Quadrilateral Plate Element Connection

CQUAD4

Defines an isoparametric membrane-bending or plane strain quadrilateral plate element.

Format:

1	2	3	4	5	6	7	8	9	10
CQUAD4	EID	PID	G1	G2	G3	G4	THETA or MCID	ZOFFS	
			T1	T2	T3	T4			

Example:

CQUAD4	111	203	31	74	75	32	2.6	0.3	
			1.77	2.04	2.09	1.80			

Field	Contents
EID	Element identification number. (Integer > 0)
PID	Property identification number of a PSHELL, PCOMP, or PLPLANE entry. (Integer > 0; Default = EID)
Gi	Grid point identification numbers of connection points. (Integers > 0, all unique.)
THETA	Material property orientation angle in degrees. THETA is ignored for hyperelastic elements. See Figure 2. (Real; Default = 0.0)
MCID	Material coordinate system identification number. The x-axis of the material coordinate system is determined by projecting the x-axis of the MCID coordinate system (defined by the CORDij entry or zero for the basic coordinate system) onto the surface of the element. MCID is ignored for hyperelastic

elements. (Integer ≥ 0; If blank, then THETA = 0.0 is assumed.)

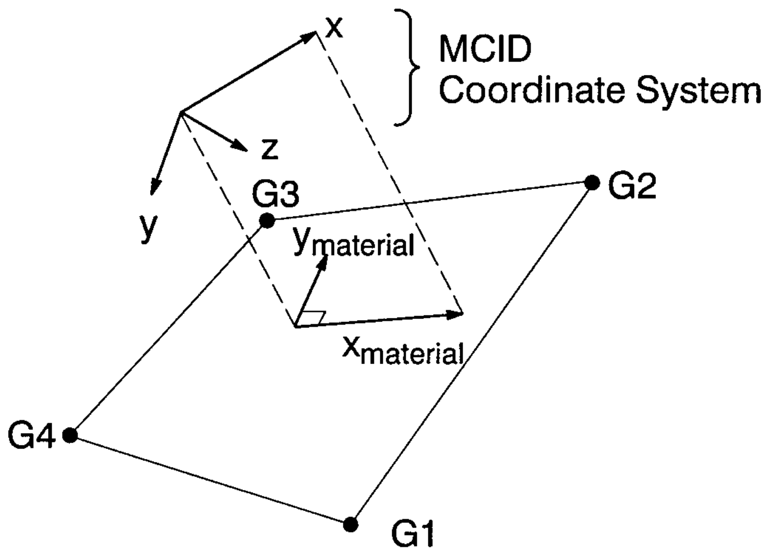


Figure 1. MCID Coordinate System Definition.

ZOFFS	Offset from the surface of grid points to the element reference plane. ZOFFS is ignored for hyperelastic elements. See Remark 6. (Real)
Ti	Membrane thickness of element at grid points G1 through G4. Ti are ignored for hyperelastic elements. (Real ≥ 0.0 or blank, not all zero. See Remark 4 for default.)

Quadrilateral Plate Element Connection **CQUAD4**

Remarks:

1. Element identification numbers should be unique with respect to all other element identification numbers.
2. Grid points G1 through G4 must be ordered consecutively around the perimeter of the element.

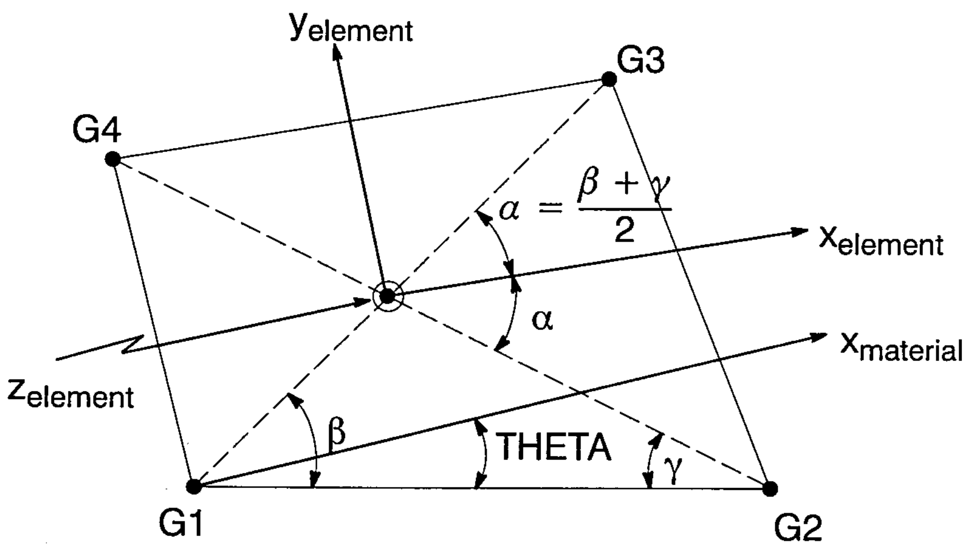


Figure 2. CQUAD4 Element Geometry and Coordinate Systems.

3. All the interior angles must be less than 180°.
4. The continuation is optional. If it is not supplied, then T1 through T4 will be set equal to the value of T on the PSHELL entry.

Quadrilateral Plate Element Connection **CQUAD4**

5. The reference coordinate system for the output of stress, strain and element force depends on the element type.

   a. For CQUAD4 elements which are not p-elements and not hyperelastic, the reference coordinate system for output is the element coordinate system.

   b. For CQUAD4 elements is referenced by a PSET or PVAL entry, the stresses, strains and element forces are output in the local tangent plane of the element. The local tangents are oriented in a user defined direction which is uniform across a set of elements. By default, the local tangent x-direction is oriented in the positive x-direction of the basic coordinate system. See the OUTRCV Bulk Data entry for user defined output coordinate systems.

   c. For hyperelastic elements the stress and strain are output according to CID on the PLPLANE entry.

6. Elements may be offset from the connection points by means of ZOFFS. Other data, such as material matrices and stress fiber locations, are given relative to the reference plane. A positive value of ZOFFS implies that the element reference plane is offset a distance of ZOFFS along the positive Z-axis of the element coordinate system. If the ZOFFS field is used, then the MID1 and MID2 fields must be specified on the PSHELL entry referenced by PID.

   The specification of offset vectors gives wrong results in solution sequences that compute differential stiffness: linear buckling analysis provided in SOLs 5, 16, 105 and 200, and geometric nonlinear analysis provided in SOLs 106, 129, 153, and 159 with PARAM,LGDISP,1.

Quadrilateral Plate Element Connection **CQUAD4**

7. For finite deformation hyperelastic analysis, the plot codes are given by the CQUADFD element name in the *MSC/NASTRAN Quick Reference Guide*, Appendix A.

8. If a CQUAD4 element is referenced by a PSET or PVAL entry, then a p-version formulation is used and the element can have curved edges.

   a. If a curved edge of a p-element is shared by an h-element without midside nodes, the geometry of the edge is ignored and set straight.

   b. Edges with midside nodes cannot be shared by p-elements.

9. By default, all of the four edges of the element are considered straight unless the element is a p-element and the edge is associated to curved geometry with a FEEDGE or FEFACE entry.

Curved Quadrilateral Shell Element Connection **CQUAD8**

Defines a curved quadrilateral shell or plane strain element with eight grid points.

Format:

1	2	3	4	5	6	7	8	9	10
CQUAD8	EID	PID	G1	G2	G3	G4	G5	G6	
	G7	G8	T1	T2	T3	T4	THETA or MCID	ZOFFS	

Example:

CQUAD8	207	3	31	33	73	71	32	51	
	53	72	0.125	0.025	0.030	.025	30.	.03	

Field	Contents
EID	Element identification number. (Integer > 0)
PID	Property identification number of a PSHELL, PCOMP, or PLPLANE entry. (Integer > 0)
G1, G2, G3, G4	Identification numbers of connected corner grid points. Required data for all four grid points. (Unique Integers > 0)
G5, G6, G7, G8	Identification numbers of connected edge grid points. Optional data for any or all four grid points. (Integer ≥ 0 or blank)
Ti	Membrane thickness of element at corner grid points. Ti are ignored for hyperelastic elements. (Real ≥ 0.0 or blank, not all zero. See Remark 4 for default.)

Curved Quadrilateral Shell Element Connection — CQUAD8

THETA	Material property orientation angle in degrees. See Figure 2. THETA is ignored for hyperelastic elements. (Real; Default = 0.0)
MCID	Material coordinate system identification number. The x-axis of the material coordinate system is determined by projecting the x-axis of the MCID coordinate system (defined by the CORDij entry or zero for the basic coordinate system) onto the surface of the element. MCID is ignored for hyperelastic elements. (Integer ≥ 0; If blank, then THETA = 0.0 is assumed.)

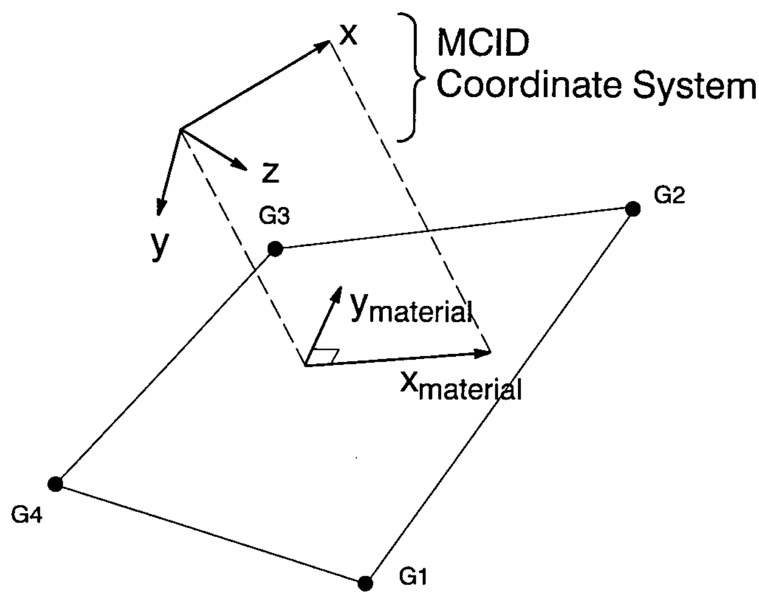


Figure 1. MCID Coordinate System Definition.

ZOFFS	Offset from the surface of grid points to the element reference plane. See Remark 6. ZOFFS is ignored for hyperelastic elements. (Real)

Remarks:

1. Element identification numbers should be unique with respect to all other element IDs of any kind.
2. Grid points G1 to G8 must be numbered as shown.
3. The orientation of the material property coordinate system is defined locally at each interior integration point by THETA, which is the angle between $x_{material}$ and the line of constant η.
4. T1, T2, T3 and T4 are optional. If they are not supplied, they will be set equal to the value of T on the PSHELL entry.
5. It is recommended that the midside grid points be located within the middle third of the edge. If the edge point is located at the quarter point, the program may fail with a divide-by-zero error or the calculated stresses will be meaningless.
6. Elements may be offset from the connection points by means of the ZOFFS field. Other data, such as material matrices and stress fiber locations, are given relative to the reference plane. A positive value of ZOFFS implies that the element reference plane is offset a distance of ZOFFS along the positive z-axis of the element coordinate system. If the ZOFFS field is used, then the MID1 and MID2 fields must be specified on the PSHELL entry referenced by PID.

   The specification of offset vectors gives wrong results in solution sequences with linear buckling analysis, SOLs 5, 16, 105 and 200.

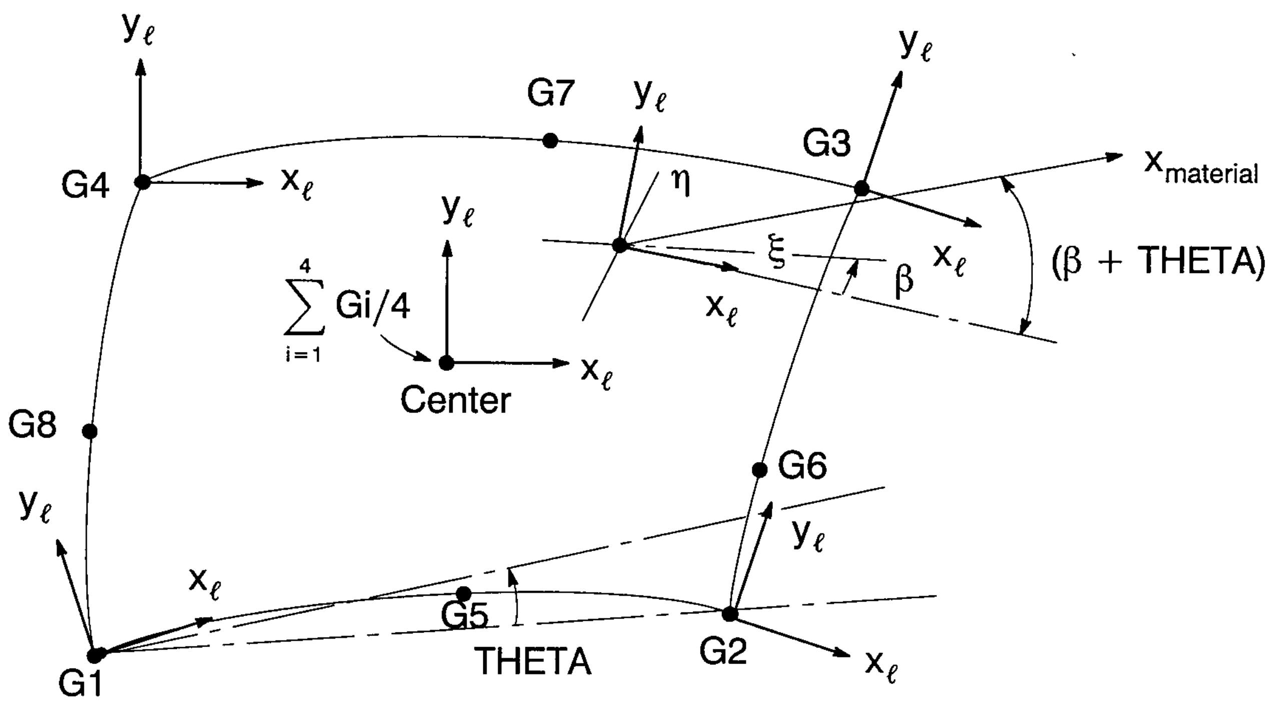
7. If all midside grid points are deleted, then the element will be excessively stiff and the transverse shear forces incorrect. A User Warning Message is printed, and a CQUAD4 element is recommended instead. If the element is hyperelastic, then it is processed identically to the hyperelastic CQUAD4 element.

8. For a description of the element coordinate system, see the *MSC/NASTRAN Reference Manual,* Section 5.3.2. Stresses and strains are output in the local coordinate system identified by x_ℓ and y_ℓ in Figure 2. However, for hyperelastic elements the stress and strain are output in the coordinate system identified by the CID field on the PLPLANE entry.

9. For hyperelastic elements the plot codes are specified under the CQUADFD element name in the *MSC/NASTRAN Quick Reference Guide*, Appendix A.

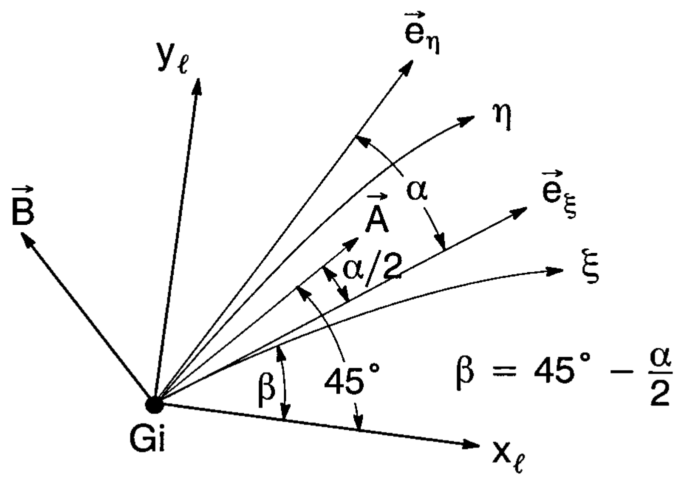
Curved Quadrilateral Shell Element Connection

CQUAD8



Stress output at each Gi local system

Curved Quadrilateral Shell Element Connection **CQUAD8**



where	$\vec{e}_\eta$	is tangent to h at Gi
	$\vec{e}_\xi$	is tangent to c at Gi
	$\vec{A}$	is formed by bisection of $\vec{e}_\eta$ and $\vec{e}_\xi$
	$\vec{B}$ and $\vec{A}$	are perpendicular
	y_ℓ	is formed by bisection of $\vec{A}$ and $\vec{B}$
	x_ℓ	is perpendicular to y_ℓ

Figure 2. CQUAD8 Element Geometry and Coordinate Systems.

Layered Composite Element Property **PCOMP**

Defines the properties of an n-ply composite material laminate.

Format:

1	2	3	4	5	6	7	8	9	10
PCOMP	PID	Z0	NSM	SB	FT	TREF	GE	LAM	
	MID1	T1	THETA1	SOUT1	MID2	T2	THETA2	SOUT2	
	MID3	T3	THETA3	SOUT3	-etc.-				

Example:

PCOMP	181	−0.224	7.45	10000.0	HOFF				
	171	0.056	0.	YES			45.		
			−45.				90.		

Field	Contents
PID	Property identification number. (0 < Integer < 1000000)
Z0	Distance from the reference plane to the bottom surface. See Remark 10. (Real; Default = −0.5 times the element thickness.)
NSM	Nonstructural mass per unit area. (Real)
SB	Allowable shear stress of the bonding material (allowable interlaminar shear stress). Required if FT is also specified. (Real > 0.0)
FT	Failure theory. The following theories are allowed (Character or blank. If blank, then no failure calculation will be performed):

Layered Composite Element Property **PCOMP**

	"HILL" for the Hill theory.
	"HOFF" for the Hoffman theory.
	"TSAI" for the Tsai-Wu theory.
	"STRN" for the Maximum Strain theory.
TREF	Reference temperature. See Remark 3.. (Real; Default = 0.0)
GE	Damping coefficient. See Remarks 4. and 12. (Real; Default = 0.0)
LAM	Symmetric lamination option. If LAM = "SYM", only plies on one side of the element centerline are specified. The plies are numbered starting with 1 for the bottom layer. If an odd number of plies is desired with LAM = "SYM", then the center ply thickness (Ti) should be half the actual thickness. (Character or blank. If blank, all plies must be specified.)
MIDi	Material ID of the various plies. The plies are identified by serially numbering them from 1 at the bottom layer. The MIDs must refer to MAT1, MAT2, or MAT8 Bulk Data entries. See Remark 1. (Integer > 0 or blank, except MID1 must be specified.)
Ti	Thicknesses of the various plies. See Remark 1. (Real or blank, except T1 must be specified.)
THETAi	Orientation angle of the longitudinal direction of each ply with the material axis of the element. (If the material angle on the element connection entry is 0.0, the material axis and side 1-2 of the element coincide.) The plies are to be numbered serially starting with 1 at the bottom layer. The bottom layer

Layered Composite Element Property **PCOMP**

	is defined as the surface with the largest $-Z$ value in the element coordinate system. (Real; Default = 0.0)
SOUTi	Stress or strain output request. See Remarks 5 and 6. (Character: "YES" or "NO"; Default = "NO")

Remarks:

1. The default for MID2, ..., MIDn is the last defined MIDi. In the example above, MID1 is the default for MID2, MID3, and MID4. The same logic applies to Ti.

2. At least one of the four values (MIDi, Ti, THETAi, SOUTi) must be present for a ply to exist. The minimum number of plies is one.

3. The TREF specified on the material entries referenced by plies are not used. Instead TREF on the PCOMP entry is used for all plies of the element. If not specified, it defaults to "0.0."

   If the PCOMP references temperature dependent material properties, then the TREF given on the PCOMP will be used as the temperature to determine material properties. Note that MAT8 data entries do not support temperature dependent material properties.

   TEMPERATURE Case Control commands are ignored for deriving the equivalent PSHELL and MATi entries used to describe the composite element.

4. GE given on the PCOMP entry will be used for the element and the values supplied on material entries for individual plies are ignored. The user is responsible for supplying the equivalent damping value on the PCOMP entry. If PARAM,W4 is not specified GE is ignored in transient analysis. See the *MSC/NASTRAN Quick Reference Guide*, Section 6.

Layered Composite Element Property **PCOMP**

5. Stress and strain output for individual plies are available in all superelement static and normal modes analysis and requested by the STRESS and STRAIN Case Control commands.

6. If PARAM,NOCOMPS is set to -1, stress and strain output for individual plies will be suppressed and the homogeneous stress and strain output will be printed. See also Remark 9.

7. ELFORCE and ELSTRESS requests must be present for all elements for which ply stress or failure index output is desired.

8. A function of this entry is to derive equivalent internal PSHELL and MATi entries to describe the composite element. Any sorted echo request will also cause printout and/or punch of the derived PSHELL and MATi entries in User Information Message 4379 and/or the punch file. (See the *MSC/NASTRAN Reference Manual*, Chapter 15 for proper interpretation of the output from User Information Message 4379.) However, if these equivalent PSHELL and MAT2 entries are input, then stress or strain output for individual plies is not available and PARAM,NOCOMPS,-1 must be supplied.

9. The failure index for the boundary material is calculated as Failure Index $= max\left(\tau_{1z},\ \tau_{2z}\right)/SB$.

10. If the value specified for Z0 is not equal to -0.5 times the thickness of the element and PARAM,NOCOMPS,-1 is specified, then the homogeneous element stresses are incorrect, while lamina stresses and element forces and strains are correct. For correct homogeneous stresses, use ZOFFS on the corresponding connection entry.

Layered Composite Element Property **PCOMP**

11. An unsymmetric layup is not recommended in buckling analysis or the calculation of differential stiffness. Also, Z0 should not be used to specify an unsymmetric layup.
12. To obtain the damping coefficient GE, multiply the critical damping ratio C/C by 2.0.

Shell Element Orthotropic Material Property Definition **MAT8**

Defines the material property for an orthotropic material for isoparametric shell elements.

Format:

1	2	3	4	5	6	7	8	9	10
MAT8	MID	E1	E2	NU12	G12	G1Z	G2Z	RHO	
	A1	A2	TREF	Xt	Xc	Yt	Yc	S	
	GE	F12	STRN						

Example:

MAT8	171	30.+6	1.+6	0.3	2.+6	3.+6	1.5+6	0.056	
	28.−6	1.5−6	155.0	1.+4	1.5+4	2.+2	8.+2	1.+3	
	1.−4		1.0						

Field	Contents
MID	Material identification number. Referenced on a PSHELL or PCOMP entry only. (0 < Integer < 1000000)
E1	Modulus of elasticity in longitudinal direction, also defined as the fiber direction or 1-direction. (Real ≠ 0.0)
E2	Modulus of elasticity in lateral direction, also defined as the matrix direction or 2-direction. (Real ≠ 0.0)

Shell Element Orthotropic Material Property Definition **MAT8**

NU12	Poisson's ratio (ϵ_2/ϵ_1 for uniaxial loading in 1-direction). Note that $\nu_{21} = \epsilon_1/\epsilon_2$ for uniaxial loading in 2-direction is related to ν_{12}, E_1, and E_2 by the relation $\nu_{12} E_2 = \nu_{21} E_1$. (Real)
G12	In-plane shear modulus. (Real ≥ 0.0; Default = 0.0)
G1Z	Transverse shear modulus for shear in 1-Z plane. (Real > 0.0; Default implies infinite shear modulus.)
G2Z	Transverse shear modulus for shear in 2-Z plane. (Real > 0.0; Default implies infinite shear modulus.)
RHO	Mass density. (Real)
Ai	Thermal expansion coefficient in i-direction. (Real)
TREF	Reference temperature for the calculation of thermal loads, or a temperature-dependent thermal expansion coefficient. See Remarks 4. and 5. (Real or blank)
Xt, Xc	Allowable stresses or strains in tension and compression, respectively, in the longitudinal direction. Required if failure index is desired. See the FT field on the PCOMP entry. (Real > 0.0; Default value for Xc is Xt.)
Yt, Yc	Allowable stresses or strains in tension and compression, respectively, in the lateral direction. Required if failure index is desired. (Real > 0.0; Default value for Yc is Yt.)
S	Allowable stress or strain for in-plane shear. See the FT field on the PCOMP entry. (Real > 0.0)
GE	Structural damping coefficient. See Remarks 4. and 6. (Real)

Shell Element Orthotropic Material Property Definition — MAT8

F12	Interaction term in the tensor polynomial theory of Tsai-Wu. Required if failure index by Tsai-Wu theory is desired and if value of F12 is different from 0.0. See the FT field on the PCOMP entry. (Real)
STRN	For the maximum strain theory only (see STRN in PCOMP entry). Indicates whether Xt, Xc, Yt, Yc, and S are stress or strain allowables. [Real = 1.0 for strain allowables; blank (Default) for stress allowables.]

Remarks:

1. If G1Z and G2Z values are specified as zero or blank, then transverse shear flexibility calculations will not be performed, which is equivalent to zero shear flexibility (i.e., infinite shear stiffness).
2. An approximate value for G1Z and G2Z is the in-plane shear modulus G12. If test data are not available to accurately determine G1Z and G2Z for the material and transverse shear calculations are deemed essential; the value of G12 may be supplied for G1Z and G2Z. In SOL 106, linear and nonlinear elastic material properties in the residual structure will be updated as prescribed in the TEMPERATURE Case Control command.
3. Xt, Yt, and S are required for composite element failure calculations when requested in the FT field of the PCOMP entry. Xc and Yc are also used but not required.
4. TREF and GE are ignored if this entry is referenced by a PCOMP entry.
5. TREF is used in two different ways:

a. In nonlinear static analysis (SOL 106), TREF is used only for the calculation of a temperature-dependent thermal expansion coefficient. The reference temperature for the calculation of thermal loads is obtained from the TEMPERATURE(INITIAL) set selection. See Figure 1 in Remark 10. in the MAT1 description.

b. In all SOLs except 106, TREF is used only as the reference temperature for the calculation of thermal loads. TEMPERATURE(INITIAL) may be used for this purpose, but TREF must then be blank.

6. If PARAM,W4 is not specified, GE is ignored in transient analysis. See the *MSC/NASTRAN Quick Reference Guide*, Section 6.

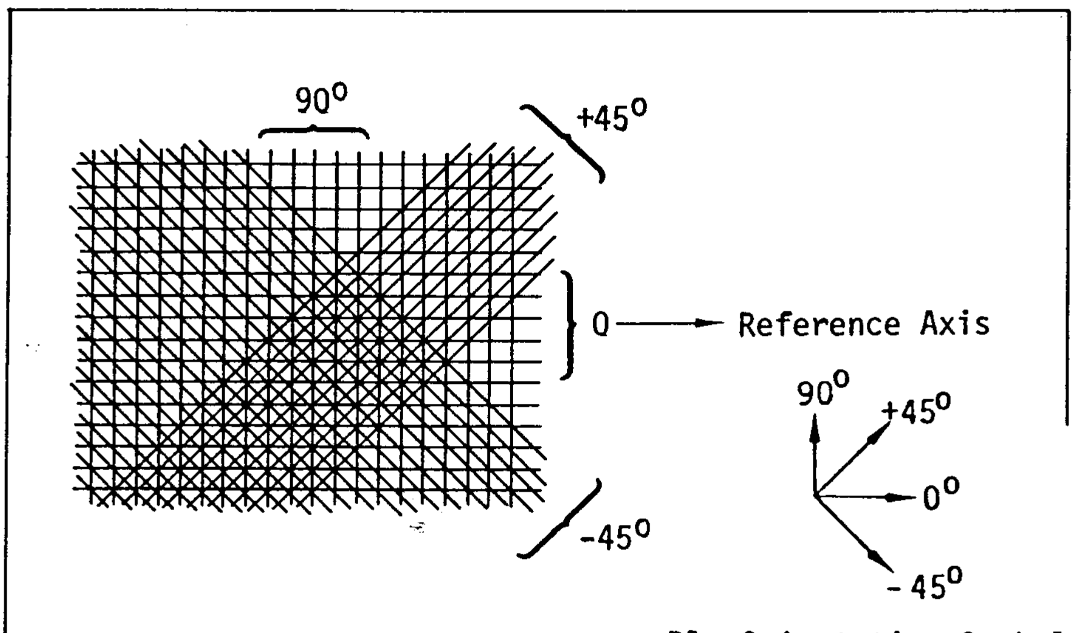
Ply Orientation Code for Design Studies and Layouts R

One of the chief advantages in using advanced composite structure is the feasibility of orienting material fibers to match load requirements. This requires that the designer show (in addition to material) how the structure is constructed:

- o Fiber Direction (termed "ply orientation")
- o Ply Stackup

A "shorthand" code for ply fiber directions and ply stackup may be used on layouts or in studies, where actual part fabrication is not involved:

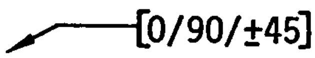
- o Each ply is shown by a number representing the direction of fibers, in degrees, with respect to a reference axis. 0^{o} fibers of either tape or fabric are normally aligned with the largest axial load.



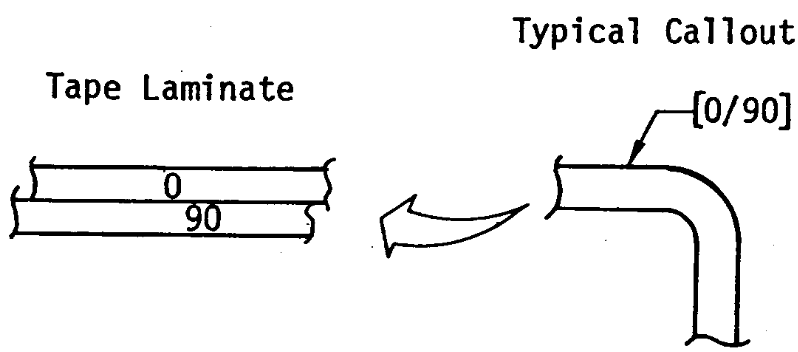
Ply Orientation Symbol
(See Section 6)

ont'd)

B. The plies are listed in sequence, set off by brackets, starting from the side indicated by the code arrow.



C. Adjacent plies with different angles of orientation are separated by a slash.

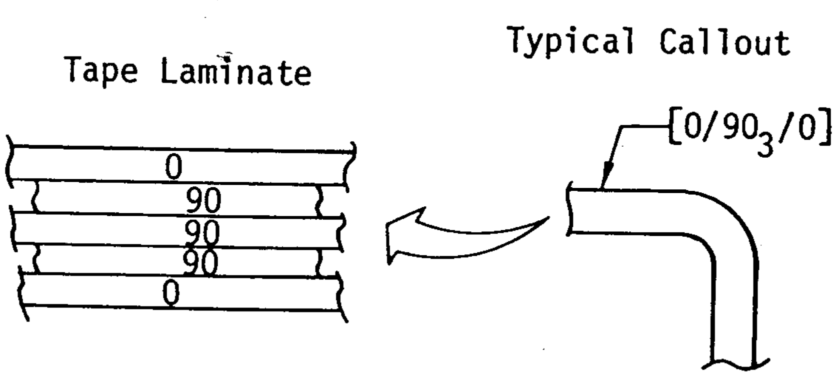


D. Callouts of fabric plies are differentiated from tape plies by parentheses.

Example: [(±45)/(0,90)] Code for fabric plies

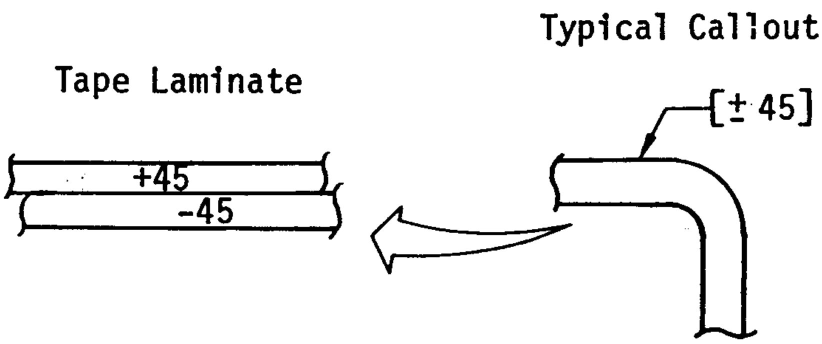
[0/(±45)/90] Code for tape and fabric plies.

E. Adjacent plies of the same angle of orientation are shown by a numerical subscript.

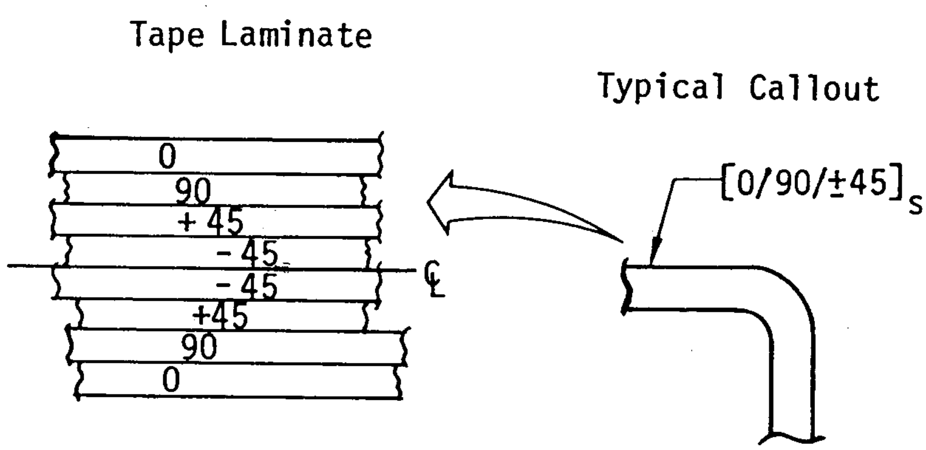


(Cont'd)

F. For tape only: when ± is used, two adjacent plies are indicated, with the top symbol being the first of the two.



G. Symmetric laminates with an even number of plies are listed in sequence, starting at one face and stopping at the mid-point of the material (the plane of symmetry). A subscript "S" following the bracket indicates that only one-half of the code is shown, with the other half being a mirror-image.

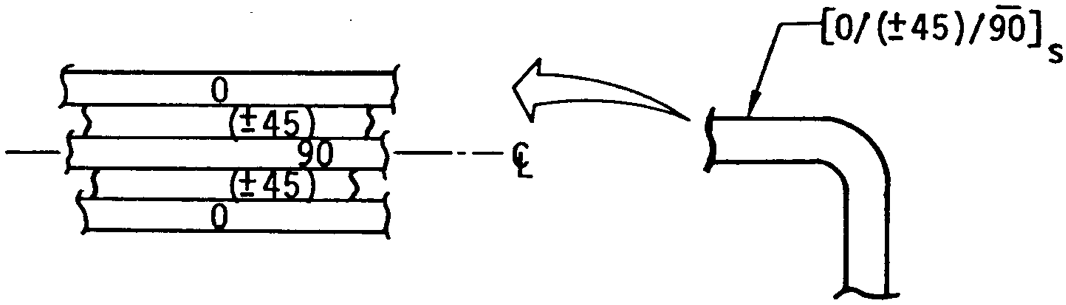


d)

1. Symmetric laminates with an odd number of plies have the center one overlined to indicate this condition. Starting with this ply, the rest of the code would be a mirror-image to that part shown.

Typical Callout

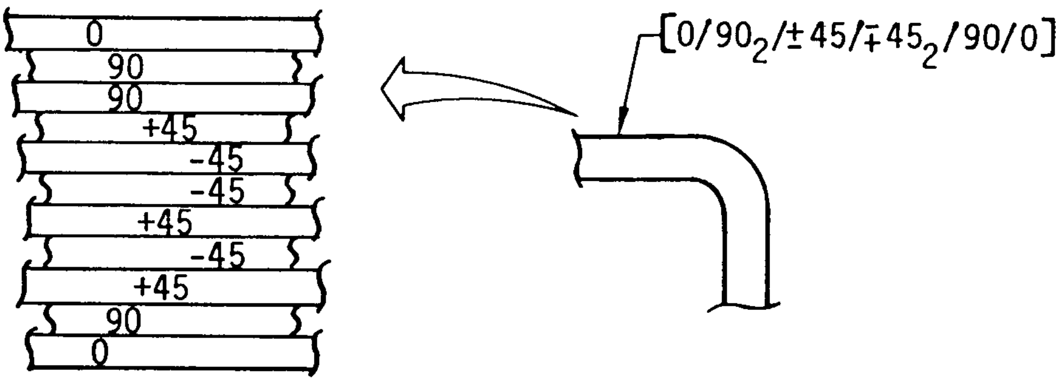
Tape and Fabric Laminate



. A subscript "T" has been used in various industry and Air Force documents to show that the total laminate is indicated. This is not recommended for Boeing use.

. Typical callout examples -

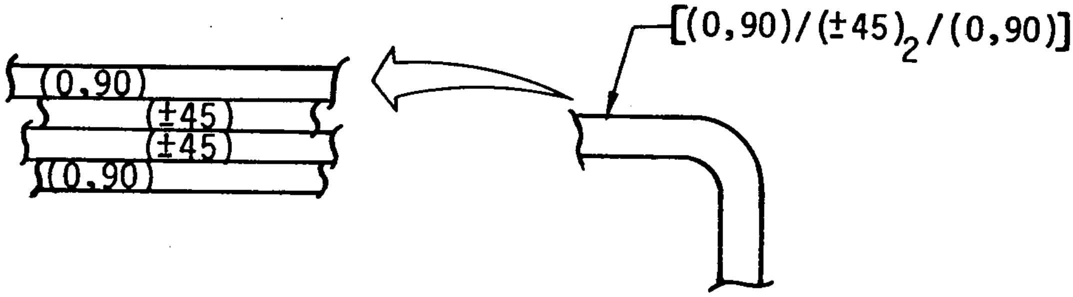
Tape Laminate



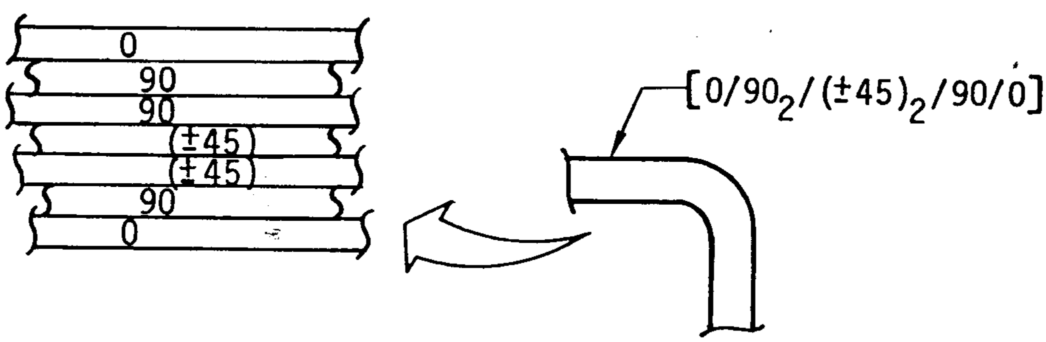
ont'd)

J. (Cont'd)

Fabric Laminate

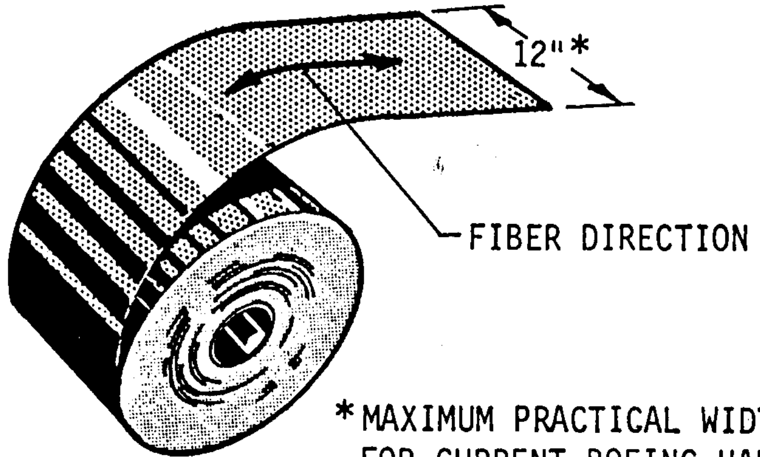


Tape and Fabric Laminate



MATERIAL SELECTION - Class 1 vs. Class 2

CLASS 1 : TAPE

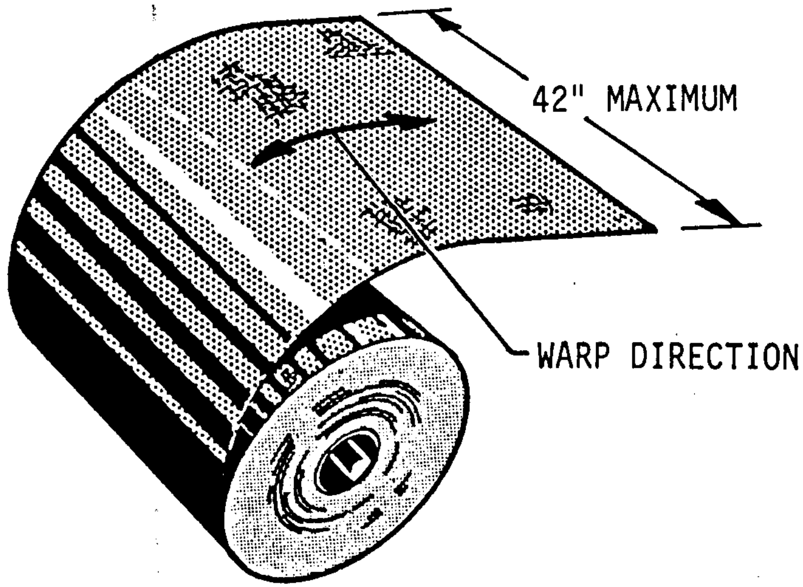


*MAXIMUM PRACTICAL WIDTH FOR CURRENT BOEING HAND LAYUP OPERATIONS.

DESIGN CONSIDERATIONS

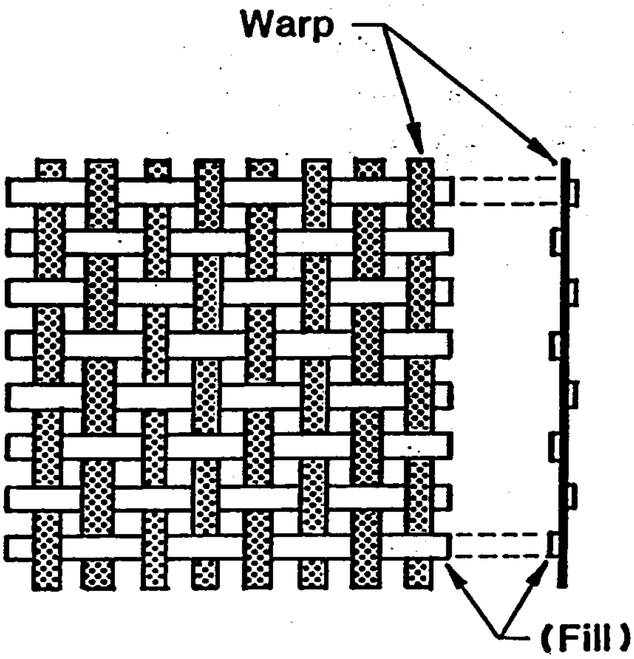
- ADVANTAGES OF TAPE
  - CAN BE TAILORED MORE EASILY TO MATCH LOADS
  - AERODYNAMICALLY ACCEPTABLE SURFACE. (REQUIRES LESS SURFACE TREATMENT THAN FABRIC)*
  - NO SPLICE OVERLAPS PARALLEL TO FIBERS
  - UP TO 10% HIGHER ALLOWABLE STRENGTH AND STIFFNESS
  - LOWER RAW MATERIAL COST
- RECOMMENDED FOR USE WHERE ADVANTAGES JUSTIFY USE AND WHERE CONTOURS PERMIT

CLASS 2 : WOVEN FABRIC

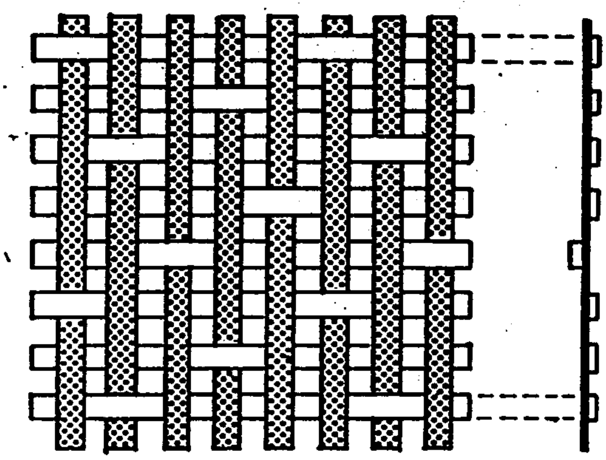


- ADVANTAGES OF FABRIC
  - LESS MATERIAL HANDLING DAMAGE
  - EASIER FORMING ON CONTOURS AND CORNERS
  - MORE RESISTANT TO SURFACE BREAKOUT AND DELAMINATION
- AN ACCEPTABLE AERODYNAMIC SURFACE IS HARDER TO OBTAIN ON FABRIC THAN ON TAPE *

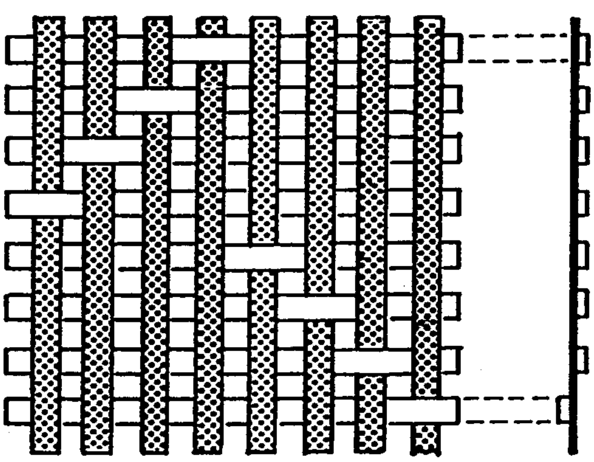
Fabric Weave Patterns



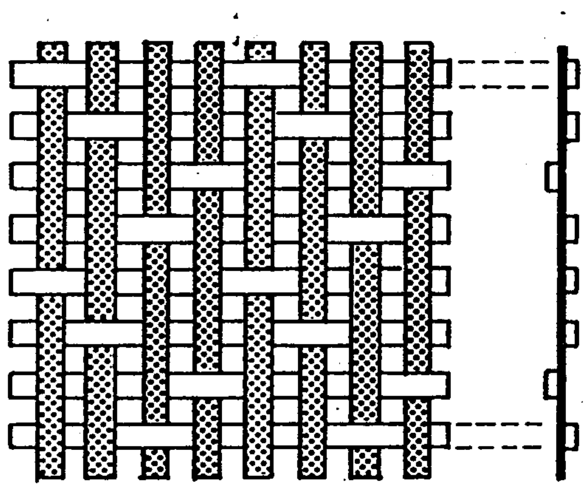
Plain Weave
(3K-70-P OR 3K-70-PW)



5 Harness Satin Weave
(1K-50-5H)



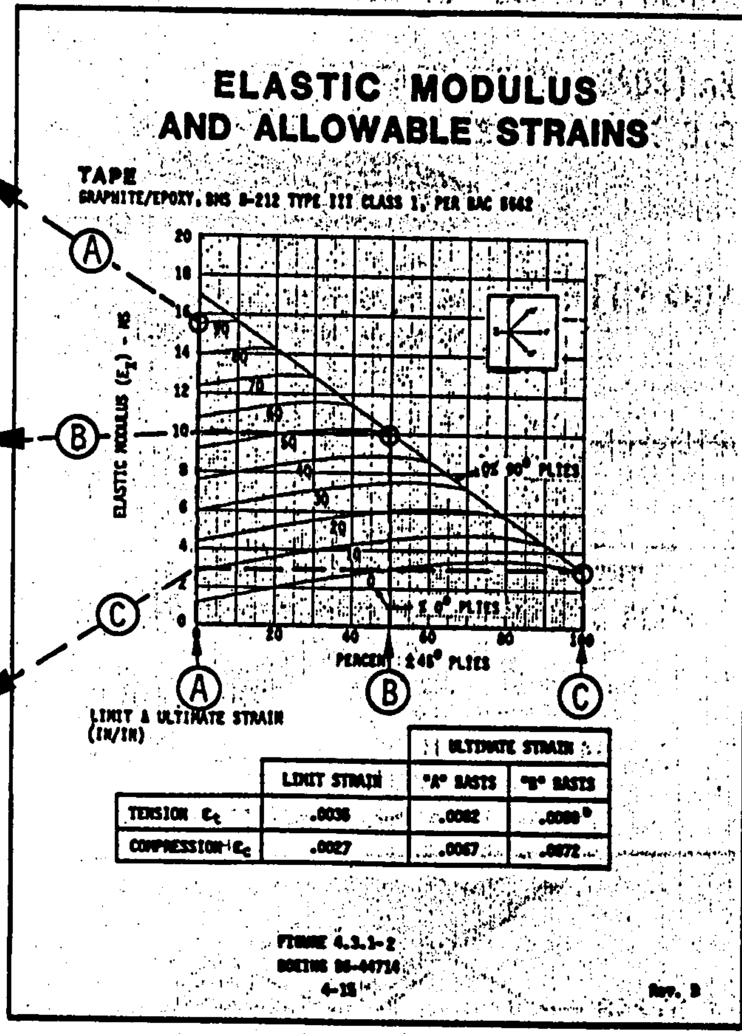
8 Harness Satin Weave
(3K-135-8H)



Crowfoot Satin Weave
(3K-70-CSW)

INFLUENCE OF FIBER ORIENTION ON STIFFNESS AND STRENGTH

ROUGHLY EQUIVALENT MATERIAL	STRENGTH (F_{tu}) $F_{tu} = E_x \varepsilon_{tu}$	STIFFNESS (E_x)
TITANIUM F_{tu} = 135 KSI E_x = 16 MSI	$F_{tu} = (15.5\times10^6)(.0090)^*$ = 139.5 Ksi	E_x = 15.5 MSI
ALUMINUM F_{tu} = 70 KSI E_x = 10.3 MSI	$F_{tu} = (10\times10^6)(.0090)^*$ = 90 Ksi	E_x = 10 MSI
FIBER GLASS F_{tu} = 22.5 KSI E_x = 3 MSI	$F_{tu} = (3\times10^6)(.0090)^*$ = 27 Ksi	E_x = 3 MSI



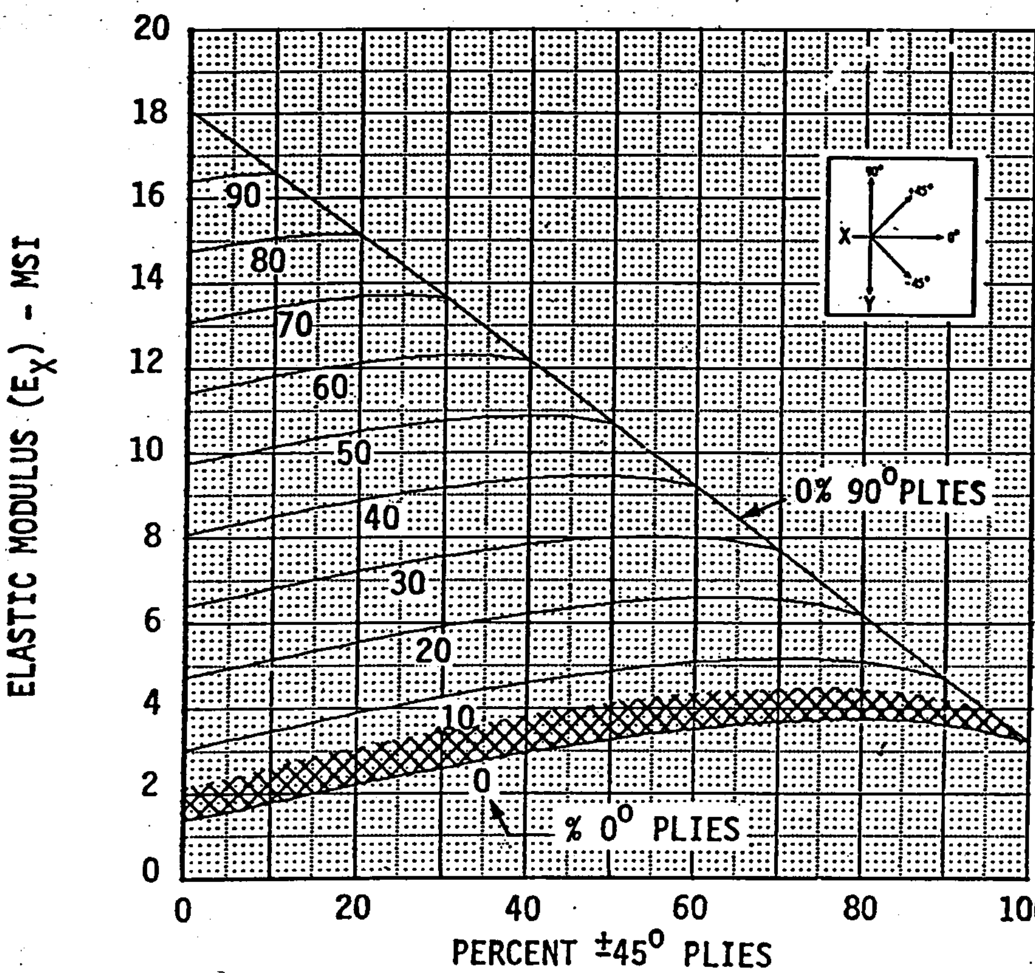
Note that values are for the "x" direction; properties for the "y" direction are different.

MATERIAL A - 90% - 0° PLIES, 10% - 90° PLIES
MATERIAL B - 50% - 0° PLIES, 50% - 45° PLIES
MATERIAL C - 100% - 45° PLIES

ELASTIC MODULUS AND ALLOWABLE STRAINS

TAPE

GRAPHITE/EPOXY, BMS 8-212 TYPE II CLASS 1, PER BAC 5562



CAUTION : PROPERTIES IN THE CROSS - HATCHED REGION MAY BE ADVERSELY AFFECTED BY TEMPERATURE AND HUMIDITY.

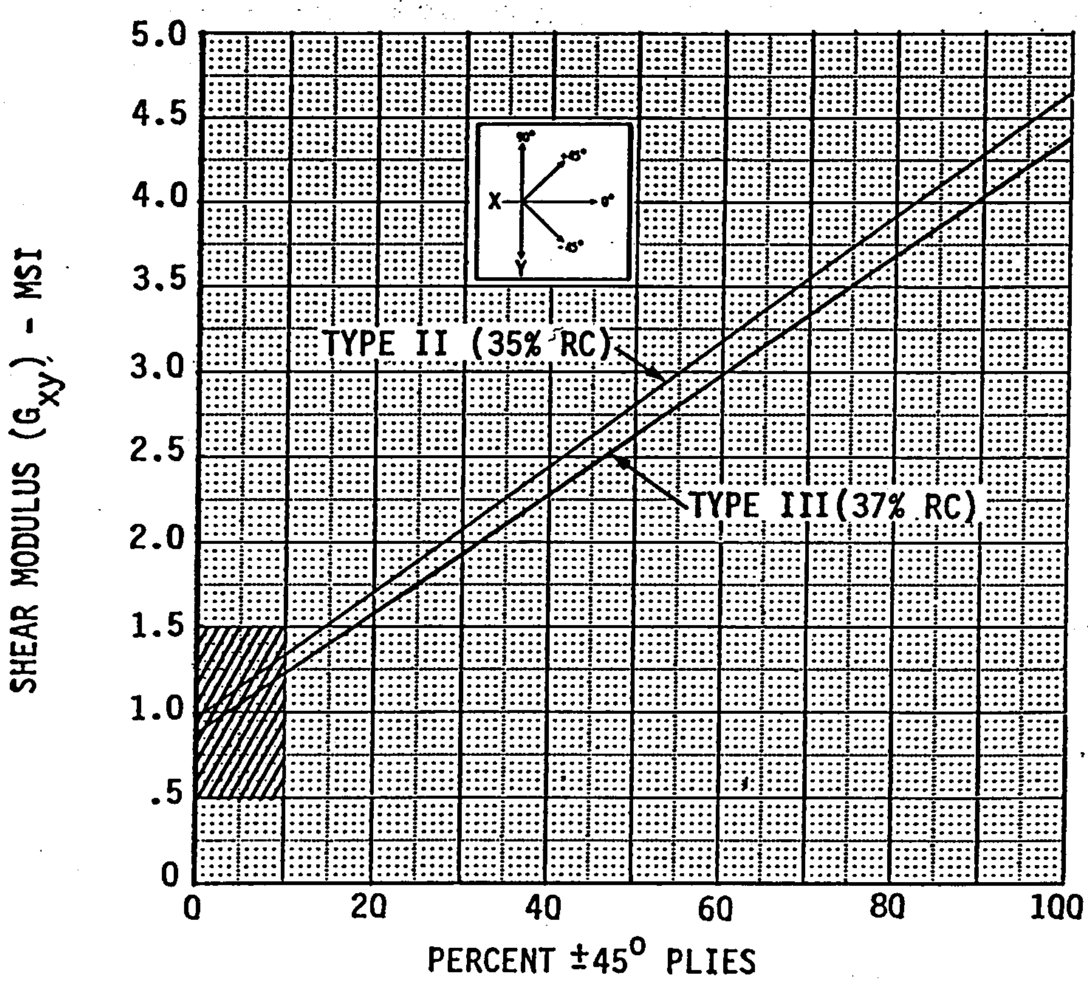
LIMIT & ULTIMATE STRAIN (IN/IN)

	LIMIT STRAIN	ULTIMATE STRAIN "A" BASIS	ULTIMATE STRAIN "B" BASIS
TENSION ε_t	.0035	.0082	.0090
COMPRESSION ε_c	.0027	.0067	.0072

IN-PLANE SHEAR MODULUS AND ALLOWABLE STRAINS

TAPE

GRAPHITE/EPOXY, BMS 8-212 CLASS 1, PER BAC 5562



RC-NOMINAL RESIN CONTENT PER BMS 8-212

LIMIT & ULTIMATE STRAIN (IN/IN)

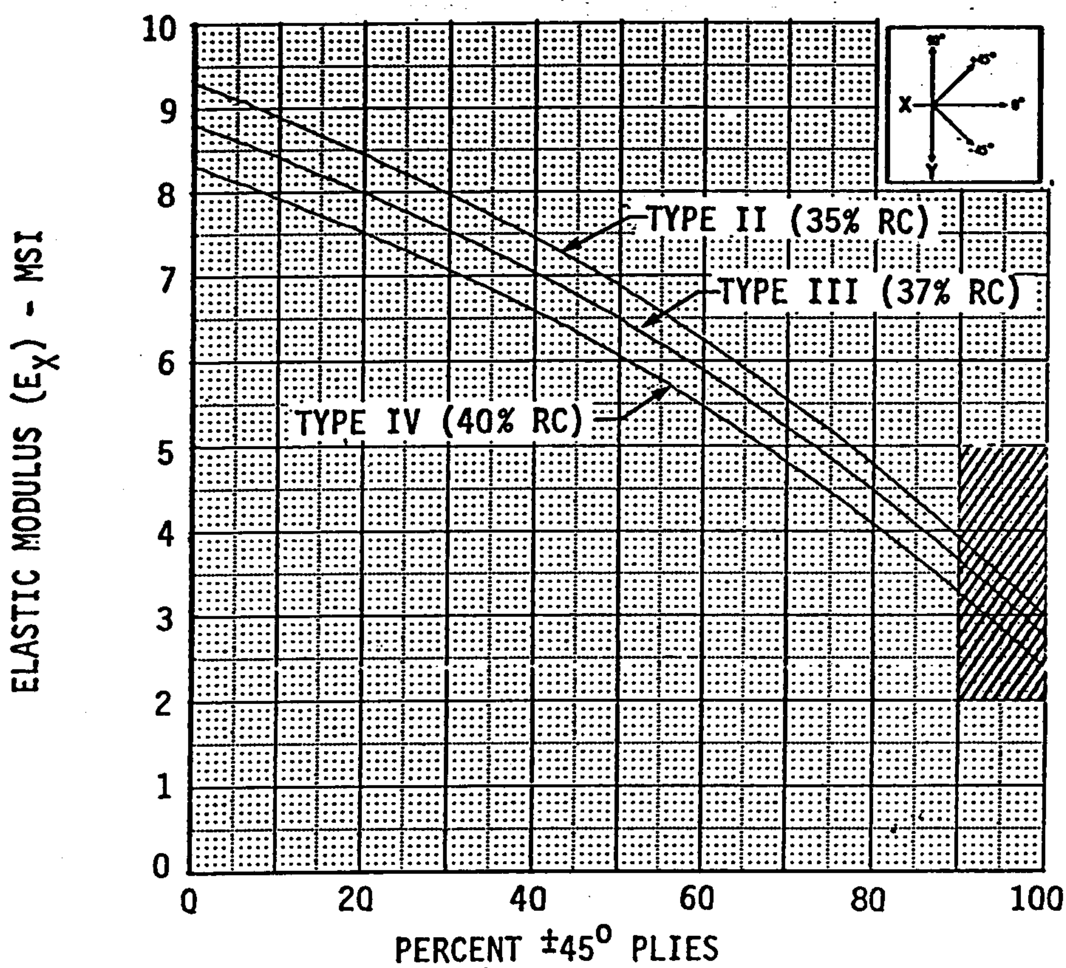
	LIMIT STRAIN	ULTIMATE STRAIN "A" BASIS	ULTIMATE STRAIN "B" BASIS
SHEAR ε_{xy}	.0053	.0133	.0144

CAUTION : PROPERTIES IN THE CROSS - HATCHED REGION MAY BE ADVERSELY AFFECTED BY TEMPERATURE AND HUMIDITY.

ELASTIC MODULUS AND ALLOWABLE STRAINS

FABRIC

GRAPHITE/EPOXY, BMS 8-212 CLASS 2, PER BAC 5562



CAUTION: PROPERTIES IN THE CROSS-HATCHED REGION MAY BE ADVERSELY AFFECTED BY TEMPERATURE AND HUMIDITY.

RC-NOM RESIN CONTENT PER BMS 8-212

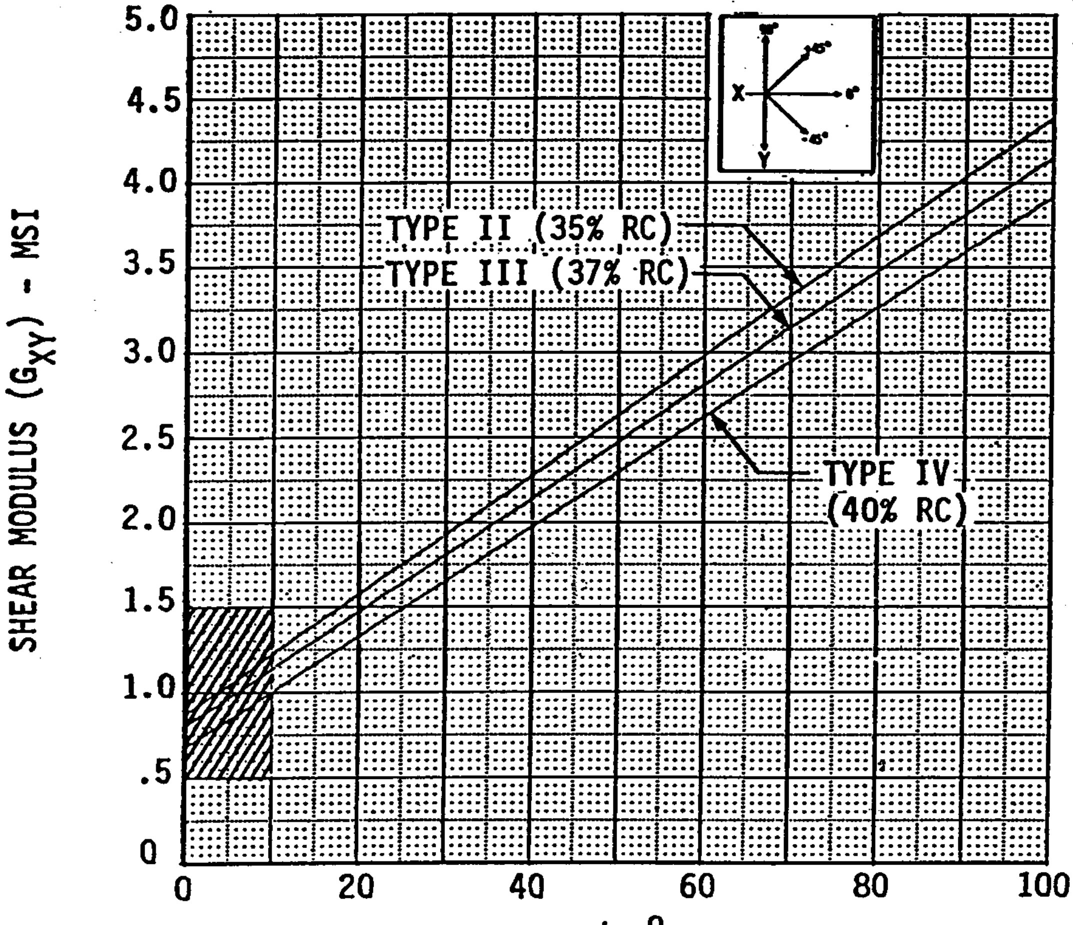
LIMIT & ULTIMATE STRAIN (IN/IN)	LIMIT STRAIN	ULTIMATE STRAIN	
		"A" BASIS	"B" BASIS
LAMINATE TENSION ε_t	.0035	.0059	.0064
LAMINATE COMPRESSION ε_c	.0027	.0054	.0059
SANDWICH TENSION ε_t	①	.0043	.0046
SANDWICH COMPRESSION ε_c	①	.0035	.0038

① ULTIMATE STRAIN ÷ 1.5

IN-PLANE SHEAR MODULUS AND ALLOWABLE STRAINS

FABRIC

GRAPHITE/EPOXY, BMS 8-212 CLASS 2, PER BAC 5562



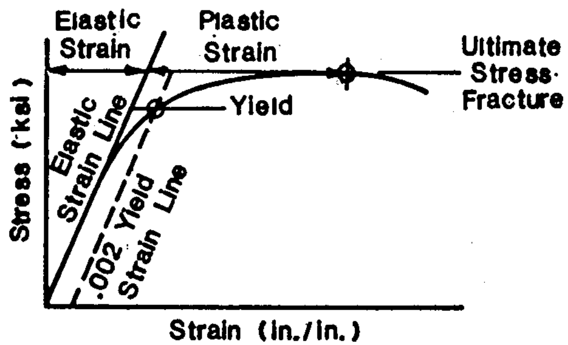
CAUTION : PROPERTIES IN THE CROSS - HATCHED REGION MAY BE ADVERSELY AFFECTED BY TEMPERATURE AND HUMIDITY.

RC= NOMINAL RESIN CONTENT PER BMS 8-212

LIMIT & ULTIMATE STRAIN (IN/IN)		ULTIMATE STRAIN	
	LIMIT STRAIN	"A" BASIS	"B" BASIS
LAMINATE SHEAR ε_{xy}	.0053	.0108	.0117
SANDWICH SHEAR ε_{xy}	①	.0070	.0076

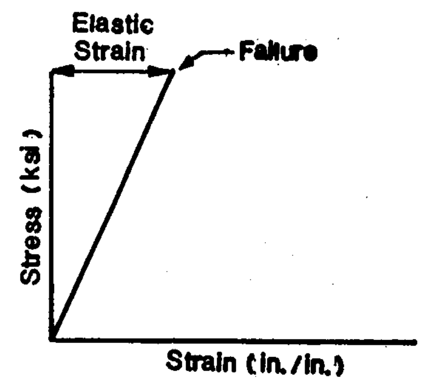
① ULTIMATE STRAIN ÷ 1.5

TYPICAL STRESS - STRAIN CURVE

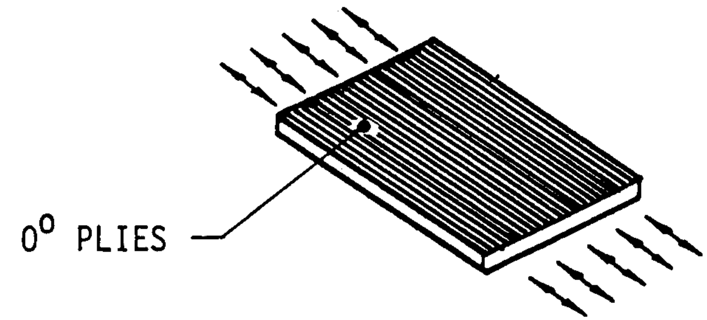
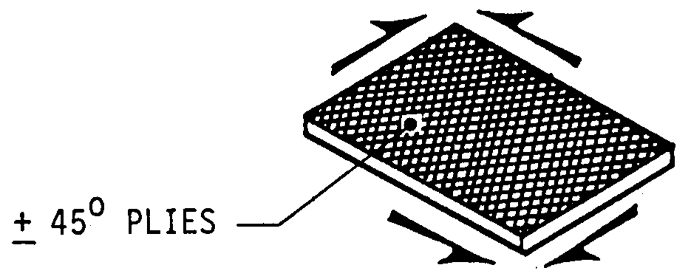
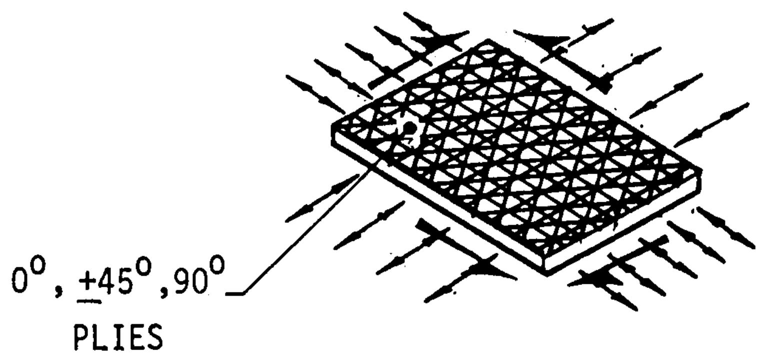


- ELASTIC PROPERTIES UP TO YIELD STRESS
- DUCTILE ULTIMATE FAILURE
- ALLOWABLE LIMIT STRESS IS DETERMINED BY YIELD POINT

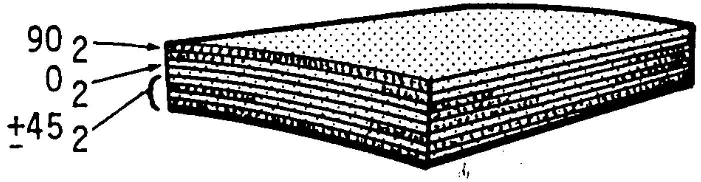
TYPICAL STRESS - STRAIN CURVE

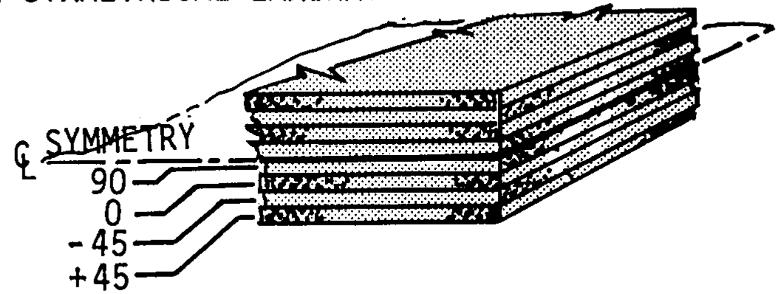


- ELASTIC PROPERTIES UP TO ULTIMATE STRESS
- BRITTLE OR "SUDDEN" ULTIMATE FAILURE
- ALLOWABLE LIMIT STRESS IS DETERMINED BY MICROCRACKING OF THE RESIN.

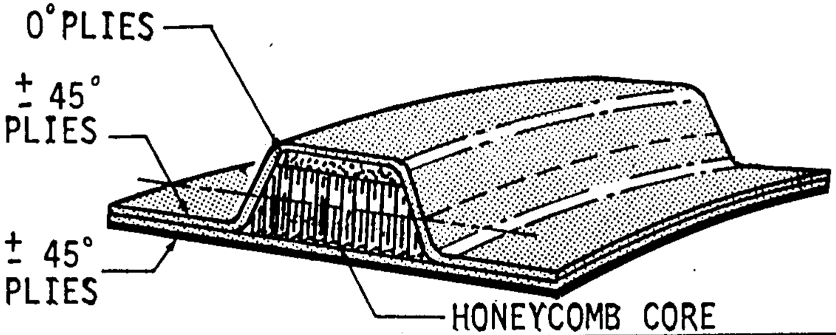
LOADING CONDITION	SOLUTION - ORIENT PLIES TO LOAD
AXIAL (TENSION OR COMPRESSION)	
SHEAR	
BIAXIAL + SHEAR	

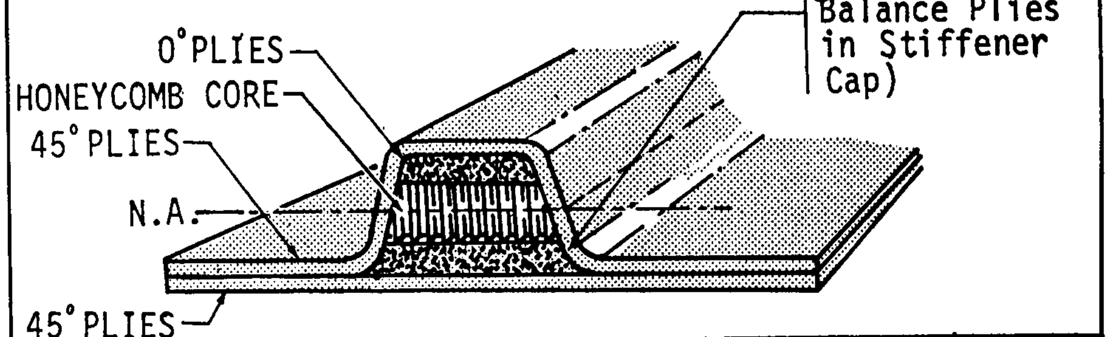
PROBLEM	RECOMMENDATIONS
PROBLEM AN UNBALANCED LAYUP CAN WARP DURING CURE CYCLE DUE TO THE DIFFERENCE IN THERMAL EXPANSION BETWEEN THE PLIES	USE SYMMETRICAL LAMINATE LAYUP



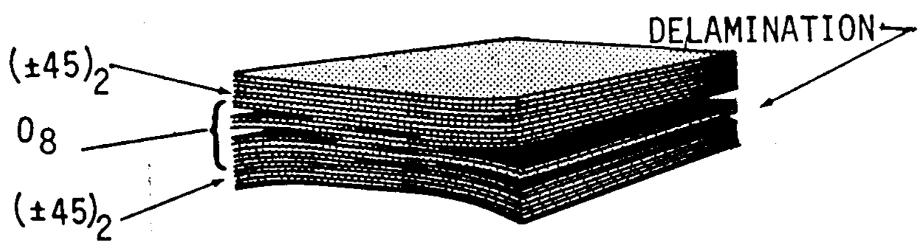


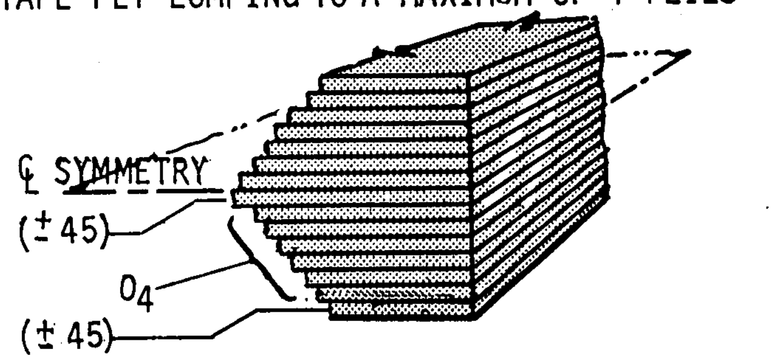
PROBLEM	RECOMMENDATIONS
PROBLEM AN UNBALANCED ASSEMBLY CAN WARP DURING CURE CYCLE.	BALANCE LAYUP IN UNSYMMETRICAL AREA AROUND THE NEUTRAL AXIS (N.A.)(SECT. 5.27)

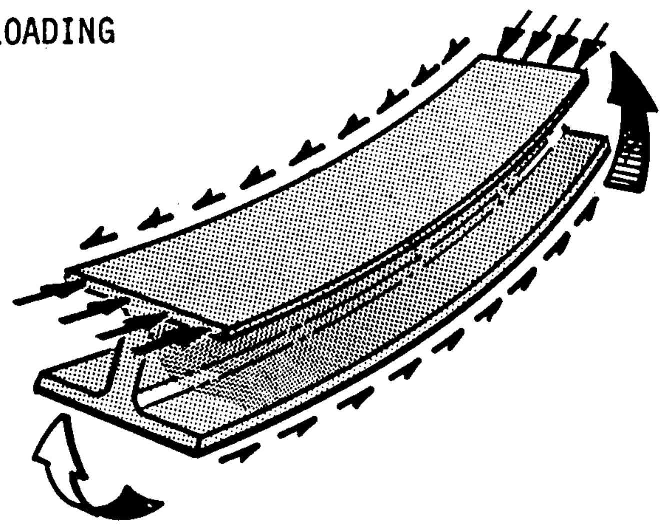
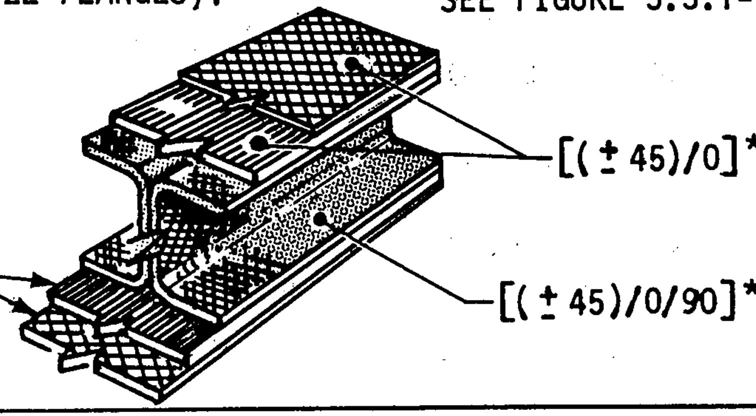
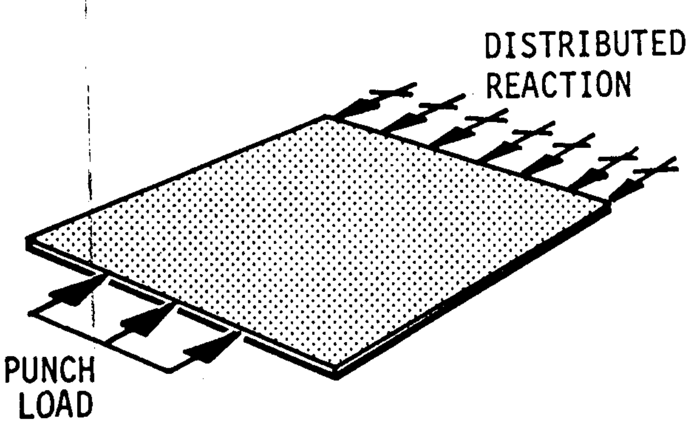
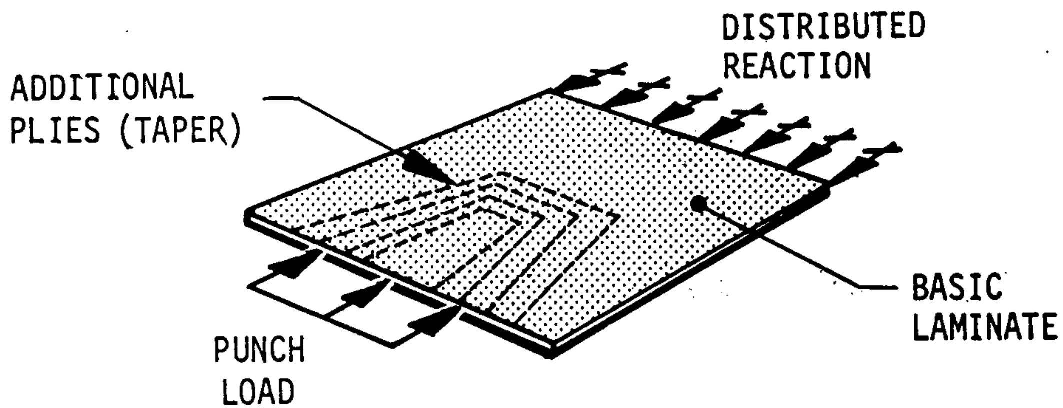




PROBLEM	RECOMMENDATIONS
PROBLEM LUMPING TOO MANY TAPE PLIES OF SAME ORIENTATION CAN RESULT IN DELAMINATION.	• INTERSPERSE PLY ORIENTATIONS • LIMIT TAPE PLY LUMPING TO A MAXIMUM OF 4 PLIES

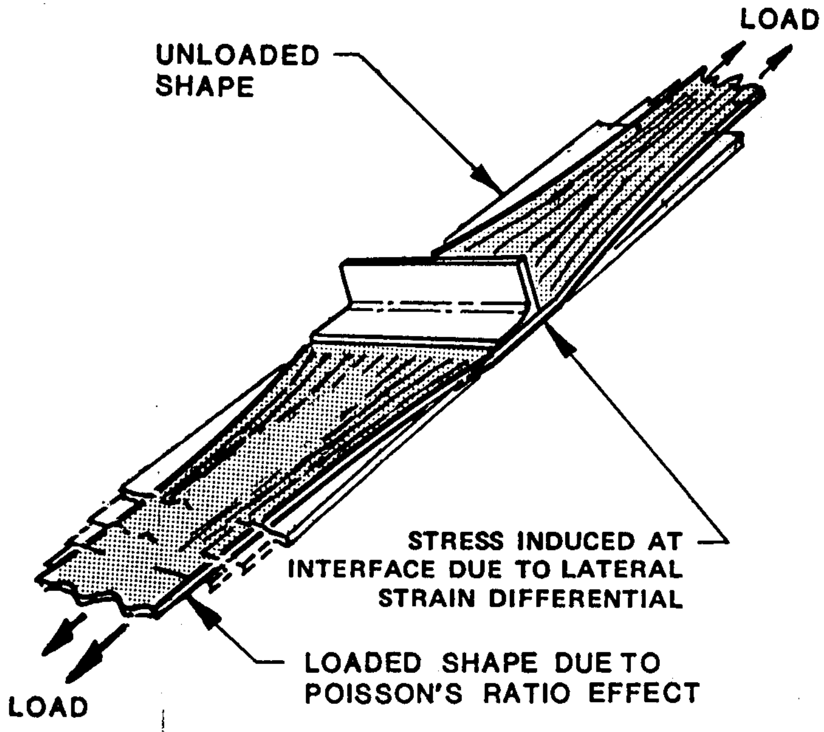




DESIGN CONDITION	SOLUTION
• BEAM SUBJECT TO BENDING, SHEAR AND END LOADING 	• INCLUDE 0^o PLIES IN CAP TO EFFICIENTLY SUSTAIN BENDING INDUCED AXIAL LOADS • INCLUDE $\pm 45^o$ PLIES IN WEB (TO SUSTAIN SHEAR LOADS) AND IN CAPS (TO STABILIZE FLANGES). SEE FIGURE 3.3.1-8 *Intersperse ply orientations (Fig.3.2.3-4) 
• PANELS SUBJECT TO LOCAL PUNCH LOAD  	• WITHIN THE BASIC LAMINATE, DISTRIBUTE SUFFICIENT ADDITIONAL 0^o PLIES TO REACT LOCAL PUNCH LOAD, PLUS $\pm 45^o$ PLIES TO REDISTRIBUTE THIS LOAD 

PROBLEM

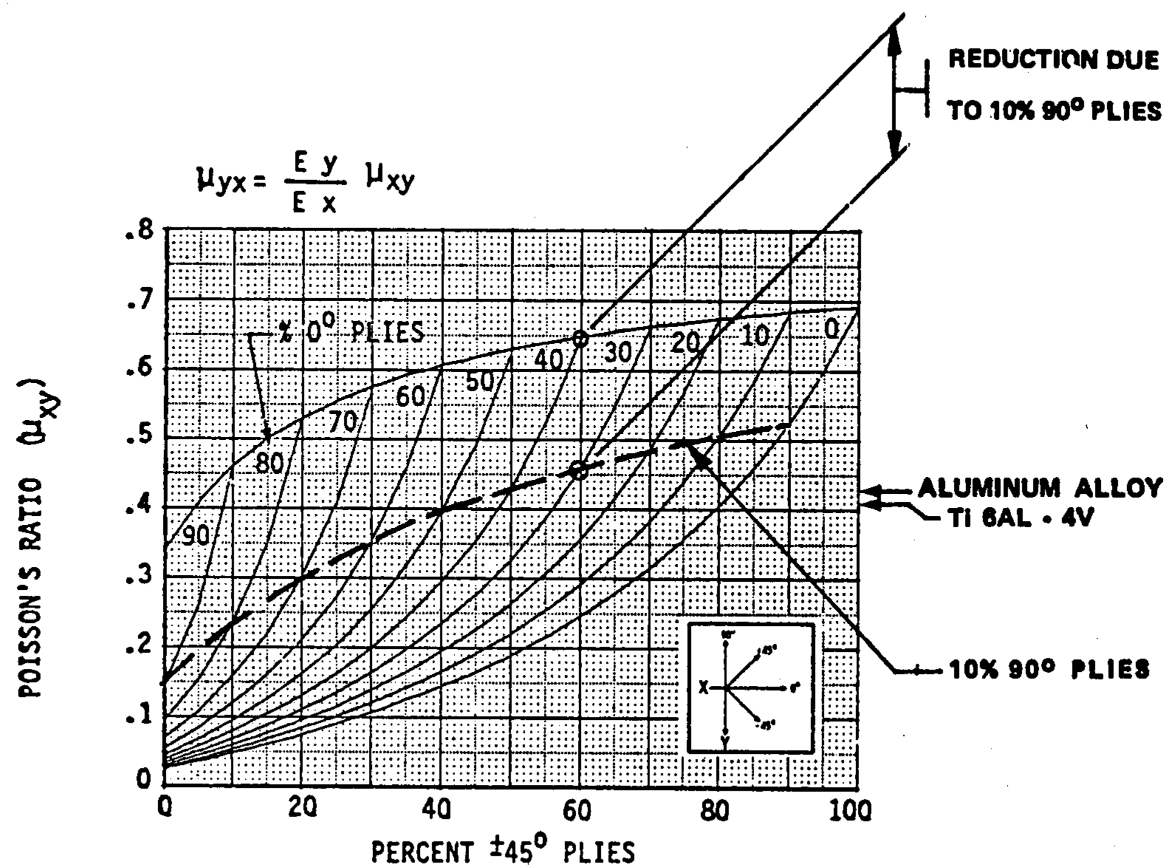
LATERALLY INDUCED LOADS



GR/EP PANELS MAY INDUCE LARGER LOADS TO ADJOINING STRUCTURE THAN METAL PANELS BECAUSE OF A LARGER POISSON'S RATIO.

RECOMMENDATIONS

- ANALYZE EFFECTS OF INDUCED LOADS ON ASSY
- CONSIDER USE OF 90° PLIES TO REDUCE POISSON'S RATIO



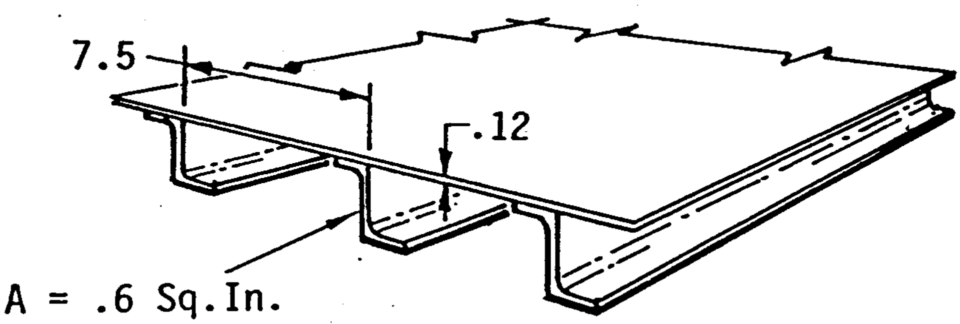
REFERENCE FIGURE 4.3.5-1

Stiffness Matching

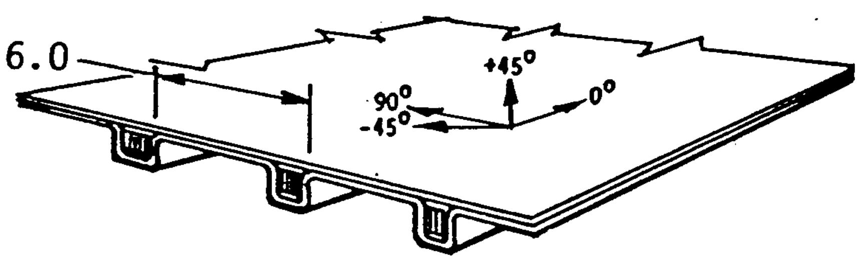
Many applications of composite structure are to replace certain existing aluminum components. When this occurs, it is often necessary to match the stiffness, both bending and torsional, of the existing aluminum structure. This is especially true for aerodynamic surfaces. Bending stiffness depends upon EI and torsional stiffness depends upon JG. See DM86B1, Section 12, for a discussion of torsion.

Example Problem

In this example, a portion of an aerodynamic surface is being replaced by composite structure. Since the basic aerodynamic cross section of the surface is unchanged, I is proportional to changes in area while J is proportional to changes in the skin thickness.



EXISTING ALUMINUM SECTION



PROPOSED COMPOSITE SECTION

.3

Calculate the existing aluminum panel stiffness (7075-T6)

t_{skin} = .12 in

$A_{stiffener}$ = .6 sq.in. at 7.5 inch spacing

Convert the stiffener area to equivalent skin thickness

$\frac{.6}{7.5} = .08$ in

Equivalent skin thickness = .12 + .08 = .20

E = 10.3×10^6 psi (DM8481, Figure 83.1-3-1)

G = 3.9×10^6 psi (DM8481, Figure 83.1-3-1)

The composite panel stiffener spacing is 6.0 inches; therefore, the area of the existing aluminum panel that must be matched is 6.0 x .20 = 1.2 sq.in.

AE = 1.2 $(10.3 \times 10^6) = 12.36 \times 10^6$ pounds

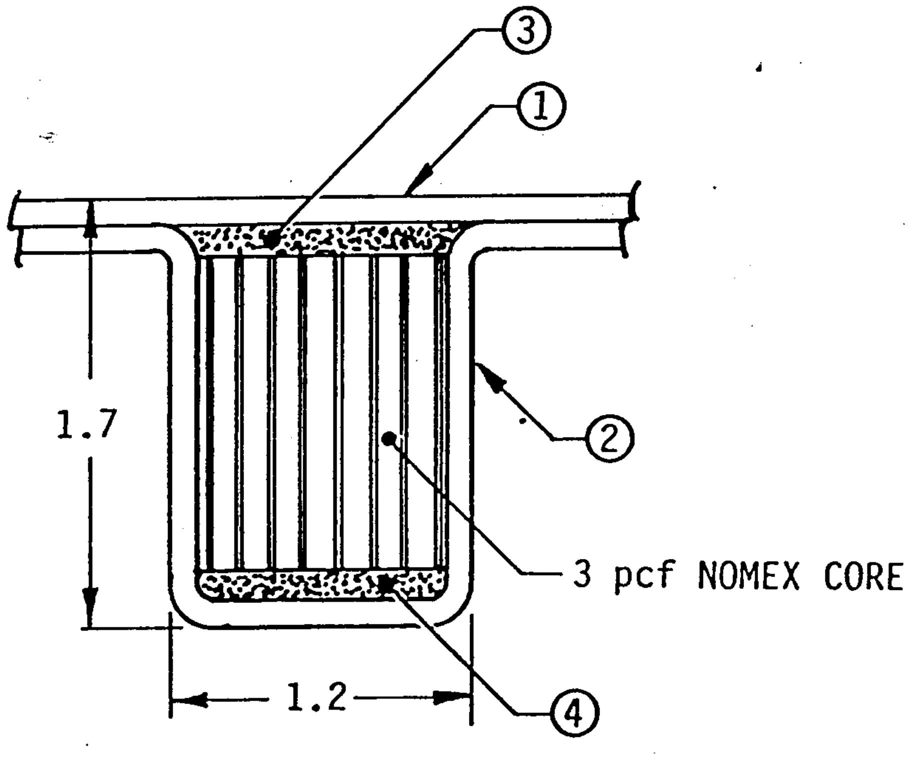
Since the shear is reacted by the skin only, find AG for the skin.

AG = 6 (.12) $(3.9 \times 10^6) = 2.81 \times 10^6$ pounds

.4

Composite panel configuration

The sizing of the composite panel to match the existing panel stiffness is an iterative process. It is unlikely that the designer will be able to determine the acceptable thickness and ply orientation during the first trial solution. This example shows only the final calculations.



3

Find the required AE and AG for the new composite panel assuming the following configuration for elements 2, 3 and 4:

Element	Number of plies 0°	45°	90°	Total	t	Percentage 0°	45°	90°	E* (psi)	** G(psi)
2	2	14	2	18	.1062	11	78	11	4.9×10^6	3.6×10^6
3	16	4	2	22	.1298	73	18	9	13.3×10^6	1.5×10^6
4	20	4	2	26	.1534	77	15	8	13.9×10^6	1.4×10^6

Elements	Layup
2	$[\pm 45_2/0/90/\pm 45/\pm \overline{45}]_s$
3	$[0_4/\pm 45/0_4/90]_s$
4	$[0_5/\pm 45/0_5/90]_s$

* See Figure 4.3.1-2

** See Figure 4.3.1-4

AE (Skin and Stiffener)

Element		A	E	AE
2	.1062 (6+1.7+1.7) =	.9983	4.9×10^6	4.89×10^6
3	.1298 (1.2-.1062-.1062) =	.1282	13.3×10^6	1.71×10^6
4	.1534 (1.2-.1062-.1062) =	.1515	13.9×10^6	2.11×10^6
				8.71×10^6

AG (Skin Only)

Element		A	G	AG
2	.1062 [6-(1.2-.1062-.1062)]	.532	3.6×10^6	1.92×10^6
3	.1298 [1.2-.1062-.1062]	.128	1.5×10^6	$.19 \times 10^6$
4				—
				2.11×10^6

Required $AE_{skin} = AE_{alum} - AE_{2+3+4} = 12.36 - 8.71 = 3.65 \times 10^6$

Required $AG_{skin} = AG_{alum} - AG_{2+3} = 2.81 - 2.11 = .7 \times 10^6$

1.

Find the thickness of the skin

Assume that the skin is made from 11 (total) plies:

$t_{skin} = 11(.0059) = .0649$ inches

$A_{skin} = 6\ (.0649) = .3894$ sq. in.

Required $E_{skin} = \frac{3.65 \times 10^6}{.3894} = 9.37 \times 10^6$

Required $G_{skin} = \frac{.70 \times 10^6}{.3894} = 1.80 \times 10^6$

Since G depends on the % of 45^o plies only, from Figure 4.3.1-4:

For G = 1.80, % of $\pm 45^o$ plies = 26% ,which yields 3 plies.

However, for a balanced layup, use 4 plies of $\pm 45^o$ for 4/11 = 36%

From Figure 4.3.1-2 ,for $E = 9.33 \times 10^6$, & % of ± 45% plies = 36%;

% of 0^o plies = 44% ⟶ 5 plies

Make the skin as follows:

Ply quantity				%								
0^o	45^o	90^o	Total	0^o	45^o	90^o	t	A	E*	AE*	G*	AG*
5	4	2	11	45	36	19	.0649	.3894	9.6	3.74	2.15	.79

* times 10^6

Layup: $[0, 90, \pm 45, \overline{0_3}]_s$

.4

Check if JG matches for the aluminum and composite structure

Since $J = \frac{4A^2}{\sum b/t}$ and A (the enclosed area of aerodynamic surface) is unchanged, find $\frac{G}{\sum b/t}$

where b is the length of the line enclosing area A, and t is the skin thickness (see DM 86B1 Section 12.1.3)

For the composite panel, G of elements 2 and 3 is not equal to G of the skin, therefore, find an equivalent thickness of elements 2 and 3. (Since element 4 is remote from the skin it is not considered.)

$$t_{el_2} = \frac{G_2 t_2}{G_{skin}} = \frac{3.6(.1062)}{2.15} = 1.78 \text{ in.} \qquad t_{el_3} = \frac{1.5(.1298)}{2.15} = .091$$

$$\frac{G}{\sum b/t}(\text{composite}) = \frac{2.15 \times 10^6}{\left(\frac{1.0}{.091+.0649}\right)+\left(\frac{5.0}{.178+.0649}\right)} = .080 \times 10^6$$

$$\frac{G}{\sum b/t}(\text{aluminum}) = \frac{3.9 \times 10^6}{6.0/.12} = .078 \times 10^6$$

} CHECKS WITHIN 5%