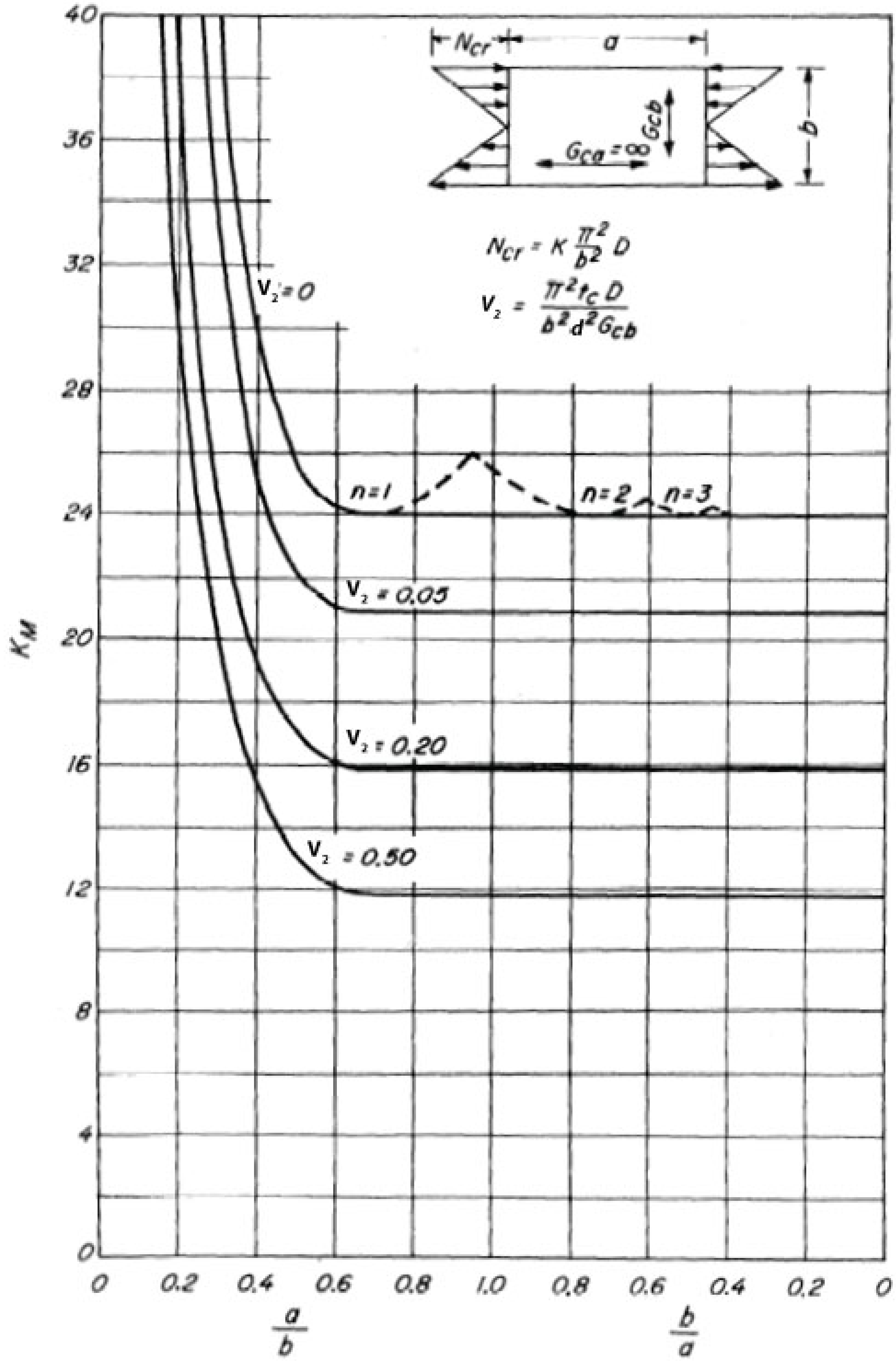


FIGURE 4.9.5.3(d) *Coefficient K_M for simply supported sandwich panels with corrugated core under edgewise bending. Core corrugation flutes parallel to side a.*

4.10 DESIGN OF FLAT RECTANGULAR SANDWICH PANELS UNDER COMBINED LOADS

Assuming that a design begins with chosen design stresses and a given design load to transmit, a flat rectangular panel of sandwich construction under edgewise loads, with or without loads directed normal to the plane of the panel, shall be designed to comply with the basic design principles summarized in Section 4.2.1. These conditions must be met. This section addresses global buckling. The other failure modes listed in Section 4.4 should be checked separately.

Face sheet stresses shall be determined for each load applied separately, and the effects of combining the loads and stresses shall be assessed by appropriate interaction equations for the face sheet materials, as given in References 4.10(a) and (b), wherein design values of these stresses are established.

Overall buckling of the sandwich, or local instabilities such as dimpling or wrinkling of the face sheets, may lead to total collapse of the panel. Equations for estimating intracell buckling (dimpling) and face sheet wrinkling under combined loads have been given in Equations 4.6.5.4 and 4.6.6.5(a), respectively.

Overall buckling of sandwich panels under combined loads is given by interaction equations in terms of ratios, R, of the applied stress or load under combined loading to the buckling stress or load under separate loading ($R = N/N_{cr}$). Appropriate subscripts are given to R to denote stress or load direction.

4.10.1 Combined load buckling

4.10.1.1 Biaxial compression

Overall buckling of sandwich panels under biaxial compression can be estimated by the interaction formula

$$R_{cx} + R_{cy} = 1 \qquad 4.10.1.1$$

This equation is correct for square, isotropic sandwich panels where shear stiffness is much larger than the bending stiffness ($V \approx 0$). It can be exceedingly conservative for long panels with negligible shear stiffness ($V >> 0$). V is a parameter that compares bending and shear rigidities, and is defined in Section 4.7.2.1. For more accurate analyses including sandwiches with corrugated core consult References 4.10.1.1(a) through 4.10.1.1(e).

4.10.1.2 Bending and compression

Overall buckling of sandwich panels under edgewise bending and compression can be estimated by the interaction formula

$$R_{cx} + \left(R_{bx}\right)^{3/2} = 1 \qquad 4.10.1.2$$

Approximate values obtained from this equation may be conservative. For more accurate analyses including sandwiches with corrugated core consult References 4.10.1.1(b), 4.10.1.1(e), and 4.10.1.2.

4.10.1.3 Compression and shear

Overall buckling of sandwich panels under edgewise compression and shear can be estimated by the interaction formula

$$R_c + \left(R_s\right)^2 = 1 \qquad 4.10.1.3$$

References 4.10.1.1 (b), (d), and (e) contain more complete information.

4.10.1.4 Bending and shear

Overall buckling of sandwich panels under edgewise bending and shear can be closely approximated by the interaction formula

$$\left(R_b\right)^2 + \left(R_s\right)^2 = 1 \qquad 4.10.1.4$$

Details of the analysis leading to these interaction curves are given in References 4.10.1.1(b) and 4.10.1.2.

4.10.2 Combined in-plane and transverse loads

The combination of edge loads with loads directed normal to the plane of a sandwich panel can greatly magnify deflections and stresses due to the normal load only. Design information for panels under normal load only is given in Section 4.7. The deflections and stresses under combined loads can be closely approximated by the formula

$$\psi = \frac{\psi_0}{1 - \dfrac{N}{N_{cr}}} \qquad 4.10.2$$

where Ψ is the deflection or stress due to edgewise load combined with normal load, ψ_0 is deflection or stress due to normal load only, N is edgewise loading (single or combined), and N_{cr} is overall edgewise buckling load (single or combined). Details concerning this equation are given in References 4.10.1.1(e) and 4.10.2.

4.11 DESIGN OF SANDWICH CYLINDERS

4.11.1 Introduction

Assuming that a design begins with chosen design stresses and a given external loading, a circular cylinder with walls of sandwich construction shall be designed to comply with the basic design principles summarized in Section 4.2.1. These conditions must be met. In addition, if the cylinder is extremely long, it shall have sufficient bending stiffness so that sideways buckling will not occur. The other failure modes listed in Section 4.4 should be checked separately.

Overall buckling of the sandwich, or local instabilities such as dimpling or wrinkling of the face sheets, may lead to total collapse of the cylinder. Detailed procedures, theoretical equations, and graphs for determining dimensions of the face sheets and core, as well as necessary core properties, are given in the following paragraphs. Double equations are given, one equation for sandwich with face sheets of different materials and thicknesses and another for sandwich with both face sheets of the same material and of equal thickness.

This section covers sandwich cylinders under different types of loading: external radial pressure in Section 4.11.2, torsion in Section 4.11.3, axial compression or bending in Section 4.11.4, and combined loads in Section 4.11.5.

4.11.2 Sandwich cylinders under external radial pressure

The face sheet modulus of elasticity E', and the stress values, F_c, shall be compression values at the conditions of use, that is, if application is at elevated temperature, then the face sheet properties at elevated temperature shall be used in design. The face sheet modulus of elasticity is the effective value at the face sheet stress. If the stress Is beyond the proportional limit value, an appropriate tangent, reduced or modified compression modulus of elasticity shall be used (Reference 4.11.2).

4.11.2.1 Determining face sheet thickness, core thickness, and core shear modulus for sandwich cylinders under external radial pressure.

This section presents equations and a design procedure to determine the sandwich face sheet thickness, core thickness, and core shear modulus such that overall buckling of a sandwich cylinder will not occur at the chosen face sheet design stresses. The equations and procedure presented apply to sandwich cylinders having face sheets of isotropic materials and an isotropic or orthotropic core. Ends of the cylinder are assumed to be simply supported on rigid plates that hold the ends circular. The face sheet stresses are related to the applied external radial pressure (no axial load) by the equation:

$$t_{UPR}\ F_{C\,UPR} + t_{LWR}\ F_{C\,LWR} = r\,q \qquad 4.11.2.1(a)$$

$$t = \frac{r\,q}{2F_c} \quad \text{(for equal face sheets)}$$

where t is the face sheet thickness, F_c is the chosen design hoop compressive face sheet stress, q is the design value of the external radial pressure, r is the mean radius of the cylinder, and UPR, LWR are subscripts denoting the upper and lower face sheets, respectively. If the core can support hoop compression loads, $r\,q$ should be replaced by $(rq - F_{c\ core}\ t_c)$, where $F_{c\ core}$ is the core stress in the hoop direction.

In determining thickness of face sheets for a sandwich with face sheets of different materials, Equation 4.11.2.1(a) must be satisfied. In addition, to avoid overstressing either face sheet, the design stresses $F_{c\ UPR}$ and $F_{c\ LWR}$ must be chosen so that $F_{c\ UPR}/E'_{UPR} = F_{c\ LWR}/E'_{LWR}$ (where E' is the effective compression modulus of elasticity of the face sheet, and beyond the proportional limit this should be taken as the secant modulus). For example, if the upper face sheet is a material such that the ratio $F_{c\ UPR}/E'_{UPR} = 0.005$ and the lower face sheet is a material such that the ratio $F_{c\ LWR}/E'_{LWR} = 0.002$, then the design must be based on a ratio of 0.002; otherwise the lower face sheet will be overstressed. In order to accomplish this, the chosen design stress for the upper face sheet must be reduced to 0.002 E'_{UPR}. For many combinations of materials, it will be found advantageous to choose thicknesses such that $E'_{UPR}\,t_{UPR} = E'_{LWR}\,t_{LWR}$.

The load per unit of length due to an applied external radial pressure at which buckling of a sandwich cylinder occurs is given by the theoretical equation (Reference 4.11.2.1(a)).

$$r\,q = \left(\frac{E'_{UPR}\,t_{UPR}}{\lambda_{UPR}} + \frac{E'_{LWR}\,t_{LWR}}{\lambda_{LWR}}\right)K \qquad 4.11.2.1(b)$$

$$r\,q = \left(\frac{2E'\,t}{\lambda}\right)K$$

where E' is the effective compression modulus of elasticity of the face sheets, $\lambda = 1 - \nu^2$, ν is Poisson's ratio of the face sheets, and K is a theoretical coefficient. Combining Equations 4.11.2.1(a) and (b), and requiring that the strain is the same in the two face sheets results in

$$K = \frac{F_{cUPR}\lambda_{UPR}}{E'_{UPR}} \frac{\left[E'_{UPR}\, t_{UPR} + E'_{LWR}\, t_{LWR}\right]}{\left[E'_{UPR}\, t_{UPR} + \left(\frac{\lambda_{UPR}}{\lambda_{LWR}}\right) E'_{LWR}\, t_{LWR}\right]} \qquad 4.11.2.1(c)$$

If the face sheets have the same Poisson's Ratio, ν, and thus the same λ, then this simplifies to

$$K = \frac{F_{cUPR}\lambda}{E'_{UPR}} = \frac{F_{cLWR}\lambda}{E'_{LWR}}$$

Since the strains are the same in the two face sheets, K may be related to stress and modulus in either the upper or lower face sheet. If the two face sheets are equal (same modulus, Poisson's ratio, and thickness):

$$\frac{F_c\lambda}{E'} = K \qquad \text{(for equal face sheets)}$$

The coefficient K is dependent upon cylinder dimensions and sandwich bending and shear rigidities. Convenient nondimensional parameters for determining K are d/r, L/r, $E'_{UPR}\, t_{UPR}/E'_{LWR}\, t_{LWR}$ and $V = D/r^2S$ where d is the distance between face sheet centroids, L is cylinder length, D is sandwich bending stiffness, and S is sandwich shear stiffness. For a cylinder where the face sheets are thin compared to the core ($tc \approx d$):

$$D = \frac{E'_{UPR}\, t_{UPR}\, E'_{LWR}\, t_{LWR}\, d^2}{\left(E'_{UPR}\, t_{UPR} + E'_{LWR}\, t_{LWR}\right)\lambda}$$

$$S = \frac{G_c\, d^2}{t_c}$$

Substitution of expressions for D and S into the parameter V results in

$$V = \frac{D}{r^2 S}$$

$$V = \frac{E'_{UPR}\, t_{UPR}\, E'_{LWR}\, t_{LWR}\, d}{\left(E'_{UPR}\, t_{UPR} + E'_{LWR}\, t_{LWR}\right)\lambda r^2 G_c} \qquad 4.11.2.1(d)$$

$$V = \frac{E' t d}{2\lambda r^2 G_c} \qquad \text{(for equal face sheets)}$$

where G_c is the core shear modulus associated with shear distortion in the radial and circumferential directions.

A minimum value of the required distance between face sheet centroids, d, will be determined by assuming $V = 0$ for a first approximation. This value of d is minimum because $V = 0$ only if the core shear modulus is infinite. For any actual core the shear modulus is not infinite; hence a thicker core must be used.

The chart of Figure 4.11.2.1 gives minimum values of d/r for sandwich with isotropic face sheets. Parameters needed for use of the chart are:

1. Face sheet properties $\frac{F_{c\,UPR}\lambda_{UPR}}{E'_{UPR}}$ and $\frac{F_{c\,LWR}\lambda_{LWR}}{E'_{LWR}}$
2. Cylinder length-to-radius ratio L/r
3. Ratio of $\frac{E'_{UPR}\ t_{UPR}}{E'_{LWR}\ t_{LWR}}$

From the value of d the core thickness is computed by the formula

$$t_c = d - \frac{t_{UPR} + t_{LWR}}{2} \qquad 4.11.2.1(e)$$

$t_c = d - t$ (for equal face sheets)

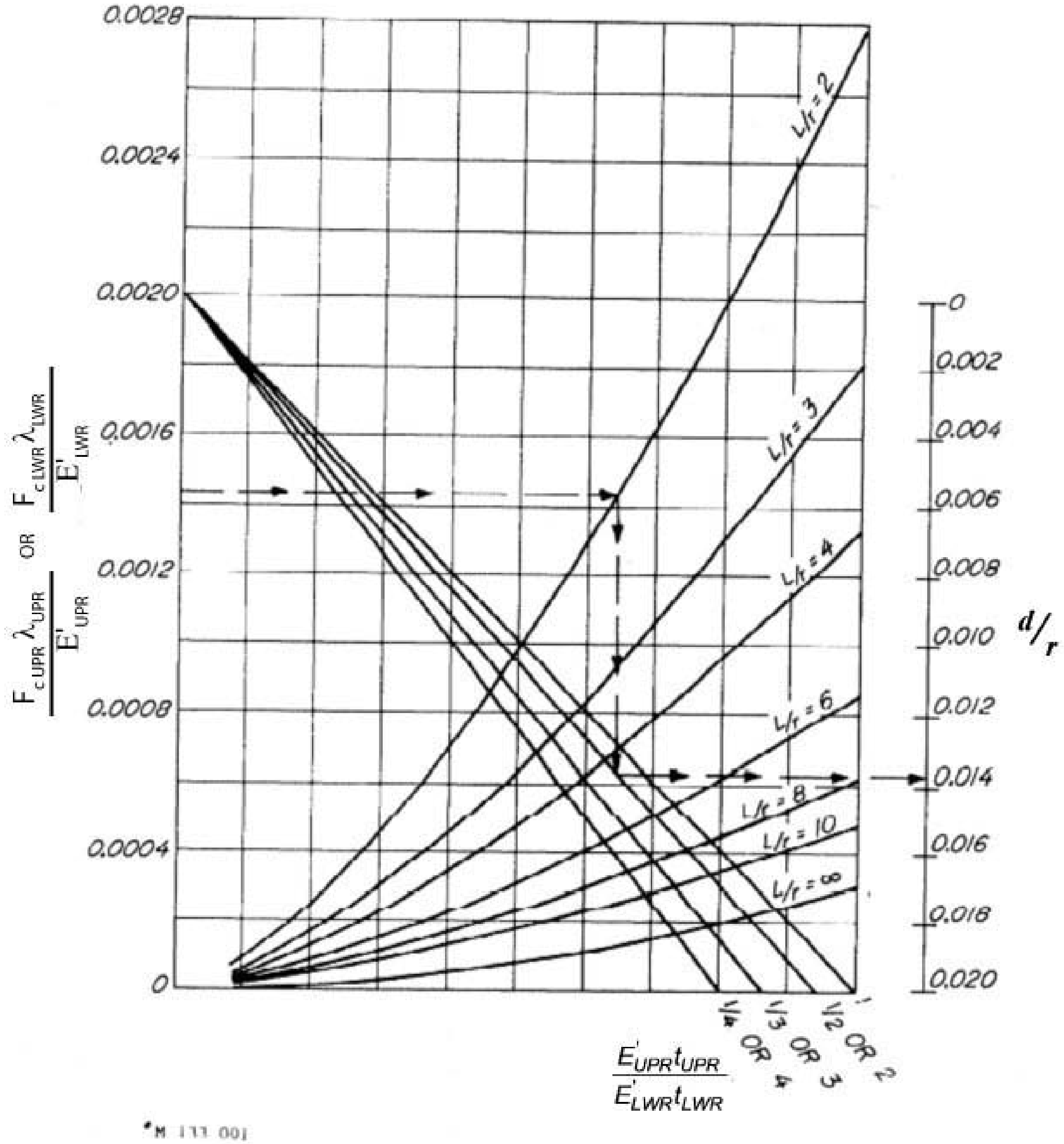


FIGURE 4.11.2.1 *Chart for determining minimum d/r ratio (V = 0) such that the walls of a sandwich cylinder with isotropic face sheets will not buckle under external radial pressure (no axial load).*

4.11.2.2 Final design

The final sandwich design is arrived at by assuming a slightly thicker core than determined above and using checking curves of Figure 4.11.2.2(a) for V = 0, Figure 4.11.2.2(b) for V = 0.05, and Figure 4.11.2.2(c) for V = 0.10. The final design shall be based on a buckling coefficient of 0.95 times the values given by Figures 4.11.2.2(a) through 4.11.2.2(c) (Reference 4.11.2.2(a)). Several iterations may be necessary because the parameter V is dependent upon sandwich thickness and core shear modulus. Interpolation for values of V other than those given in the figures can be done graphically.

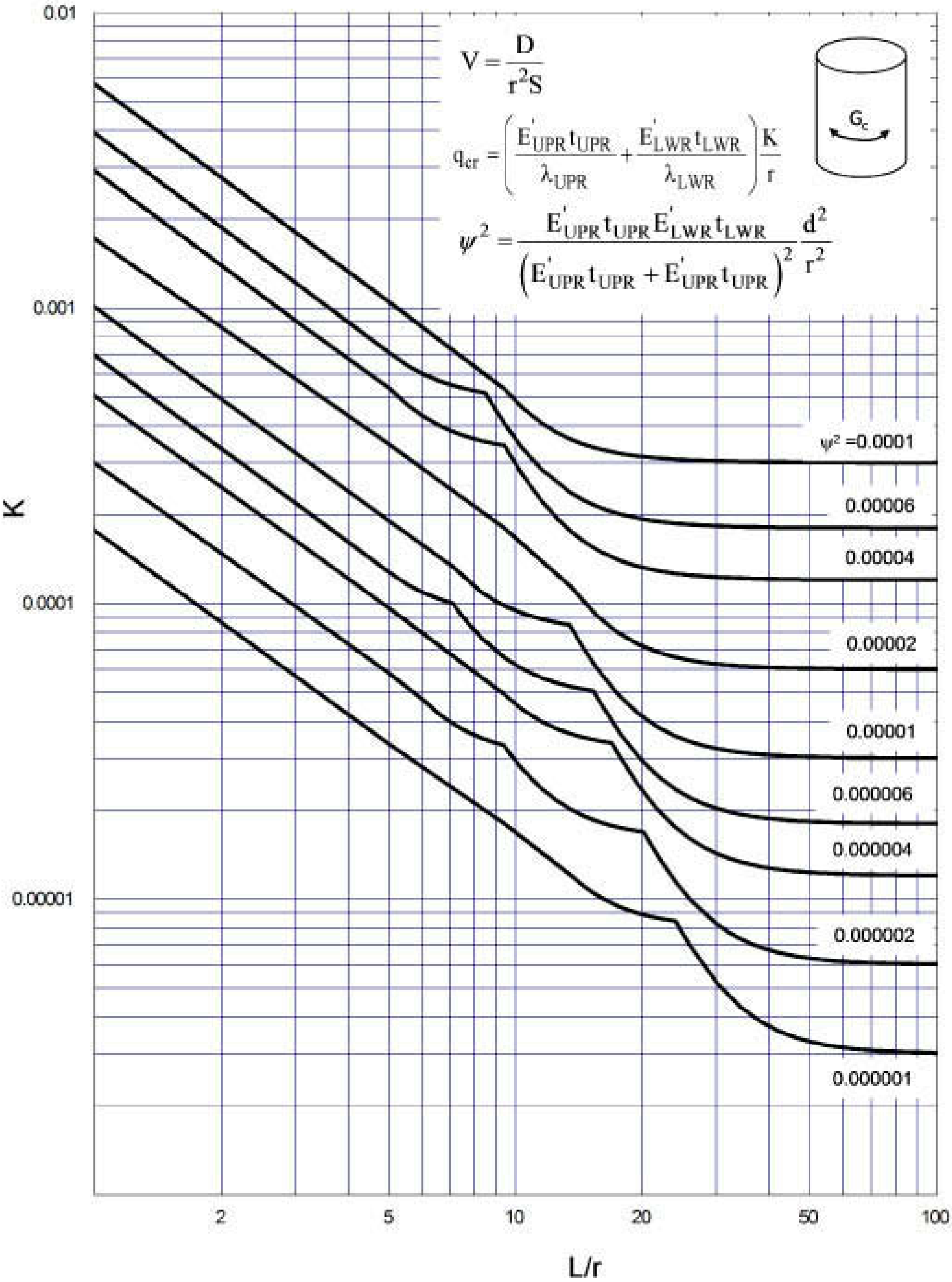
If a more accurate determination of K is desired, the approximate equation given in Equation 4.11.2.2 can be solved (Reference 4.11.2.1(a)). This equation is

$$K = \frac{\psi^2\left(n^2-1\right)\left(3+\dfrac{n^2L^2}{\pi^2r^2}\right)\left[\left(\dfrac{n^2L^2}{\pi^2r^2}-\dfrac{1}{3}\right)\left(n^2-1+\dfrac{\pi^2r^2}{L^2}\right)-\dfrac{2}{3}\right]+\dfrac{8}{9}\left[1+\left(n^2+\dfrac{\pi^2r^2}{3L^2}\right)V\right]}{\left[\left(\dfrac{n^2L^2}{\pi^2r^2}+1\right)^2\left(n^2-1\right)+\dfrac{1}{3}\right]\left[1+\left(n^2+\dfrac{\pi^2r^2}{3L^2}\right)V\right]} \qquad 4.11.2.2$$

where $\psi^2 = \dfrac{E'_{UPR}t_{UPR}E'_{LWR}t_{LWR}d^2}{\left(E'_{UPR}t_{UPR}+E'_{LWR}t_{LWR}\right)^2 r^2}$ or $\psi^2 = \dfrac{d^2}{4r^2}$ for equal face sheets.

Values of the number of circumferential buckles, n, are chosen to produce minimum values of K. This approximate equation does not contain terms with core shear moduli in the radial-axial directions because these terms have little influence on cylinders longer than about one diameter. Thus the curves given approximate the behavior of cylinders with orthotropic as well as isotropic cores.

The family of curves on Figures 4.11.2.2(a) can be approximated very closely by a single curve for cylinders of short and moderate length. This single curve shown in Figure 4.11.2.2(d) is obtained by modifying the coordinate axes of Figure 4.11.2.2(a). For long cylinders, the single curve branches into a family of lines of steeper slope, as shown in the upper right hand portion of Figure 4.11.2.2(d). This family of lines is dependent upon the ratio r/d, as well as the graph abscissa. If the value of the abscissa is such that the branched lines are shown, the value of the ordinate shall be picked from the branched lines rather than the bottom straight line. Thus, for an abscissa of 10^5 and a value of 100 for the parameter $\dfrac{\left(E'_{UPR}\,t_{UPR}+E'_{LWR}\,t_{LWR}\right)r\sqrt{\lambda}}{d\sqrt{E'_{UPR}\,t_{UPR}\,E'_{LWR}\,t_{LWR}}}$, the proper value of the ordinate to produce the least buckling pressure would be 305 (upper branch), not 164 (bottom straight line).



$$V = \frac{D}{r^2 S}$$

$$q_{cr} = \left(\frac{E'_{UPR} t_{UPR}}{\lambda_{UPR}} + \frac{E'_{LWR} t_{LWR}}{\lambda_{LWR}} \right) \frac{K}{r}$$

$$\psi^2 = \frac{E'_{UPR} t_{UPR} E'_{LWR} t_{LWR}}{\left(E'_{UPR} t_{UPR} + E'_{UPR} t_{UPR} \right)^2} \frac{d^2}{r^2}$$

FIGURE 4.11.2.2(a) *Buckling coefficient K for sandwich cylinder under external radial pressure. Isotropic face sheets; isotropic or orthotropic core; V = 0.0.*

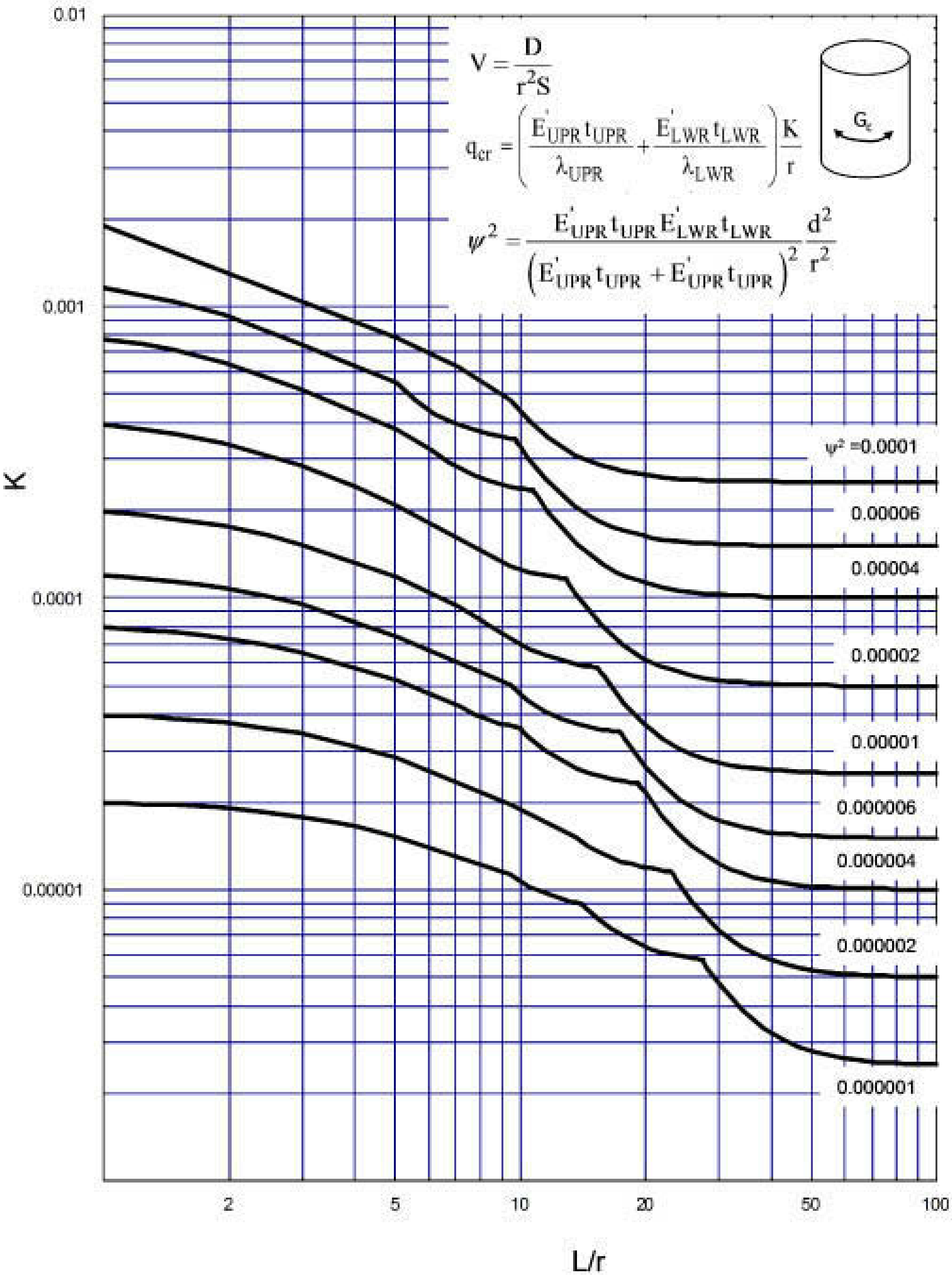


FIGURE 4.11.2.2(b) *Buckling coefficient K for sandwich cylinder under external radial pressure. Isotropic face sheets; isotropic or orthotropic core; V = 0.05.*

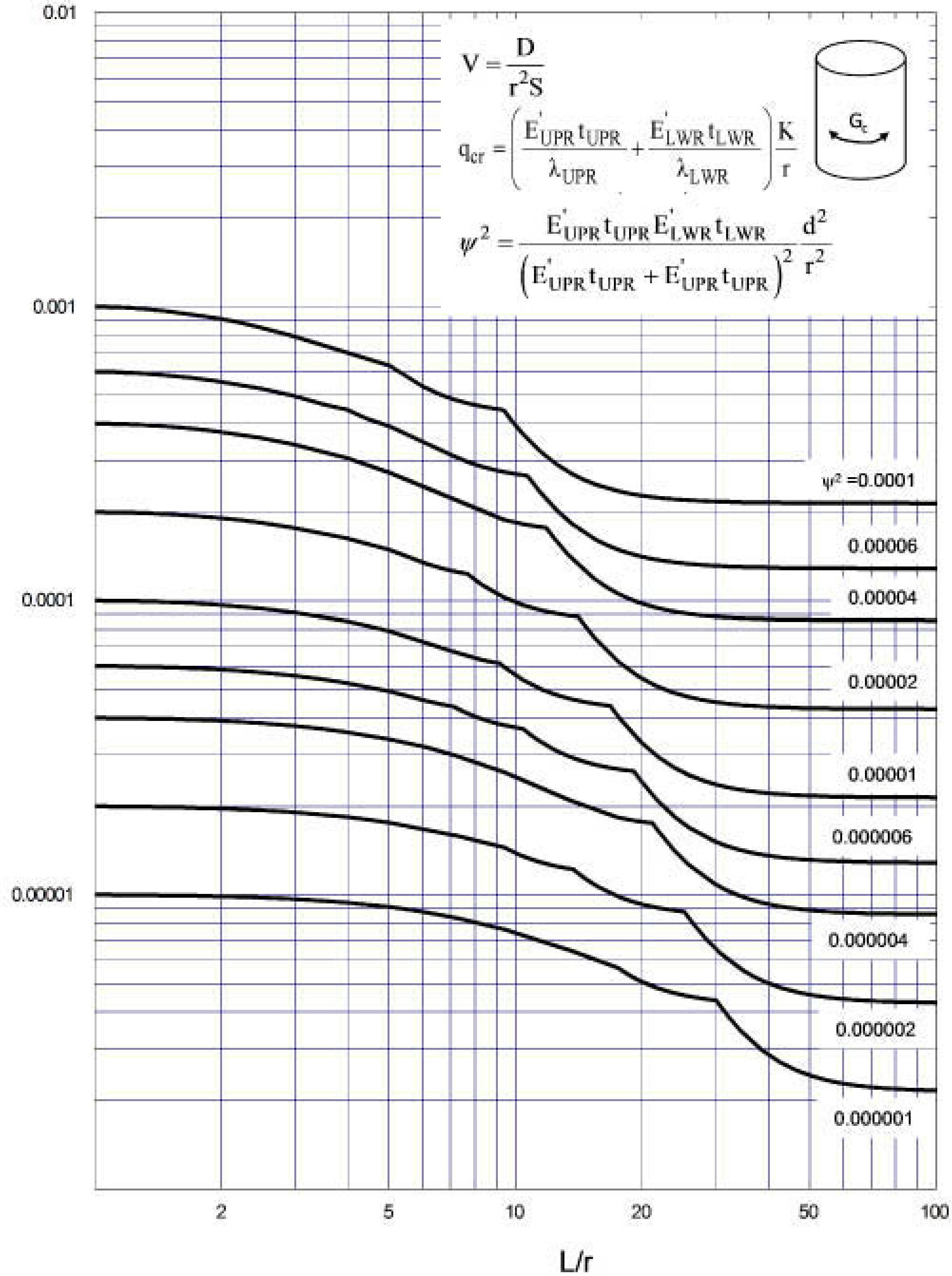


FIGURE 4.11.2.2(c) *Buckling coefficient K for sandwich cylinder under external radial pressure. Isotropic face sheets; isotropic or orthotropic core; V = 0.10 .*

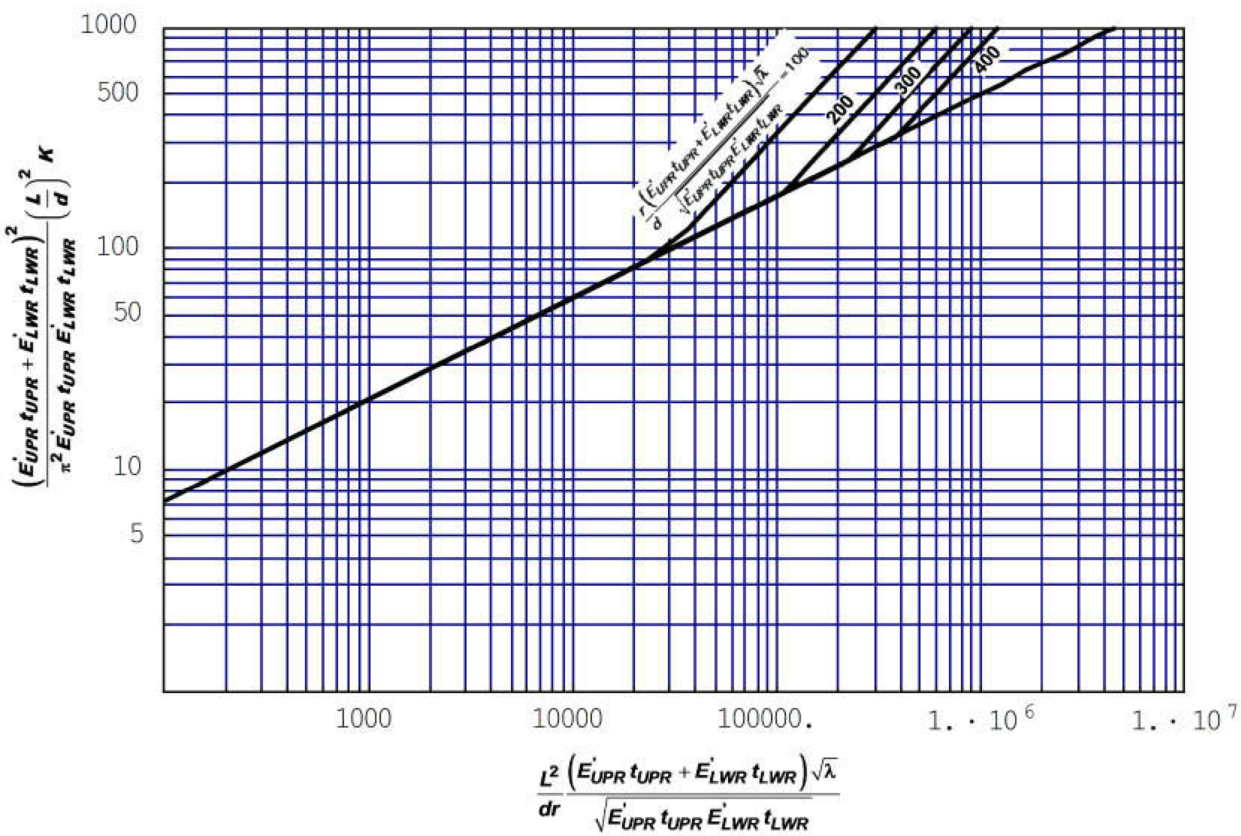


FIGURE 4.11.2.2(d) *Modified buckling coefficient for sandwich cylinders under external radial pressure. Isotropic face sheets; isotropic or orthotropic core; V = 0.*

4.11.3 Sandwich cylinders under torsion

The face sheet modulus of elasticity E', shear modulus G', and the stress values, F_s, shall be values at the conditions of use. That is, if application is at elevated temperature, then the face sheet properties at elevated temperature shall be used in design. The effective elastic modulus shall be the lower of either the compressive or tensile value of the face sheet material in a direction 45° to the cylinder axis. (The compression and tension stresses, F_c and F_t, are equal to the shear stress, F_s for an isotropic tube in torsion.) If the stress is beyond the proportional limit values, an appropriate tangent, reduced or modified compression modulus of elasticity shall be used.

4.11.3.1 Determining face sheet thickness for sandwich cylinders under torsion

The face sheet stresses are related to the applied external torsion load by the equation:

$$t_{UPR} F_{S,UPR} + t_{LWR} F_{S,LWR} = N \qquad \text{4.11.3.1(a)}$$

$$t = \frac{N}{2F_s} \quad \text{(for equal face sheets)}$$

where t is the face sheet thickness, F_S is the chosen design face sheet shear stress, and UPR, LWR are subscripts denoting the upper and lower face sheets, respectively. N is determined from the design torque, T, by

$$N = \frac{T}{2\pi r^2} \qquad 4.11.3.1(b)$$

where r is the mean radius of curvature of the cylinder wall.

In determining thickness of face sheets for a sandwich with face sheets of different materials, the Equation 4.11.3.1(a) must be satisfied. In addition, to avoid overstressing either face sheet, the design stresses $F_{S\,UPR}$ and $F_{S\,LWR}$ must be chosen so that $F_{S\,UPR}/G_{S\,UPR} = F_{S\,LWR}/G_{S\,LWR}$ (where G_S is the shear modulus of the face sheet, and beyond the proportional limit this should be taken as the secant shear modulus). For example, if the upper face sheet is a material such that the ratio $F_{S\,UPR}/G_{S\,UPR} = 0.005$ and the lower face sheet is a material such that the ratio $F_{S\,LWR}/G_{S\,LWR} = 0.002$, then the design must be based on a ratio of 0.002; otherwise the lower face sheet will be overstressed. In order to accomplish this, the chosen design stress for the upper face sheet must be reduced to 0.002 $G_{S\,UPR}$. For many combinations of materials, it will be found advantageous to choose thicknesses such that $G_{S,UPR}\, t_{UPR} = G_{S,LWR}\, t_{LWR}$ or $E'_{UPR}\, t_{UPR} = E'_{LWR}\, t_{LWR}$.

If the cylinder is long and slender and the radius is limited, the face sheet thicknesses may be increased in order to prevent sideways buckling as covered by Section 4.11.3.3.

4.11.3.2 Determining core thickness and core shear modulus for sandwich cylinders under torsion

This section gives procedures for determining core thickness and core shear modulus so that overall buckling of the sandwich walls of the cylinder will not occur (References 4.11.3.2(a) through 4.11.3.2(c)). The face sheet stress at buckling is given by the formula:

$$\begin{aligned} F_{s\,UPR} &= KE'_{UPR}\frac{h}{r} \\ F_{s\,LWR} &= KE'_{LWR}\frac{h}{r} \qquad 4.11.3.2(a) \\ F_s &= KE'\frac{h}{r} \qquad \text{(for equal face sheets)} \end{aligned}$$

where E' is the effective elastic modulus at stress F_s, h is the sandwich thickness, r is the radius of curvature, and K is a theoretical buckling coefficient dependent on cylinder dimensions and sandwich bending and shear rigidities.

Values of the coefficient K are given by ordinates of the curves in the upper portion of Figures 4.11.3.2(a) through 4.11.3.2(f) for sandwich with isotropic face sheets and isotropic or orthotropic cores. Determination of the coefficients was based on the assumption that the Poisson's ratio of the face sheets was 0.25. Figures 4.11.3.2(a) through 4.11.3.2(c) are for sandwich with thin face sheets (t_c/h = 1) and Figures 4.11.3.2(d) through 4.11.3.2(f) for sandwich with moderately thick face sheets (t_c/h = 0.7). Figures 4.11.3.2(a) and (d) have isotropic core, Figures 4.11.3.2(b) and (e) have orthotropic core with the core shear modulus in the cylinder hoop direction 0.4 times that in the cylinder axial direction, and Figures 4.11.3.2(c) and (f) have orthotropic core with the core shear modulus in the hoop direction 2.5 times that in the axial direction. The curves give approximate values for cylinders of sandwich construction with corrugated cores. More accurate data for such sandwich cylinders are given in References 4.11.3.2(a) and (b).

Final design values of K shall be 0.75 times the values given by Figures 4.11.3.2(a) through 4.11.3.2(f) (Reference 4.11.3.2(d)).

Figure 4.11.3.2	Core	Core Shear Ratio *	t_ch
(a)	Isotropic	1.0	1.0
(b)	Orthotropic	0.4	1.0
(c)	Orthotropic	2.5	1.0
(d)	Isotropic	1.0	0.7
(e)	Orthotropic	0.4	0.7
(f)	Orthotropic	2.5	0.7

* Core Shear Ratio = Shear modulus in hoop direction / Shear modulus in axial direction

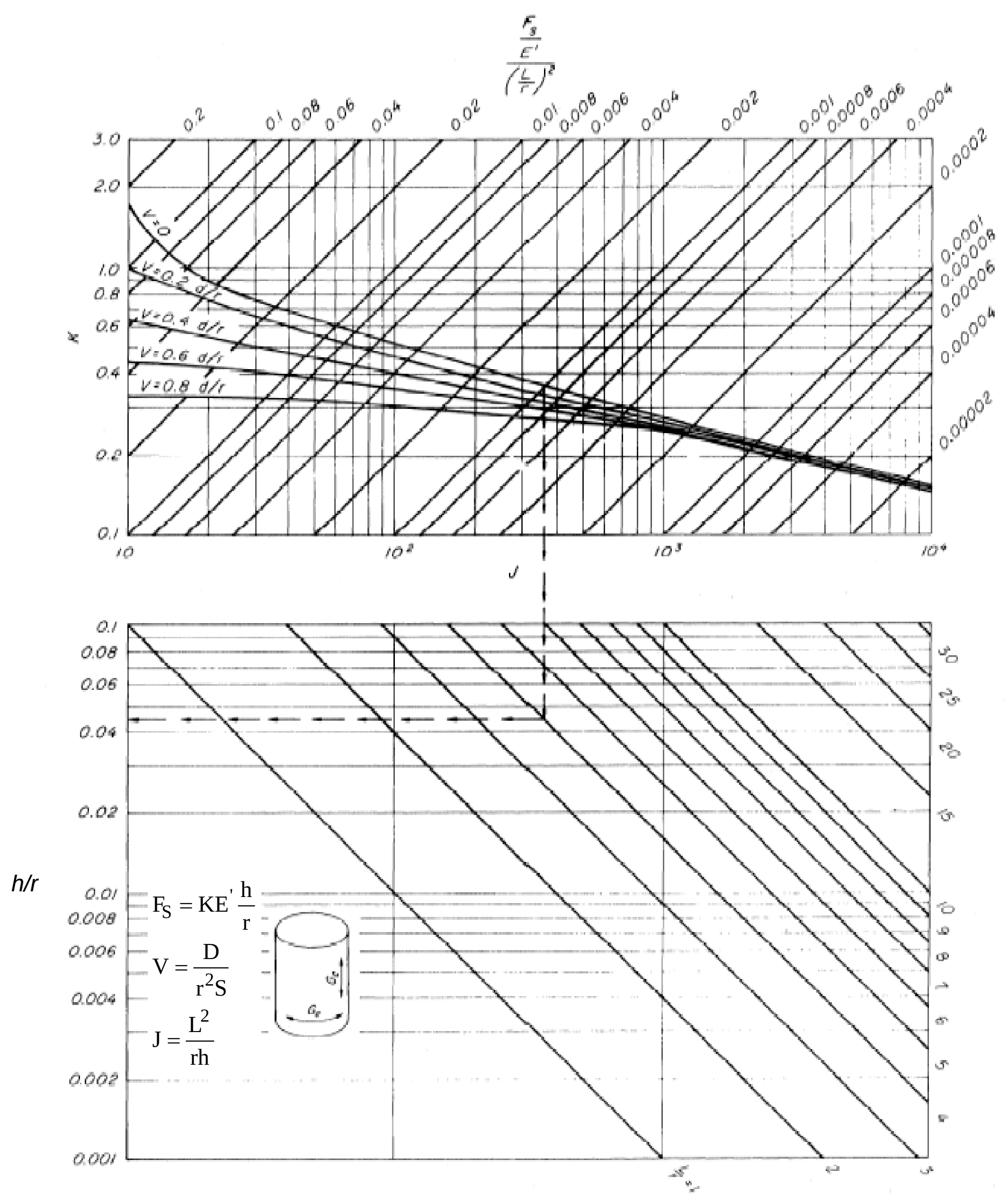


FIGURE 4.11.3.2(a) *Chart for determining h/r ratio such that a sandwich cylinder with isotropic core will not buckle; t_c/h = 1.0.*

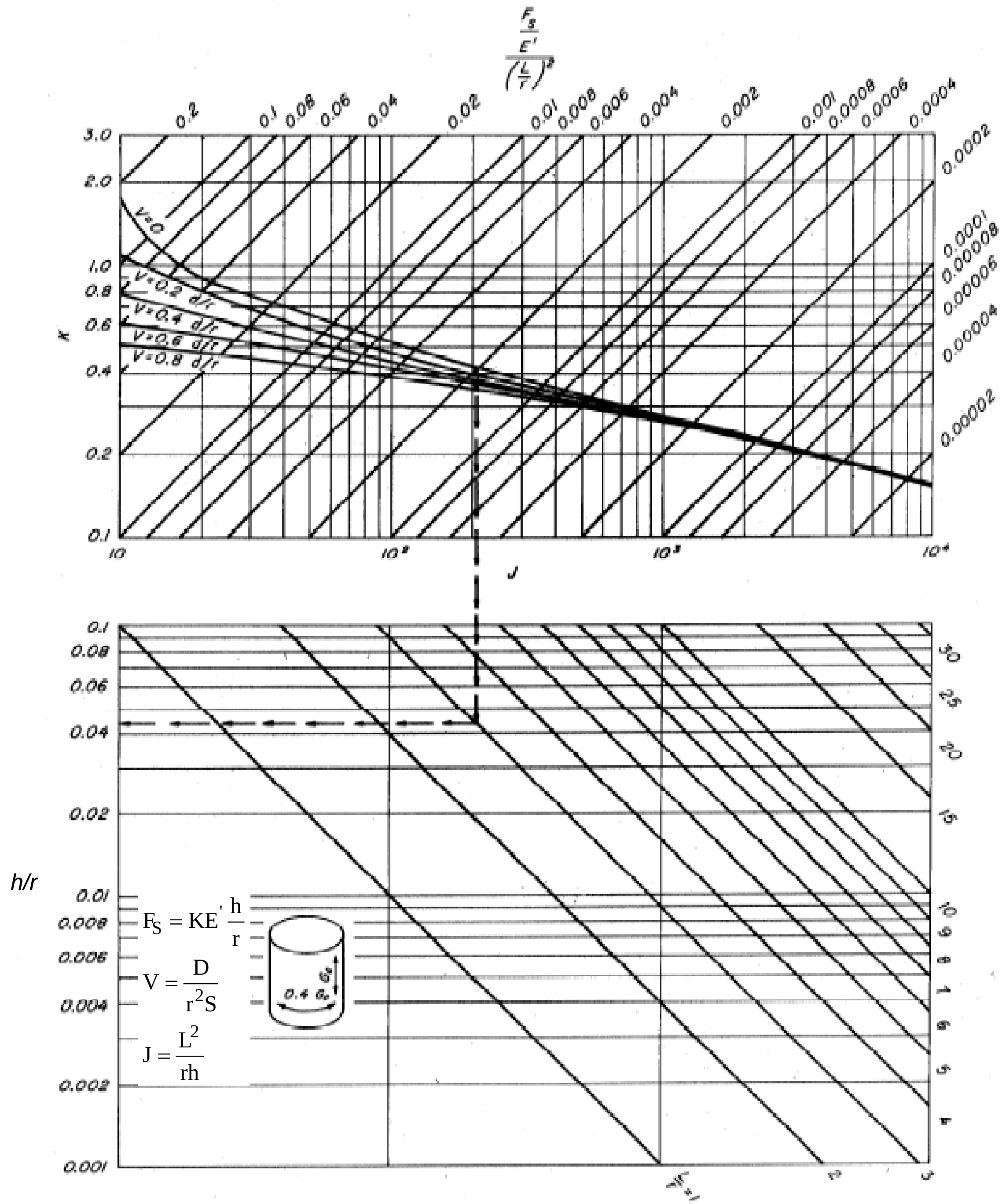


FIGURE 4.11.3.2(b) *Chart for determining h/r ratio such that a sandwich cylinder with orthotropic core will not buckle; $t_c/h = 1.0$.*

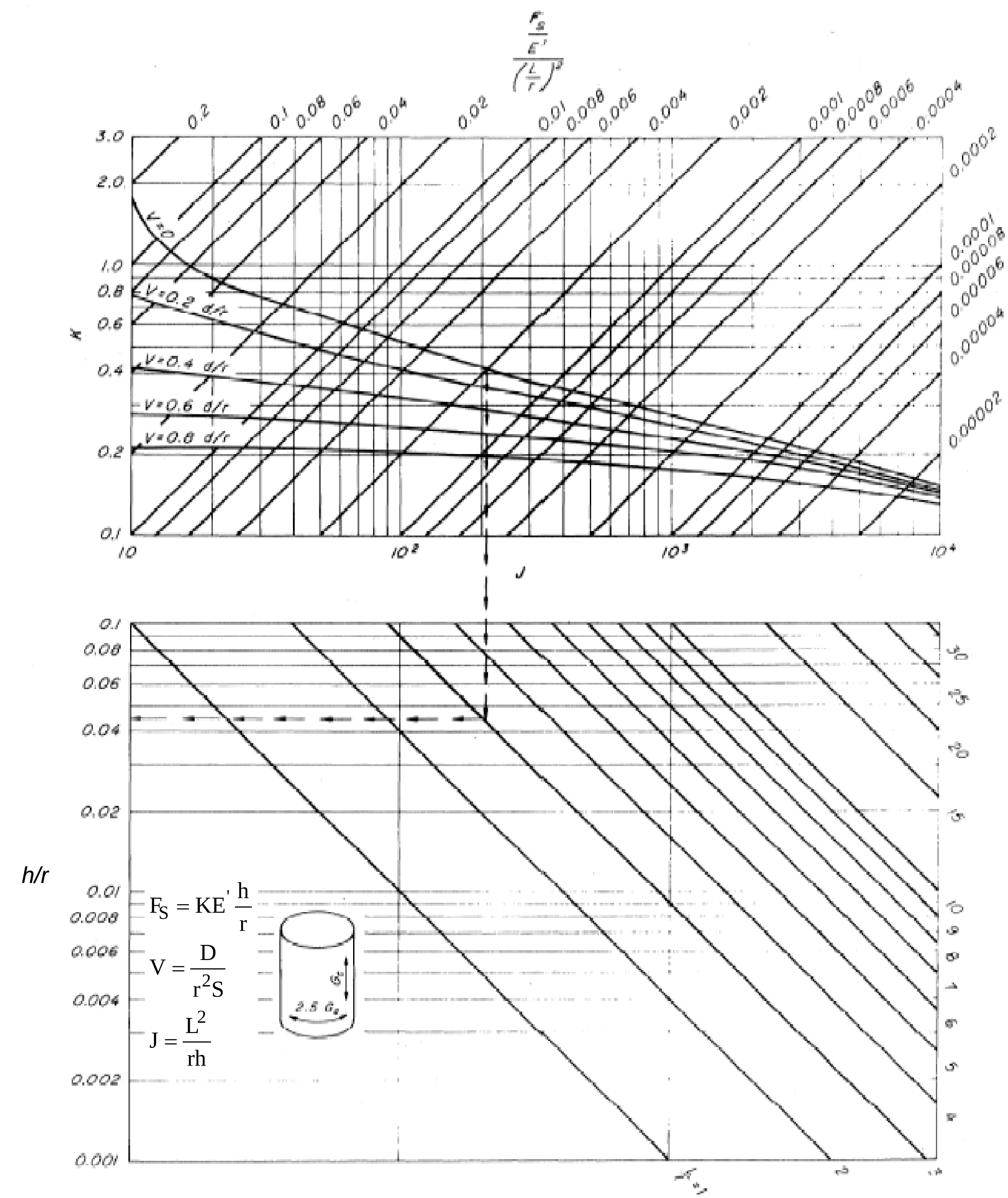


FIGURE 4.11.3.2(c) *Chart for determining h/r ratio such that a sandwich cylinder with orthotropic core will not buckle; t_c/h = 1.0.*

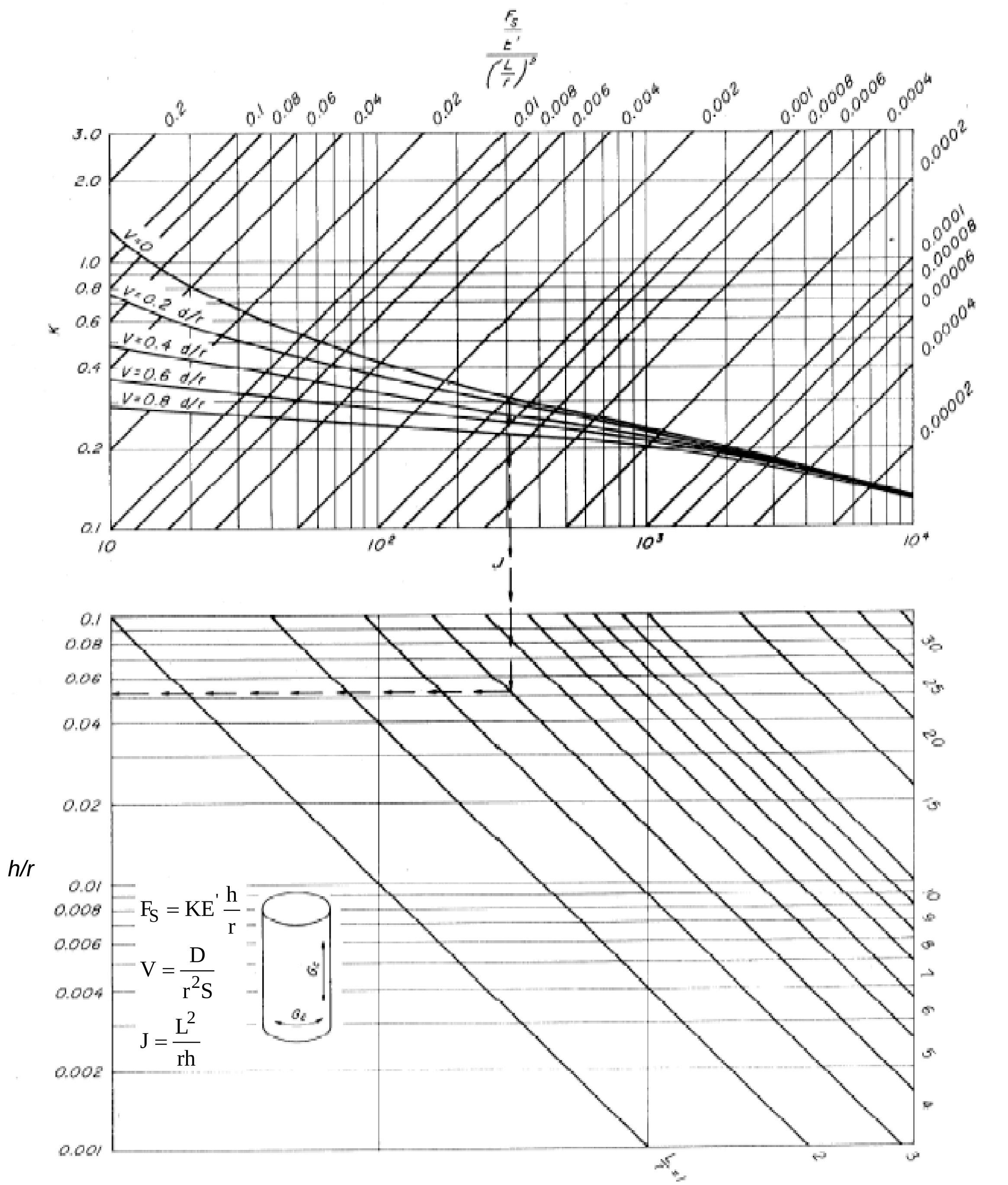


FIGURE 4.11.3.2(d) *Chart for determining a h/r ratio such that a sandwich cylinder with isotropic core will not buckle; t_c/h = 0.7.*

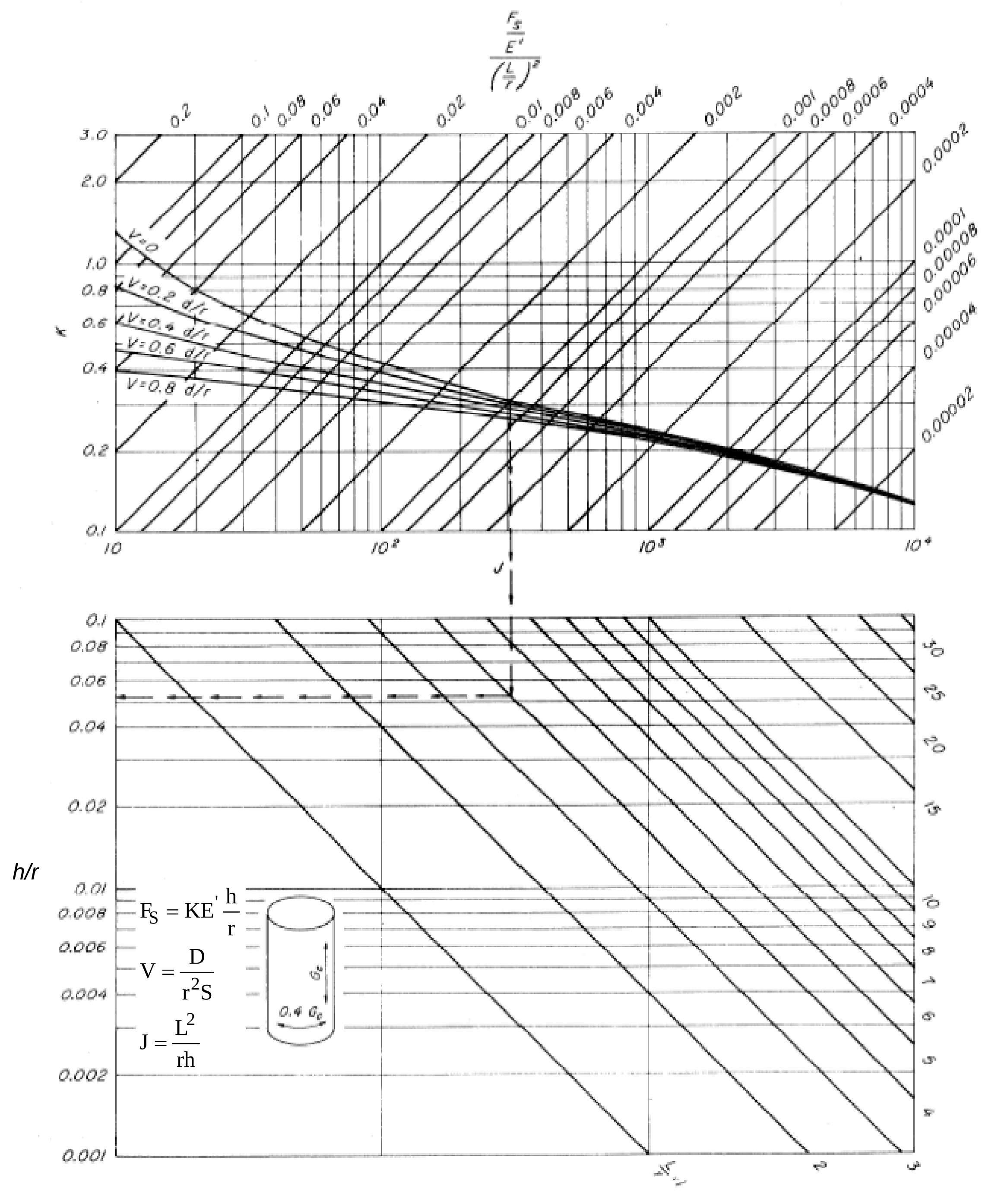


FIGURE 4.11.3.2(e) *Chart for determining h/r ratio such that a sandwich cylinder with orthotropic core will not buckle; t_c/h = 0.7.*

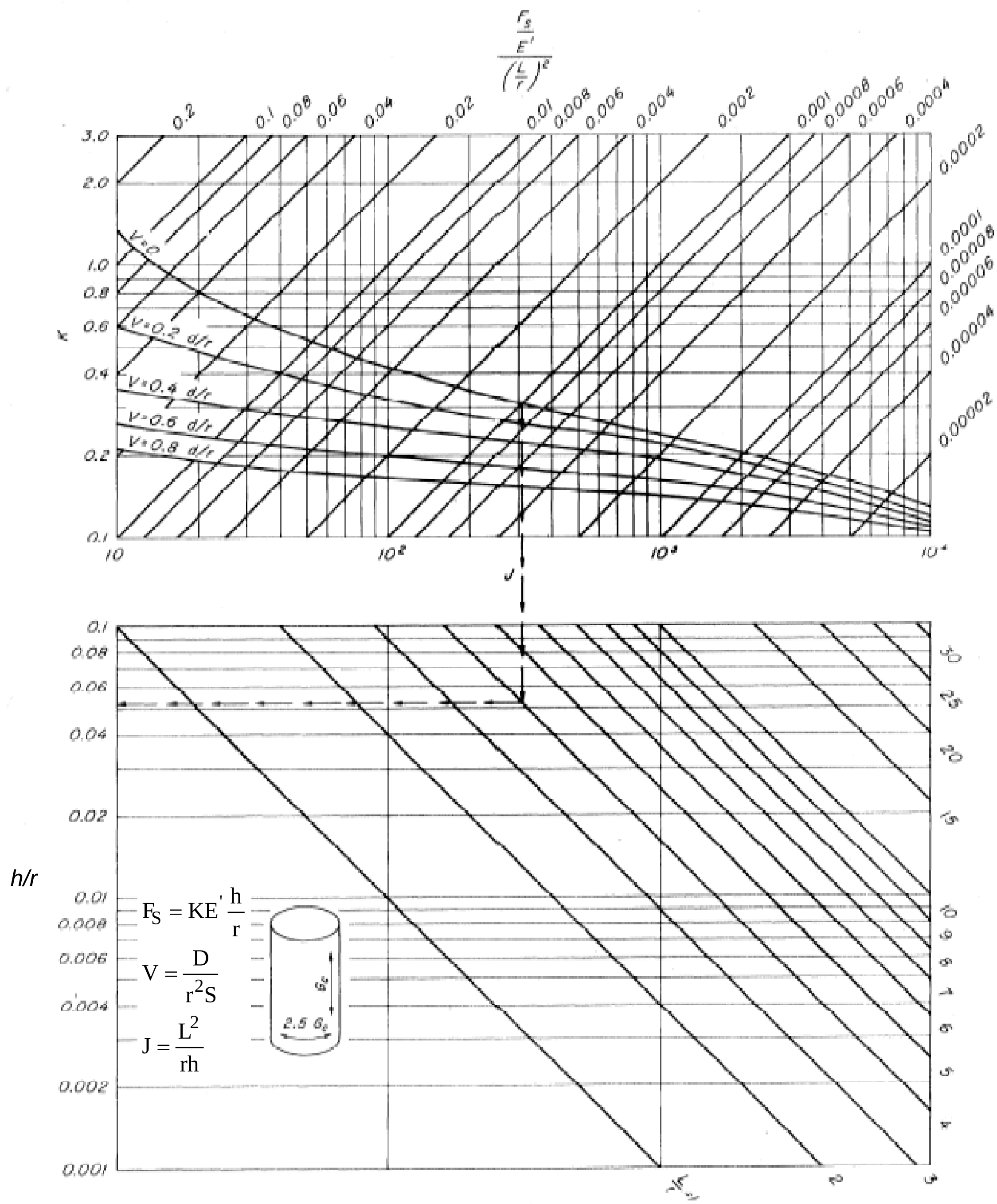


FIGURE 4.11.3.2(f) *Chart for determining h/r ratio such that a sandwich cylinder with orthotropic core will not buckle;* $t_c/h = 0.7$.

Solving Equation 4.11.3.2(a) for h/r gives:

$$\frac{h}{r} = \frac{F_{s\,UPR}}{E'_{UPR}}\left(\frac{1}{K}\right)$$

$$\frac{h}{r} = \frac{F_{s\,LWR}}{E'_{LWR}}\left(\frac{1}{K}\right) \qquad 4.11.3.2(b)$$

$$\frac{h}{r} = \frac{F_s}{E'}\left(\frac{1}{K}\right) \text{ (for equal face sheets)}$$

Therefore, if K is known, these equations can be solved directly to obtain h because all other quantities are known. After h is obtained, the core thickness, t_c, is computed from the equations

$$t_c = h - (t_1 + t_2) \qquad 4.11.3.2(c)$$

$$t_c = h - 2t \text{ (for equal face sheets)}$$

Values of K depend upon the bending stiffness, D, and shear stiffness, U, of the sandwich as incorporated in the parameter

$$V = \frac{D}{r^2 U} = \frac{E'_{UPR}\, t_{UPR}\, E'_{LWR}\, t_{LWR}\, d}{\left(E'_{UPR}\, t_{UPR} + E'_{LWR}\, t_{LWR}\right)\lambda r^2 G_c} \qquad 4.11.3.2(d)$$

$$V = \frac{E' t d}{2\lambda r^2 G_c} \quad \text{(for equal face sheets)}$$

where d is the distance between face sheet centroids, $\lambda = 1 - \nu^2$, ν is Poisson's ratio of the face sheets, and G_c is the core shear modulus associated with shear distortion in the radial and circumferential directions. Values of K are also dependent upon cylinder geometry as represented conveniently by dimensionless parameters L/r, h/r, and $J = L^2/hr$.

A minimum value of h required will be determined by assuming $V = 0$ for a first approximation. The value of h is minimum because $V = 0$ only if the core shear modulus is infinite. For any actual core the shear modulus is not infinite; hence, a thicker core must be used.

Charts of Figures 4.11.3.2(a) through 4.11.3.2(c) give values for h/r for sandwich with thin, equal, isotropic face sheets or for sandwich with thin isotropic face sheets, such that or $E'_{UPR}\, t_{UPR} = E'_{LWR}\, t_{LWR}$. The charts of Figures 4.11.3.2(d) through 4.11.3.2(f) apply to similar sandwich with moderately thick face sheets ($t_c/h = 0.7$). The upper portion of the charts is entered with the appropriate line represented by the known value of a parameter

$$\frac{F_s / E'}{(L/r)^2} \qquad 4.11.3.2(e)$$

The intersection of this appropriate line with the curve for $V = 0$ in the upper graph occurs at an abscissa value of J which is solved graphically for any particular L/r ratio to give the minimum h/r ratio in the lower graph of the figures.

Since actual core shear modulus are not very large, a value of h somewhat greater than the minimum calculated must be used. Figures 4.11.3.2(a) through 4.11.3.2(f) are entered at curves with values of V as computed by the definition of V in Equation 4.11.3.2(d). Figures 4.11.3.2(a) and 4.11.3.2(d) apply to sandwich with isotropic cores for which the circumferential shear modulus is equal to the axial shear modulus. Figures 4.11.3.2(b) and 4.11.3.2(e), and Figures 4.11.3.2(c) and 4.11.3.2(f) apply to sandwich with orthotropic cores for which the circumferential shear modulus is equal to 0.40 and 2.50 times, respectively, the axial shear modulus.

NOTE: For honeycomb cores with core ribbons parallel to the cylinder axis, $G_c = G_{TL}$ and the circumferential shear modulus is G_{TW}. For honeycomb cores with core ribbons circumferential, $G_c = G_{TW}$ and the circumferential shear modulus is G_{TL}. If core ribbons are at an angle θ to the panel length a,

$$G_c = \frac{G_{TL} G_{TW}}{\left(G_{TL}\sin^2\theta + G_{TW}\cos^2\theta\right)}.$$

In using Figures 4.11.3.2(a) through 4.11.3.2(f), it is necessary to iterate because V is directly proportional to the core thickness, t_c. As an aid to determining final values of t_c and G_c, Figure 4.11.3.2(g) presents a number of lines representing V for various values of G_c with V ranging from 0.01 to 2.0 and G_c ranging from 1,000 to 1,000,000 psi. The following procedure is suggested:

1. Determine thickness d from Figures 4.11.3.2(a) through 4.11.3.2(f) using a value of 0.01 for V.

2. Compute the constant relating V to G_c

$$VG_c = \left[\frac{E'_{UPR} t_{UPR} E'_{LWR} t_{LWR} d}{\left(E'_{UPR} t_{UPR} + E'_{LWR} t_{LWR}\right)\lambda r^2}\right]$$

$$VG_c = \left[\frac{E' t d}{2\lambda r^2}\right] \text{ (for equal face sheets)}$$

3. With this constant, enter Figure 4.11.3.2(g) and determine necessary G_c.

4. If the shear modulus is outside the range of values for the materials available, slide up the appropriate line of Figure 4.11.3.2(g) and pick up a new value of V, for a reasonable value of core shear modulus.

5. Reenter Figures 4.11.3.2(a) through 4.11.3.2(f) with the new value of V and repeat steps 1, 2, and 3.

The design shall be checked by using graphs of Figures 4.11.3.2(a) through 4.11.3.2(f) to determine the values of K to compute the actual buckling stress, F_{scr}, from Equation 4.11.3.2(a).

4.11.3.3 Check to determine whether sideways buckling will occur

If the sandwich cylinder is fairly long it may buckle sideways similar to the way a column buckles under end compression. The load per unit length of circumference at which sideways buckling will occur is given approximately by the formula:

$$N_{cr} = \frac{\pi\left(E'_{UPR} t_{UPR} + E'_{LWR} t_{LWR}\right) r}{2L} \qquad 4.11.3.3$$

$$N_{cr} = \frac{\pi E' t r}{L} \text{ (for equal face sheets)}$$

If the value of N_{cr} as computed by this equation is less than the design load, the cylinder will have to be redesigned by using a larger radius, shorter length, or stiffer face sheets. This equation was derived for thin-walled cylinders and is about 3% in error for $d/r = 0.2$. For $d/r < 0.2$, the error is less than 3%.

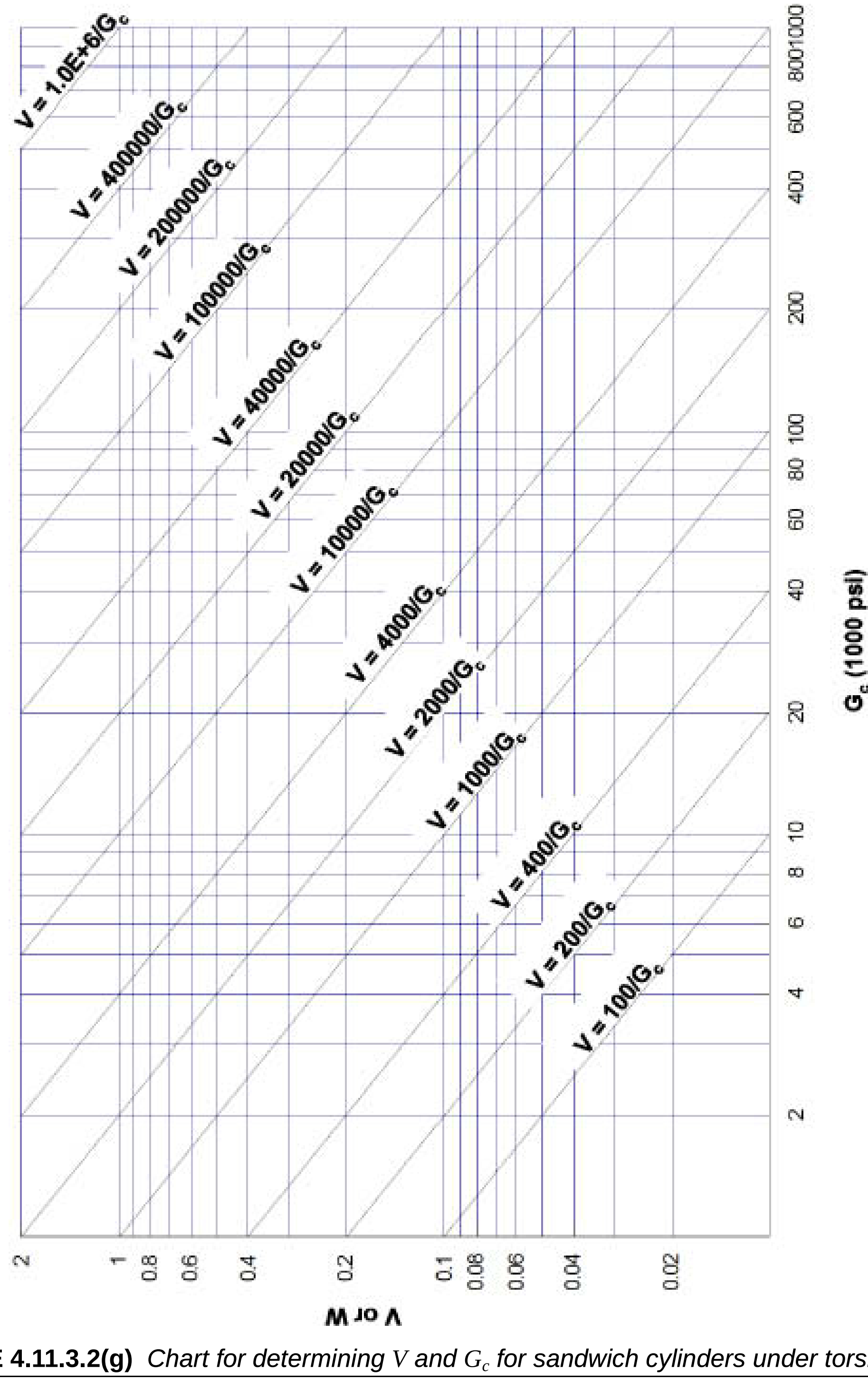


FIGURE 4.11.3.2(g) *Chart for determining V and G_c for sandwich cylinders under torsion load.*

4.11.4 Sandwich cylinders under axial compression or bending

The face sheet modulus of elasticity E', and the stress values, F_c, shall be compression values at the conditions of use. That is, if application is at elevated temperature, then the face sheet properties at elevated temperature shall be used in design. The face sheet modulus of elasticity is the effective value of the face sheet stress. If the stress is beyond the proportional limit values, an appropriate tangent, reduced or modified compression modulus of elasticity shall be used.

4.11.4.1 Determining face sheet thickness, core thickness, and core shear modulus

This section gives procedures for determining face sheet and core thickness and core shear modulus so that overall buckling of the sandwich walls of the cylinder will not occur (References 4.11.3.2(a), (b), and (d), and 4.11.4.1(a) through (c).

The face sheet stresses are related to the axial load by the equations:

$$t_{UPR} F_{cUPR} + t_{LWR} F_{cLWR} = N \qquad 4.11.4.1(a)$$

$$t = \frac{N}{2F_c} \quad \text{(for equal face sheets)}$$

where t is the face sheet thickness, F_c is the chosen design face sheet compressive stress, and UPR, LWR are subscripts denoting the upper and lower face sheets. If the load is produced by bending moment, the relationship between maximum N and bending moment, M, for a cylinder of mean radius, r, is given by $N = M/\pi r^2$.

In determining thickness of face sheets for a sandwich with face sheets of different materials, Equation 4.11.4.1(a) must be satisfied. In addition, to avoid overstressing either face sheet, the design stresses $F_{c\ UPR}$ and $F_{c\ LWR}$ must be chosen so that $F_{c\ UPR}/E'_{UPR} = F_{c\ LWR}/E'_{LWR}$ (where E' is the effective compression modulus of elasticity of the face sheet, and beyond the proportional limit this should be taken as the secant modulus). For example, if the upper face sheet is a material such that the ratio $F_{c\ UPR}/E'_{UPR} = 0.005$ and the lower face sheet is a material such that the ratio $F_{c\ LWR}/E'_{LWR} = 0.002$, then the design must be based on a ratio of 0.002; otherwise the lower face sheet will be overstressed. In order to accomplish this, the chosen design stress for the upper face sheet must be reduced to 0.002 E'_{UPR}. For many combinations of materials, it will be found advantageous to choose thicknesses such that $E'_{UPR}\, t_{UPR} = E'_{LWR}\, t_{LWR}$. If the core can support axial compression load, N should be replaced by $(N - F_c t_c)$.

If an axially compressed cylinder is long and slender and the radius is limited, the face sheet thicknesses may have to be increased in order to prevent column buckling, as covered by Section 4.11.4.3.

Theoretical equations are based on buckling load for classical sine wave buckling. The theory defines the parameters involved rather than determines exact coefficients for computing buckling loads. Large discrepancies exist between theory and tests, and unfortunately the test values for buckling of thin-walled cylinders in axial compression or bending are much lower than expected by classical theory (References 4.11.4.1(d) and (e)). Previous design information based on large-deflection theory and diamond-shaped buckles gave results less than one-half the buckling loads given by classical theory. Continued efforts in shell analysis have shown that the post-buckling behavior tends to approach much lower limits.

Until sufficient test data are available, reduction factors must be applied to theoretical buckling coefficients. These reduction factors attempt to account for effects of initial shell irregularities and thicker shells have less reduction from classical theory than thinner shells (References 4.11.3.2(d) and 4.11.4.1(d)). Reduction factors, k, which are 95% of factors given in Reference 4.11.3.2(d), are presented in Figure 4.11.4.1(a) as a function of the ratio of mean cylinder radius, r, to the cylinder wall radius of gyration, ρ.

The following procedures are applicable to cylinders longer than the length of one ideal buckle, such as would form in the wall of a long cylinder. If the core shear modulus is high, the ideal buckle length is generally about equal to the radius of the cylinder. The buckle length becomes shorter than the radius as the core shear modulus decreases.

The load per unit cylinder circumference at which buckling of the sandwich wall will occur is given by the formula

$$N_{cr} = 2kK \frac{\sqrt{\left(E'_{UPR} t_{UPR} + E'_{LWR} t_{LWR}\right) D}}{r} \qquad 4.11.4.1(b)$$

where k is the reduction factor given in Figure 4.11.4.1(a), D is the sandwich bending stiffness, r is the mean radius of curvature, and K is a theoretical buckling coefficient dependent on sandwich bending and shear rigidities. This equation solved for the face sheet stress becomes:

$$\begin{aligned} F_{cUPR} &= \frac{kKE'_{UPR}}{\sqrt{\lambda}} \frac{2\sqrt{E'_{UPR} t_{UPR} E'_{LWR} t_{LWR}}}{E'_{UPR} t_{UPR} + E'_{LWR} t_{LWR}} \left(\frac{d}{r}\right) \\ F_{cLWR} &= \frac{kKE'_{LWR}}{\sqrt{\lambda}} \frac{2\sqrt{E'_{UPR} t_{UPR} E'_{LWR} t_{LWR}}}{E'_{UPR} t_{UPR} + E'_{LWR} t_{LWR}} \left(\frac{d}{r}\right) \qquad 4.11.4.1(c) \\ F_c &= \frac{kKE'}{\sqrt{\lambda}} \left(\frac{d}{r}\right) \text{ (for equal face sheets)} \end{aligned}$$

where E' is effective compressive modulus of elasticity at stress F_c, $\lambda = 1 - \nu^2$, ν is Poisson's ratio of the face sheets (here it is assumed $\nu = \nu_{UPR} = \nu_{LWR}$), and d is distance between face sheet centroids.

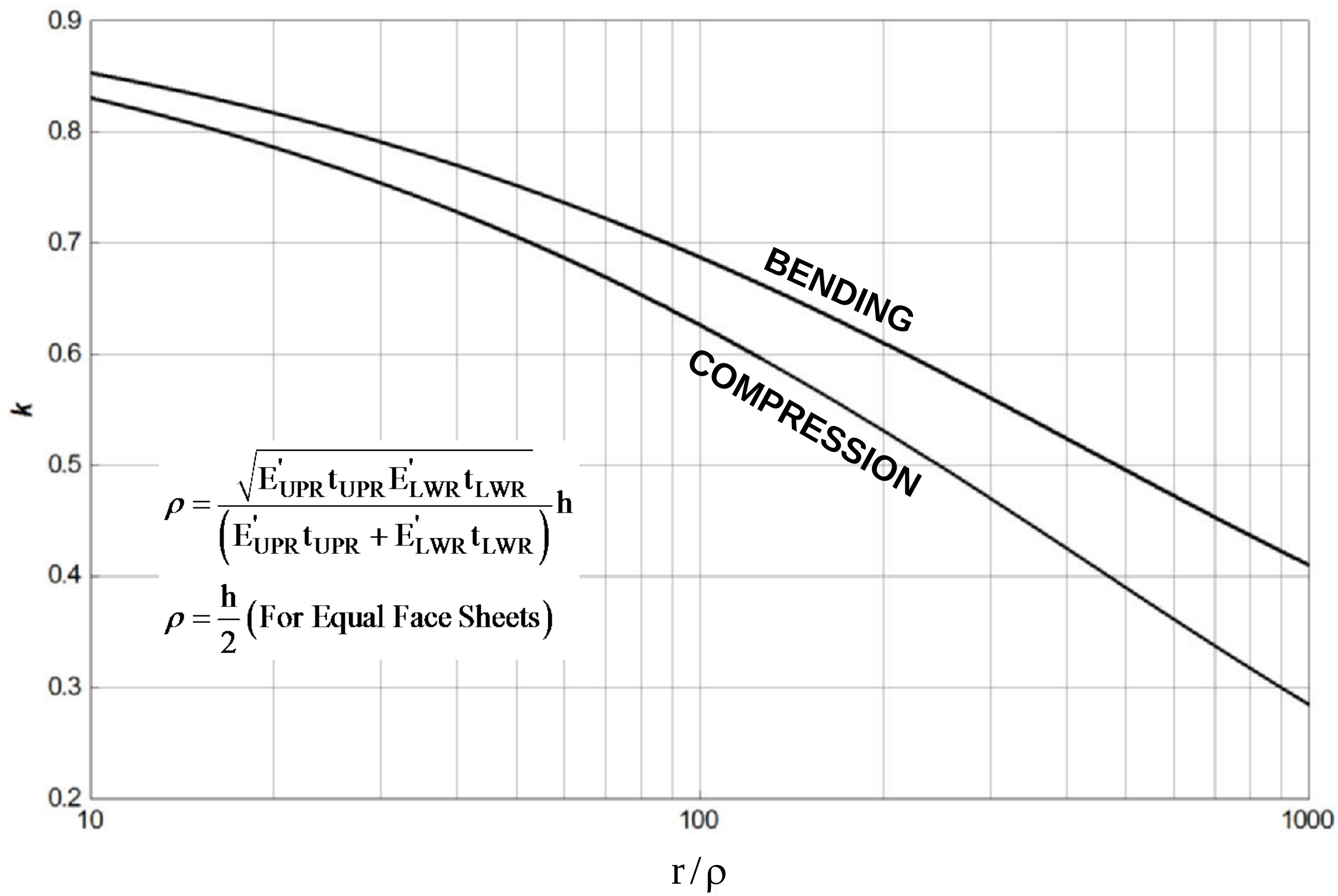


FIGURE 4.11.4.1(a) *Reduction factor, k, for the buckling of sandwich cylinders in axial compression or bending.*

Values of the coefficient K are given by the following approximate equations:

For sandwich with isotropic or honeycomb core or corrugated core with flutes circumferential –

For $\left(\frac{1+R}{2\sqrt{R}}\right)\left(\frac{r}{d}\right)V \leq \frac{1}{2}$

$$K = 1 - \left(\frac{1+R}{2\sqrt{R}}\right)\left(\frac{r}{d}\right)V$$

$$K = 1 - \left(\frac{r}{d}\right)V \quad \text{(for equal facings)}$$

For $\left(\frac{1+R}{2\sqrt{R}}\right)\left(\frac{r}{d}\right)V > \frac{1}{2}$

$$K = \frac{1}{4\left(\frac{1+R}{2\sqrt{R}}\right)\left(\frac{r}{d}\right)V}$$

$$K = \frac{1}{4\left(\frac{r}{d}\right)V} \quad \text{(for equal facings)}$$

For sandwich with corrugated core having flutes axial –

$$K = 1 - \frac{1}{4}\left(\frac{1+R}{2\sqrt{R}}\right)\left(\frac{r}{d}\right)W \qquad 4.11.4.1(d)$$

$$K = 1 - \frac{1}{4}\left(\frac{r}{d}\right)W \quad \text{(for equal face sheets)}$$

where

$$R = \frac{E'_{LWR} t_{LWR}}{E'_{UPR} t_{UPR}}$$

$$V = \frac{E'_{UPR} t_{UPR} \, E'_{LWR} t_{LWR} \, d}{\left(E'_{UPR} t_{UPR} + E'_{LWR} t_{LWR}\right)\lambda r^2 G_c} \qquad 4.11.4.1(e)$$

$$V = \frac{E' t \, d}{2\lambda r^2 G_c} \quad \text{(for equal face sheets)}$$

$$W = \frac{E'_{UPR} t_{UPR} \, E'_{LWR} t_{LWR} \, d}{\left(E'_{UPR} t_{UPR} + E'_{LWR} t_{LWR}\right)\lambda r^2 G'_c} \qquad 4.11.4.1(f)$$

$$W = \frac{E' t \, d}{2\lambda r^2 G'_c} \quad \text{(for equal face sheets)}$$

G_c is the core shear modulus associated with directions axial to the cylinder and perpendicular to the wall of the cylinder and G'_c is the core shear modulus associated with the directions circumferential to the cylinder and perpendicular to the wall of the cylinder. As values of core shear modulus decrease, values of V or W increase and values of K gradually decrease.

Solution of the face sheet stress equation for h/r after substitution of expression for K from these equations results in

$$\frac{d}{r} = \left(\frac{1+R}{2\sqrt{R}}\right)\left[\frac{F_{cUPR}\lambda_{UPR}}{kE'_{UPR}} + V\right]$$
$$\frac{d}{r} = \left(\frac{1+R}{2\sqrt{R}}\right)\left[\frac{F_{cLWR}\lambda_{LWR}}{kE'_{LWR}} + V\right] \qquad 4.11.4.1(g)$$
$$\frac{h}{r} = \frac{F_c\lambda}{kE'} + V \text{ (for equal face sheets)}$$

or

$$\frac{d}{r} = \left(\frac{1+R}{2\sqrt{R}}\right)\left[\frac{F_{cUPR}\lambda_{UPR}}{kE'_{UPR}} + \frac{W}{4}\right]$$
$$\frac{d}{r} = \left(\frac{1+R}{2\sqrt{R}}\right)\left[\frac{F_{cLWR}\lambda_{LWR}}{kE'_{LWR}} + \frac{W}{4}\right] \qquad 4.11.4.1(h)$$
$$\frac{h}{r} = \frac{F_c\lambda}{kE'} + \frac{W}{4} \text{ (for equal face sheets)}$$

To determine values of d, it is necessary to iterate because k and V or W are dependent upon d. A first iteration to determine a minimum value of d can be made from these equations and the graph of Figure 4.11.4.1(a) by assuming $V = 0$ or $W = 0$. This value of d is minimum because $V = 0$ or $W = 0$ only if the core shear modulus is infinite. For any actual core, the shear modulus is not infinite, hence a thicker core must be used. Values of V or W are also dependent upon core shear modulus, G_c. As an aid to finally determining d and G'_c, Figure 4.11.4.1(b) presents lines representing V or W for various values of G_c.

The following procedure is suggested:

Use V for sandwich with isotropic or honeycomb core or corrugated core with flutes circumferential, and use W for sandwich with corrugated core with flutes axial.

1. Assume V or W equal to zero, k equal to 0.6, and compute a value of d from Equations 4.11.4.1(g) or (h).

2. Enter Figure 4.11.4.1(a) with a value of r/ρ based on the computed value of d, and determine a new value of k. (ρ is defined in Figure 4.11.4.1(a).)

3. Recompute the value of d using the value of k determined in Step 2.

4. Repeat Steps 2 and 3 until the value of d computed in Step 3 agrees with that used in Step 2.

5. Assume a small value for V or W and repeat previous steps to determine a somewhat larger value of d.

6. Compute the constant relating V or W to G_c:

$$\left[\frac{E'_{UPR}t_{UPR}E'_{LWR}t_{LWR}d}{\left(E'_{UPR}t_{UPR} + E'_{LWR}t_{LWR}\right)\lambda r^2}\right] \text{ or } \left[\frac{E'td}{2\lambda r^2}\right]\left(\text{for equal facings}\right) = VG_c \text{ or } WG_c$$

7. With this constant, enter Figure 4.11.4.1(b) and determine necessary G_c.

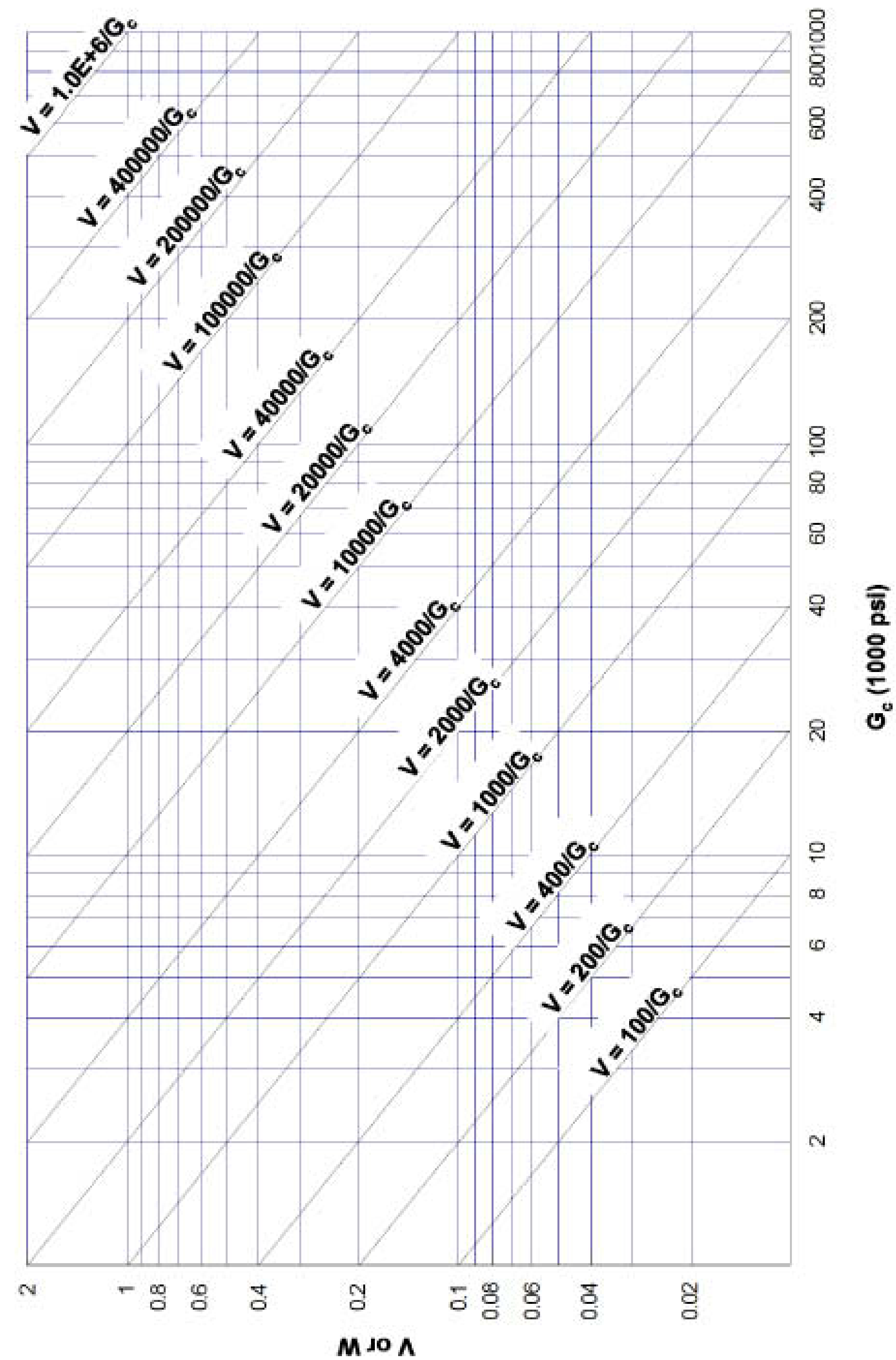


FIGURE 4.11.4.1(b) *Chart for determining V and G_c for sandwich cylinders under axial compression or bending load.*

8. If the shear modulus is outside the range of values for materials available, slide up the appropriate line of Figure 4.11.4.1(b) and pick a new value of V or W for a reasonable value of core shear modulus.

9. Recompute d with a new value of V or W and repeat steps until the value of d is obtained.

10. Calculate core thickness t_c from the equations

$$t_c = d - \frac{t_{UPR} + t_{LWR}}{2} \qquad 4.11.4.1(i)$$
$$t_c = d - t \text{ (for equal face sheets)}$$

4.11.4.2 Checking procedure for determining cylinder wall buckling stress, F_{cr}

The design shall be checked by using the graph of Figure 4.11.4.1(a) to obtain k values, and Equation 4.11.4.1(d) to obtain K to substitute into Equations 4.11.4.1(b) or (c) to compute the end load N_{cr} or the buckling stress F_{cr}, respectively. The equations apply to sandwich cylinders having isotropic face sheets and isotropic honeycomb or corrugated cores as noted. It should be understood that if the desired F_{cr} is above the proportional limit values, the value of E' shall be an effective value used in computing V and in computing F_{cr}.

4.11.4.3 Check to determine whether column buckling will occur

If an axially compressed sandwich cylinder is fairly long, it may buckle as a column. The face sheet stress at which Euler column buckling will occur, if the ends of the cylinder are hinged, is given by the formula:

$$F_{eUPR} = \frac{\pi^2 r^2 E'_{UPR}}{2L^2}$$
$$F_{eLWR} = \frac{\pi^2 r^2 E'_{LWR}}{2L^2} \qquad 4.11.4.3(a)$$

where L is the unsupported column length and the subscript e denotes Euler. The load per unit length of circumference of the cylinder is given by:

$$N_e = \frac{\pi^2 r^2 \left(E'_{UPR} t_{UPR} + E'_{LWR} t_{LWR}\right)}{2L^2} \qquad 4.11.4.3(b)$$
$$N_e = \frac{\pi^2 r^2 E' t}{L^2} \text{ (for equal face sheets)}$$

If the value of N_e as computed by the above equation is less than the design load, the cylinder will have to be redesigned by using a larger radius, shorter length, or stiffer face sheets. The equations above were derived for thin-walled cylinders and are about 3% in error for d/r = 0.2. For d/r < 0.2 the error is less than 3%.

4.11.5 Sandwich cylinders under combined loads

Face sheet stresses shall be determined for each load applied separately and the effects of combining the loads and stresses shall be assessed by appropriate interaction equations for the face sheet materials.

Overall buckling, or local instabilities such as dimpling or wrinkling of the face sheets, may lead to collapse of the cylinder. Equations for estimating wrinkling and dimpling under combined loads have been given in Sections 4.6.6.5 and 4.6.5.4, respectively. If estimates based on information given in Section 4.6 indicate that wrinkling or dimpling could be expected, this behavior should be confirmed by test.

Overall buckling of cylindrical sandwich panels under combined loads is given by interaction equations in terms of ratios, R, wherein R denotes the ratio of the applied stress or load under combined loading to the buckling stress or load under separate loading ($R - N/N_{cr}$). Appropriate subscripts are given to R to denote stress or load direction.

4.11.5.1 Axial compression and external lateral pressure

Overall buckling of sandwich walls of a circular cylinder under axial compression and external lateral pressure can be estimated by the interaction formula

$$R_{cx} + R_{py} = 1 \qquad 4.11.5.1$$

This equation is usually somewhat conservative for most sandwich cylinders. It can be exceedingly conservative for sandwich cylinders with $V >> 0$. For more accurate analyses, including sandwich walls with corrugated core, consult references (References 4.11.3.2(a) and (b), and 4.11.5.1(a) and (b)).

4.11.5.2 Axial compression and torsion

Overall buckling of the sandwich walls of a circular cylinder under axial compression and torsion can be estimated by the interaction formula

$$R_c + R_s = 1 \qquad 4.11.5.2$$

This equation is conservative for short and thick-walled cylinders for which the torsion term should have an exponent of 2. The equation can be very conservative for sandwiches with $V >> 0$. For more accurate analysis of sandwich walls with corrugated core consult (References 4.11.3.2(a) and 4.11.5.1(a)).

4.11.5.3 Torsion and lateral external or internal pressure

Overall buckling of sandwich walls of a circular cylinder under torsion and external or internal pressure can be estimated by the interaction formula

$$R_p + R_s^2 = 1 \qquad 4.11.5.3$$

For external pressure R_p is positive, and for internal pressure R_p is negative. Details of the derivation and resultant interaction curves are given in References 4.11.3.2(a) and (b).

4.12 FINITE ELEMENT MODELING OF SANDWICH STRUCTURE

4.12.1 Introduction

Finite element (FE) modeling provides the most general and versatile engineering analysis tool for design of structures, and many commercially available FE programs are available. However, knowledge of the finite element method and an understanding of the mechanical behavior of sandwich structures are recommended before design of a sandwich construction with finite elements is attempted.

Numerous texts on the finite element method are available which discuss its theory and use in detail. Therefore, only a few general comments will be made here with respect to special considerations for sandwich structures.

Sandwich panel finite element modeling can take different approaches depending on panel geometry and constituent materials. For practical reasons, most of the FE analyses for sandwich panels are based

on two-dimensional (2-D) plate and shell elements. These 2-D approaches generally fall into one of the classifications discussed in the subsections below.

It should be noted that global response quantities can be predicted accurately using any of the model types described below, provided that they are implemented properly. However, accurate determination of detailed responses, e.g., through the thickness distributions of transverse shear and normal stresses, requires the use of higher-order layered shell models or models including solid elements.

In general, the following points should also be kept in mind when analyzing sandwich constructions with the finite element method.

Shear deformation of the core has to be considered, and general plate and shell finite elements often do not include shear deformation since it is negligible for metallic sheets. The user should select elements that are adequate for sandwich analysis and design. This is particularly important for buckling and eigenfrequency analyses, where neglecting or misrepresenting the shear deformation of the core can result in unconservative results. Specific problems may occur for cores with very low shear modulus, e.g., low-density Polyurethane (PUR) cores. For elements that do include transverse shear stiffness, the formulation may result in the shear stiffness of the face sheets dominating over the shear stiffness of the core, even when the face sheets are very thin. The user is advised to check this by comparing with analytical solutions or full three-dimensional (3-D) FE models.

Due to the low stiffness and strength in the thickness direction of the sandwich, local load introductions, corners, and joints must be checked by 2-D or 3-D analyses to a far greater extent than would be the case for solid structures. For the same reasons, curved sandwich panels with small radii of curvature (less than 10 times the sandwich thickness) will have to be analyzed using 2-D plane strain or 3-D elements to account for the transverse normal stresses not included in shell elements.

When one or more the components of the sandwich construction is anisotropic, e.g., composite face sheets, honeycomb or balsa core, the anisotropy of material must be included in the analysis. With composite face sheets or honeycomb core, if a very fine mesh is employed to study the area near a stress concentration, care should be taken that mesh sizes are not smaller than a representative volume element.

References 4.12.1(a) and (b) give extended overviews and references for analytical and computational procedures for sandwich structures. Noor, et al., (Reference 4.12.1(b)) contains an exhaustive reference list (over 1300 citations) of analytical and computational procedures for sandwich structures. A further survey can be found in Librescu and Hause (Reference 4.12.1(c)), which includes an extended formulation to include buckling and postbuckling response of flat and curved sandwich structures subjected to mechanical and thermal loads.

4.12.2 Global models

In global approximation models, the sandwich is replaced by an equivalent single-layer plate or shell element with global through the thickness approximations for the displacements, strains and/or stresses. By its very nature, a global finite element model does not include all the geometric details or capture local stress concentrations. If local stress states are needed, then additional analysis techniques must be employed.

A common approach is to extract the element forces and moments from the global FE model results, and use this information as input for further analysis. The FE results can be post-processed to calculate local responses and check the failure modes listed in Section 4.6 (face sheet ply-by-ply margins of safety, core shear strength, flatwise strength, core crushing, and instability modes). The FE results may also be used as input to “hand analysis” to determine stresses at joints, attachments, or other details.

A more detailed “local” FE model of a relatively small area of the structure may be developed, to determine the behavior in the region of a cutout, attachment, or other structural features. Displacements

from the global model can be transferred onto the boundary of the local model. Alternatively, the local model may be integrated into the global model by tying the boundaries together using constraint equations. However, in this case, the transfer of load at the model boundaries requires careful examination to ensure continuous stress and strain fields.

4.12.3 Layered models

The layered shell or discrete layer modeling approach exploits standard shell finite elements. The sandwich is divided into three or more layers, and for thin sandwich panels may provide a first approximation of the global behavior. This approach gives an equivalent single-layer result using classical lamination theory to compute the sandwich stiffness coefficients. In this model all layers have a common, unique rotation through the cross section of the sandwich. Because the core material generally only offers shear stiffness, shear-flexible C^0 shell elements are typically used in layered shell models. Shell elements based on the classical Kirchhoff-Love theory (C^1 shell elements) can be used; however, such an approach ignores the transverse shear flexibility offered by the sandwich structure. Figure 4.12.3 shows a simple example of a layered model, with one layer of elements for each face sheet and another layer for the core. More complex models may be constructed, using additional elements through the thickness. A variation on this approach uses three (or more) 2-D elements layered over each other and sharing nodes.

A similar approach, the method of element-layering, is similar to a method used to compute fracture parameters and interlaminar stresses in skin-stiffener debond problems. Element-layering is designed to provide the needed higher order in transverse displacements. The method simply consists of modeling the laminated composite shell with more than one thick-shell element through the thickness, each element layer comprising a sublaminate of similar components. The layers are tied together in the model with multi-point constraint equations (MPC) that enforce displacement compatibility at the layer interfaces. Element-layering introduces two additional degrees of freedom per node for each additional layer. If a conventional model has six degrees of freedom per node, a two-layer model has eight, and a four-layer model has twelve. Note that that element-layering does not require shear correction factors since the transverse shear distribution is nearly constant in each layer.

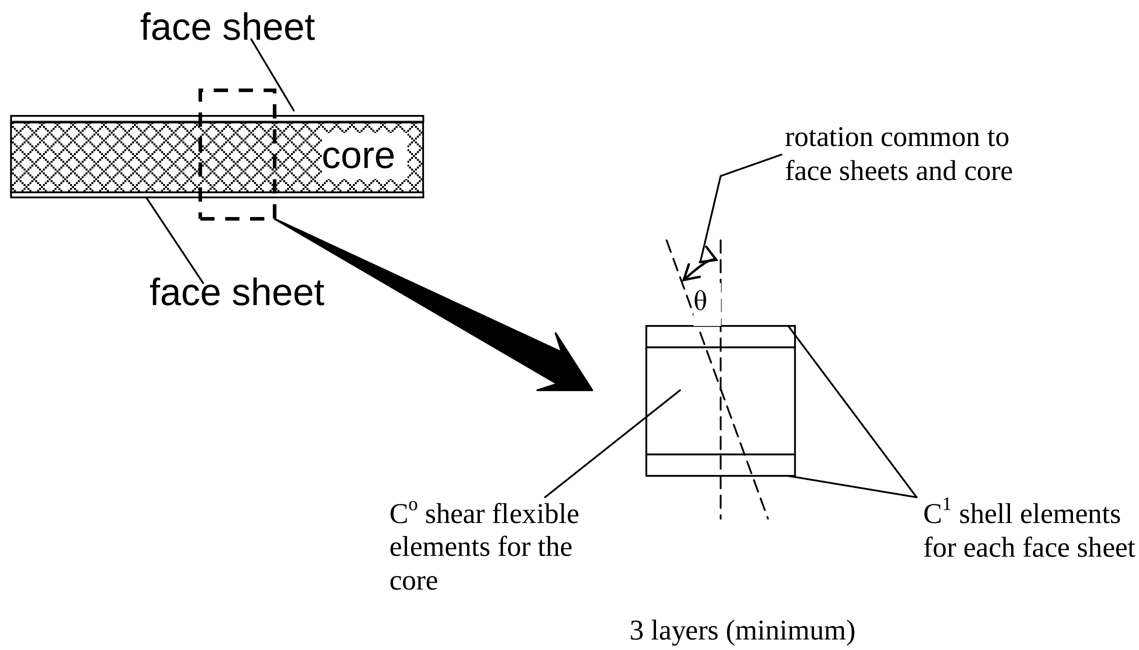


FIGURE 4.12.3 *Simple layered model for a sandwich.*

4.12.4 Solid models

In addition to the plate and shell element approaches, more detailed and computationally expensive approaches fall into the following categories.

Shell/solid models use shell elements to model each individual face sheet and solid elements to model the core material. This provides a modeling approach for both general and local response predictions. The accuracy of this approach, as well as its computational cost, is related to the through the thickness modeling of the core material. Each face sheet may be a multi-layer laminate and is modeled using shell elements, while the core is modeled using solid elements. Multiple solid elements through the core thickness may be required to represent the core deformation accurately. The shell element reference surface is defined to coincide with the bounding surface of the adjacent solid element. This may be achieved by defining an "offset" between the shell element reference surface (the centerline of the face sheet) and the nodes, which are at the corners of the solid elements and therefore at the interface between the face sheet and the core.

For the shell/solid models, displacement-field compatibility between the shell element and the solid element must be considered during the modeling process. For example, the use of 4-node C^1 shell elements for the face sheets with an 8-node solid element leads to an incompatibility for the normal displacements. Lagrangian shell elements based on a C^0 formulation, used with standard solid elements of the same order, give displacement-field compatibility for the translational degrees of freedom. Combinations of the 4-node, 8-node or 9-node C^0 shell element with the 8-node, 20-node or 27-node solid element, respectively, give translational displacement compatibility.

In a full three-dimensional (3-D) finite element model, solid three-dimensional elements are used to model both the face sheets and the core. These models are referred to as 3-D solid models and they are typically reserved for detailed local modeling because of their computational cost.

4.12.5 Sandwich element models

Recently, specially formulated sandwich elements have been implemented in some specialized commercial codes, allowing models to be made using specialty elements developed specifically for the analysis of sandwich structures. Sandwich element models embody the kinematics and stiffness of the sandwich structure for less computational cost than the layered shell/solid models. However, these elements are not widely available and their formulations vary so the user should be careful to check the element documentation to understand what assumptions are used. References 4.12.5(a) and (b) show examples of this approach.

4.13 OPTIMUM SANDWICH

The concept of sandwich construction combining thin, strong face sheets on lightweight, thick cores suggests possibilities of deriving constructions so proportioned that minimum-weight constructions, often called “optimum” constructions, for a given stiffness or load capability are attained. It is important to realize that the minimum-weight construction derived may not be practical because “optimization” may require unusually thin face sheets that may not be available, or unusually lightweight cores of great thickness. Constraints may be imposed on the optimization, such as limits on core thickness or minimum gage requirements (including limiting face sheet thickness to multiples of ply thickness or readily available sheet metal gages).

Previous chapters for design of specific sandwich components will give correct sandwich proportions regarding stresses in face sheets and core, buckling, or deflections, but these sandwiches may not be minimum weight. Examples will be given to illustrate this point.

Direct optimization without examination of the resultant designs may lead to erroneous conclusions when comparing material requirements with constructions other than sandwich because the “optimized” sandwich may not be a realistic one.

Intuitive optimization, such as requiring that all parts be fully stressed or the failure occur in all modes simultaneously, does not necessarily produce minimum-weight structural components (Reference 4.13).

4.13.1 Sandwich weight

The weight of a sandwich is given by the formula

$$W = w_{UPR} t_{UPR} + w_{LWR} t_{LWR} + w_c t_c + W_B \qquad 4.13.1(a)$$
$$W = 2wt + w_c t_c + W_B \text{ (for equal face sheets)}$$

where W is sandwich weight (per unit area), w is density, t is thickness, subscripts UPR and LWR denote the upper and lower face sheets, subscript c denotes core, and W_B is the total weight of the bond (per unit area) between face sheet and core. This bond may be an adhesive or braze material. If it is assumed the bond weight is the same for all sandwich of the type considered, then the weight comparisons can be made on the basis of (W- W_B).

It is also convenient to express t_c as (d - (t_{UPR} + t_{LWR})/2) where d is distance between face sheet centroids. Then the equations for weight can be rewritten as

$$\left(W - W_B\right) = \phi_{UPR} t_{UPR} + \phi_{LWR} t_{LWR} + w_c d \qquad 4.13.1(b)$$
$$\left(W - W_B\right) = 2\phi t + w_c d \text{ (for equal face sheets)}$$

where

$$\phi_{UPR} = w_{UPR} - \frac{w_c}{2}, \quad \phi_{LWR} = w_{LWR} - \frac{w_c}{2}, \quad \phi = w - \frac{w_c}{2}, \qquad 4.13.1(c)$$

It is essential that the weight units be consistent in using the equations. Thus if w is density in pounds per cubic inch, t and d must be in inches and then $(W - W_B)$ is weight in pounds per square inch of sandwich area.

Example: Compute $(W - W_B)$ for a sandwich with aluminum face sheets, 0.032 in. thick and density 0.100 pci, on a ¾ in. thick honeycomb core having a density of 6 pcf (0.0035 pci).

From Equation 4.13.1(a):

$$(W - W_B) = 2(0.100)(0.032) + 0.0035(0.75)$$

$$(W - W_B) = 0.00640 + 0.00263 = 0.00903 \text{ psi}$$

or

$$(W - W_B) = 144(0.00903) = 1.30 \text{ psf}$$

4.13.2 Sandwich bending stiffness

Since the primary purpose of structural sandwich is to provide stiffness, hence low deflection under transverse load and high resistance to buckling under edgewise (in-plane) load, a minimum-weight sandwich to provide a specific bending stiffness can be determined.

The bending stiffness of a sandwich, per unit width, is given by the formula

$$D = \frac{\frac{E_{UPR} t_{UPR}}{\lambda_{UPR}} \frac{E_{LWR} t_{LWR}}{\lambda_{LWR}}}{\frac{E_{UPR} t_{UPR}}{\lambda_{UPR}} + \frac{E_{LWR} t_{LWR}}{\lambda_{LWR}}} d^2 \qquad 4.13.2(a)$$

$$D = \frac{Et}{2\lambda} d^2 \text{ (for equal face sheets)}$$

where D is the bending stiffness, subscripts UPR and LWR denote the upper and lower face sheets, E is the face sheet elastic modulus, $\lambda = 1 - \nu^2$, ν is Poisson's ratio, t is face sheet thickness, and d is distance between face sheet centroids.

Substitution of the stiffness expression into the weight equation and minimizing weight by calculus (Reference 4.13.2(a)), results in the following expressions for d and t to produce minimum-weight sandwich for a required stiffness D

$$d^3 = \frac{2D}{w_c} \left[\sqrt{\frac{\phi_{UPR} \lambda_{UPR}}{E_{UPR}}} + \sqrt{\frac{\phi_{LWR} \lambda_{LWR}}{E_{LWR}}} \right]^2 \qquad 4.13.2(b)$$

$$d^3 = \frac{8D\phi\lambda}{w_c E} \text{ (for equal face sheets)}$$

and

$$t_{UPR} = \frac{d}{2}\frac{w_c}{\phi_{UPR}}\frac{\sqrt{\frac{\phi_{UPR}\lambda_{UPR}}{E_{UPR}}}}{\left(\sqrt{\frac{\phi_{UPR}\lambda_{UPR}}{E_{UPR}}}+\sqrt{\frac{\phi_{LWR}\lambda_{LWR}}{E_{LWR}}}\right)}, \qquad t_{LWR} = \frac{d}{2}\frac{w_c}{\phi_{LWR}}\frac{\sqrt{\frac{\phi_{LWR}\lambda_{LWR}}{E_{LWR}}}}{\left(\sqrt{\frac{\phi_{UPR}\lambda_{UPR}}{E_{UPR}}}+\sqrt{\frac{\phi_{LWR}\lambda_{LWR}}{E_{LWR}}}\right)}$$

4.13.2(c)

$$t = \frac{d}{4}\frac{w_c}{\phi} \text{ (for equal face sheets)}$$

The resultant construction will be found to be proportioned so that approximately two-thirds of the sandwich weight will be in the core (References 4.13.2(a) through (e)).

<u>Example</u>: Determine dimensions of sandwich components so that the resultant composite will have a bending stiffness $D = 3.0 \times 10^6$ lb-in^2 per inch of width. The face sheet properties are $E_{UPR}/\lambda_{UPR} = 10^7$ psi, w_{UPR}=0.100 pci, $E_{LWR}/\lambda_{LWR} = 3 \times 10^6$ psi, $w_{LWR} = 0.061$ pci, and core weight $w_c = 0.0034$ pci.

Minimum weight design -

From Equation 4.13.2(b)

$$d = \left\{\frac{2(3.0)\left(10^6\right)}{0.0034}\left[\sqrt{\frac{0.0983}{10^7}}+\sqrt{\frac{0.0593}{3\left(10^6\right)}}\right]^2\right\}^{\frac{1}{3}} = 4.66 \text{ in.}$$

And from Equation 4.13.2(c)

$$t_{UPR} = 2.33\frac{0.0034\sqrt{\frac{0.0983}{10^7}}}{0.0983\left(\sqrt{\frac{0.0983}{10^7}}+\sqrt{\frac{0.0593}{3\left(10^6\right)}}\right)} = 0.033 \text{ inch}, \quad t_{LWR} = 2.33\frac{0.0034\sqrt{\frac{0.0593}{3\left(10^6\right)}}}{0.0593\left(\sqrt{\frac{0.0983}{10^7}}+\sqrt{\frac{0.0593}{3\left(10^6\right)}}\right)} = 0.078 \text{ inch}$$

With these dimensions the sandwich weight (minus bond weight) is 0.0237 psi of which 0.0156 psi is core, 0.0033 psi is the upper face sheet and 0.0048 psi is the lower face sheet. Although the core density is less than that of the face sheets, about 2/3 of the sandwich weight is in the core.

Minimum weight design for sandwich with equal face sheets –

Sandwich with both face sheets of type 1 –

From Equation 4.13.2(b)

$$d = \left\{\frac{8(3)\left(10^6\right)(0.0983)}{0.0034\left(10^7\right)}\right\}^{\frac{1}{3}} = 4.11 \text{ inch}$$

And from Equation 4.13.2(c)

$$t = \frac{4.10(0.0034)}{4(0.0983)} = 0.035\ \text{inch}$$

The sandwich weight (minus bond weight) is 0.0209 psi of which 0.0139 psi is in the core and 0.0070 psi is in the face sheets. The core weight is 66% of the sandwich weight.

Sandwich with both face sheets of type 2 –

From Equation 4.13.2(b)

$$d = \left\{\frac{8(3)\left(10^6\right)(0.0593)}{0.0034(3)\left(10^6\right)}\right\}^{\frac{1}{3}} = 5.19\ \text{inch}$$

And from Equation 4.13.2(c)

$$t = \frac{5.19(0.0034)}{4(0.0593)} = 0.074\ \text{inch}$$

The sandwich weight (minus bond weight) is 0.0262 psi of which 0.0171 psi, or 65%, is in core weight.

A summary of the dimensions is shown in Table 4.13.2.

The values in Table 4.13.2 show that the lightest sandwich is that with both face sheets of type 1 material (face sheet with the lower value of $\phi\lambda/E$). The thinnest face sheet of type 2 material is obtained when both face sheets are made of type 2 , but this produces a sandwich about 10% heavier than one of minimum weight.

TABLE 4.13.2. *Summary table of component dimensions for minimum weight sandwich example.*

Face Sheets	Thickness			Sandwich Weight*
	d	t_{UPR}	t_{LWR}	
	inch	inch	inch	psi
Both face sheets type 1	4.11	0.035	0.035	0.0209
Types 1 and 2	4.66	0.033	0.078	0.0237
Both face sheets type 2	5.19	0.074	0.074	0.0262

* Does not include bond weight.

4.13.3 Sandwich bending moment capacity

The bending moment resistance of a sandwich with thin, equal face sheets on a core of negligible bending stiffness is given by the formula

$$M = Fth \qquad \text{4.13.3(a)}$$

where M is bending moment resistance per unit width, F is face sheet stress, t is face sheet thickness, and d is distance between face sheet centroids. Solving this equation for t and substitution into the weight equation (Equation 4.13.1(b)), then minimizing with respect to h, d (Reference 4.13.2(a)) results in

$$d^2 = \frac{2M\phi}{Fw_c} \qquad 4.13.3(b)$$

and finally

$$t = \frac{dw_c}{2\phi} \qquad 4.13.3(c)$$

Comparison of these expressions with those based on stiffness criteria show that face sheets could be about twice as thick for moment resistance criteria as for stiffness criteria and that d values are dependent on relative stiffness and bending moment requirements. The resultant construction will have about half of its weight in the core.

Example: Determine dimensions of sandwich components so that the resultant sandwich will have a bending moment resistance of 7,000 in.-lb/in. of width. The face sheet properties are F = 45,000 psi design stress (in this case, the yield stress of the metal face sheets) and w = 0.100 pci, and the core weight w_c = 0.0034 pci.

From the equations given above:

$$d = \left\{ \frac{2(7,000)(0.0983)}{45,000(0.0034)} \right\}^{\frac{1}{2}} = 3.00 \text{ inch}$$

$$t = \frac{3.00(0.0034)}{2(0.0983)} = 0.052 \text{ inch}$$

With these dimensions the sandwich weight (minus bond weight) is 0.0204 psi of which 0.0101 psi or about 50% is in the core.

Example: Determine dimensions of sandwich components so that the resultant sandwich will have a bending moment resistance of at least 7,000 in.-lb/in. of width and a bending stiffness of at least $D = 3 \times 10^6$ lb-in^2 per inch of width. The face sheet properties are $E/\lambda = 10^7$ psi, w = 0.100 pci, and w_c = 0.0034 pci.

From the example in Section 4.13.2, the minimum weight sandwich for the required stiffness will have d = 4.11 inch, t = 0.035 inch, and weight W = 0.0209 psi. From Equation 4.13.3(a) the face sheet stress in this sandwich due to the bending moment is

$$F = \frac{7,000}{0.035(4.11)} = 48,600 \text{ psi}$$

and a higher strength material must be used in the face sheets.

If a face sheet material with an allowable design stress of only 20,000 psi must be used, the design must be changed as follows, based on the bending moment criteria

$$d = \left\{ \frac{2(7,000)(0.0983)}{20,000(0.0034)} \right\}^{\frac{1}{2}} = 4.50 \text{ inch}$$

$$t = \frac{4.50(0.0034)}{2(0.0983)} = 0.078 \text{ inch}$$

These dimensions are larger than those necessary for required stiffness, thus the stiffness will be much more than needed (nearly three times as stiff). The sandwich weight (minus bond weight) is 0.0306 psi, which is about 46% heavier than needed for the stiffness criteria only. Thus the use of a stronger face sheet material would be distinctly advantageous.

The face sheet stress, F, should not exceed dimpling or wrinkling stresses given by procedures in Sections 4.6.5 and 4.6.6.

4.13.4 Sandwich panel buckling

The load at which general buckling of sandwich panels occurs is given by the formula

$$N = K \frac{\pi^2}{b^2} D \qquad \text{4.13.4(a)}$$

where N is buckling load per unit panel width, K is a coefficient dependent upon type of loading, type of edge support, panel aspect ratio, and shear parameter V, while b is panel width, D is sandwich bending stiffness per unit width. The parameter $V = \pi^2 D/b^2 U$ where U is the sandwich shear stiffness. V is usually quite small and the dependence of K upon V is of a secondary nature; thus the proportion of sandwich components to produce a minimum-weight construction having a given value of N will be very nearly the same as for a sandwich designed to have a minimum weight and a specified stiffness. It is possible to minimize weight based on panel buckling criteria, but most often so difficult that first approximations based on stiffness criteria suffice. An example is given to demonstrate the procedure.

Example: Determine dimensions of sandwich components such that a simply supported panel 40 inch wide and 80 inch long will not buckle under a load (applied on the 40 inch side) of 40,000 lb. The face sheet properties are $E/\lambda = 10^7$ psi and $w = 0.100$ pci and core properties $w_c = 0.0034$ pci and $G_c = 20,000$ psi.

For this panel the buckling coefficient is

$$K = \frac{4}{(1+V)^2} \qquad \text{4.13.4(b)}$$

and combining this expression with Equation 4.13.4(a) and the weight Equation (4.13.1(a)) and minimizing the weight (Reference 4.13.2(a)) results in:

$$d^3 = \frac{Nb^2(1+V)^2}{2\pi^2 E/\lambda (1-V)\frac{w_c}{4w}} \qquad \text{4.13.4(c)}$$

$$t = d(1-V)\frac{w_c}{4w}$$

For this example, $w_c/4w = 0.0085$, so

$$d = \left\{ \frac{40,000(40)(1+V)^2}{2\pi^2\left(10^7\right)(0.0085)(1-V)} \right\}^{\frac{1}{3}} = \left\{ 0.954\frac{(1+V)^2}{(1-V)} \right\}^{\frac{1}{3}}$$

$$t = 0.0085d(1-V)$$

Assuming values of V and computing d and t, then

V	d (inch)	t (inch)
0	0.984	0.0084
0.05	1.035	0.0084
0.10	1.086	0.0083
0.15	1.141	0.0082

This tabulation shows that variations in V have little influence on d and practically no influence on t.

Assume t = 0.0085 inch, then the face sheet stress is

$$F = \frac{N}{2t} = \frac{1,000}{.0170} = 59,000 \text{ psi}$$

Thus a strong face sheet material must be used; and if the core is honeycomb, it must have a very small cell size to prevent face sheet dimpling. Assuming these are possible, the actual value of V can be calculated. Assume d = 1 inch. Then

$$V = \frac{\pi^2 tdE/\lambda}{2b^2G_c} = \frac{\pi^2(0.0085)(1)\left(10^7\right)}{2(1,600)(20,000)} = 0.0131$$

This value of V is so small that the effect of V is indeed negligible.

The weight of the panel is

$$W = 1(0.0034) + 0.017(0.100) = 0.0051 \text{ psi}$$

Choose a thicker face sheet to lower the stress –

for t = 0.020 inch, F = 1,000/0.04 = 25,000 psi

Then solving the buckling criteria for D and finally d results in

d = 0.64 in. for V = 0

d = 0.65 inch for V = 0.0197 for t = 0.020 inch

The weight of this panel is

$$W = 0.65(0.0034) + 0.040(0.100) = 0.0062 \text{ psi}$$

which is a reasonable value; but is about 20% heavier than sandwich with 0.0085 inch face sheets.

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CHAPTER 5 FABRICATION OF SANDWICH STRUCTURES (MATERIALS AND PROCESSES)

5.1 INTRODUCTION

This chapter covers the manufacturing aspects of sandwich panels and parts. A generic sandwich structure with the primary components labeled can be seen in Figure 5.1(a). *Figures 5.1(b) is a series of photographs briefly showing some of the steps in fabricating a simple honeycomb panel, with a diagram of the final configuration and some important aspects seen in Figure 5.1(c).* ARP 5606 (Reference 5.1) also provides a series of photographs briefly describing the fabrication of sandwich composite nondestructive inspection standards. While the materials and processes shown are not extensive, they can be used as a starting point for considering manufacture of simple sandwich structures.

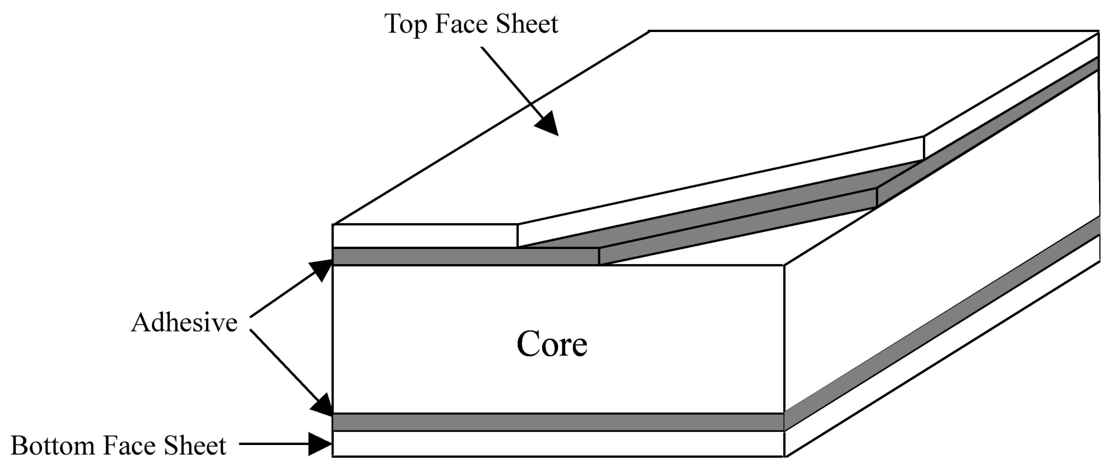


FIGURE 5.1(a) *Generic sandwich composite.*

Processing methods used for solid laminate composites usually need to be modified for sandwich structure, due to the presence of core and an adhesive for the core-to-face sheet interface Aspects such as toughness, durability, environmental aging, and failure modes may change as a result of processing issues.

For honeycomb core, processing plays an additional critical step, as fillets must be formed between the face sheets and the core to obtain a quality bond. The formation of fillets is highly process dependent and will be discussed in more detail later. Attempting to achieve the stiffness to weight advantages that sandwich structure can offer without taking these processing differences into account can result in problems in manufacturing and service, and ultimately increased costs.

There are a number of terms applied to describe various uses of adhesives and bonding in composite structure. Unfortunately, there is inconsistent usage in the composites industry for some of the following terms: bond, adhesive bond, secondary bond, co-cure, and co-bond. In addition, usage of these terms can also differ when applied to conventional laminates, resin infusion laminates, sandwich structure, and repair. The correct term and meaning may also be different depending on the focus of the discussion: an interface (e.g., face sheet to core), a part cross section during a single cure cycle, the part as a whole during a single cure cycle, or a complex assembly which undergoes a number of these processes during multiple cure cycles.

a) Tool

b) Raw materials used

c) Laying up a face sheet from prepreg

d) Applying film adhesive

e) Inserting honeycomb core

f) Bagging assembly

FIGURE 5.1(b) *Basic steps in processing a sandwich panel with honeycomb core and prepreg face sheets.*

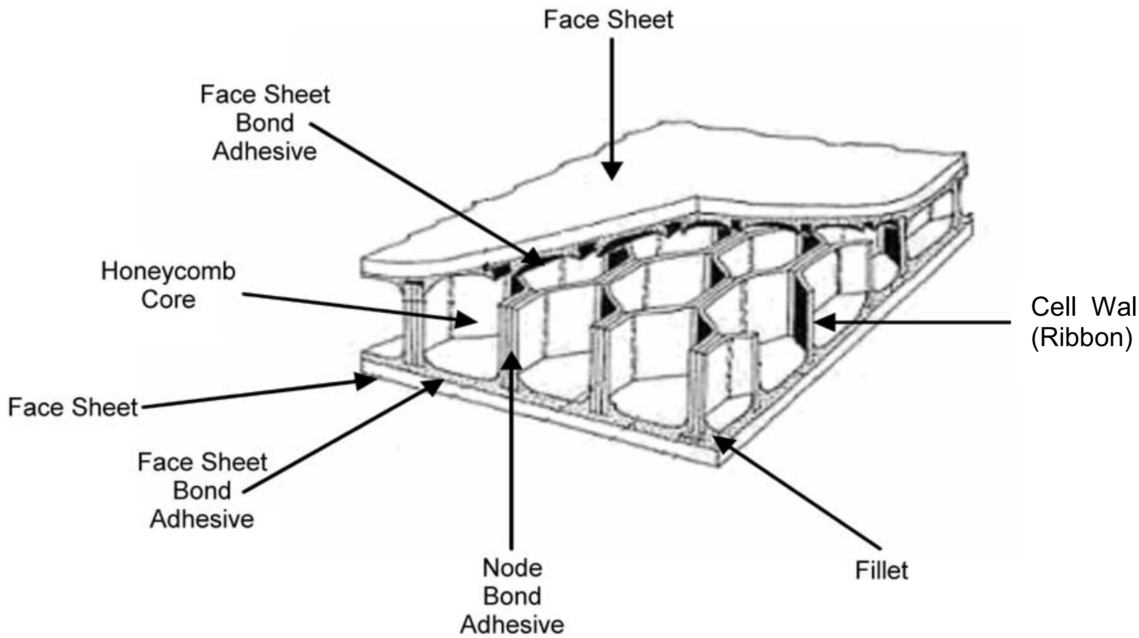


FIGURE 5.1(c) *Honeycomb sandwich composite.*

For this sandwich materials and processes chapter, the following set of terms are defined as follows, using the point of view of the sandwich as a whole undergoing one cure cycle:

- Bond – The adhesion of one surface to another, with or without the use of an adhesive as a bonding agent (same as Volume 1 definition),
- Adhesive Bonding – Bonding of two or more solid substrates using a distinct adhesive material,
- Secondary Bonding – Adhesive bonding when using one or more precured composite face sheets. Also applies to bonding metallic substrates such as aluminum face sheets on aluminum honeycomb core,
- Sandwich Co-cure – Both face sheets curing at same time as the adhesive (except there may not be any adhesive if using self-adhesive prepreg), and
- Sandwich Co-bond – One precured face sheet bonded to the core with the second face sheet cured and bonded to the core at the same time.

5.2 MATERIALS

5.2.1 Cores

Core material forms commonly used for sandwich structure include honeycomb, open and closed cell rigid foams, thermoplastic shapes, and wood. Honeycomb materials include kraft and aramid paper, aluminum, glass and carbon fiber materials, and to a lesser extent, titanium and stainless steel. Wood is still a common commercial core material, and continues to have some application in aerospace such as flooring, but costs have climbed until other materials may be competitive. Chapter 3 contains a detailed discussion of core materials.

Any directionality to the core properties, such as the core ribbon direction for honeycomb, should be controlled as required by the design, analysis, and manufacturing documentation. Some cores outgas during cure, which can present processing challenges, and non-metallic core materials can pick up mois-

ture making for poor core-to-face sheet bonds. These issues and others are discussed in more detail in the following sections.

5.2.2 Face sheets

The same composite materials used to fabricate composite laminates can usually be used to fabricate sandwich structures, especially if the face sheets are pre-cured. It is important to recognize that composite face sheets used on sandwich structure may have substantially different cosmetic and mechanical properties than solid laminates if the face sheets are cured at the same time as they are bonded to the core (co-cured).

Among other issues, dimpling of the face sheets may occur as a result of co-curing, where the impression of the core cell walls telegraphs through the co-cured face sheets to the surface, as seen in Figure 5.2.2(a). A diagram of this condition from the side can be seen in Figure 5.2.2(b). This condition tends to be most pronounced with very thin face sheets and large-celled honeycomb cores. If smooth surfaces are required on the face sheet, substantial and expensive finishing operations may be required for such a dimpled face sheet.

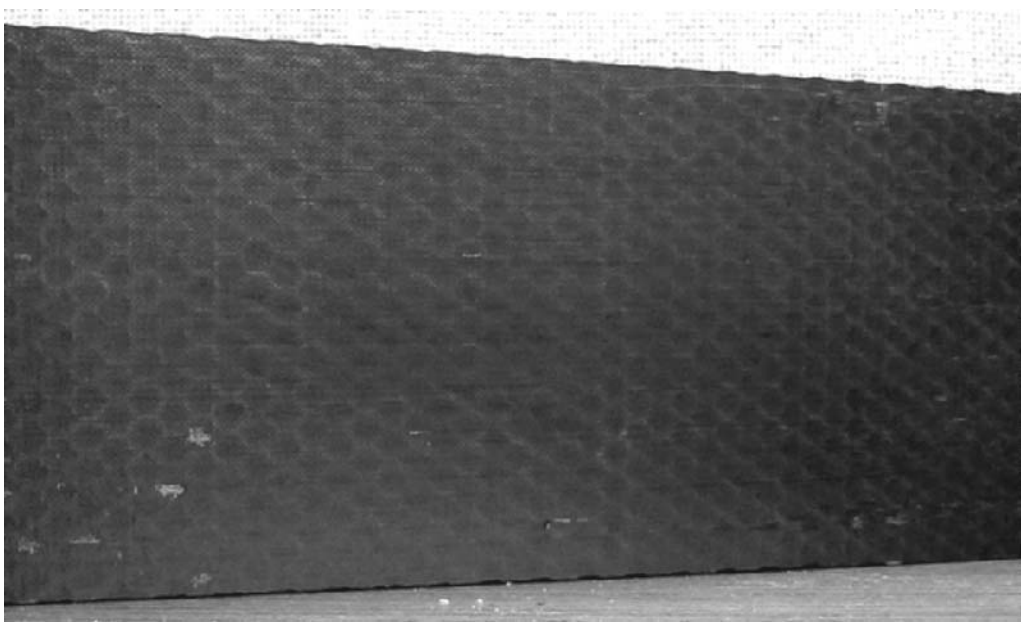


FIGURE 5.2.2(a) *Telegraphing of honeycomb core through a co-cured face sheet.*

In addition, the mechanical properties of a co-cured face sheet may be substantially lower than those for a conventional laminate for reasons in addition to the geometric effects such as dimpling. If open cell honeycomb is used, the core may contact as little as 1-5% of the face sheet surface area. When the face sheet composite laminate is cured up against core it tends to drape into the cells. This results in waviness and makes it very difficult to maintain resin pressure in the composite laminate during cure. Many prepregs may need to experience higher than atmospheric pressure and the accompanying resin pressure in order to prevent excessive porosity or even delamination. Figure 5.2.2(c) shows an example of the “waviness” that can result from a co-cure versus an identical lay-up and honeycomb that was manufactured using pre-cured face sheets. Note the substantially larger adhesive voids on the co-cured face sheet.

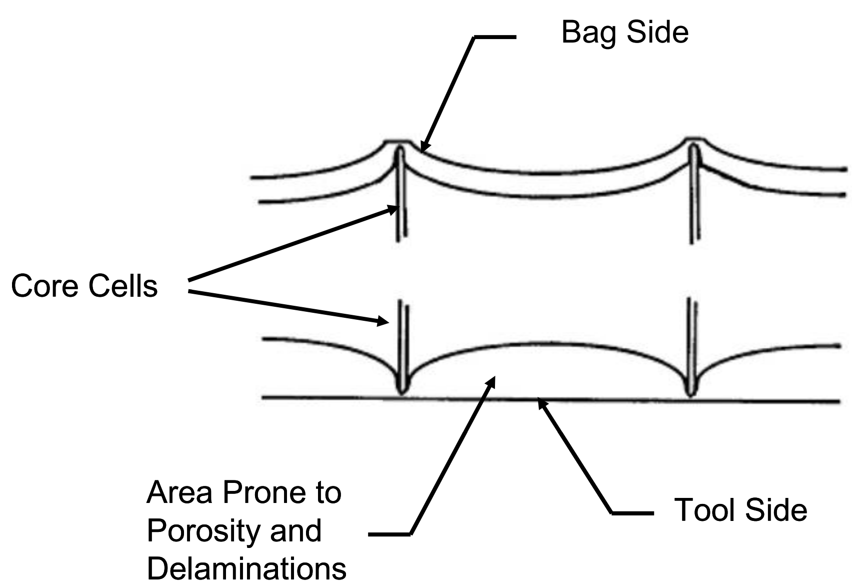
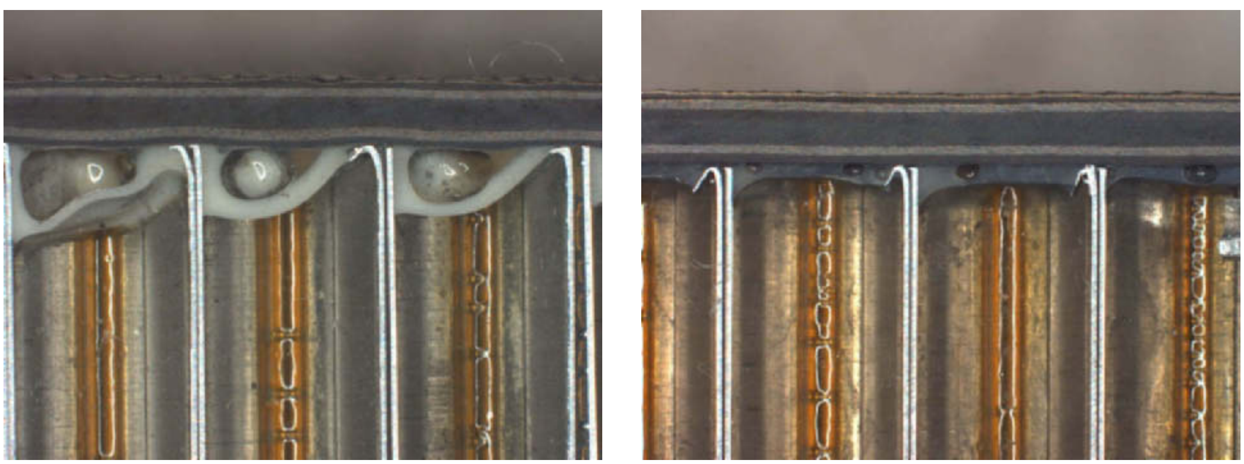


FIGURE 5.2.2(b) *Example of dimpling of co-cured composite face sheet.*



Co-Bonded

Pre-Cured

FIGURE 5.2.2(c) *Waviness of co-cured face sheet versus identical sandwich with pre-cured face sheet.*

As a result, even though the autoclave pressure may be 40 psi, the resin pressure in the face sheet might be just a few psi - or nothing at all. Thus the co-cured face sheet laminates may have lower mechanical properties, similar to a vacuum bag cure or even a contact pressure-cured laminate. If co-curing is intended for fabrication of sandwich structure, composite prepregs are often initially screened for suitability by co-curing representative sandwich test panels or even parts, often before any other testing is performed. It is also possible to co-cure representative face sheets against the core, machine away the core, then mechanically test the laminates to determine actual co-cured face sheet properties. The knockdowns seen for co-cured face sheets can vary depending on the structure and density of the core, prepreg and adhesive formulation, as well as process differences. The knockdowns tend to be most severe in compression.

If thin face sheets are used, again especially if they are co-cured to honeycomb core, permeability of the face sheets after cure may be an issue. Permeability is defined as an interconnecting network of voids, porosity, microcracks and/or delaminations, which provide a direct path between the core cells and the environment. When structures with permeable face sheets are cycled between low and high pressures, such as during an aircraft ground-air-ground cycle, they can breathe moisture laden air into the core. When cooled, the moisture condenses and collects in the core, causing further damage and weight increase. This and the susceptibility to impact damage (such as tool drop and hail) may require the use of face sheets that are thicker than essential for carrying structural loads. If only elimination of permeability is required, films such as a bondable Tedlar or Mylar may be included in the face sheet lay-up, although other service and repair issues may be associated with these materials.

To finish fabrication of a sandwich structure, the face sheets must be joined to the core. For metal and pre-cured composite face sheets, an adhesive is required. For co-cured composite face sheets, the composite resin may also function as the adhesive, but the face sheet laminate properties may be adversely affected because of the reduced resin content. Slightly higher resin content prepreg may be required adjacent to the core for the prepreg to successfully perform this double function. The prepreg resin may function as a less than optimum adhesive, but it still may be adequate for many applications.

For the most part, especially with pre-cured face sheets, the types of composite materials, environmental knockdowns, and application trade-offs for composite face sheets used for sandwich structure are similar to those for conventional composite laminates.

5.2.3 Adhesives

The requirements for adhesive used for bonding face sheets to core can be similar to those for secondary bonding, co-curing, or co-bonding laminates, with some additions or modifications. Adhesives intended for use on sandwich structures (particularly those with honeycomb core) differ slightly from those commonly used for other joining processes in that the flow is better controlled to facilitate the formation of fillets which are necessary for a quality face sheet-to-core bond. It is critical that an adhesive be chosen that is compatible with this intended use. Some composite parts with honeycomb core have failed because they used a film adhesive intended for metal-to-metal bonds that did not flow and form fillets well.

If the adhesive is going to be cured at the same time as a prepreg face sheet, compatibility between these materials must be established, assuring that a weak interface is not created. This weak interface may involve mixing of the adhesive and prepreg resins, which could create a third, weaker material.

For honeycomb sandwich structure, the formation of the face sheet-to-core fillet involves properties such as resin flow, surface tension, and wetting out the face sheet and core materials during the pre-gel portion of the cure cycle. The amount of flow must be limited to prevent the adhesive from running down the cell walls, where it contributes to weight but not the face sheet-to-core bond strength.

There are other reasons to be judicious when selecting a film adhesive for use in a sandwich structure. It is important that the adhesive will flow and bond well under the reduced pressure present when bonding face sheets to open cell core, as discussed in Section 5.2.2. The adhesive must flow some distance into the cell to form fillets, but not run down so much that there is no resin left for filleting at the cell wall-to-face sheet interface. Many adhesives may need to experience higher than atmospheric pressure and the accompanying resin pressure in order to prevent excessive porosity and allow for proper flow. In processing sandwich structures, it is critical to realize that adhesives, even of the same class (e.g., epoxy), can have different chemical formulations, which will cause them to behave differently during cure.

Since nonmetallic core materials can be very hygroscopic, it is important that the adhesive not be excessively affected by prebond moisture. Some adhesives still will volatilize solvents or actually chemically form gasses during cure, which can internally pressurize the core, forming poor face sheet-to-core bonds which may appear as blisters. Trapped gas at too high a pressure can also split or push the core as the

gas attempts to escape to a lower pressure region. Honeycomb core with perforations in the cell walls can be used to allow this gas to escape (venting), and is common for space applications.

Small changes in the surface condition of the core can have a substantial influence on the effectiveness of the adhesive. For example, the presence of fuzz on the tops of the cells after a machining operation, if still firmly attached to the core, can either help anchor the adhesive to the core, or prevent proper fillet formation.

For evaluating bonding to the core, it is common to test similar but higher density core to further stress the adhesive and help identify any deficiencies in filleting or other bonding aspects. Testing alternate materials with less contact area can also be used to provide higher stress for evaluation.

Film adhesives are typically used for bonding the face sheet to the core. Typical uncured film adhesive thicknesses vary from 3 to 15 mils (0.075 mm - 0.38 mm). Adhesive films usually have a loosely woven polyester, glass or nylon mesh or mat embedded in them to facilitate handling and provide bondline thickness control. This fibrous material in a film adhesive is called a carrier or scrim; such a film is referred to as supported. Unsupported adhesive films are discussed below. If the carrier is on the surface of the adhesive film instead of embedded within, it is critical that the carrier face the core, or the bond may be weak at the adhesive-face sheet interface. Film adhesive typically comes sandwiched between a coated release paper and a plastic film. In order to obtain the scrim facing the core, use the “paper-side-to-face sheet” rule.

Film adhesives also come in unsupported form. These adhesives are used where extremely lightweight adhesive is necessary and they are usually reticulated. Reticulation involves blowing hot air through the cells of the honeycomb core to “melt” the film adhesive so that it draws back to the edges of the cells, resulting in the largest possible fillet without any extra adhesive on the inside of the face sheet in the center of the cell. For some applications this adhesive on the face sheet away from the cell walls may have no value and simply adds weight. Since the adhesive tends to gel before the prepreg resin, for some applications it may provide a surface for the face sheet to push against prior to gel, allowing development of some resin pressure.

Adhesive is not always used between a face sheet and the core when a part is co-cured. These self-adhesive face sheets rely on excess resin from the prepreg to form the fillets necessary for a good bond. This technique can save weight on a structure, but is usually limited to very lightweight cores where a secondary adhesive is not needed to give the core-to-face sheet bondline higher strength than the core itself.

5.2.4 Surfacing and sealing

Because of the dimpling, surface sealing, and other issues associated with sandwich composite face sheets, surfacing materials are commonly used with co-cured face sheets. While film adhesive was initially used for this purpose, materials specifically designed for this function are now commercially available. They are similar in form and handling to film adhesive but with decreased density, and improved surface appearance and sandability. They are cured with the prepreg face sheets, and may also have a contribution in reducing core crush. A surfacing ply of a fine weave glass prepreg is also common. MIL-HDBK-349 outlines the steps typically taken in final surface processing (Reference 5.2.4).

For parts that have already been cured, a resin wash with a low viscosity resin can be used to smooth the surface and seal pinhole porosity. A resin wash can be performed as a part of the preparation for paint, although with a definite hit to part weight, especially if multiple coats are required to completely seal and smooth the surface. The part may be initially warmed slightly to help draw the resin into any discontinuities. Conventional fill and fair can also be used, but can have an even greater weight contribution, and a substantial penalty in labor required to smooth and sand the material. For these reasons the required finish and seal for the sandwich face sheets should be seriously reviewed when planning the manufacturing process. Any labor, weight and/or cost savings planned with co-bonding the face sheets may

be lost with this cosmetic smoothing and sealing to meet surface finish and leak requirements, in addition to other issues such as core crush.

Sealing of various areas of sandwich composite structures is frequently required for environmental and mechanical protection. While developed for the fabrication of sandwich composite NDI standards, the methods described in ARP 5606 (Reference 5.1) are representative of some sealing processes typically performed in a production environment.

5.3 PROCESSES

There are a substantial number of processes that are specific to or tailored for manufacture of sandwich structures. While there are some more exotic techniques for sandwich construction such as using resin injection methods, most aerospace composite sandwich structures built today are fabricated with either pre-cured or co-cured prepreg face sheets. Since the main function of the core is to separate the face sheets and carry shear loads with a very lightweight material, most core materials are relatively weak. This presents unique problems during processing, in many cases limiting the pressure that can be applied to the part during cure. When manufacturing with lightweight core, especially honeycomb, extra precautions need to be taken to protect the core since simple finger pressure can damage the material. Many of these processes involve elevated temperature exposure for some period of time. Each user must validate that their combination of multiple heat exposures does not appreciably affect their core and other bond detail properties for their particular application.

5.3.1 Core

A number of core processes are associated with fabrication of sandwich structure, all of which involve some sort of handling of the material. As relatively lightweight materials, core should be handled only when necessary and as little as possible to avoid damage and distortion. Like any other structural adhesive bond detail, clean cotton gloves should be worn when handling core to help protect against contamination. The core should be fully supported and should not be twisted during handling and operations to prevent core damage. It should be packaged for shipping and storage to prevent core damage or contamination. Core material should usually be stored in the original shipping container. If the container has been opened, the core is normally wrapped in kraft paper and returned to the original container if possible. Never stack pieces of core material directly atop one another as this can damage the surface. Separate stacked pieces with non-waxed cardboard or similar materials. When handling metallic honeycomb core, care must be taken at the edges as the cut cells are very sharp.

5.3.1.1 Cleaning

Core that has been delivered clean and has remained in an unopened container during shipping and storage, and is processed in a clean area, may not require further cleaning before bonding. Clean core which is not being used should be wrapped in non-waxed paper or other non-coated covering to protect it from contamination.

In spite of these precautions, some cleaning may be necessary prior to bonding. If the core does require cleaning, spray or immersion in solvent or cleaning agent is frequently used for metallic honeycomb core. Although less effective, localized cleaning with a solvent soaked wipe is sometimes used. Vapor degreasing can be effective for some metallic honeycomb core materials, but environmental regulations may preclude or limit its use. In this case the affected surfaces may be flushed with solvent, and/or solvent wiped. If only dust removal is required, it is frequently performed using filtered compressed air and/or vacuum.

If cleaning is required, some of the guidelines contained in ARP 4916 (Reference 5.3.1.1) may be applicable. Again, although this specification is written for composite repair applications, much of the content is still applicable to original manufacture.

Solvents are increasingly being replaced with water-based solutions, and may be preferred for solvent sensitive materials such as foams and thermoplastics. Use of water-based solutions with lower vapor pressures brings the need for drying. Each combination of core, solvent or cleaning solution, and set of process parameters should be confirmed through testing as effective for removing the contaminants in question, adequately preparing the core surface for bond, and ensuring that the core properties of concern are not significantly affected. Any cleaning material should not leave any kind of residue on a surface that will be subsequently bonded.

For spray, immerse or degrease processes using either solvent or cleaning solution, the core is racked on a gridded carrier, preferably with any open cells in a vertical position to permit rapid drainage of the liquid. Do not stack the core details. Spray or immerse the core with solvent or cleaning solution, then allow solvent to drain. After removal from the chamber, dry the core as specified. Position the core so the cell cavities are fully exposed to air circulation and adequate drainage of entrapped solvent or cleaning solution is provided. If cleaning of aluminum honeycomb core is required some of the guidelines contained in Chapter 5 of MIL-HDBK-349 (Reference 5.2.4) may be applicable.

For localized cleaning (less than 10 percent of the total area), core is commonly cleaned as follows. Wipe the core with clean cheesecloth moistened with solvent or cleaning solution. Do not saturate the core with solvent or cleaning solution, using only the minimal amount. Before installing the core, dry as specified. When core is only soiled with dust, remove the dust and residue using filtered (oil-free) compressed air and/or vacuum.

Foam cores and areas containing cured foaming adhesive (splice joints) or potted areas can be cleaned by lightly sanding with 240- to 320-grit sandpaper. Optionally they can then be wiped with clean cheesecloth moistened with solvent or cleaning solution.

When using solvent or cleaning solution on bond details, care should be taken to assure that adequate time is allowed for the solvent to fully flash off the surfaces. Some newer, more environmental regulation-compliant cleaners may take substantially longer to flash than older solvents. Any solvents or cleaning solutions not eliminated at room temperature should be completely removed by drying in an oven or with a heat gun limited typically to about 150°F (66°C). Extended drying at room temperature may be successful, but any residual solvent which volatilizes during bond of the sandwich structure may cause problems.

Although honeycomb that has been externally contaminated requires cleaning, commonly accomplished using solvents, it should be noted that polymers can contain low molecular weight compounds that can be leached to the surface by solvents and interfere with bonding if concentrated at the bondline. While technically not contaminants, they can interfere with bonding in the same manner. This issue is commonly dealt with by heat conditioning the non-metallic core before use, as covered in the following section. It can also be handled with additional cleaning cycles, perhaps with other cleaning processes and materials.

The out-time for core in a particular environment prior to requiring cleaning, re-cleaning, and/or drying needs to be established for each core material, bond process, and application.

5.3.1.2 Drying

Removal of moisture absorbed into the core material is a general issue for sandwich composite manufacturing, as prebond moisture exposure can be detrimental to the cured bondline properties and internal pressure (steam pressure) inside the core cells for open cell honeycomb can damage or destroy the core.

An example of honeycomb core, which has blown due to internal pressure, can be seen in Figure 5.3.1.2. Nonmetallic core assemblies or details should be oven dried (and kept dry) before splicing, beveling, applying film adhesive, stabilizing, potting, heat-forming, machining, or assembly operations.

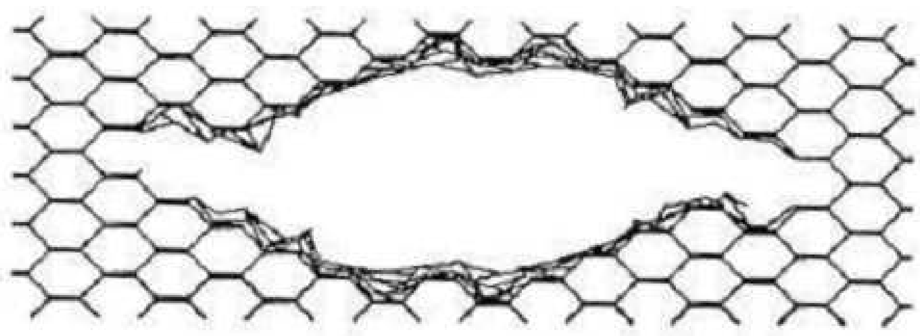


FIGURE 5.3.1.2 *Blown core.*

Nonmetallic core should be dried after all cleaning operations, then packaged in moisture-proof or moisture-resistant sealed packaging, preferably with a desiccant. Packaging materials that do not contact the details should not offgas any materials detrimental to bonding. Packaging materials that do contact the bond details should not transfer any materials detrimental to bonding. Drying after solvent cleaning may help remove residual solvents or cleaning agents from cleaning operations as well as absorbed moisture. Drying of composite materials is addressed by ARP 4977 (Reference 5.3.1.2). Although written for repair of composite structure, most of the content is applicable to original manufacture as well.

Common drying temperatures range up to 250°F (121°C) for core alone, and from 140 to 180°F (60 to 82°C) for longer time periods for bond details including core. Drying of core and other bond details in a production setting will typically be performed in an oven, although a variety of heat sources, such as radiant heaters, could potentially be used. Most cores can be placed in a hot oven, but delicate details or subassemblies may need to be heated to the final drying temperature at a controlled rate to avoid damage.

Some cores, bond details, or subassemblies may be dried at temperatures as high as the cure final dwell temperature, but may require extensive tooling for support and specialized processes to avoid distortion or other damage. Core may also be heat conditioned at the maximum temperature the core will be exposed to in subsequent processing, but this is usually intended to drive off residual core material constituents (low molecular weight compounds) rather than just moisture.

A typical process for drying nonmetallic core is as follows. Place the core in the oven, then raise the temperature to the highest subsequent processing temperature, based on the free-air temperature. Hold the temperature for a minimum of about 120 minutes, beginning the hold when the free-air temperature reaches the bottom end of the temperature dwell range. After completing the dwell, cool to 150°F (66°C) or lower before removing the core from the oven.

The following nonmetallic honeycomb core receiving and storage requirements are typical. Before any processing or assembly operations, nonmetallic core that has been dried is stored in a clean room for no longer than the following maximum times: 72 hours when wrapped in kraft paper, 14 days when sealed in moisture-proof bagging/packaging material, or 90 days when sealed in a moisture-proof bagging/packaging material with desiccant. When the listed storage times have been exceeded, the core is re-dried and reprocessed for storage as described above. Desiccant bags are re-dried if exposed to ambient environment conditions for more than 8 hours, typically at 230 to 260°F (110 to 127°C) for a minimum of 16 hours.

Common in-process composite procedures for protecting nonmetallic core from moisture pickup are as follows. During lay-up procedures, the core should be stored as specified until ready to install in the lay-up. Core laid-up on a tool that is exposed to ambient clean room environmental conditions should be covered with adhesive film or prepreg within 24 hours. Exposed core should be covered with a vacuum

bag sealed to the tool to limit exposure to ambient conditions for a maximum of 72 hours if it has not been covered with adhesive film or prepreg.

The out-time for core in a particular environment prior to requiring drying, re-drying, and/or cleaning needs to be established for each core material, bond process, and application.

5.3.1.3 Forming

Metallic honeycomb can be mechanically formed using a brake or rolls. The core surfaces may need to be protected from direct contact with the rollers using a thin sheet.

Some forming for nonmetallic core, such as introducing a complex contour, may be performed by heat forming core that is flexible enough to avoid being damaged by the process. The higher the density of the core, the thinner the sheet of core must be to avoid damage.

Nonmetallic honeycomb core material is commonly heat-formed as described below, where the nominal density does not exceed 6.0 pounds per cubic foot (96 kg/m^3) and the thickness is one inch (25 mm) or less. Additional flexibility for thicker or heavier core sections may be obtained by forming “green” (partially cured) core. Depending on the processing temperature and degree of curvature used, heat-forming may alter the mechanical properties of the core; testing may be required to determine this effect.

Heat-forming is critical enough to be specified on the engineering drawing or applicable fabrication planning. Drying is typically performed before heat forming, and heat forming is typically not performed after core stabilization of any kind. There may be instances where it is necessary to stabilize core before machining and subsequently heat form after machining. Use of double-backed tape to stabilize the core will not interfere with subsequent heat forming operations.

The core section is placed in an oven at a temperature which will soften but not permanently damage the resin. This temperature may vary by application, and the heat damage (if any) should be assessed by each user. A forming tool may be used in the oven, perhaps with pressure or mechanical weight (e.g., heavy chain mesh) to force core contact with the tool, or this may be quickly performed outside the oven after the core has been removed but is still warm. The forming tool may need to accommodate spring-back of the core. The time and forces required for this forming are dependent on the core material density and stiffness, the contour being formed, and the temperatures used. If pressure or mechanical force is applied to the core, care needs to be taken to not physically damage the details.

Nonmetallic honeycomb core is commonly heat-formed as follows.

a. Cut a sheet of nonporous, Tetrafluoroethylene (TFE) coated fiberglass cloth release film to the same dimensions as the part.
b. Tape the release film to the tool with nonsilicone flashbreaker tape.
c. Position the honeycomb core on the tool and form with hand pressure using nonsilicone tape to hold the core in place. All areas of the core do not have to be in direct contact with the tool.
d. Cut a sheet of porous, TFE coated fiberglass cloth release film approximately 2 to 4 inches (50 to 100 mm) larger than the core surface.
e. Place the release film over the top of the honeycomb core.
f. Cover the entire lay-up with one layer of dry cloth material. Pad all sharp corners with additional dry cloth material.
g. Place a nylon vacuum bag over the entire lay-up, and extend several inches beyond the part edge.
h. Locate the vacuum and static lines adjacent to the lay-up and inside the sealed edge of the vacuum bag.
i. Seal the nylon vacuum bag to the tool with vacuum bag sealant.
j. Position the assembly (tool and bagged lay-up) in an oven preheated to 450 to 475°F (232 to 246°C).
k. Hold the oven temperature at 450 to 475°F (232 to 246°C) for 5 to 10 minutes,

l. Slowly apply a vacuum at 0.5 to 1.0 inch of mercury (1.7 to 3.4 kPa) per minute until 20 to 29 inches of mercury (70 to 100 kPa) is reached.
m. Hold the vacuum at 20 to 29 inches of mercury (70 to 100 kPa) and temperature at 450 to 475°F (232 to 246°C) for 15 to 25 minutes.
n. Cool the oven to a minimum temperature of 150°F (66°C) before venting the vacuum. Cooling may be accelerated by opening the oven doors.

After an open cell core section has been formed to a desired shape, the perimeter may be potted to stabilize the shape. Core which has been heat formed or distorted in handling but not stabilized in some manner may attempt to revert to its previous shape to some degree during subsequent heat exposures, such as cure of the face sheet-to-core bond.

5.3.1.4 Splicing

Due to design and use constraints, different areas of a structure with sandwich core may use different or modified cores. Multiple sections of the same type of core may be required for a large part. Splicing is used to locally change the core to one with different properties, or attach smaller pieces of core into a larger piece. The splicing can be done prior to bond of the face sheets to the core, or during the same cure.

If a core insert is going to be completely surrounded by another section of core, the insert is typically cut one cell larger than measurement would indicate for a snug fit. Even if not specifically controlled by the design, analysis, or manufacturing documentation, it is good practice to match the ribbon direction or other core directionality when splicing sections of core together.

Once the individual core sections for a continuous structure have been cut and formed to size, they must be bonded (or spliced) together. These bonds can be formed at the same time as the face sheets are bonded for simple flat parts, but the complexity of many structures demands that the core details be bonded in advance, perhaps even before completing core machining. Typical criteria for a honeycomb edge to be spliced (bonded) are shown in Figure 5.3.1.4(a).

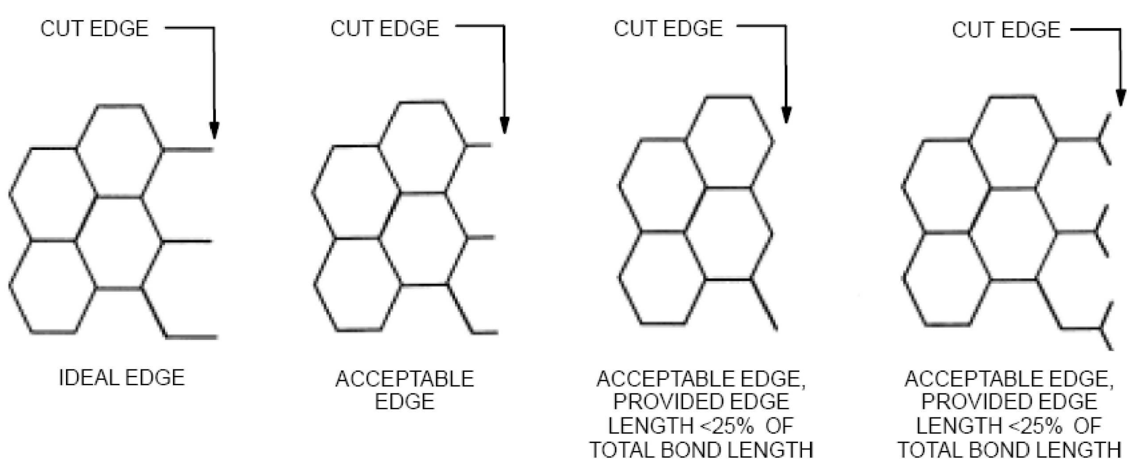


FIGURE 5.3.1.4(a) *Typical criteria for honeycomb cut edges to be spliced.*

These bonds are usually formed with adhesives specifically designed for core splice. The adhesives used may be heat curing film adhesives, either foaming or not. If the adhesive does not expand, multiple layers may be required to establish an adequate bond. For small jobs, two-part paste adhesives can also

be used. A bond may also be formed between the edge of the core and any close-out detail using foaming adhesive, or just to stabilize the edge of the core. Regardless of what adhesive is used, storage time and temperature and cumulative out-time must be in accordance with established requirements.

When splicing together two flat pieces of core where further core machining is not planned, care needs to be taken to keep the surface planes in line so that there is not a step between the pieces across the splice. If one core piece is significantly more compliant than the other, care is usually taken to have the more compliant piece slightly higher than the other piece.

Recurring splicing of core material should be specified on the engineering drawing or applicable fabrication planning. Rules for splicing that manufacturing personnel can perform as needed should also be documented. Core material must be clean and dry, and stored as specified before any splicing operations. Splicing operations should be performed in an area meeting the clean room requirements.

Core splices should be interlocking butt splices as shown in Figure 5.3.1.4(b), and may be parallel or perpendicular to the ribbon direction. Splicing (foaming) adhesive for assemblies needs to be compatible with the cure temperature range and time established for the face sheet and adhesive materials.

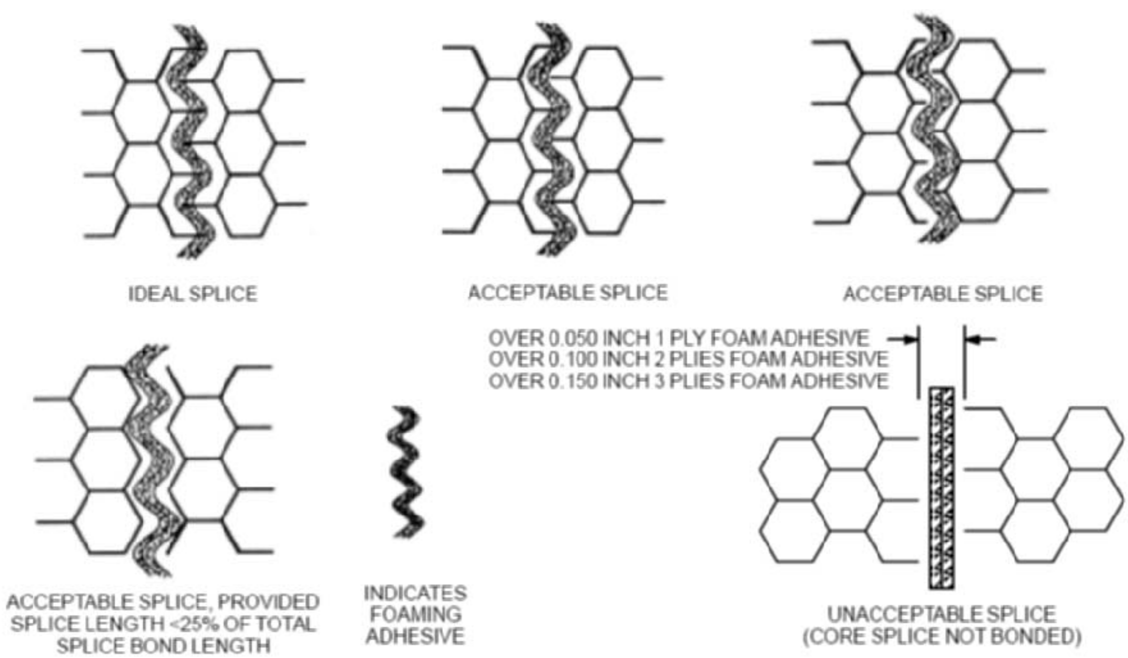


FIGURE 5.3.1.4(b) *Interlocking butt splices.*

Splicing adhesive does not require curing before the spliced honeycomb details are placed in composite lay-up assemblies. When specified by the fabrication planning, splicing adhesive may be co-bonded during the assembly face sheet-to-core cure cycle.

Core is spliced as follows. Only splice core that has been cleaned in accordance with the procedures outlined in Section 5.3.1.1. Butt splice the two core details as shown in Figure 5.3.1.4(b) when core edges are cut to an acceptable condition shown in Figure 5.3.1.4(a) for the majority of the splice area. Place the required layers of foaming adhesive in the joint as indicated in Figure 5.3.1.4(b), which assumes a nominal foaming adhesive thickness of 0.050 inch (1.27 mm). Double the number of layers when the nominal foaming adhesive thickness is 0.025 inch (0.64 mm).

After allowing the foaming adhesive to warm to room temperature, cut the foaming adhesive into strips to fit the core edges to be joined. Remove the protective backing from one side of the foaming adhesive strip and place the foaming adhesive on one of the core edges to be bonded. Then remove the protective backing on the other side of the adhesive strip and butt the core sections together, taking care not to significantly bend (or crush) the edges of the core. It is imperative that all paper backing be removed from the foaming adhesive used.

If the core splice is to be cured at the same time as the face sheet bond, then the core detail with foaming adhesive can be installed into the sandwich part assembly. If the core splice is to be pre-cured before it is placed in the composite lay-up assembly, typically the core detail is bagged as follows:

1. Place one layer of nonporous, TFE coated, fiberglass release film or nonporous TFE release film on the project plate or tool. Extend the release film a minimum of 2 inches (50 mm) beyond the bond line. When required, tape the release film in place with nonsilicone flash breaker or release tape.
2. Place the honeycomb core assembly on the release film. Secure the honeycomb core in place with nonsilicone adhesive tape so the core remains stationary during cure.
3. Place another layer of nonporous, TFE coated, fiberglass release film or nonporous TFE release film over the joint. Extend the release film a minimum of 2 inches (50 mm) beyond the joint bond line.
4. A caul plate may be placed over the release film if required to control adhesive flow.
5. Place one or two layers of dry cloth breather material over the top of the lay-up. Cover all sharp corners and other projections on the lay-up with additional dry cloth material to protect the vacuum bag from puncture during cure.
6. Place a nylon vacuum bag over the entire assembly, and seal to the tool with sealing tape.
7. Locate the vacuum and static lines adjacent to the honeycomb core and inside the sealed edge of the vacuum bag.

Place the core assembly in the oven. Cure the foaming adhesive using the manufacturer's recommended cure cycle unless another has been established, typically applying a vacuum of 6 to 12 inches of mercury (20 to 40 kPa). Raise the oven temperature to the dwell temperature range in 30 to 300 minutes, based on free-air temperature.

The dwell temperature is typically the cure temperature used when bonding the face sheets to the core, which includes the splice. Hold the oven temperature for 120 to 150 minutes. For assemblies that will be bonded at a suitable temperature range and time established for the splice material, the splice cure may be terminated once the core splice is stable, as the cure will be completed during the subsequent face sheet bond cure cycle.

Cool the oven to 150°F (66°C) a minimum of 40 minutes before removing the core from the oven. Release the vacuum after the temperature reaches 150°F (66°C). Cool the core to room temperature and remove the spliced core from the plate or tool using extreme caution to not damage the core. Visually inspect the splice(s) for excessive voids or other defects.

When co-curing the face sheets, the entire sandwich panel will be fabricated in one cure as described later. The foam splicing material must be compatible with the cure cycle of the face sheet material.

Fiberglass honeycomb has the additional alternative of a crush splice, where short overlaps between the two core pieces are driven together. Splicing thicker sections of core in this manner should use beveled edges. Crush splicing may be possible but less effective with some aluminum honeycomb cores.

5.3.1.5 Potting

Potting (filling) is used to selectively reinforce sections of sandwich core for hardpoints, fasteners, or other tasks where the base core properties are not sufficient over a limited area. For relatively lightly loaded areas, the same foaming adhesive used for splicing may be adequate. Higher density core may also be used. Foam in the core cells may also slow the intrusion of water, stabilize the walls for machining, create a better thermal insulator or meet other objectives, but it may not make the core detail stronger or stiffer than an equivalent increase in weight from a higher density core.

However, if heavier loads are anticipated, syntactic foam has a good balance of light weight and improved properties. If very heavy loads are anticipated, epoxy filled with chopped fibers is commonly used, or even solid laminates or metal inserts for highly stressed point locations. See Figures 2.7.1(a) and (b) for some examples of reinforcement methods.

When specified, core cells or areas are potted in accordance with established requirements. Before potting, foaming adhesive is usually removed from areas to be potted (especially if foaming adhesive is not being used for the potting). Core may or may not be removed from areas to be potted. If the core is removed, the core at the periphery should not become distorted as a result of the removal or the potting operation.

Clean the core prior to potting and allow the potting compound to warm to room temperature if refrigerated. Mix the potting compound according to the manufacturer's recommendation and apply the compound using an injection gun, spatula, or trowel. Protect the core surfaces around the area that will not be potted to minimize the chance of damaging the core with either stray compound or the tools used to apply the potting compound. A clean sheet of thin aluminum with a cutout for the potted area works well. Cure the potting compound according to the manufacturer's recommended cure cycle if no other cure cycle has been established. If co-curing, make certain that the potting compound is compatible with the cure cycle that will be used to bond and cure the face sheets.

Although ARP 4991 (Reference 5.3.1.5) is written for the repair of damaged sandwich structure, many of the techniques are applicable or comparable to those used for the potting of core details during original manufacture for bonding inserts, creating hardpoints in the core, splicing in sections of core, or other purposes. The specification covers multiple procedures for filling core areas with resin or potting material. ARP 5606 (Reference 5.1) also covers procedures for potting cored areas and, again, although intended for the fabrication of inspection standards, is representative of manufacturing procedures.

Closeouts can consist of edge filler, or extruded details in a C, U, or Z configuration. Inserts can be custom fabricated, or be as simple as hard points created using foaming adhesive, potting material, or even solid laminate inserts.

Honeycomb is sometimes filled with foam primarily to make it a better thermal insulator, such as foamed directly into the cells using foaming adhesive. Very low density friable foams can also be press crushed into honeycomb cells, using rubber pads to press the foam below the core wall so the face sheet fillets can be formed.

5.3.1.6 Septums

Septums are used to splice sections of core through the thickness, such as when a two inch thick core is desired and only one inch thick core is available. The core pieces are bonded together using film adhesive in a manner similar to the core-to-face sheet bond. Septums may also be used for some acoustic and Radar Cross Section (RCS) functions.

One manner of septum use is described in ARP 4991 (Reference 5.3.1.5). A glass fabric prepreg ply is used between the two layers of core, as described under the Thick Core Plug Preparation method. Usually, there is not an attempt to match the cells between the two layers, but some special applications

may require that to some degree. Some will use film adhesive on both sides of the glass ply, or in place of the prepreg, depending on experience and application.

Some destructive test validation of the septum bonding process employed should be performed at the scale it will be used, especially since very low pressure is typically used during cure of the septum bond. Thick core layers may also insulate the septum bond during cure, so the actual bondline temperature may be substantially lower than anticipated or desired. The actual bondline temperature should be confirmed through measurement, although a partial cure may be acceptable if the cure can be successfully completed during subsequent bond operations.

5.3.1.7 Core stabilization for machining

Machining of honeycomb core usually requires some form of stabilization for the core to keep its shape during the machining operations. Polyethylene glycol solutions, vacuum chucks, or even ice have been used for machining stabilization. Core may also be stabilized for machining using materials such as double-sided pressure-sensitive adhesive tapes. In addition, layers of film adhesive or temporary face sheets may be bonded to the side opposite that being machined and held down with vacuum. In any case, the materials used for stabilization should not introduce contamination, which could interfere with subsequent bond operations.

5.3.1.8 Machining

There are many processes that core may undergo to prepare it as a bond detail. Machining or cutting is an initial step if complex core detail shapes are required. Low density or honeycomb core may need to be stabilized as described above during cutting and machining operations. Core material should also be clean and dry before any machining operations.

Best practice for machining and forming core details prior to bond is to prevent contamination to the greatest degree possible. Core should be machined in a clean area that precludes contamination of the core details. Core details should be handled like any other structural adhesive bond detail, avoiding contact with surfaces to be bonded, and using clean cotton gloves when contact is required.

Other than stabilization materials, no other process materials are used, especially oils which would be difficult to remove after machining and would affect subsequent bonding to the core. Lubricating oil, grease, or other foreign material should not be used on the core or cutting tools unless specifically authorized. It is a common practice to dedicate forming machines and work cells to core fabrication processes to eliminate the potential for contamination.

If no contamination is being introduced as a part of the machining operation, then vacuuming or blowing off the residual dust with compressed air may be sufficient for cleaning after machining. Compressed air used should be trapped and filtered to prevent compressor oil or any other contamination from being distributed on the core. Note that traps need to periodically be drained in order to retain their effectiveness. After machining, all dust and other residue must be removed and drying may be performed.

The machining procedures and hardware used for fabrication of composite structure are fairly straightforward, with much in common with those used for wood or light metal-working. While numerically controlled machines can perform almost every machining operation, they may not be required for lesser complexities, although the tolerances may not be as tight. Core details can be fabricated using saws (band, jig, circular, or cut-off) and drills (hand-held or press). Hand-held routers may be used for simple reliefs. Blades specifically designed for the core and specialized bits may be required for optimum results, such as diamond grit or tungsten carbide edged tools.

Abrasive methods such as grinding and sanding may be used on glass-reinforced honeycomb and many foam cores. If the greater (and nonuniform) density and potentially lower mechanical properties are acceptable, the core may be crushed to shape (at room or elevated temperature). In addition, honeycomb HOBE (Honeycomb Before Expansion) may be cut or machined prior to expansion, although thick-

ness transitions may not be as smooth as when machined after expansion. Unexpanded aluminum honeycomb HOBE and expanded core can be seen in Figure 5.3.1.8(a).

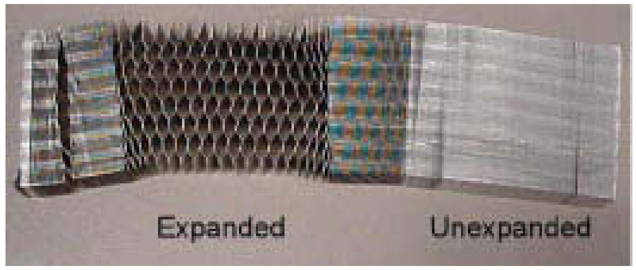


FIGURE 5.3.1.8(a) *Expanded and unexpanded HOBE aluminum honeycomb core.*

The machining tools can be hand-held for prototype or low rate production, with cutting and drilling templates for moderate rate production, or computer-controlled for high rate production. Feed and speed rates should be evaluated and optimized for the core materials, the bit or blade materials and construction, the detail or part configuration, and the quality level and tolerances required. The type of core, required dimensions and tolerances, and density will typically dictate the type of appropriate cutter and control. Thicknesses less than 1/8" are difficult to handle and not common, and thickness tolerances also vary by thickness.

Nonmetallic core may be machined (edge beveling or cutout) as follows. First stabilize honeycomb core using one of the methods described in Section 5.3.1.7. Machine the core as specified on the engineering drawing. When the core is to fit against closure members, cut the core slightly oversize to ensure a snug fit. Remove any burrs greater than 0.005 inch (0.13 mm). After machining, remove all dust and residue using filtered air and/or vacuum. Honeycomb core beveling may be rounded as a manufacturing option (see Figure 5.3.1.8(b)).

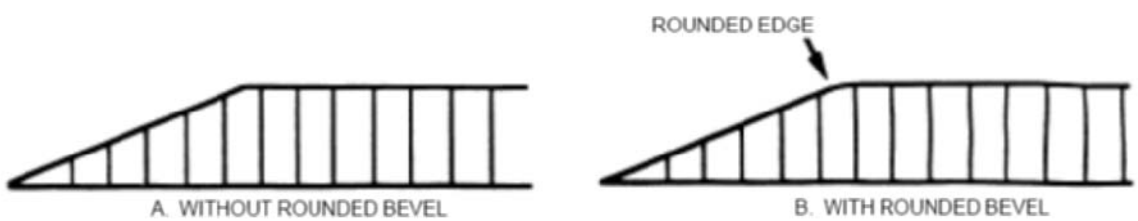


FIGURE 5.3.1.8(b) *Beveled honeycomb core.*

After machining, core details should be visually inspected for any evidence of damage such as delamination of the node bonds or contamination. After machining, the core material should be cleaned and stored to keep clean and dry.

For manufacturing cleanup of core with small damaged areas, foaming adhesive or comparable materials are commonly applied within the following limits. The maximum area is typically 0.25 square inch (1.6 cm^2), and the maximum length is 0.50 inch (13 mm) perpendicular to the core edge. The maximum frequency is that these areas be no closer than 12 inches (300 mm) from another. Areas exceeding the established limits are typically reworked by removing the affected area and splicing in a new piece of

core. Spliced pieces should be beveled as required, and should extend a minimum of 1 inch (25 mm) past the bevel.

Early honeycomb core had to be perforated because of volatiles exuded during the cure by adhesives in use at that time; this configuration is still preferred for some space applications. If the open cell core may be exposed to rapid temperature increases or pressure changes, venting of the core may be required to prevent internal cell pressure from damaging the core or the core-to-face sheet bonds. If the core has not been procured perforated for this purpose, sometimes it is further machined with holes through the cell walls, or slots machined in the tops of the cell walls. The slots machined must be deeper than the face sheet-to-core bond fillets, but not usually deeper than twice the slot width. Use of the slots should be characterized for effects on mechanical properties, especially compression.

5.3.1.9 Tolerances

Sandwich structure core materials are frequently much less dimensionally stable compared to solid laminate and metallic materials. Tolerances appropriate for non-cellular materials should not be arbitrarily applied to sandwich core materials, or extreme costs may be incurred. Dimensional tolerances on core can be substantially larger than those for other bond details. These tolerances, and the compliance of the core during bonding operations, should be taken into consideration.

Thickness tolerances for core bond details are typically much more than ±5 mil (0.13 mm) for overall thicknesses up to two inches (50 mm), and at least twice that up to four inches (100 mm). Substantially looser tolerances may considerably reduce costs. Solid or less flexible bond details adjacent to core details should have proportionally tighter tolerances, approaching those used with metal-to-metal bonding, certainly no more than ±5 mil (0.13 mm), and perhaps less than ±3 mil (0.08 mm) in order to consistently meet desired bondline thicknesses.

Tolerances for the core are typically more critical locally, where the core interfaces with rigid bond details, such as a closeout. Given deflection of the core and relatively thin bondlines between the face sheet and core (compared to the fillets along the cell walls), the core is typically machined somewhat oversize, commonly about 10 mils (0.25 mm), so that the core bites into the film adhesive.

Fraying or burrs (soft edges) on the core can also be a problem, but ultrasonic cutters, which leave a sharp edge, may actually require different stackup tolerances than conventional machining operations, which leave softer edges.

To meet bondline thickness requirements, bond details (including core with curvature) require not only that close tolerances for the details be maintained, but also registration of the details with respect to each other becomes critical.

5.3.2 Face sheets - co-cure vs. pre-cure and resin pressure

When the face sheets are cured at the same time as they are bonded to honeycomb core, a serious quality effect is that the prepreg is only supported during cure at the tops of the cell walls. The thinner the face sheet, the more prominent any drooping in the unsupported areas between the cells is likely to be. This surface effect is known as telegraphing (or dimpling) as mentioned in Section 5.2.2. The larger the cell size, the greater span that must be covered and increased droop may be seen. If the face sheets are co-cured then it may not be possible to optimize the cell size for structural considerations, but necessary to reduce it for manufacturing considerations. The co-cured face sheet may appear resin starved because of increased porosity, especially on the tool side.

An additional quality effect when the face sheets are cured at the same time as they are bonded to the core is that the resin pressure during cure, which directly affects laminate quality, is at or near zero at points above the cells. This can lead to quality problems such as porosity. The thinner the face sheets, the more likely that a void network path may be formed in the face sheet, which directly connects the cells

to the exterior environment. As moist air is drawn into the cells and the moisture condenses as the air is cooled (dew), liquid water collects in the core.

Because many manufacturers experienced chronic quality problems with co-cured face sheets, several studies on the factors that affected part structural and cosmetic quality were undertaken (see reference list in Reference 5.3.2(a)). To measure resin pressure, pressure transducers were mounted into a flat aluminum tool. The transducers were recessed, and the recesses filled with uncatalyzed resin so only hydrostatic resin pressure was measured, not mechanical forces. A multiple channel data collector was used to monitor the transducers. Hydrostatic resin pressure was monitored for both laminates and thin-skinned sandwich structures in the autoclave. Figure 5.3.2(a) shows that adequate resin pressure is fairly easy to maintain with flat laminates. Hydrostatic resin pressure at the corner of the laminate is about ten percent less than the applied autoclave pressure. It was noted that at the center of the laminate, the resin pressure loss is greater than ten percent. The cause of the reduction in resin pressure for this particular run was a small resin leak at the center transducer. However, the resin leak did not affect the resin pressure at the corner location, 6 inches (15 cm) away. This implies that laminates with high fiber volumes can have relatively independent resin pressure zones in close proximity. This partially explains why composite structures can be manufactured with gross variations in quality related to resin pressure in close proximity.

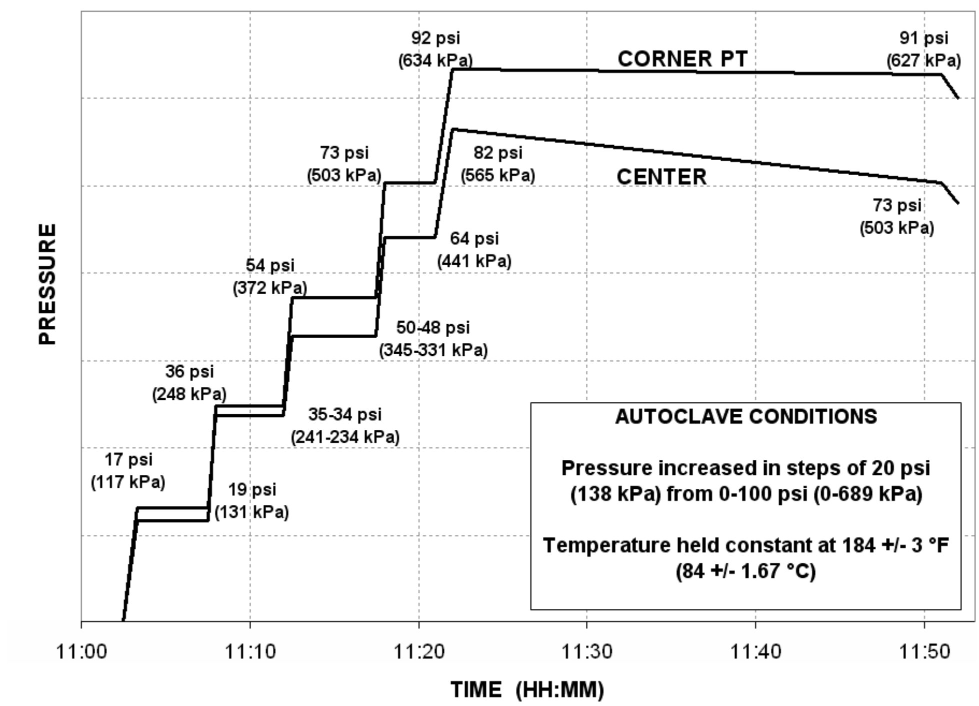


FIGURE 5.3.2(a) *Resin pressure readings at center and corner of a solid laminate panel (Reference 5.3.2(b)).* (Note that different scales are used for the locations)

Figure 5.3.2(b) shows that resin pressure is difficult to maintain with honeycomb sandwich structures. This partially explains the additional difficulty in manufacturing high quality honeycomb structures in comparison to laminates. In such a case, it would be necessary to maintain a long intermediate dwell to get rid of any moisture in the lay-up and allow for adequate flow and consolidation to occur. In spite of the extremely low resin pressure under the honeycomb, Figure 5.3.2(c) shows that the edge band area still maintained adequate resin pressure, preventing moisture-induced void formation. This reinforces the concept of independent zones of resin pressure (and resulting laminate quality).

An additional issue when face sheets are co-cured is core crush, which is much less likely with pre-cured face sheets compared to co-cured face sheets. Core crush is discussed in Section 5.4.

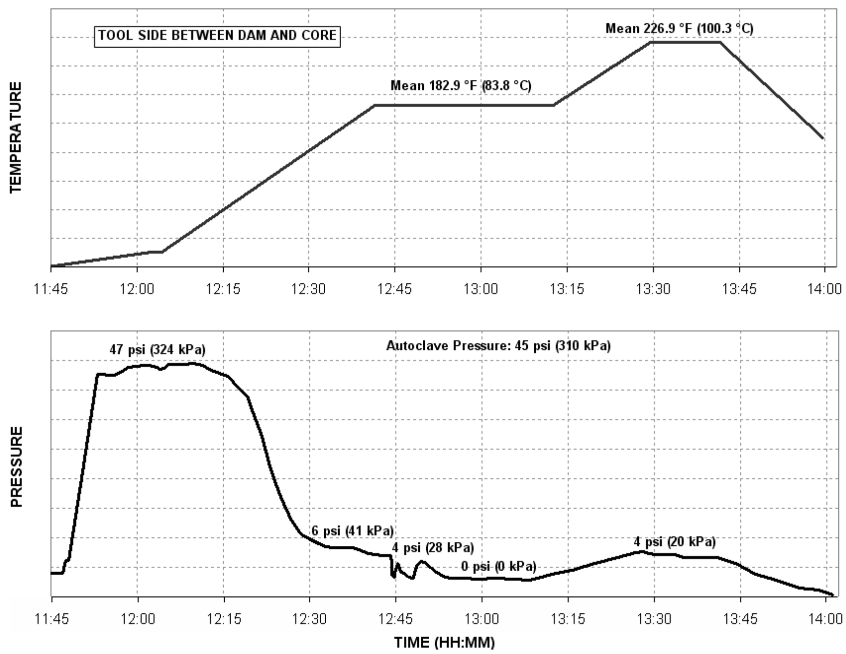


FIGURE 5.3.2(b) *Resin pressure readings for honeycomb sandwich structure (Reference 5.3.2(b)).*

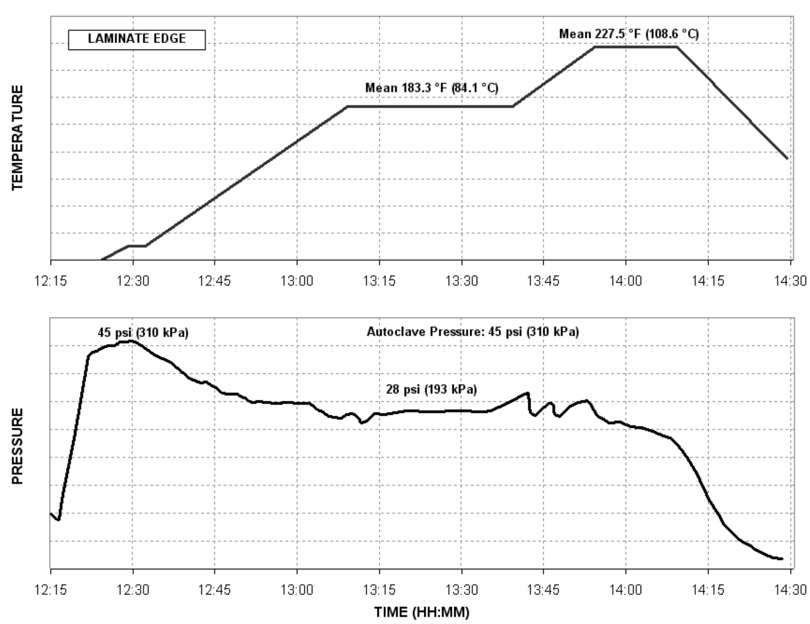


FIGURE 5.3.2(c) *Resin pressure readings for edge band (Reference 5.3.2(b)).*

5.3.3 Adhesive

The adhesive should be specifically evaluated for its suitability for bonding of the sandwich core material. This is especially important if open cell honeycomb of any type is being used as a core material; for fillets to form well, the adhesive must have a specific rheology. The adhesive first needs to be evaluated for its suitability for bonding of the sandwich core material at the coupon level. Flatwise tension and climbing drum peel are most commonly used for this evaluation. Like any material evaluation, they must be evaluated at all of the critical environmental conditions, such as cold/dry and hot/wet. By following the building block approach, problems with bonding should be eliminated before any large structure is actually made.

Non-metallic core material can be fairly hygroscopic, especially aramid materials. This moisture can affect the properties of the core directly. Prebond moisture can also affect the adhesive used to bond the face sheets to the core, and even the face sheet prepreg if the face sheets are co-cured. Especially since nonmetallic core may absorb more moisture by weight than most adhesive bond details, honeycomb bond details should be dried prior to lay-up as noted in Section 5.3.1.2. The ensuing lay-up and cure should be performed as expeditiously as possible to avoid moisture re-uptake.

The core moisture can affect the cure reaction of the adhesive and prepreg, but the moisture vaporized during cure can also induce porosity, voids, and other defects in the adhesive and co-cured face sheets. The knockdown for the core from moisture absorption is fairly straightforward, and can be handled in a manner similar to that used for other composite materials. Assessing prebond moisture effects for composite materials and adhesives can be much more difficult. While most manufacturers prefer to bond their core after drying, the effect on some adhesives and prepregs can be minimal. If the margins for a particular application are generous enough, the effect may not require drying of the core prior to bonding.

If film adhesive is used for bonding the face sheet to open celled core, little of the adhesive is actually in contact with the cell walls. If weight is critical, lighter weight film adhesive may be reticulated so that it is wholly applied to the tops of the cell walls, or adhesive can be carefully roller coated on the tops of the cell walls. In any case, the adhesive used should be checked for compatibility with open celled core bonding. The adhesive must remain at the face sheet-to-core interfaces forming healthy and strong fillets, which dramatically increases the available bond area for open cell cores, such as honeycomb, rather than just running down the cell walls. This is typically checked using a flatwise tensile strength test, where a bond which is stronger than the core itself is desired, resulting in failure within the core, away from the face sheet-to-core bond.

Larger dimensional tolerances are usually required for cost effective sandwich composite construction compared to solid substrate structural adhesive bond details. As a result, the adhesive may be tasked with performing well with larger than normal bondline thicknesses, filling gaps without a severe strength knockdown.

5.3.3.1 Impression check

A sandwich composite structure is formed by bonding sandwich core details, similar to any other bonded assembly. Bonding pre-cured laminates to core is quite similar to bonding metallic face sheets to a core where a prefit operation is usually performed. All bond faying surfaces should be pre-fit to assure that tolerances are close enough to allow adequate bonding to be performed. If the bond details to be joined are both rigid, then very close tolerances may be required, approaching half or even a third of the acceptable bondline thickness range. When softer or uncured details are to be joined, then more relaxed tolerances can frequently be allowed for adequate bonding.

A dry fit is typically performed initially as a preliminary check. If more than light hand pressure is required to bring corresponding faying surfaces into contact, additional effort should be expended to assure that the intended cure pressure will be adequate to bond the details satisfactorily. While simply checking

the core-to-face sheet interface for gaps used during a dry fit may be acceptable, the best way to perform a pre-fit is an impression check.

The impression check is a kind of dry run for bonding of the structure. The details should be dimensionally identical to those to be used for production. Since the details will not be bonded during the impression check, details which have not had their faying surfaces prepared (and for metal details, primed) can be used if the surface preparations do not appreciably affect the dimensional tolerances or other aspects that would affect the bonding process.

Instead of the film adhesive to be used in production, an impression check film product similar in dimensions can be used. The thermal, pressure and time response of the intended adhesive should be reproduced, but the impression check film does not bond the surfaces and hopefully does not contaminate the details. If such an impression check material is not available, or to better capture the response of the adhesive intended for use, frequently the intended film adhesive is sandwiched between two layers of a thin release film. The cure cycle intended for production is used, with the same tolerances on the heat up rates, dwell temperatures, and vacuum and pressure profiles, although the final dwell is frequently truncated after gel of the adhesive has been accomplished.

It should be recognized that since the impression check material is not allowed to wet out the core material or other faying surfaces, it provides an indication of adequate pressure, but that alone does not assure that adequate filleting of the film adhesive on the core will be accomplished in an actual cure. Filleting and other additional bond aspects must be validated through destructive test of actual assemblies and mechanical test of coupons. Co-cured potting materials and foaming adhesives are frequently omitted at the impression check stage, and would be validated at the later destructive test phase.

After the impression check part is cooled, it is disassembled, the impression check material is removed and the effective bondline thickness is measured. Due to differences between impression check and issues such as wet out and flow of the adhesive during an actual cure, the impression check effective bondline thicknesses that are measured may be shifted from the actual bondline thicknesses that are measured during later destructive testing.

In the core area, the film is inspected for uniformity of the impression of the cells indicating adequate and even pressure. Details should be inspected for any signs of damage such as core crush, distortion or movement. Core bond areas with faint or no core imprint indicate inadequate pressure application to the bond. For more rigid details this may imply inadequate control of dimensional tolerances for the bond details.

For areas between solid substrates, the impression check material is inspected for uniformity of thickness, areas where inadequate pressure may be indicated by the presence of excessive adhesive thickness, voids or porosity. The film can be held up to a bright light to aid in inspection. The range of film thicknesses for all areas of interest should be measured with a tolerance less than an order of magnitude smaller than the nominal acceptable bondline thickness.

Because of the insulating properties of most sandwich core materials, a proper heat survey which determines all actual bondline temperatures (and for the face sheets, as well, if co-cured) is even more critical than normal. Large thermal gradients in the part during cure may also contribute to increased internal stresses leading to warping or even the structure tearing itself apart during cool down or low temperature service.

Since lower pressures are typically required for sandwich structure cures than for solid laminates, the tooling may be proportionally less massive, tending to reduce heavy tooling induced thermal gradients. Within the flow (or rheology) limits of the adhesive and the prepreg resin, a slower heat up rate will help to limit the maximum temperature difference across the part during cure. Preliminary heat surveys can be performed as a part of the impression check, but are usually repeated as a part of later destructive testing.

This impression check is less practical with increasing levels of co-curing or co-bonding being performed, and so for these manufacturing approaches additional emphasis is placed on the destructive test aspects discussed elsewhere.

5.3.3.2 Bonding

Sandwich structure fabrication by its very nature involves adhesive bonding, with all the same strengths and constraints as conventional adhesive bonding. Sandwich core bonding deserves the same attention as any other structural adhesive bonding. Like any bond detail, sandwich core is subject to contamination. The same issues associated with structural adhesive bonding are present for sandwich structures including loading, surface preparation and contamination, and environmental effects. Bond failures with sandwich structures typically arise from poor application of materials, inappropriate or poorly controlled processes, contaminated or incompatible materials, and poor design practices. The full properties of the core material may not be realized unless the core first fails. Sandwich core bonding is critical to the structure since if failure of the core-to-face sheet bond does occur, the result can be a catastrophic "unzipping" of large areas of the face sheet from the core.

Pre-cured face sheets and other bond details need to be prepared for bonding. Surface preparation for lightly loaded pre-cured composite face sheets is sometimes limited to removal of peel ply. For anything more demanding, removal of the release film or peel ply would be followed by a light grit blast or at least a thorough scuff sand. Any surface gloss on a pre-cured laminate needs to be removed with 180 grit, or finer paper or alumina. Do not sand or grit blast heavily, just enough to remove any glaze. Too much abrasion will excessively expose fibers on the surface. The sanding residue must then be wiped off with acetone, isopropyl alcohol, or other approved solvent. Clean, white, lint-free, cotton gloves should be worn when handling bond details or adhesive and during all bonding operations or procedures described.

MIL-HDBK-349 (Reference 5.2.4) contains detailed information on the preparation of aluminum face sheets, which is more complex than pre-cured laminate face sheets. A water break test as described in ARP 4916 (Reference 5.3.1.1) can also be used to determine the cleanliness and readiness for bond of bond details such as pre-cured or metal face sheets.

The following requirements should be met for film adhesive application. Adhesive material should be as specified in the applicable material or process specification. Adhesive properties should be confirmed through coupon level tests with the core and face sheet materials that will be used. Adhesive application operations shall be performed in an area meeting clean room requirements.

After removal from refrigerated storage, sealed containers of adhesive should not be opened until the adhesive warms to room temperature and all moisture stops condensing on the container or package. Moisture on or in an adhesive can greatly weaken the core-to-face sheet bond. Condensation should never be allowed to form on the film adhesive. If moisture or other volatiles are present in the adhesive then it must have an escape path (the part must be well vented). This is covered in more detail in Volume 3, Section 5.7.8.

Film adhesive should be applied to core as follows:

a. If not at room temperature, remove the adhesive from refrigeration and warm to room temperature in the sealed container or package until all moisture stops condensing on the container or package. Reseal in the original container (or an equivalent) before returning to refrigerated storage.
b. Suspend each roll of film adhesive horizontally through the roll axis, free from contact with other rolls or objects.
c. On a clean table or table covered with clean Mylar or untreated kraft paper, cut the adhesive to the approximate shape of the surface to be bonded.
d. Record the batch and roll number of the adhesive and the cumulative time the adhesive is at room temperature until start of the curing operation.

e. Remove the protective wrapping from one side of the adhesive film, and smoothly apply the film to the core surface. Make certain that, if applicable, the correct side (usually film covered) is against the honeycomb. In some instances it may be advantageous to apply the adhesive to the face sheet first. This will aid in working out air bubbles after application.
f. Tack the film adhesive in place using hand pressure. When necessary, a hot-air gun or heat iron (usually limited to 150°F (66°C)) may be used to aid in tacking the adhesive.
g. Remove the remaining protective sheet from the adhesive film immediately before subsequent assembly operations and no sooner.
h. Trim the film adhesive flush with or larger than the perimeter of the bond joint surface. Butt joints are allowed, but do not overlap-splice the adhesive unless specifically authorized. As an example, MIL-HDBK-349 (Reference 5.2.4) instructs to overlap 1/16-inch (1.6 mm).

If the core does not fail first, then the adhesion failure mode should be considered. In contrast to the preferred cohesion failure mode, adhesion failure modes are poorly understood and growth can be very unpredictable, usually worsening with environmental exposure and/or repeated loading. As a result, adhesion failure modes are always suspect, even if the strength values are high. Special attention and care, perhaps also based on a history of usage, should be used when considering acceptance of an adhesion failure mode.

When the adhesive begins to flow during the cure cycle (and the resin in the prepreg face sheets, as well, if co-cured), if the same pressure is to be maintained, there must be some float in the tooling to account for the adhesive and prepreg bulk factors. The honeycomb material is fully cured and does not flow or mix. A path must be provided for trapped gas, especially as open celled core materials are filled with air during lay-up.

Because of substantial strength and stiffness differences between cored areas and surrounding solid areas, use of caul plates must be applied with flexibility and discretion. The caul thickness is typically limited to two or three times the thickness (or stiffness) of the face sheet materials. Thicker and stiffer cauls may be desired for cosmetic appearances, but require considerably more control of bond detail tolerances or else more substantial problems may result.

Like any other lower level bond assembly, cured core bond details should be debagged in a clean room environment to avoid contamination. All materials such as flashbreaker tape that could contact a faying surface, including the bond surface of the core materials, should be investigated to ensure it will not transfer residue that is not conducive to bonding.

Like many bonded structures, doublers may be used in transitions from high stress to surrounding areas. These doublers are typically external to the face sheet, but for tightly toleranced structures they can be built between the face sheet and core, although at greater expense. They are typically no thicker than the face sheet, and the same guidelines for staggering and edge geometry would be in place. Externally applied doublers may be secondarily bonded.

The edges of sandwich structures can be finished by filling with edge filling compounds, metal extrusions, or even specialized tapes. Attach points and hard points are fabricated by building in or secondarily bonding inserts. See Sections 2.7 and 2.8 for some examples.

Sandwich panel bond details or bonded subassemblies can be joined using variants of the same sort of bond configurations used for solid substrate adhesive bond joints. Different pieces can be joined using butt joints (for lightly loaded joints). There are also several variants of double strap type configurations, with either doublers on the exterior (either separate or in the form of an extruded detail joining both faces), or in the form of interior details similar to a woodworking biscuit joint. A third section of core can be used to fit into cutouts between the adjoining sections.

The lay-up of the sandwich structure bond details is handled in much the same manner as any structural bonding operation, using clean room facilities and practices as described under bonding operations.

The bonding of the face sheets to core, and perhaps curing of the face sheet laminate, as well, is typically performed in a press (if flat) or vacuum bagged in an autoclave or oven (if curved).

5.3.3.3 Filleting

Structural adhesives need to be specifically evaluated for their suitability for bonding of sandwich structures. For honeycomb in particular, the adhesive filleting down the cell walls provides much of the bond strength. If only the top surface of the cells is bonded, even if the bonding is performed well and cohesive failure is obtained, the bond will be weaker than a filleted bond because the area bonded is so low. An adhesive requirement for bonding face sheets to core with open cells is that the adhesive have proper flow so it stays at the face sheet-to-core bond and wicks onto the cell walls enough to form fillets, but not so much as to flow into the entire thickness of the cell. Many adhesives are formulated specifically for honeycomb bonding applications in which the rheology is well controlled.

Fillet formation is critical to the strength and durability of the core bond. An example of honeycomb core fillets can be seen in Figure 5.3.3.3(a). A schematic showing good adhesive fillet formation can be seen in Figure 5.3.3.3(b). A photograph of a well-formed fillet is shown in Figure 5.3.3.3(c). A diagram showing a number of fillet formation problems can be seen in Figure 5.3.3.3(d).

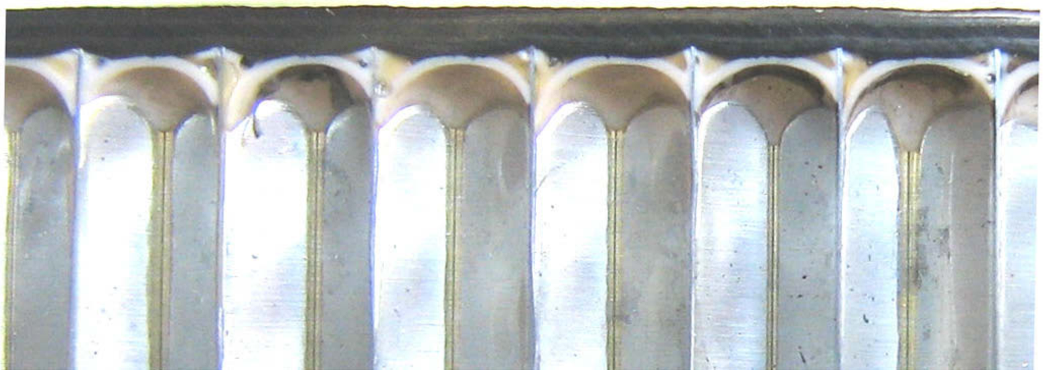


FIGURE 5.3.3.3(a) *Adhesive bond fillet on honeycomb core.*

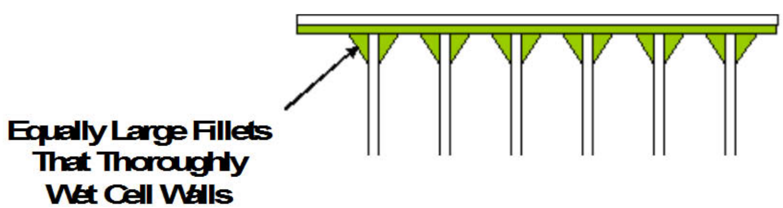


FIGURE 5.3.3.3(b) *Good honeycomb core-to-face sheet bond fillets.*

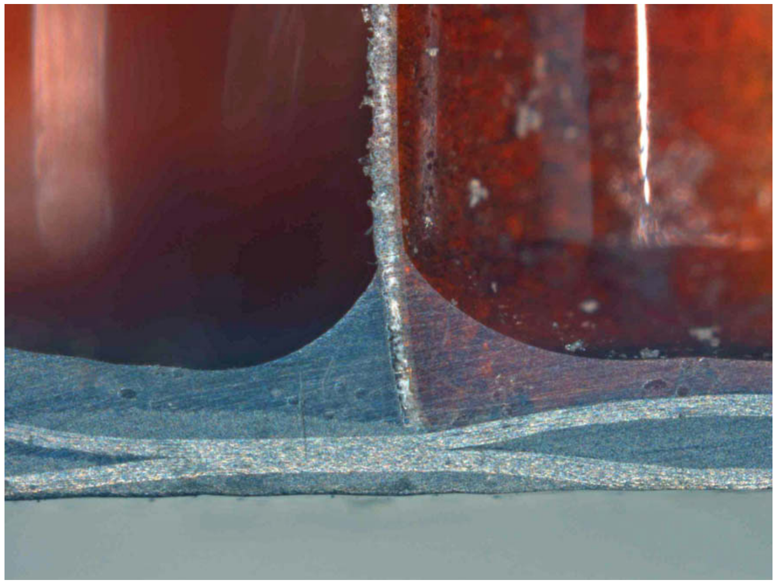


FIGURE 5.3.3.3(c) *A well formed fillet.*

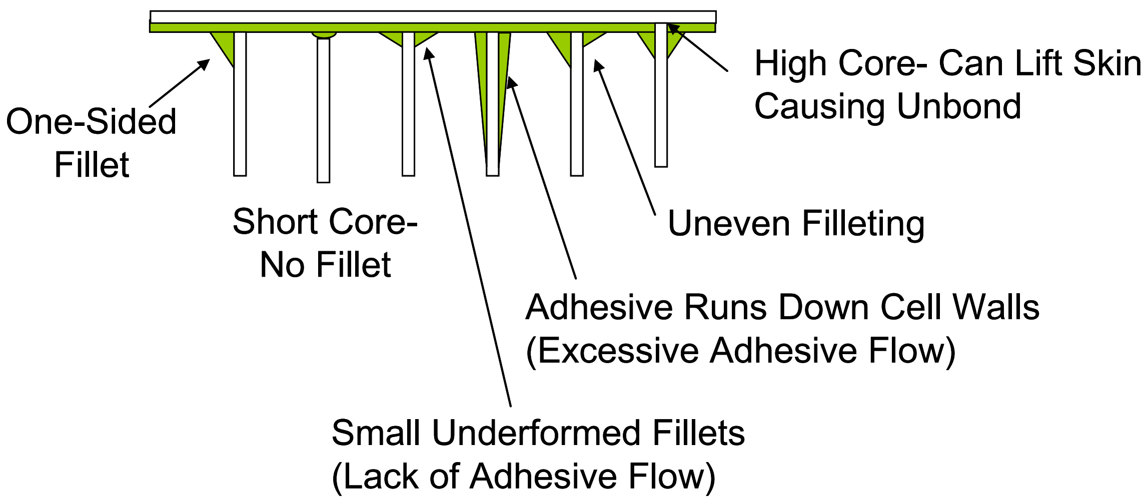


FIGURE 5.3.3.3(d) *Fillet formation problems.*

Figure 5.3.3.3(e) demonstrates what can happen if the surface tension of the adhesive is too high and the cell walls do not wet out. Figure 5.3.3.3(f) shows a poor fillet due to adhesive curing before volatiles were allowed to escape (i.e., too rapid of a heat up). Figure 5.3.3.3(g) demonstrates a total lack of filleting due to the film adhesive exceeding its out-time and losing its flow characteristics.

Reticulating the adhesive involves placing the adhesive on the core, then melting the adhesive using hot air. The goal is for the adhesive spanning the cells to be pulled to the cell walls so that all the adhesive is being used for the face sheet-to-core bond.

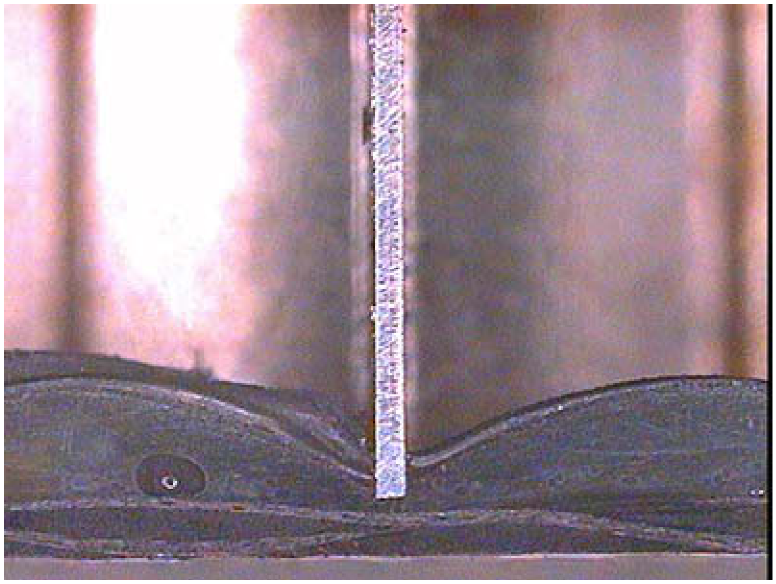


FIGURE 5.3.3.3(e) *Poor fillet due to moisture in adhesive causing high surface tension.*

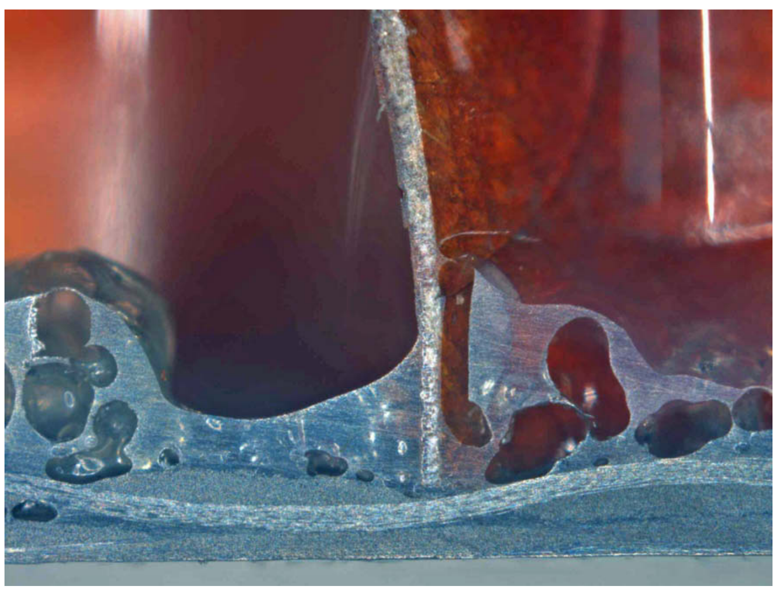


FIGURE 5.3.3.3(f) *Poor fillet due to adhesive curing before volatiles were allowed to escape.*

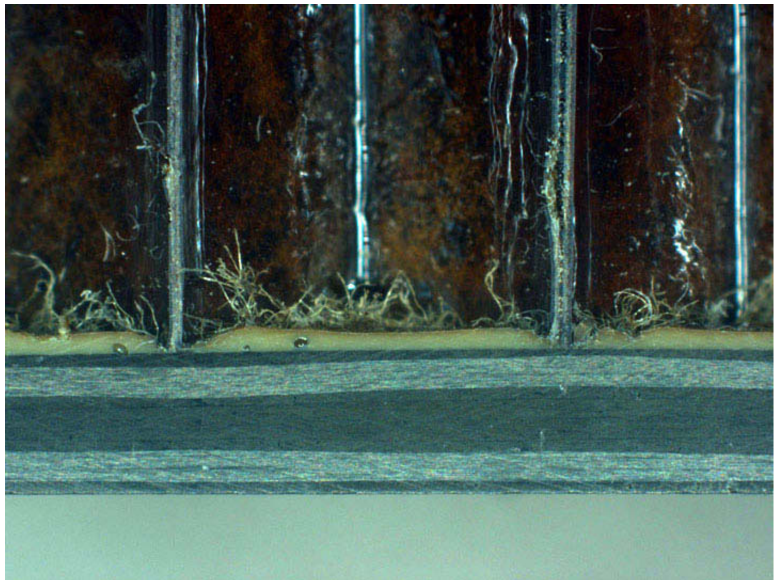


FIGURE 5.3.3.3(g) *No filleting due to excessive adhesive out-time.*

5.4 HONEYCOMB CORE CRUSH

5.4.1 Core crush during cure

Core crush is generally considered to be a measure of distortion and movement of honeycomb core details that happens during cure. Core crush is most often associated with beveled areas, where the autoclave pressure can cause distortion and movement in the plane of the core, as illustrated in Figure 5.4.1(a).

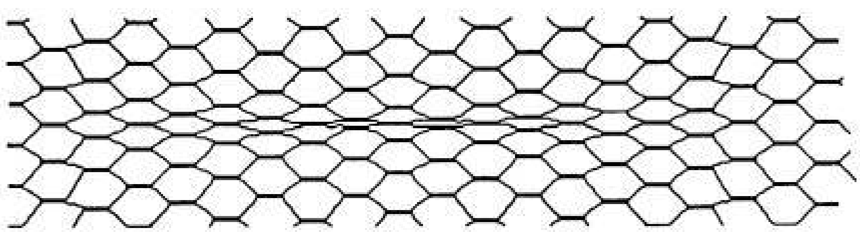


FIGURE 5.4.1(a) *An example of honeycomb core crush.*

A metric for quantifying core crush can be the maximum distance the core is deflected (usually at the midpoint between corners) from the original machined straight edge for the bevel. For all but the slightest crush, the condition is usually not acceptable and not repairable. A side view of honeycomb core distorted during cure through core crush can be seen in Figure 5.4.1(b). A top view of a honeycomb sandwich panel, which originally had a square core detail, can be seen in Figure 5.4.1(c). Because the core crush involves slippage of the surface plies, wrinkling is a secondary but very objectionable condition.

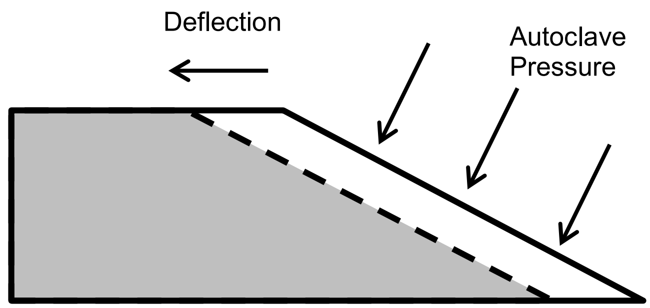


FIGURE 5.4.1(b) *An example of honeycomb core crush (side view).*

FIGURE 5.4.1(c) *Panel exhibiting substantial core crush.*

To prevent core crush, the horizontal component of the force being applied to the core (in the plane of the sandwich, perpendicular to the cell wall direction) must be resisted by some combination of effects. These include mechanical resistance (mechanical tie-down or friction grip strip), the resistance from friction sliding in the edge-band (which is increased linearly with edge-band width) and any resistance that the core detail itself can provide (usually minimal unless stabilized in some manner as discussed). In some cases, core crush may also be minimized by reducing the bevel angle.

The forces that initiate core crush are the horizontal forces that are applied to the thickness of the core across the bevel, and these forces must be resisted by the plies with either adequate friction or mechanical restraint to prevent slippage of the plies. Typically, the plies closest to the core exhibit the most displacement. Factors such as edge support, performed in the vacuum bagging process, can also resist the core crush forces. However, completely eliminating horizontal forces on the face of the bevel completely can result in other quality problems. Close-outs are details which can be used to protect the core from side forces. The core-to-close-out bond is typically handled in a manner similar to a core splice, while the remainder of the close-out bonding is handled like any other bond detail.

Many of these factors are subject to processing variability, which can make it even more frustrating when troubleshooting a core crush issue. Of course, many of these core crush factors also interact, mak-

ing the problem substantially more complicated. The coefficient of friction is a function of prepreg properties, which in turn may be a function of cure time and temperature profile, and the effective force is obviously a function of pressure.

It is important to characterize and, if possible, quantify exactly the type of core crush that is being experienced. The core almost always experiences some sort of deformation and/or displacement. The upper face prepreg plies are also experiencing some level of slip or movement, which may also result in wrinkling as a side effect. Some level of lower face prepreg movement may be seen, from none to displacement of the entire part on the tool.

5.4.2 Core crush - theoretical discussion

Conceptualizing what is happening to create core crush comes down to a couple of basic factors - forces and friction. Part of what makes core crush seem somewhat unpredictable is the friction factor. Friction is broken down into static and dynamic components, where the static friction is higher than the dynamic. Prior to reaching the static friction threshold, very little displacement is possible. Once the static threshold has been exceeded, the lower dynamic friction allows substantial and rapid displacements. This nonlinear response can complicate troubleshooting of existing manufacturing operations, or in developing new parts or manufacturing processes.

As an example of what can happen in a manufacturing environment, it is possible that a number of sometimes interacting factors contributing to core crush can be varying over time, but not to a degree which is recognized as significant at the time. As long as the static friction threshold is not exceeded, core crush is not perceived to be a significant factor. Once exceeded, the quality problems with core crush rapidly become substantial, but the manufacturing perception may be that nothing has changed.

There are a number of forces at work, some of which counter others. The most obvious forces result from the cure vessel pressure on the geometry of the part. While changes to autoclave applied pressure are clearly related to core crush, geometry changes such as changing the thickness or bevel angle can also change the resultant forces on the core and, as a result, change the core crush response.

The core stiffness can resist the cure vessel forces to some degree, usually well in the thickness direction, but honeycomb tends to be substantially less stiff perpendicular to the cell direction. Any geometric changes can also alter the core detail's ability to resist these forces. Changing to a higher density core can assist in resisting core crush.

The prepreg face sheet is a less obvious contributor in resisting the cure forces. The prepreg has a characteristic drape or hand as a measure of the prepreg stiffness. This is largely driven by the fabric architecture, finish and sizing of the reinforcement; the total resistance is proportional to the number of plies. The face sheet contribution to resistance to core crush is maximized if the face sheet is pre-cured.

A less obvious pressure that can react the cure vessel pressure is from any gas inside open core cells. While the open cells start with atmospheric pressure, autoclave cured parts will be put under a vacuum bag in the clean room, although the vacuum drawn on sandwich structures is normally limited to less than full vacuum. The primary variable that determines whether the core pressure changes from atmospheric to that of the vacuum system, or somewhere between, is the permeability of the face sheet. If the face sheet is metal or pre-cured composite, then the permeability is zero. Barring other leak paths, atmospheric pressure in the cells is maintained at room temperature. The higher the permeability and the longer the time exposure to vacuum, the closer the cell pressure will approach the vacuum level drawn under the bag.

After reaching a given autoclave cure pressure, the vacuum bag is typically vented to atmospheric pressure. Any air remaining in the cells during the cure cycle will expand with increased temperature and push back against the pressure being applied by the autoclave. This reduces the effective force pushing the face sheet onto the core (but helps develop resin pressure for a co-cured face sheet). As shown in

the resin pressure discussion, if the gas in the cells is allowed to leak out of the cells during the cure then the effective pressure will be changing during the cure.

Prepreg permeability is a function of a number of reinforcement and resin parameters. There are several reinforcement parameters that are influential in determining the prepreg permeability. The tightness or looseness of the weave is important, as is the weight of the tows and fabric, as well as the twist condition of tows or rovings. The twist condition can be never twisted, twisted, or untwisted (twisted then untwisted). The size and/or finish on the fiber and/or fabric can also alter the prepreg permeability.

The primary resin parameter that influences the prepreg permeability is resin viscosity, which is in turn a strong function of the resin chemistry and formulation. In addition, the resin viscosity also varies as a function of time and temperature during shelf-life to out-time and then cure cycle. The amount of resin in the prepreg, or resin content, also influences permeability. Less obvious is the distribution of resin through the reinforcement produced as a result of the prepreg manufacturing process. For various reasons some prepregs are manufactured where the reinforcement is not completely wetted with resin. Subsequent calendaring (or working) of the prepreg can help completely distribute the resin. Because of all of this complexity, it may be easiest to measure permeability for a representative face sheet going through a simulated cure cycle.

Friction is the other primary aspect for core crush, and is determined by many of the same factors that have already been discussed under forces and permeability. The resin viscosity is a critical parameter, and also a function of other parameters as discussed above. The prepreg resin functions as a kind of lubricant in facilitating plies sliding past each other; each resin formulation has a characteristic lubricity, which may also change as a function of viscosity. The resin content and resin distribution then also become measures of the amount of lubricant available for ply slippage. Finally, the reinforcement material, size, finish, weave, and twist all influence the characteristic coefficient of friction for the reinforcement. Because of this complexity, a seemingly small or perhaps unobserved change can dramatically influence the core crush response for a given part.

The common manufacturing core crush responses can be evaluated in terms of the above discussion. The use of mechanical ties and/or grip strips are an external attempt to dramatically increase the friction between the plies, preventing them from the slipping which then leads to core crush. The use of a ply of pre-cured film adhesive and/or prepreg on the surface of the core detail can have a couple of positive effects. It obviously increases the rigidity of the core detail so it may better resist the cure forces. Less obviously, it also becomes an additional permeability barrier between the core cells and the outside. Again, precuring the face sheet maximizes all the face sheet stiffness, friction and permeability responses to resist core crush.

5.4.3 Core crush stabilization for cure

Until a core detail is bonded to the face sheets, it can be very easy to deform with the forces present during cure. To firm up relatively lightweight core during co-cure with face sheets, the core frequently needs to be stabilized. The core weight needed for service may be lighter than what can be used in successful co-cure fabrication without these additional procedures. For severe core crush cases, more than one of the manufacturing actions may be required. Sometimes the most straightforward and effective way to address core crush is to pre-cure the face sheets.

The following is primarily applicable to honeycomb core bond details. While foam core can also be subject to core crush, thermoplastic foams are typically handled through a densification process which crushes the outside surfaces at an elevated temperature exceeding the face sheet-to-core bond cure temperature. This creates a higher density “skin” that resists further and non-uniform crush during cure. The core crush discussion is still applicable to foam core, as are the pre-cured film adhesive and/or prepreg ply stabilization methods shown.

Core crush can be minimized by the use of tie downs or mechanisms which prevent ply slippage during cure. An example is the use of friction or grip strips (or simply wider edge bands that may be trimmed after cure), which increase the effective friction which must be overcome in order for the plies to slip and allow the core to crush. Adhesive-backed strips of abrasive paper or tack (grip) strips are applied to the lay-up tool around the periphery of the part, and then laminate plies are placed, staggered, onto the strips. A minimum of two plies (next to and underlying, and next to and overlying the core detail) should be staggered onto the grip (also known as grit) strips when using this method.

Another common method for responding to core crush is to pre-cure a ply of film adhesive (and/or prepreg) on the surface (or both surfaces) of the core detail. This makes the core detail much more capable of resisting the core crush forces encountered during cure. Core material should be stored and dried as specified in Section 5.3.1.2 before any stabilizing operations. The engineering drawing or fabrication planning should define application areas for the adhesive or prepreg. Like all bonding operations, cure crush prevention procedures should be performed in a clean room area meeting the requirements. After applying the plies, the core detail assembly is typically bagged as shown in Figure 5.4.3 and as follows:

1. When required, place one layer of nonporous release film or equivalent on the project plate or tool. Extend the release film 2 to 6 inches (50 to 150 mm) beyond the core edge.
2. Place the honeycomb core assembly on the release film.
3. Place another layer of nonporous release film on the core assembly.
4. Place one or two layers of dry cloth breather material or equivalent over the top of the lay-up. Cover all sharp corners and other projections on the lay-up with additional dry cloth material to protect the vacuum bag from puncture during cure.
5. Place a nylon vacuum bag over the entire assembly, and seal to the tool with sealing tape.

The final dwell cure time for the film adhesive and/or prepreg can frequently be truncated if the assembly will undergo an additional full cure cycle during cure of a higher level assembly.

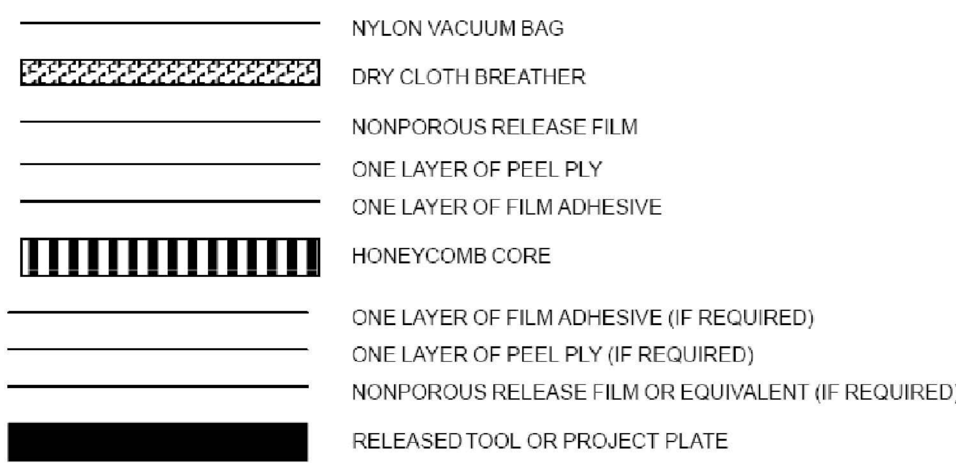


FIGURE 5.4.3 *Bagging the assembly for stabilization.*

5.4.4 Core material characteristics and core crush

The core is not particularly dimensionally stable, and can easily be deformed by pressure gradients. While the core has some substantial strength parallel to the core cell direction through the thickness, it is considerably more compliant in the length and width if not otherwise restrained. The ability of the core

(especially the lighter densities) to resist deformation can be substantially lower than the horizontal forces applied to the chamfer by the bag during cure, and the core may get weaker and softer at elevated temperature during the cure cycle.

In addition, some materials are more prone to core crush than others. Materials intended for co-curing face sheets over core should be screened, ideally using the most demanding part intended for use, but at least with an effective discriminator panel which is larger than lab scale. The panel should be at least 2 feet by 2 feet (0.6 m by 0.6 m), and all the criteria that represent the quality requirements that are present should be applied. All the production materials should be used including the core with at least the extremes of bevel angles (nominally 20 to 30 degrees), face sheet thickness, and edge band widths represented. The core used should be at least as thick as the maximum used, or 0.50 - 0.75 inch (13 - 19 mm), whichever is greater. The smoothness of the cut of the core may also reduce the coefficient of friction between the core and the face sheet, potentially affecting core crush.

5.4.5 Prepreg and adhesive material characteristics and core crush

The viscosity of the prepreg resin and the film adhesive can substantially influence core crush based on friction. Fillers or other rheology modifiers can be used to modify the thermal viscosity profile, adding stiffness to the prepreg or film adhesive. The coefficient of prepreg friction can be measured for given temperatures and viscosities (as a function of temperature history).

The prepreg resin viscosity profile over the cure cycles used is also a factor, as well as the lubricity of the resin for the fiber plies at the range of temperatures and viscosities. The resin chemistry and formulation aspects which can affect these properties can also be altered to influence core crush, and may explain some of the differences in core crush with different resin systems using the same fiber or fabric.

Because the viscosity profile affects both the ability to resist forces and the lubricity of the resin, aging of the prepreg in the freezer or at room temperature and the number of hot debulks can affect the degree of cure and thus the viscosity profile. Some systems are much more sensitive to this thermal history than others, thus the processing variability can be even more extensive. As a result, deliberately staging the prepreg may significantly affect the level of core crush observed. Increased thixotropy would be most valuable, providing increased viscosity at low shear rates but allowing some movement at higher rates.

Due to temperature dependent viscosity effects, the prepreg permeability may change substantially at elevated temperatures during cure. Unfortunately, these primary friction and permeability factors are not usually controlled or even measured during manufacture of the prepreg. They also may be cure cycle, part geometry and fabrication technique dependent.

The property referred to as the hand, drape, body or stiffness of the prepreg is also a factor in resisting slippage. The same resin and fiber can be substantially affected by the sizing applied during manufacture of the fiber and/or fabric, and can in turn also resist the resin flow, increasing the effective viscosity, and contributing to core crush resistance.

This means that the weave and weight of the fabric by itself can also have an influence. A relatively hard fiber sizing does increase the tow integrity, and also can increase the breathability of the prepreg, which is usually good for allowing a path for trapped gas to escape. Conversely, the trapped gas in the core cells will then be exposed directly to vacuum. The stiffer the tow or fabric from sizing or finishes, the stiffer the prepreg will be.

The impregnation process itself can also influence core crush by altering the physical distribution of the resin through the prepreg. Light impregnation, which leaves most of the resin on the surface, will provide more resin for lubrication, and can result in higher core crush. If more advanced impregnation puts more resin in the interior of the fiber pack, then less resin for lubrication is available and, everything else being equal, the friction between plies should increase. A resin impregnation index may be in use by the prepreg manufacturer as a quantifiable measure of this property.

The level of compaction is a related, but somewhat independent, factor that is also performed during the prepregging process. Compacting the material decreases the coefficient of friction over the noncompacted version, but also may decrease the permeability, with the consequences as noted elsewhere in this section. The more open the weave, the easier it is to get more resin to the interior of the reinforcement and off the surface. Given these relationships, and also given the other adverse aspects that may be caused, lower resin content also provides less resin to increase the core crush aspects. A more extreme measure is running at a lower tension during the impregnation, increasing slightly the void volume in the prepreg and reducing the resin on the surface.

The fiber tows and woven fabric can also be varied to maximize the ply-to-ply coefficient of friction in order to minimize crush. Fiber tow physical configurations can be never twisted (collimated), twisted, or untwisted (twisted and then untwisted). The twisted fiber usually maximizes the coefficient of friction. Increasing the entanglement of the fibers, and the roughness of the tow or fabric also increases the friction. The prepreg coefficient of friction should be evaluated at temperatures that can be seen during cure before gel. The farther the tow is from the collimated condition, the more flow resistance that will be provided by the fiber pack, increasing the effect of viscosity. The openness of the weave can also be significant.

The prepreg permeability along with the face sheet thickness and the pressure/vacuum and time profile will determine the pressure level in the cells. Temperature changes leading to expansion of residual gas is in the cells can lead to short-term slight increases in pressure that may bleed off over time. The permeability can be directly measured by measuring the rate of pressure change for the prepreg blocking the outlet of a fixed chamber, and may be performed for a number of ply stacks for the same material. Twisted material will have higher permeability, while the collimated material may have very little, especially if the prepreg has been worked (additional compaction) during material manufacture. Lower resin contents may have more difficulty in achieving low permeability.

5.4.6 Cure cycles and core crush

Cure cycles can definitely affect the degree of core crush experienced, and the time and placement of intermediate dwells and heat-up rates can be critical. More care must be taken during cure of honeycomb panels than with a laminate cure, especially in the pressure application. While a laminate cure cycle may include pressures to over 100 psi (200 inches Hg), sandwich cure cycles are typically below 50 psi (100 inches Hg). For lightweight sandwich assemblies commonly used in aerospace, the vacuum level in the vacuum bag is typically limited to 5 - 6 psi (10-12 inches Hg) max.

Sandwich core-to-face sheet cures put a substantial constraint on a critical cure parameter, the applied pressure. Crush of the core will limit the maximum pressure that can be applied. The cure pressure applied at the maximum cure temperature must not be high enough to damage the core material. For lighter density cores, this can be a substantial constraint which further reduces the pressure seen by co-cured face sheets or face sheet-to-core adhesives during cure, with accompanying effects on quality. Even with these reduced pressures, the cores such as honeycomb may be even more sensitive to side pressure. Side loads induced during cure must be reacted by tooling or fixturing to prevent collapse of the core.

Initial cure cycles are typically suggested by the material manufacturer, but these are frequently modified by the user because of facility constraints, manufacturing practices, or to address quality-specific issues. Intermediate dwells also have the effect of advancing the resin degree of cure to some extent which, at a given temperature, results in a higher viscosity, in turn providing greater resistance to core crush. The concept of a flow number is applicable because of the window in which the viscosity is low enough to prevent resistance of the forces (Ref. 5.4.6). As such, it is seen that, everything being equal, a shorter gel time is better for resisting core crush. The material properties can be dramatically affected by these kinds of changes, so it is important that the actual manufacturing cure process be characterized.

The gas pressure profile in the core during cure, which in turn is reflected in the resin pressure for the prepreg resin in the co-cured face sheet, can also be a significant factor. See the discussion of resin

pressure in Section 5.2.2, as the pressure of gas in the core affects the resin pressure. If there is actually some level of vacuum in the core, then these effects can worsen as it may force the face sheets into the cells even more and make dimpling more prominent.

Internal bag pressure techniques may be used to increase the cell pressure in order to increase the quality of the co-cured face sheet. While having some degree of gas pressure in the cells is of value so that the face sheets can have some small degree of resin pressure, if the pressure is too high then the gas can pick up some of the load, reducing the friction between the core and the face sheet. This allows the core detail to “float” in the lay-up during cure and can result in core crush and/or poor bonding between the core and face sheet.

5.5 QUALITY ISSUES INCLUDING NONDESTRUCTIVE EVALUATION (NDI)

It is important that composite laminate visual and nondestructive inspection criteria are not arbitrarily applied to sandwich structure. Sandwich structure surface quality may differ substantially from some high quality laminates. Acceptable defect and damage sizes can frequently be larger for sandwich structure that is designed primarily for stiffness, compared to solid laminates that are designed primarily for strength. Standards used for NDI should definitely represent sandwich structure, including the critical defect sizes. The sandwich structure itself may limit the NDI options due to increased attenuation by the core.

Honeycomb core details should be visually examined after machining, heat–forming, or core splicing operations for discrepancies. Some discrepancies may fall within acceptance limits established for the part. Discrepancies that exceed acceptance limits may fall within correctable limits. If so, they would be reworked as specified. Discrepancies exceeding the established acceptance and correctable limits should be rejected and submitted to the Material Review Board (MRB) for disposition. Examples of potential honeycomb core detail discrepancies and acceptance limits are contained in Table 5.5, along with possible correctable limits and procedures. All of these defects, limits and procedures would potentially be different depending on the application.

Although it depends on the specific design, voids or separation in the core-to-face sheet bond that encompass an area less than that for a few cells, are frequently considered to be acceptable. Even acceptable voids may require a minimum separation to the nearest other void of a few inches, and separation from other aspects such as edges geometry changes, fasteners or other features.

A complete plan needs to be developed that will cover the visual and dimensional inspection of a given part, and the application of nondestructive inspection that will be used. This would include a full characterization of the NDI process, usually including baselining with the NDI standard built into a development part, and the nature, categorization, sizing and acceptance limits for all of the defects and anomalies that may be encountered in production. If control of ribbon direction or other directionality of the core is required, then a tolerance must be established for that as well.

A complete assessment of what is allowable per the design and trade-offs for ranges of defects between performance and cost needs to be made. If enough scrutiny is applied, some anomaly can usually be identified in even the highest quality structure. Even if these anomalies are acceptable, if their presence in the original effort to develop and qualify the assembly is not documented and accepted, then their “discovery” can be disruptive if revealed subsequently in production.

An initial bonded assembly is typically fabricated using all of the materials, processes, documentation, and personnel intended for production. This bonded assembly, commonly referred to as a destructive test article, is destroyed as a part of evaluation. If possible, test specimens or elements are machined out of the part for mechanical test. For sandwich structure, these would typically be flatwise tension or climbing drum peel tests. For solid substrate bonds, typically a floating roller or T-peel and a lap shear test would be performed, such as for the bonded edge band around the periphery of the core. Tests for co-cured

face sheets would be similar to those used in Volumes 1 and 3 for composite laminate characterization, typically including glass transition temperature, ply thickness, and short beam shear.

TABLE 5.5 *Types of possible honeycomb core discrepancies and possible acceptance limits and correction procedures.*

(Acceptance limits and correctable limits are examples only and are not recommendations; any such limits must be validated for the specific application and material system combinations in the sandwich structure.)

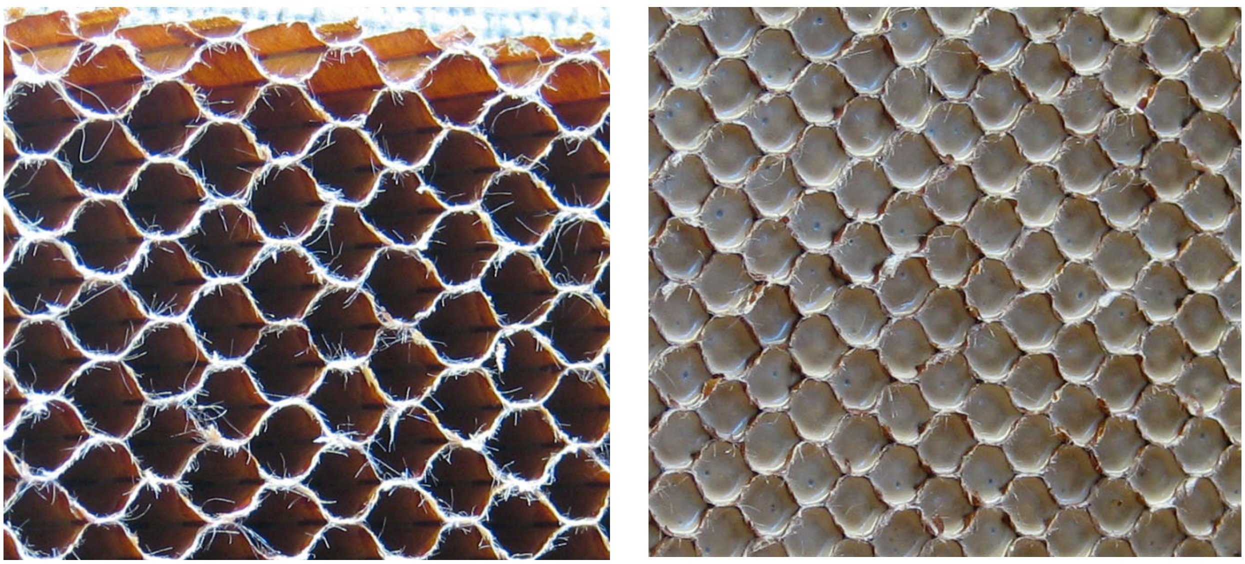
Possible Discrepancy	Acceptance Limit Example	Potential Correctable Limits	Possible Correction Procedures
Edge Cell Tear-Out	- Maximum Area: 0.25 in^2 - Maximum Length:0.5 in - Maximum Frequency: Once in any 12 in	Unlimited as long as weight gain less than 10%	Bond in new core sections
Frayed or Burred Edges	Machined fiber fuss less than 0.10 in	Unlimited as long as core dimensions maintained	Sand with 180 or finer sandpaper
Core Splice: Shrinkage of Foaming Adh.	Not to exceed 10% of splice depth	Unlimited	Submit to MRB
Core Splice: Thickness Mismatch	Not to exceed 0.006 in	Unlimited as long as core dimensions maintained	Lightly sand mismatch areas
Core Splice: Unbonded	Must be a continuously bonded/spliced joint	Limited to 20% of core splice	Repair bond or bond in new core sections
Contour	Must maintain position in layup tool when tacked in place	May be reformed only once	Reform core as specified
Prepotted Core	Shrinkage less than 0.03 in from core surface	Unlimited	Fill with additional potting compound
Slashed core	None	None	Replace core
Surface Depressions	Max dimensions: 0.02 in deep by 0.5 in long, at least 6 in from another	Unlimited	Fill depression with prepreg
Surface irregularities, protrusions, or flat areas	Must not affect final assembly contour	Unlimited when core dimensions maintained	Sand with 180 or finer sandpaper
Node separation: complete	No more than two in any 12 in diameter. Not adjacent in ribbon direction	Unlimited except max of 25% or core may be corrected	Splice or bond in new core
Node separation: Partial	No more than 12 in any 12 in diameter. Not adjacent in ribbon direction	Unlimited except max of 25% or core may be corrected	Splice or bond in new core

The climbing drum peel test is commonly used as a quality control test to gauge the effectiveness of materials and processes used, providing feedback on the suitability and amount of the adhesive, core and face sheet materials, construction, tooling and processing such as surface preparation and cure. Climbing drum peel is useful as a quality control test, but due to the influence of geometry and other factors, less effective for making comparisons between materials. The flatwise tension test is typically used for

comparison between materials, and can also be used for quality control for configurations that cannot be tested using the climbing drum peel.

Given that the samples machined from parts may not be exactly as would be used for coupon tests, the failure mode is at least as important as the load value obtained. If failure of the core (not prematurely induced by test specimen configuration) can be obtained, this is a very positive reflection on the sandwich construction.

An example of desirable core failure on a climbing drum peel test specimen is shown in Figure 5.5(a). Note the core that is still bonded to the peeled face sheet. An example of adhesion failure on a climbing drum peel test is shown in Figure 5.5(b). Note that no core is evident on the peeled face sheet and the core still has much adhesive on it. Adhesion failure is not desirable because it indicates the adhesive is not well-bonded to the core (or face sheet), and may indicate contamination or problems in surface preparation.

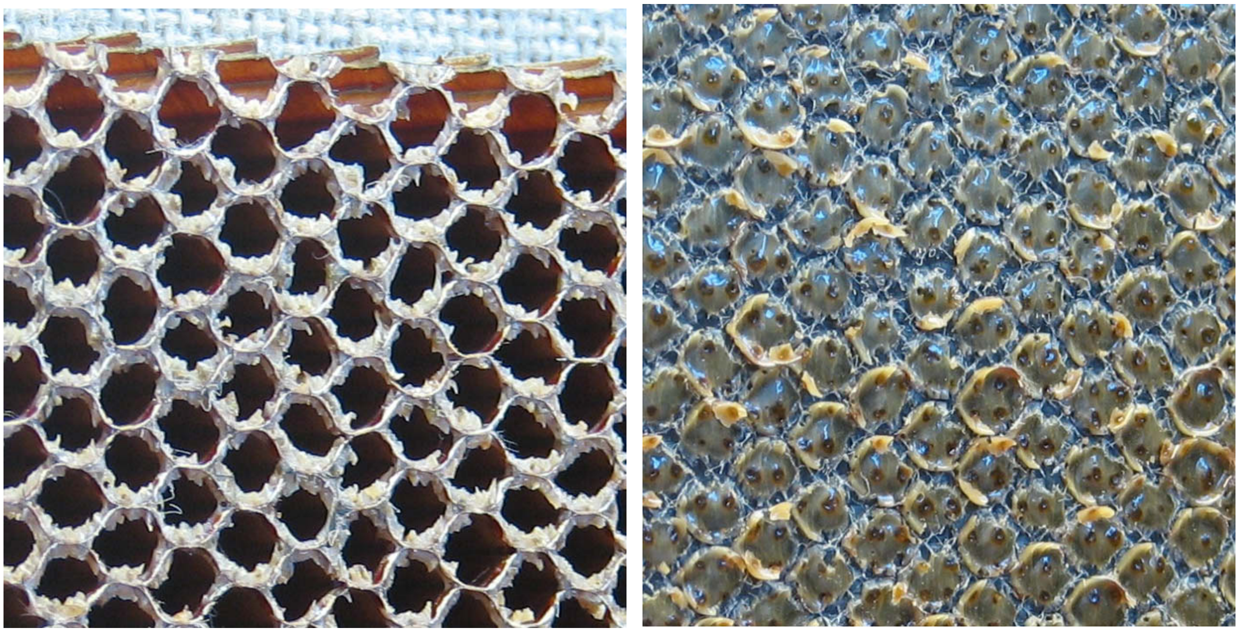


Core side Face sheet side

FIGURE 5.5(a) *Example of core failure on a climbing drum peel specimen.*

Part of this initial destructive test article is commonly fabricated with specific inserts or damage that can be cut from the part and used as a standard for nondestructive inspection. Fabrication of more generic sandwich composite NDI standards is discussed by ARP 5606 (Reference 5.1) to be used for damage assessment and repair evaluation, but is largely applicable to inspection of original manufacture of structures. The standards are potentially useable for ultrasonic, resonance, and tap test NDI procedures looking for disbonds and delaminations. Some assessment needs to be made on how representative these or other available NDI standards are to inspection for a specific structure's materials and geometries.

After removal of any test specimens and/or nondestructive inspection standards, the remainder of the structure typically has the face sheets peeled from the core. The core and face sheet are then visually evaluated for indications of "good" bonding in the form of a cohesion failure mode, any unintended damage to the core in the form of crush or distortion or movement, or inadequate pressure.



Core Side　　　　Face sheet side

FIGURE 5.5(b) *Example of adhesion failure on a climbing drum peel specimen.*

Defects that would not be acceptable in a final assembly are usually acceptable in a subassembly or other bond detail, if that defect will be removed as part of a later manufacturing step. An example of this would be a void along the outside of the edge band in a sandwich structure that is built to have the edges later trimmed, which would completely remove the defect as a part of the trimming.

For more information regarding processing of sandwich composites, see Reference 5.5.

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5.3.1.1 SAE Aerospace Recommended Practice, ARP 4916, “Masking and Cleaning of Epoxy and Polyester Matrix Thermosetting Composite Materials,” March 1997.

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5.3.1.5 SAE Aerospace Recommended Practice, ARP 4991, Core Restoration of Thermosetting Composite Components,” February 2007.

5.3.2a Frank-Susich, D., Ruffner, D., and Laananen, D., “Computer Aided Cure Optimization,” Proceedings of the 37th International SAMPE Symposium (A93-15726 04-23), March 1992, pp. 1075-1088.

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5.4.6 Hoisington, M. A., Seferis, J. C, and Thompson, D., “Scale-up for Hot Melt Prepreg Manufacturing,” Proceedings of the International SAMPE Symposium and Exhibition, 37th, Anaheim, CA, Mar. 9-12, 1992, (A93-15726 04-23), p. 264-277.

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CHAPTER 6 QUALITY CONTROL

6.1 INTRODUCTION

Quality control is required to ensure that the sandwich structure will meet the requirements to achieve its design purpose. It typically includes inspection of each batch of the core, face sheets, and adhesives to ensure that they meet certain minimum requirements. It also encompasses monitoring of the process, including verification that the correct materials are used during assembly, and that the appropriate cure cycle is followed. Finally, the finished sandwich structure is inspected for dimensional tolerance and non-conformities.

Many of the issues and procedures associated with quality control in solid composite laminates also apply to sandwich structures; see Volume 3, Chapter 6. In addition, many nondestructive inspection methods suitable for sandwich structures are discussed in Volume 3, Section 6.3.2. This chapter will focus on items unique to sandwich structures.

6.2 MATERIAL PROCUREMENT QUALITY ASSURANCE PROCEDURES

6.2.1 Specifications and documentation

The specification for materials, fabrication processes, and material testing techniques must ensure compliance with the engineering requirements.

Chapter 2 in this Volume describes acceptance test methods for characterizing core materials, face sheets, and the bond between them. Section 5.11 of Volume 3 provides a brief discussion of requirements for specifications and documentation. Sections 5.11.2.4 of Volume 3 provides information on variable statistical sampling plans that control the frequency and extent of material property verification testing to achieve targeted quality levels.

6.2.2 Receiving inspection

The composite user's material specifications typically define inspection requirements on incoming materials (i.e., core, adhesives and face sheet materials for sandwich) to ensure that they meet engineering requirements. A general discussion of this topic is found in Volume 3, Section 5.11.2.4. Additional details of sampling plans that control the frequency and extent of material property verification testing to achieve targeted quality levels can be found in ANSI/ASQC-Z1.4-1993 and ANSI/ASQC-Z1.9-1993 (References 6.2.2 (a) and (b)).

Honeycomb core is typically characterized by the material from which it is made (e.g., aluminum, Nomex™, Korex™), the density, the cell size, and the cell configuration (e.g., hexagonal, overexpanded, flex). Other core materials, such as foams and wood materials, are typically characterized primarily by the constituents and density of the final core configuration.

Acceptance test requirements may vary from user to user. A typical example of acceptance test requirements for honeycomb core is shown in Table 6.2.2. Cores with heavier densities (in the 12-16 pcf range) may have different requirements or test methods.

Visual inspection checks the core for nonconformities such as distorted cells, broken or buckled cell walls, node bond separations, partially bonded nodes, and fuzzy or chipped cell edges. An example of a piece of honeycomb core with distorted cells is shown in Figure 6.2.2. Other tests which may also be required include water migration between cells, electrical transmission testing, and a flexibility test.

Perishable materials such as adhesives and prepregs typically require revalidation testing to allow additional use of the material after expiration of the normal storage or out time life (see Volume 3, Section 6.2.3.2). Core is generally stable at room temperature and, therefore, does not require revalidation testing. It may, however, require special handling and storage to prevent moisture uptake.

TABLE 6.2.2 *Typical acceptance tests required for honeycomb core.*

	TESTING REQUIRED		
PROPERTY	PRODUCTION ACCEPTANCE (Manufacturer) (1)	PRODUCTION ACCEPTANCE (User) (1)	ASTM STANDARD (References 6.2.2(c) through (f))
Visual and Dimensional	X		
Dimensional Stability (2)	X		D 6772
Cell Size	X	X	
Density	X	X	C 271
Bare Compressive Strength (3)	X	X	C 365
Stabilized Compressive Strength and Modulus (3)	X	X	C 365
Shear strength and modulus (ribbon and transverse directions)	X	X	C 273

(1) Manufacturer is defined as the manufacturer of the core material. User is defined as the sandwich part fabricator. Production acceptance tests are defined as tests performed by the supplier or user for initial acceptance.
(2) Tests for dimension changes after specimen is heated and then cooled to room temperature.
(3) Tests may be required under standard laboratory conditions, at elevated temperature, and/or following immersion in water or other fluids.

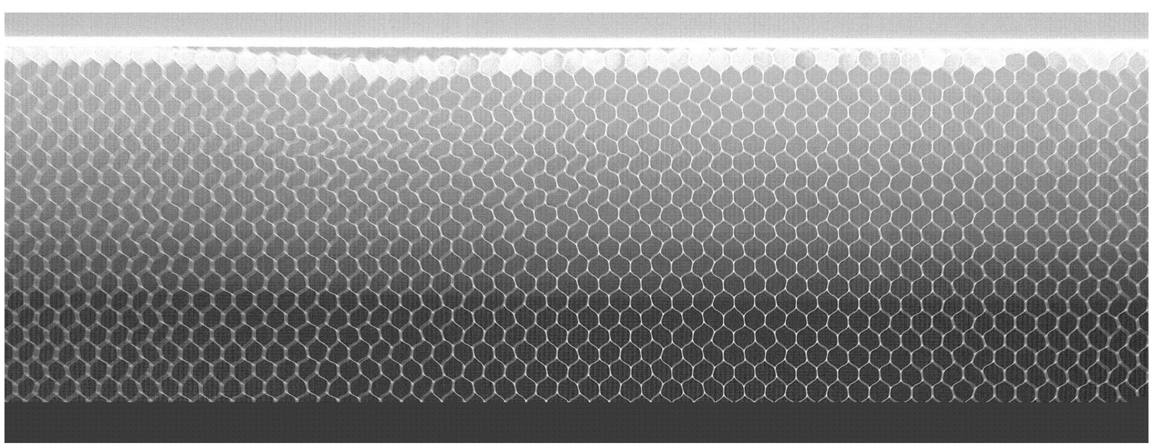

FIGURE 6.2.2 *Distorted cells in honeycomb core.*

6.3 PART FABRICATION VERIFICATION

6.3.1 Process verification

Volume 3 Chapters 5 and 6 discuss material control, material storage and handling, tooling, facilities and equipment, in-process control, part cure, and process control specimens.

Nonmetallic honeycomb core may require special storage conditions, since moisture absorption tends to reduce its strength. The core may be air-dried before machining, to help improve its stability. Once machined it may be dried at an elevated temperature and sealed in an airtight container until it is used.

The cure cycle for sandwich panels is frequently different from that for solid laminates made from the same composite material. A lower-pressure cure is typically used for sandwich structures with nonmetallic honeycomb core than for solid laminates, since higher pressures can cause the core to shift during cure, especially in areas of ramps, and excessive pressure might even cause crushing damage to the core. In addition, many manufacturers use a modified temperature profile for sandwich structures with composite face sheets, which includes a constant temperature hold during the ramp-up to the final cure temperature.

Many manufacturers use process control specimens to verify that the parts meet engineering requirements. These specimens may be taken from excess areas removed from the part itself, or a test panel laid-up and cured on the tool with the part. The mechanical and physical tests performed for solid laminates, such as strength, modulus, fiber volume, and glass transition temperature, may be performed on the face sheets. In addition, the process control specimens may be subjected to tests specific to sandwich parts, such as flatwise tension and climbing drum peel to check the core-to-face sheet bond.

6.3.2 Nondestructive inspection

Volume 3, Section 6.3.2 discusses common nondestructive inspection (NDI) techniques including visual, ultrasonic, and X-ray inspection. In addition, many NDI methods suitable for honeycomb core structures are discussed in Volume 3, Section 12.4.

Ultrasonic inspection is most commonly used for nondestructive inspection (NDI) of composite/honeycomb sandwich structures. A through-transmission C-scan can be used to inspect both face sheets, to detect crushing or other damage in the core, and to check the bond lines between the face sheets and core. A pulse echo A-scan can be used to detect porosity or other nonconformities in each face sheet separately. Inspection in the region of core splices is especially difficult. Thermography-based NDI techniques have also been demonstrated to effectively identify sandwich structure defects such as large delaminations and moisture entrapment.

NDI generally requires calibration standards specifically fabricated for sandwich structures. These standards may incorporate defects such as core crushing and core-to-face sheet disbonding. Reference 6.3.2 discusses construction variables that should be taken into account when developing the standards.

6.3.3 Destructive tests

Volume 3, Section 6.3.3 discusses types of tests, and some guidance on types of parts to be tested and frequency of testing. First article destructive testing of honeycomb core parts can be particularly useful to verify that the process produces good core to face sheet fillets and acceptable quality in the core ramp regions. Both of these features are difficult to inspect using NDI methods.

6.4 STATISTICAL PROCESS CONTROL

Volume 3, Section 6.5 discusses quality tools, control charts, process capability, process feedback adjustment, and design of experiments. That discussion applies equally to sandwiches and solid laminates.

6.5 MANAGING CHANGE IN MATERIALS AND PROCESSES

Volume 3, Section 6.4 discusses qualification of new materials or processes, divergence and risk, and production readiness.

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6.2.2(d) ASTM Standard C273, "Standard Test Method for Shear Properties of Sandwich Core Materials," American Society for Testing and Materials, West Conshohocken, PA., 2011.

6.2.2(e) ASTM Standard C365, "Standard Test Method for Flatwise Compressive Properties of Sandwich Cores," American Society for Testing and Materials, West Conshohocken, PA., 2011.

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CHAPTER 7 SUPPORTABILITY

7.1 INTRODUCTION

Supportability is an integral part of the design process that ensures support requirements are incorporated in the design. Support requirements include the skills, tools, equipment, facilities, spares, techniques, documentation, data, materials, and analysis required to ensure that a composite component maintains structural integrity over its intended lifetime.

Life cycle cost, being comprised of research and development, acquisition, operational and support, and disposal costs, is often a crucial customer requirement for any application such as a new weapon system or commercial transport.

Composite designs are usually tailored to maximize strength- and/or stiffness-to-weight performance. High performance designs are often less supportable due to increased strain levels and non-redundant load paths. Structural elements and materials should be selected for resistance to inherent and induced damage, especially delaminations and impact damage. These types of environments are especially crucial for sandwich structures as the nature of their construction (relatively thin face sheet materials) can potentially exhibit poor impact resistance characteristics. Other design considerations having an impact on supportability include durability, reliability, damage tolerance, and survivability. The information in this section is a summarization of the more comprehensive Supportability Chapter (Chapter 14) in Volume 3, which can be referred to for additional details. Furthermore, damage tolerance and durability information is briefly described in Section 7.2.4, but provided in more detail in Volume 3, Chapter 12.

7.2 DESIGN FOR SUPPORTABILITY

7.2.1 In-service experience

Composite materials were introduced into the commercial aircraft industry during the early 1960's. Advanced fibers such as boron, aramid, and carbon offered the possibility of increased strength, reduced weight, improved corrosion resistance, and greater fatigue resistance than aluminum.

The original composite parts, particularly thin-gage sandwich panels and secondary structures, experienced durability problems such as low resistance to impact, liquid ingression, and erosion. The face sheets of honeycomb sandwich parts were often only three plies or less, which was adequate for stiffness and strength, but did not consider the hostile service environment. Damage such as core crushing, impact damage, and disbonding, is often easily detected with a visual inspection due to the thin face sheets, but can also be non-detectible without the aid of a relatively sophisticated nondestructive inspection techniques. Occasionally damage is not detected initially and is, therefore, not repaired, resulting in growth of the damaged area due to cyclic loading and other physical phenomena such as liquid ingression into the core. Metallic core materials can also experience corrosion effects due to moisture ingression and galvanic effects associated with incompatible materials and/or inadequate corrosion protection.

Repair processing can also further damage a sandwich structure if not conducted properly. Most repair materials cure at temperatures above the boiling point of water, which can cause concern for a disbond at the face sheet-to-core interface during processing wherever trapped water resides. Therefore, core drying cycles are typically included prior to performing a repair. In some cases, in-service fluids such as Skydrol can cause contamination to the area intended for repair. Complete removal of Skydrol from core materials is almost impossible, where the core continues to weep the liquid so that full bonding does not occur.

Service experience data has been documented in reports from operators on parts involved in the NASA-sponsored Advanced Composites Energy Efficiency (ACEE) program, which supported the design

and fabrication of sandwich composite parts such as the B727-200 elevators and the B737 spoilers and horizontal stabilizers. Five shipsets of B727 elevators have accumulated more than 331,000 hrs. and 189,000 cycles; 108 B737 spoilers have accumulated more than 2,888,000 hrs. and 3,781,000 cycles. The service exposure data collected for these parts does not indicate any durability or corrosion problems. Thirteen airbrakes are still on aircraft, and seven have been withdrawn from service for testing to assess stiffness and residual strength.

Production carbon-epoxy sandwich parts have demonstrated weight reduction, delamination resistance, fatigue improvement, and corrosion prevention. The poor service records of some parts can be attributed to fragility, the inclusion of non-durable design details, poor processing quality, porous face sheets (insufficient thickness), and badly installed or poorly sealed fasteners.

7.2.2 Inspectability

Typical composite nondestructive inspection (NDI) methods available are: visual, through-transmission ultrasonics (TTU), pulse-echo ultrasonics, x-ray, enhanced optical schemes, and thermography. Most airlines and military operators use visual inspections supplemented with both tap test and pulse echo and low-frequency bond testing to locate damage. Because of the predominance of visual inspections, provisions should be made during the design phase for complete external and internal access for visual inspection of all components.

With a sandwich configuration there are inspection difficulties associated with potted areas, detection of fluids that have leeched into the sandwich honeycomb core, disbonds of face sheets, foam core, and damage within the core, as well as bond lines of stiffeners or frames that are bonded to internal face sheets of sandwich components.

7.2.3 Material selection

It is important to select the appropriate constituent core, face sheet and adhesive materials for a sandwich structure, considering how it is to be processed, its service environment, and its compatibility with surrounding materials. Ease of repair and the associated repair processes should also be taken into consideration when selecting the material systems.

7.2.4 Damage resistance

Components will be subject to mechanical damage from maintenance personnel, tools, runway debris, service equipment, hail, lightning, etc. Note that other types of damage may also be experienced by sandwich composite structures, but are not addressed here. Most composite components are designed to specific damage resistance, damage tolerance, and durability criteria. A supportable sandwich structure must be able to sustain a reasonable level of damage without costly rework or downtime.

Damage resistance is a measure of the relationship between the force or energy and impact footprint/shape of an associated damage event and the resulting damage characteristics. A structure with high damage resistance will incur less damage from a given event. Damage resistance levels should be such that impacts that do not create visible damage will not degrade the strength of the structure below design requirements.

Damage resistance of sandwich structures may be improved by increasing face sheet thicknesses and/or by using denser core. However, thicker face sheets that decrease damage visibility may actually lead to an increased risk of not detecting internal damage. For example, thicker face sheets are more likely to “spring back” to their original contour or shape after an impact event that fractures or locally crushes the core. This may also result in a disbond between the face sheet and core. Reinforcement fibers having high strain capability can also have a positive effect on impact resistance, and the selection of toughened matrix material can greatly enhance damage resistance. As discussed above, water ingress into a sandwich panel after impact damage is another supportability concern associated with sandwich construction.

7.2.5 Environmental compliance

Many aspects of the design, repair and maintenance of sandwich structures are impacted by environmental rules and regulations. Environmental compliance is primarily concerned with the correct disposal of hazardous wastes. A significant portion of a waste stream is made up of materials that cannot be used within their useful life. Typical heavy metal culprits include cadmium plated fasteners and chromated sealants and primers.

Consideration must be given to removal of coatings. Many chemical paint removers are not acceptable for most polymer matrix composites because the active ingredients will attack the matrix. Abrasive paint removal techniques, such as plastic media blasting, have proven successful on polymer matrix composites. Cleaning is one of the primary maintenance processes that create hazardous waste. Many of the cleaning processes that previously utilized ozone depleting solvents and other hazardous chemicals are being replaced with aqueous cleaning processes. If a component is constructed such that water intrusion is a concern, as in the case of most sandwich structures, then aqueous cleaning of the part may also be a problem.

Machining of carbon fiber laminates during cutting and trimming operations produces particulates that are considered a nuisance dust by bio-environmental engineers. TLV (Threshold Limit Values) limits were updated in 1997 by the American Conference of Government and Industrial Hygienists (ACGIH) to define loose composite fiber/dust exposure limits for composite workers. Excessive exposure may require the use of NIOSH-certified respirators with HEPA filters. Resins and adhesives used in sandwich structures may cause dermal sensitization in some workers, so that silicon-free/lint-free gloves should be mandated for use. This will also help to ensure that a contaminant-free bonded sandwich assembly is achieved.

Uncured prepregs, adhesives and resins are treated as hazardous materials. Scrap materials should be cured prior to disposal in order to reduce the HazMat disposal costs. It is important to ensure that scrap materials containing carbon fibers are sent to non-burning landfills; pyrolized carbon fibers freed by resin burn-off can represent a respiratory and electrical hazard.

7.2.6 Reliability and maintainability

Maintainability of a structure is achieved by developing appropriate customer-focused methods of inspection and maintenance during the design phase. Accessibility is an important factor when designing structures for repair. Sufficient access should be provided to properly inspect, prepare the damage area, fit and install repair parts and use repair tools and bonding equipment.

Designing for repairability is an essential element in the effective use of composite materials in aircraft and other structures. Repair considerations should influence the choice of lay-up patterns and design strain levels. The repair philosophy should be set during the conceptual design stage and the repair designs should be developed along with the component design. Candidate repair designs should be tested as part of the development test program. Repair concepts and materials should be standardized to the maximum extent possible.

7.2.7 Repairability

A component's "repairability" needs to be considered as part of the design process, starting with a reasonable/sufficient margin of safety, and should include provisions for reasonable repair size limits. Figure 7.2.7 shows the maintenance development philosophy established during the Boeing/NASA Advanced Technology Composite Aircraft Structures (ATCAS) composite fuselage program. Inspection and repair procedures must be considered during design selection. Any lightning strike protection systems that are needed on specific components should also be designed to be repairable.

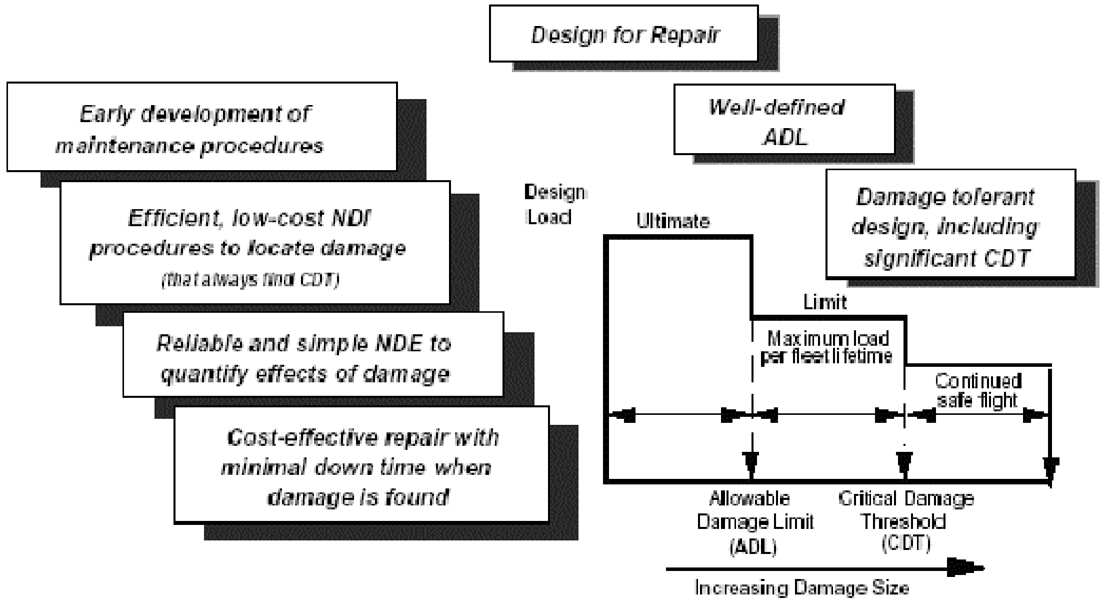

FIGURE 7.2.7 *Rules for maintainable sandwich structures (Reference 7.2.7).*

Design concept developments should include parallel efforts to establish maintenance procedures. Maintenance procedures established *after* design features are set will typically result in unnecessarily complex repairs.

Inspection techniques are integrally tied with potential damage levels, so the inspection plan must mature along with the design. Cost-effective methods to address large areas of structure are needed to initially detect damaged regions. Visual methods are most frequently used, but more sophisticated methods that can detect less severe damage are also possible. Damage that may not be found over the life of an aircraft structure, for example, with the selected inspection method must sustain Ultimate loads, while damage levels that are easily detectible during periodic inspections must be good for Limit load. (See Figure 7.2.7, and further discussion in Volume 3, Section 12.2.2 and Figure 12.2.3(a).)

Since most areas of the final design will have capability beyond that required, it is important to know what damage states reduce the strength at a specific structural location to Ultimate and Limit load capability. These damage states for aviation applications are, by definition, the Allowable Damage Limits (ADLs) and the Critical Damage Threshold (CDT), respectively. Clearly defined Allowable Damage Limits (ADLs) should be developed to allow rapid determination of the need for repair during scheduled inspection. Minimum ADLs may also be included as design requirements or objectives to reduce the number of necessary repairs.

The design of some areas of the structure can be controlled by manufacturing and durability considerations, such as minimum gage (to provide a minimum of impact damage resistance and avoid knife-edge at countersink fasteners), stiffener, rib and frame flange width, bolt spacing and edge distance requirements, and avoiding rapid ply drops and buildups. Another requirement for maintainable sandwich structure is the establishment of nondestructive inspection (NDI) and evaluation (NDE) procedures for practical damage location and quantitative assessment.

Structural repair issues. Sandwich structures are generally repaired with bonded scarf or stepped patches. Typical scarf/step taper ratios employed when repairing thin face sheets are quite shallow (e.g.,

30:1 and sometimes as shallow as 50:1). When repairing sandwich structures with thicker face sheets in more highly loaded areas, shallow taper ratios result in the removal of a large amount of undamaged material, and large patch sizes. In these situations, repairs may be combinations of scarfed and external patches. Thick face sheets require thick patches, which may require special processing to achieve proper consolidation. Patch and bondline porosity are of concern with normal field processing accomplished using vacuum pressure and heat blankets. Lower temperature cures are generally preferred due to concerns about additional damage via vaporization of water that has infiltrated the core. Also, the surrounding structure may act as a heat sink, making it difficult to achieve and control the higher temperatures with heat blankets, and may contribute to thermal gradients that can result in warpage or degradation of the surrounding structure. For thick sandwich, heat blankets on both sides of the structure may be required to control the through-thickness temperature. The shorter processing times generally associated with higher temperature cures are desirable for minimizing the out-of-service time for a damaged airplane.

Moisture ingression issues. Consideration must be given to moisture ingression when designing sandwich structures. Sandwich designs should address the effects of moisture in the core. When repairing damaged sandwich structures, a drying cycle is typically performed prior to the bonded repair so that any retained moisture does not interfere with the curing cycle. There have been numerous cases of face sheets blowing off sandwich components during the vacuum bag heating cure cycle when proper drying procedures have not been employed. Standing water that can accumulate in improperly sealed sandwich structures may add excessive weight (resulting in weight/balance issues, poor fuel economy, etc.) and can cause significant damage when it freezes. Drainage provisions are needed in some sandwich structure applications.

7.3 SUPPORT IMPLEMENTATION

A repair has the objective of restoring a damaged structure to an acceptable capability in terms of strength, durability, stiffness, functional performance, safety, cosmetic appearance, and service life. Ideally, the repair will return the structure to original capability and appearance. The design assessment of a repair for a given loading condition involves the selection of a repair concept, the choice of the appropriate repair materials and processes, then specifying the detailed configuration of the repair.

7.3.1 Part inspection

Damage in aircraft sandwich structure parts is usually found in a routine line inspection, depot inspection, or, for large damage, noticed by the pilot and/or crew members. The predominant mode of inspection is visual with more sophisticated inspections performed as needed and at the depot. Once damage is identified in-service, the damage should be characterized by measuring characteristics such as dent depth, extent of surface damage, and length of scratches, but should also consider damage that may have resulted in areas away from the impact or damage origin location. This will generally consist of tap testing to define the damage extent, followed with instrumented NDI techniques to more thoroughly characterize the damage. A good general reference on inspection methods is SAE ARP 5089 “Composite Repair NDI and NDT Handbook” (Reference 7.3.1). Inspection by visual means is by far the oldest and most economical NDI method. Fortunately, most types of damage either scorch, stain, dent, penetrate, abrade, or chip the composite surface making the damage visually identifiable. However, it is very possible for a blunt impact to create damage that is not visually obvious as a surface may spring back to its original contour with little to no indication otherwise. Chapter 6 provides a more thorough discussion regarding inspection techniques for sandwich structures.

7.3.2 Damage assessment

Damage assessment is the intermediate stage between inspection and repair and includes the decision about if and how to repair a damaged structure, the nature of the repair (permanent or temporary), and inspection after the repair and during the residual life of the repaired structure.

Mandate of the assessor - The mandate of the assessor is the authority to interpret the inspection results and to decide on the needed repair and residual life of the structure. In the field, the mandate of the assessor is limited to following the manufacturer's instructions. In a repair station, and at the manufacturer's facilities, it can be extended, provided that engineering approvals are obtained. For larger damages, experimental substantiation may be required.

Qualification of the assessor - The assessor will generally not know what the degradation of the structure due to damage will be. A qualified engineering representative should have the technical background to understand the inspection results, the available design information, should be familiar with the repair capabilities, and have the necessary skills and experience to determine the structural degradation due to damage based on calculations and/or previous data.

Information for damage assessment - The following information is needed in the assessment process: damage characterization; damage geometry; damage location; and degradation of the structure due to the damage. The available inspection and repair capability has to be evaluated at this stage as part of the decision process.

7.3.3 Repair design criteria

Repair design criteria should assure that the structural integrity and functionality of the repaired part are the same as that of the undamaged part. The repair design criteria should be established by the original manufacturer or cognizant engineering authority and used to develop repairs in the Structures Repair Manual (SRM). SRM's for specific aircraft frequently "zone" the structure to show the amount of strength restoration needed or the kinds of standard repairs that are acceptable. Zoning permits the use of simpler repairs in areas where large strength margins exist. Zoning also permits the restriction of operator repairs in areas where repairs are too complex and should be only repaired with original equipment manufacturer's (OEM's) involvement.

Repair design criteria for permanent repairs are fundamentally those that designed the part that is to be repaired. These are: restore stiffness of the original structure, withstand static strength at the expected environments up to ultimate load including stability (except for post buckled structure), assure durability for the remaining life of the component, satisfy original part damage tolerance requirements, and restore functionality of systems. Additionally there are other criteria applicable in repair situations. For aircraft structures these include: minimize aerodynamic contour changes, minimize weight penalty, minimize load path changes, and be compatible with aircraft operations schedule.

Part stiffness - First consideration in any repair is to replace structural material that is damaged. This means that especially for large repairs the stiffness and placement of repair material should match the parent material as closely as possible. This avoids any recalculations of the overall dynamic behavior of the component, such as flutter or structural load redistribution. Furthermore, many lightweight flight vehicle structures are designed to meet stiffness requirements that are more critical than their strength requirements. A repair made to a structure of this type must, therefore, maintain the required stiffness so that deflections or stability requirements are met.

Static strength and stability - Any permanent repair must be designed to support applied loads at the ultimate design load level at the extremes of temperature excursions, moisture levels, and barely visible damage levels. If the loads are not available, specific SRM repair recommendations must be strictly adhered to. In the SRM repairs, there is an implicit assumption that the specific repairs meet all static strength and stability requirements.

Load path changes are a special concern when designing repairs. When strength restoration is necessary, attention must be given to the effect of the stiffness of the repair on the load distribution in the structure. If a patch has less stiffness than the original structure, the patch may not carry its share of the load, and this causes an overload in the surrounding material. Conversely, an overly stiff patch may attract more than its share of load, causing adjacent areas to which it is attached to be overloaded. Stiff-

ness mismatch between parent material and the patch may cause peel stresses that can initiate debonding of the patch.

Durability - Durability is the ability of a structure to function effectively throughout the life of the vehicle. For commercial transport aircraft, the design life can be greater than 50,000 cycles; military fighter aircraft are designed for 4,000 to 6,000 flight hours. Included among the factors affecting durability are temperature and moisture environments. Although the parent composite structure may not be durability critical, structural repairs may be more susceptible to damage caused by repeated loads during their service lives.

Damage tolerance - Composite structures are designed to be damage tolerant to accidental damage. In practice, this is accomplished by lowering design strains so that the structure with impact caused damage can withstand ultimate load. Repairs must also be capable of tolerating a predetermined level of impact damage.

Aerodynamic smoothness - High-performance flight vehicles depend on smooth external surfaces to minimize drag. During initial fabrication, smoothness requirements are specified, usually by defining zones where different levels of aerodynamic smoothness are required. Most SRM's specify smoothness requirements for repairs consistent with initial part fabrication.

Operating temperatures - Most flight vehicles experience extremes of temperature during use. Repairs to such flight vehicles must be acceptable for the temperature extremes for which the vehicle was designed. Low temperatures result from high-altitude flight or from extremes of ground storage in cold climates. Many aircraft are designed for a minimum service temperature of -65°F (-54°C). Elevated-temperature requirements vary with the type of vehicle. The maximum temperature for commercial transport aircraft and most rotary wing vehicles is 160°F (71°C) and generally occurs during ground soak on a hot day. However, components experiencing significant loads during takeoff and initial climb may require validation of design ultimate loads at temperatures up to 200°F (93°C). Supersonic transport, fighter, and bomber aircraft typically experience aerodynamic heating of up to 220°F (104°C) or in special cases as high as 265°F (130°C), especially on the leading edges of lifting surfaces. Components exposed to engine heat, such as nacelles and thrust reversers, may be required to withstand even higher temperatures in local areas.

Environment - Repairs may be exposed to many environmental effects, including those listed below:

1. Fluids - salt water or salt spray, fuel or lubricants, hydraulic fluid, paint stripper, and humidity

2. Mechanical loading - shock, acoustic or aerodynamic vibration, and operating loads

3. Thermal cycling

Absorbed moisture can affect bonded repairs in several ways.

- Parent Laminate Blistering - As a "wet" laminate is heated to cure a bonded repair, the absorbed moisture may cause local delaminations or blisters. Pre-bond drying at lower temperatures, slow heat-up rates, and reduced cure temperatures all diminish the tendency to blister.

- Blown Face Sheets/Core of Sandwich Structure - Moisture in the cells of honeycomb sandwich structure expands when the part is heated to cure a bonded repair and develops sufficient pressure to separate the face sheet from the core, especially if the strength of the adhesive has been reduced by temperature and moisture. Similarly, this process may be sufficiently severe to rupture cell walls in low density core. Pre-drying is normally used to prevent bondline failure of this type.

- Porosity in Bondlines - As a repair is bonded to a "wet" laminate, the moisture tends to cause porosity in the bondline. This porosity can reduce the strength of the bondline. This problem can be

minimized by pre-drying, reduced temperature cure, and selection of moisture-resistant adhesives.

- It also can reduce the wet-out ability of the adhesive and actually impede the chemical bonding processes from occurring, resulting in weak/poor bond lines.

Temporary repairs - Repair design criteria for temporary or interim repairs can be less demanding, but may approach permanent repairs if the temporary repair is to be on the structure for a considerable time. Most users of aircraft prefer permanent repairs, if possible, as the temporary repairs may damage parent structure necessitating a more extensive permanent repair. Temporary repair will restore functionality but static strength requirements may be reduced to limit load or maximum load in the spectrum. Damage tolerance and durability goals are often severely reduced but are compensated by shorter inspection intervals.

7.3.4 Repair of composite structures

The task of repair begins after the extent of the damage has been established. The repair has the objective of restoring the structure to a required capability in terms of strength, stiffness, functional performance, safety, service life, and cosmetic appearance. Ideally, the repair will return the structure to original capability and appearance. To start the repair process the structural makeup of the component must be known and the appropriate design criteria should be selected from the considerations described in Section 7.3.3. The continuity in load transfer is reestablished in a damaged part by attaching new material by bolting or bonding.

7.3.4.1 Damage removal and site preparation

The first task in site preparation is the removal of coatings, typically by hand sanding or other mechanical means. The use of chemical paint stripper is normally undesirable as it can attack the composite resin system and can also become entrapped in the honeycomb core. Once the topcoat and primer are removed and the damaged plies clearly defined, the damaged plies are removed by sanding or other mechanical means. Damaged core must be cut out, with care taken not to damage the inner surface of the opposite face sheet.

Once the damage has been removed, the repair area should be thoroughly cleaned of contaminants, such as hydraulic fluids or engine oils, and should always be thoroughly dried if the repair involves a cure cycle with temperatures above 200°F (93°C). Undetected moisture will turn into steam during elevated temperature cure, causing blown core and disbonding of face sheets. For honeycomb parts cured at room temperature, presence of moisture is undesirable, particularly if the core material is aluminum. For bonded repairs, site preparation usually involves taper sanding or step cutting of plies to produce gradual introduction of load into and out of the repair material.

7.3.4.2 Bonded repairs

Bonded repairs normally use external or internal patches that are made flush with the parent material. Patches can be stepped or scarfed. Scarf angles are usually small to ease the load into the joint and to prevent adhesive from escaping, with thickness to length ratios between 1/10 to 1/40. Adhesive placed between the repair material and the parent material transfers the load from the parent material to the patch by shear. The stress concentration at the edge of the patch can be reduced by stepping or tapering the patch, as shown in Figure 7.3.4.2.

The scarf joint is more efficient from the viewpoint of load transfer as it reduces load eccentricity by closely aligning the neutral axis of the parent and the patch. However, this configuration has several drawbacks: to maintain a small taper angle, a large quantity of sound material must be removed; the replacement plies must be accurately placed in the repair joint; curing of replacement plies can result in significantly reduced strength if not cured in the autoclave; and the adhesive can run to the bottom of the

joint creating a non-uniform bond line. When properly done, this type of repair can result in part strength as strong as the original part.

The patch can be pre-cured and then secondarily bonded to the parent material, made from prepreg and then co-cured at the same time as the adhesive, or the patch can be made using dry cloth and paste resin, then co-cured (called "wet" lay-up repair).

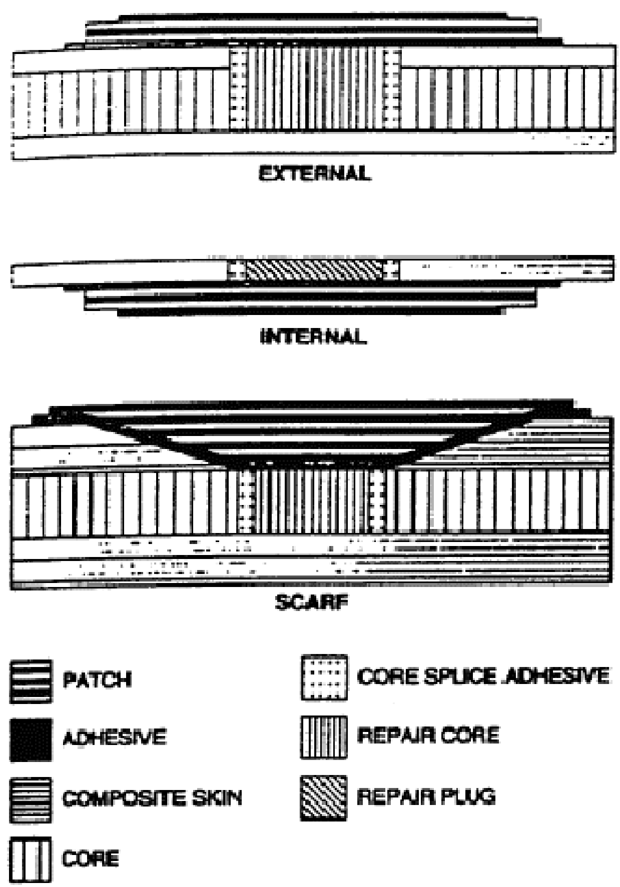

FIGURE 7.3.4.2 *Basic bonded repairs.*

Repair materials - Bonded repairs require selection of both the repair material and adhesive. A good description of materials for bonded repair is provided in Air Force TO 1-1-690 (Reference 7.3.4.2(a)) and NAVAIR 01-1A-21 (Reference 7.3.4.2(b)). Material suppliers have developed unique materials that are optimized for the repair process. Repair materials are usually lower in strength and stiffness than the original part materials.

Co-cured bonded repairs use parent material prepreg, repair material prepreg, or dry fabric with laminating resin. The resin for the wet lay-up repair often consists of two parts that do not require freezers. However, mixing of the two parts and spreading the mixed resin on the dry fabric requires strict adherence to protocol and experienced personnel to affect consistent repairs.

Two categories of adhesives are films and pastes. Films come with and without mesh carrier cloth with typical thickness between 0.0025 to 0.01 in. (0.064 to 0.25 mm). The carrier cloth provides improved

handling, results in a more uniform bondline, and helps reduce galvanic corrosion. Although film adhesives provide a more uniform bondline thickness than paste adhesives, a lack of refrigerated storage equipment often necessitates the use of paste adhesives. Wet lay-up repairs can be accomplished using paste adhesives consisting of two separate parts that have a long shelf life prior to mixing.

7.3.4.3 Repair analysis

A bonded repair is, from a structural point of view, a bonded joint. The repair geometry is often two-dimensional. If a sandwich structure is repaired, the core supports out-of-plane loads, which is why bonded repairs are very efficient for sandwich structures. The sandwich repair analysis is addressed in Volume 3, Section 14.6.

In some cases, the geometry can be approximated with lap or strap joint models. A two-dimensional finite element model can be used to calculate load distributions in the face sheet, patch, and adhesive layer. A nonlinear solution is often used to account for the nonlinear stress strain behavior of the adhesive (Volume 3, Chapter 10.4.5).

Several specially developed computer codes can be used for analyzing bonded repairs. In Reference 7.3.4.3(a), the codes PGLUE (Reference 7.3.4.3(b)), A4EI (Reference 7.3.4.3(c)) and ESDU8039 (Reference 7.3.4.3(d)) are discussed. The PGLUE program contains an automatic mesher which creates a three-dimensional finite element model of a repaired panel containing three components - a plate with a cutout, a patch, and an adhesive connecting the patch and the plate. Plasticity of the adhesive is considered in the analysis. However, the version commonly available through Air Force's Aerospace Structures Information and Analysis Center (ASIAC) does not consider peel stresses, which can be critical. Traditional bonded joint codes, such as A4EI and ESDU8039, model only a slice through the repair and do not consider the two-dimensional effects of stiffening of the sides of the repair area. Both bonded joint codes allow the patch to be stepped. A4EI considers plasticity in the adhesive shear stress but does not predict peel stress, while ESDU8039 predicts peel stress in the joint but does not consider plasticity.

Failure Analysis Considerations

- The joint should be designed in such a way that the adhesive layer is not the critical joint element.
- Peel and transverse shear stresses should be minimized by design (tapered or stepped adherends, filleting, etc.).
- Incorporation of nonlinear stress-strain behavior of the adhesive (usually approximated by elastic-plastic stress-strain curve).
- Dependence of the elastic properties of adhesive on its thickness.
- Adhesive properties as a function of environment and long term degradation.

7.3.4.4 Repair procedures

Repair procedures are generally specified by the OEM and documented in controlled documents such as commercial carrier SRMs, NAVAIR 01-1A-21 (Reference 7.3.4.2(b)), and Air Force TO 1-1-690 (Reference 7.3.4.2(a)). Bonded repairs require close control of the repair process and the repair environment. Structural integrity of the bonded joint is strongly dependent on the cleanliness of the work area and its ambient temperature and humidity.

Bonded Repair Procedures - Because sandwich structure is a bonded construction and the face sheets are thin, damage to sandwich structure is usually repaired by bonding. The following paragraphs describe various typical sandwich structure repair scenarios as they apply to some common aircraft components.

Core restoration - For full-depth core replacement there are three common methods; the core fill method, the paste adhesive method, and the film/foam method. The three methods are shown in Figure 7.3.4.4(a). The core fill method replaces the damaged honeycomb with glass reinforced paste adhesive and is limited to small damage sizes. The weight of the repairs must be calculated and compared with flight control weights and balance limits set out in the SRM. The other two methods can be used interchangeably depending on the available adhesives. However, the paste adhesive method results in a much heavier repair than the film/foam method, especially if the damage diameter is greater than 4 inches (100 mm). The foaming adhesive required to utilize the film/foam method is a thin unsupported epoxy film containing a blowing agent which is liberated during cure causing a foaming action. The expansion process needs to be performed under positive pressure to become strong, highly structured foam. Like film adhesives, foaming adhesives require high temperature cure and refrigerator storage. Core replacement is usually accomplished with a separate curing cycle and not co-cured with the patch.

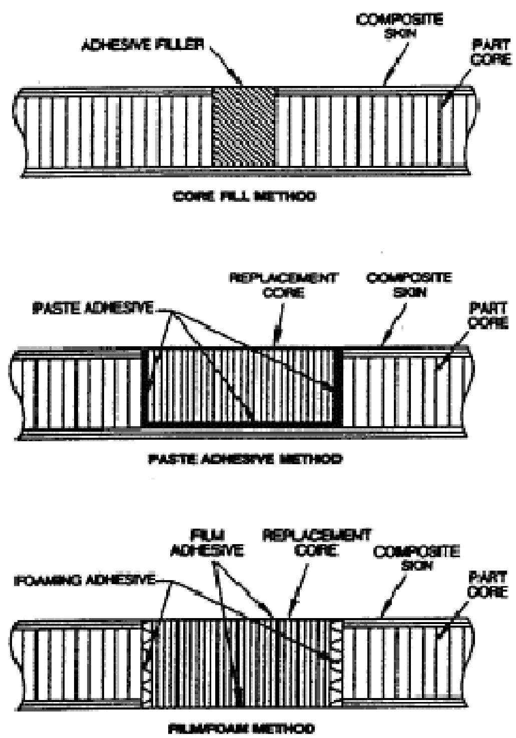

FIGURE 7.3.4.4(a) *Core replacement methods – full depth.*

For partial-depth damage, different methods can be used to attach the replacement honeycomb to the parent honeycomb as shown in Figure 7.3.4.4(b). The two methods are prepreg/film adhesive bonding and wet lay-up bonding. A description of how to perform core restorations for simple configuration is contained in SAE ARP 4991 - Core Restoration (Reference 7.3.4.4).

Following core replacement, the sandwich repair proceeds as a bonded repair of the face sheets as described above. When repairing one face sheet of the sandwich, the repaired face sheet should not have a significantly different stiffness than the base face sheet. An additional step before bonding on the face sheets is to bond a pre-cured fiberglass plug on top of the exposed core, to preserve continuity of the bond between the core and the face sheets. For bonded repair of the sandwich structure, the honeycomb must be thoroughly dried to prevent face sheet disbond during curing, and the curing pressure must be low to prevent honeycomb crushing. If it is unfeasible to dry out the honeycomb, lower temperature (200°F (93°C)) curing can be used.

Occasionally, sandwich structure is repaired using bolted external patches. In this case, the honeycomb where the bolts would pass through has to be strengthened by filling the core with the same filler as for core replacement. The diameter of this area should be at least three times the diameter of the bolt. Special bolts with limited clamping force are used for such repairs.

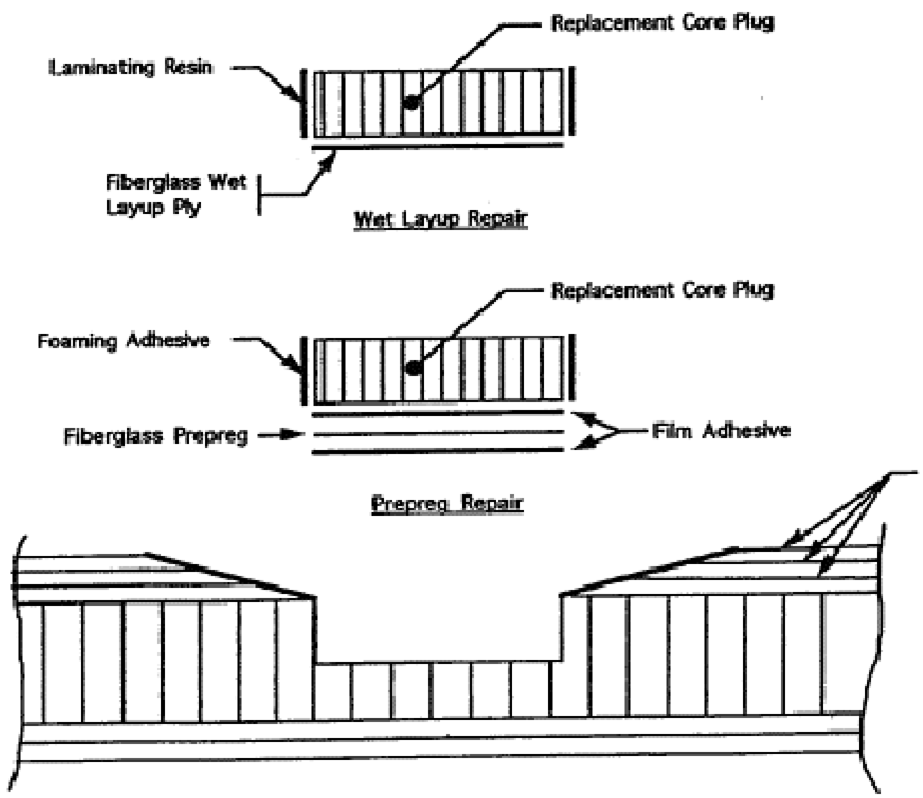

FIGURE 7.3.4.4(b) *Core replacement – partial depth.*

Sandwich repair example - The sandwich repair example is taken from NAVAIR 01-1A-21 (Reference 7.3.4.2(b)). Steps in the repair (shown in Figure 7.3.4.4(c)) consist of removing damaged material, drying the repair area, fitting replacement core, tacking the replacement core using filled paste adhesive with glass reinforcement, machining the core to match the contour of the part, installing core with foaming adhesive using a heat blanket, and installing face sheet patches using an additional cure cycle.

7.3.4.5 Repair inspection

Composite materials and adhesives require extensive record keeping to ensure they are within life, such as storage time in the refrigerator, warm-up time, and out time on the shop floor. Lay-up operations should be inspected for correct fiber orientation. Cure cycles must be monitored to assure that specifications are followed. For large repairs, a small companion panel is cured with the repair and used for coupon testing to provide confidence in the quality of the repair. Completed repairs should be inspected to determine their structural soundness using NDI methods described previously in Section 7.3.1.

7.3.4.6 Repair validation

Inspection of a repair is not sufficient to guarantee that the repair will perform as designed. Repair designs should be supported by an experimentally verified database and by analysis. Allowables can be based on the parent material properties with knock down to reflect lower cure temperatures and pressures, or allowables and material properties can be based on the repair material to be used in the repair analysis. In addition to coupon testing, a variety of elements are tested to validate repair designs. These are usually performed to support repair designs included in the SRM and range from simple joint coupons to tests of full-scale repairs. Simple two-dimensional joint coupons are used for development of repair designs; for example, a lap bonded coupon to obtain joint shear strengths. More complex elements are used to validate the repair design and repair process.

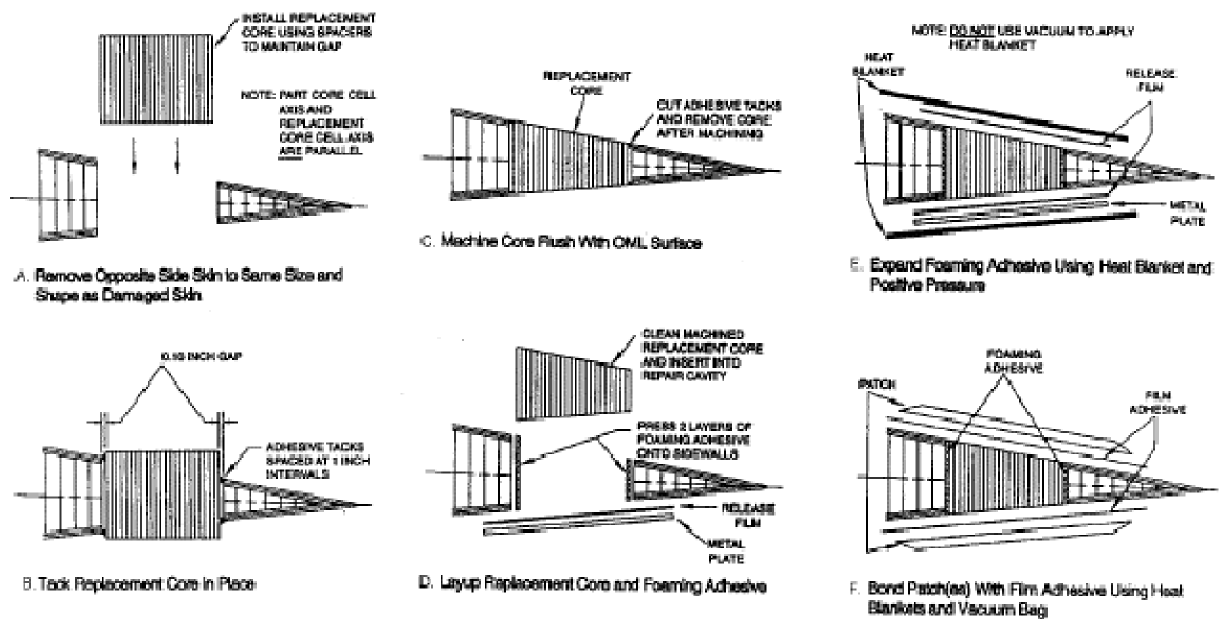

FIGURE 7.3.4.4(c) *Bonded repair of sandwich control surface.*

7.4 LOGISTICS REQUIREMENTS

Specialized knowledge and skills are required to perform sound structural composite repairs. For aircraft applications, repair technicians must be trained in a formal program for certification since they will be expected to perform bonded repairs on diverse structures with many different types of material.

Materials – Repair materials present another logistics support issue. While most structural components on an aircraft are manufactured from prepreg composite materials, support considerations may make it desirable to use different repair materials. The repair materials may be prepreg or dry cloth with a laminating resin. Film and foaming adhesives, while offering excellent structural properties, require cold storage. Laminating resins and paste adhesives present a room-temperature storable alternative, but have reduced performance. If the damaged structure has a lightning strike protection scheme, this must be restored.

Facilities – A field or depot composite repair facility consists of a lay-up area, a part preparation area, a part curing area, and a material storage area. An environmentally-controlled lay-up area is required to prevent contamination of the repair surface and materials. For on-aircraft repair or repairs conducted in a field environment, humidity and temperature control is unlikely. Some form of shelter should then be de-

vised around the repair area. Repair materials should be prepared and sealed in bags in the shop, and the bags only opened immediately prior to installation.

Because a depot repair facility must perform repairs which range up to out-right component remanufacture, the facilities should replicate those of the original equipment manufacturer. Floor space is less at a premium at the depot than it would be in a field environment; thus separate lay-up, bonding, tool manufacture, part machining, and part and tool storage areas will be available. During remanufacture, components are off-aircraft and mobile. Therefore, large industrial cold-storage, curing, machining, and inspection equipment can be used to perform repair operations at the depot.

Technical Data – Technical data in various forms is required to support composite repair to aircraft structure. Technical data ranges from structural repair manuals/military technical manuals, to part drawings and models, to loads books and finite element models.

Equipment – Heat blankets, hot bonders, heat lamps, heat guns and convection ovens are portable heating and curing equipment usually found in both the depot and field environments. These are usually used in conjunction with a vacuum bag, to expedite moisture removal prior to repair, and to provide consolidation pressure to the repair. They can be used to manufacture pre-cured composite repair patches and to bond repairs to the component. Whatever the portable heat source, generous use of thermocouples must be made to closely monitor cure temperatures. Heat blankets consist of heating elements sandwiched between temperature-tolerant flexible materials. Hot bonders are programmable heat and vacuum control units, which provide power to heat blankets automatically to an operator-specified cure cycle. Hot bonders often have a vacuum pump included. Infrared heat lamps and heat guns are also used for elevated temperature cures of composite repairs. Industrial ovens are a necessity at depots as a means of drying and curing composite parts and repairs. Autoclaves are pressurized ovens usually found at depot facilities for part repair and remanufacture. The typical 85 psi (586 kPa) pressures required to achieve maximum consolidation of a composite prepreg laminate make large structural repairs possible.

Cold storage equipment that can maintain material at 0°F (-18°C) or lower is required to maintain shelf-life on most preimpregnated materials and adhesives. For paint removal and machining of cured composite materials, down or side draft facilities separate from the bonding and curing area are required to remove dust produced during machining operations. Both field and depot facilities require equipment for nondestructive inspection of composite components. Radiography (X-ray), thermography (infra-red), ultrasonic and laser shearography inspection equipment may be required for pre-repair damage mapping, in progress inspection, and post-repair inspection.

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