An Introduction to Composite FE Analysis

July 23rd, 2009

Agenda

An Introduction to Composite FE Analysis

July 23rd, 2009
8am PDT (Seattle) / 11am EDT (New York) / 4pm BST (London)

- Welcome & Introduction (Overview of NAFEMS Activities)
  - Mr. Matthew Ladzinski, *NAFEMS North America*
- An Introduction to Composite FE Analysis
  - Mr. Tony Abbey, *FETraining*
- Q&A Session
  - Panel
- Closing

Ladzinski Abbey

THE INTERNATIONAL ASSOCIATION
FOR THE ENGINEERING ANALYSIS
COMMUNITY

An Overview of NAFEMS Activities

Matthew Ladzinski
NAFEMS
NAFEMS North America

Planned Activities

- Webinars
  - New topic each month!
  - Recent webinars:
    - Composite FE Analysis
    - 10 Ways to Increase Your Professional Value in the Engineering Industry
    - Dynamic FE Analysis
    - Modal Analysis in Virtual Prototyping and Product Validation
    - Pathways to Future CAE Technologies and their Role in Ambient Intelligent Environments
    - Computational Structural Acoustics: Technology, Trends and Challenges
    - FAM: Advances in Research and Industrial Application of Experimental Mechanics
    - CCOPPS: Power Generation: Engineering Challenges of a Low Carbon Future
    - Practical CFD Analysis
    - Complexity Management
    - CCOPPS: Creep Loading of Pressurized Components – Phenomena and Evaluation
    - Multiphysics Simulation using Implicit Sequential Coupling
    - CCOPPS: Fatigue of Welded Pressure Vessels
    - Applied Element Method as a Practical Tool for Progressive Collapse Analysis of Structures
    - A Common Sense Approach to Stress Analysis and Finite Element Modeling
    - The Interfacing of FEA with Pressure Vessel Design Codes (CCOPPS Project)
    - Multiphysics Simulation using Directly Coupled-Field Element Technology
    - Methods and Technology for the Analysis of Composite Materials
    - Simulation Process Management
    - Simulation-supported Decision Making (Stochastics)
    - Simulation Driven Design (SDD) Findings

To register for upcoming webinars, or to view a past webinar, please visit: www.nafems.org/events/webinars

e-LEARNING

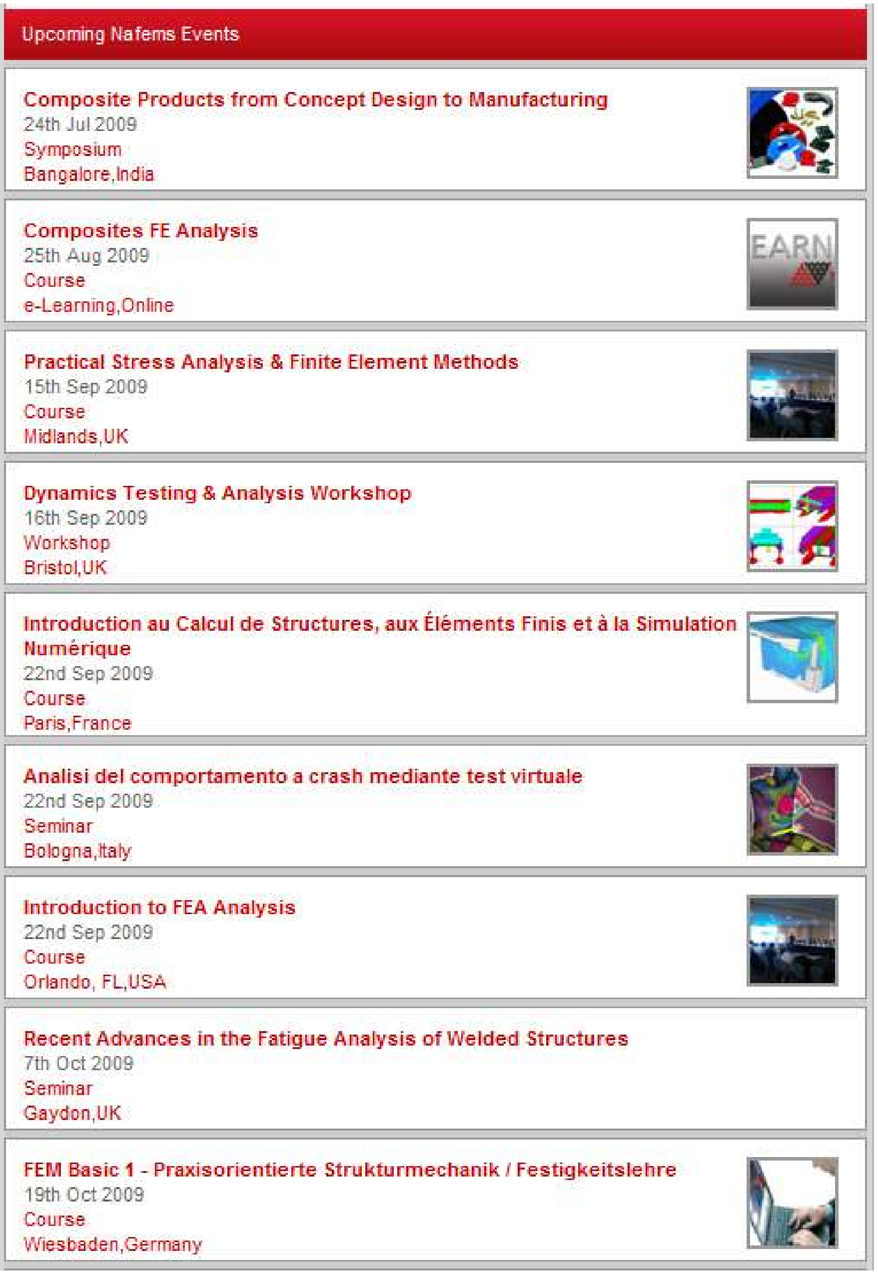
NAFEMS

- Established in 2009
- Next courses:
  - Dynamic FE Analysis – July 14th, 2009 (six-week course)
  - Composite FE Analysis – August 25th, 2009 (four-week course)
- Proposed course offerings:
  - Non-linear – Fall 2009 (four-week course)
  - Stochastics – Fall 2009
  - Verification & Validation – Fall/Winter 2009
- For more information, visit: www.nafems.org/e-learning

Currently running!

NAFEMS Events

- Multiple opportunities to attend conferences, seminars/workshops and training courses
- Let us know if you would like to schedule an on-site training course
- For more information, please visit: www.nafems.org



Upcoming Nafems Events

Composite Products from Concept Design to Manufacturing
24th Jul 2009
Symposium
Bangalore,India

Composites FE Analysis
25th Aug 2009
Course
e-Learning,Online

Practical Stress Analysis & Finite Element Methods
15th Sep 2009
Course
Midlands,UK

Dynamics Testing & Analysis Workshop
16th Sep 2009
Workshop
Bristol,UK

Introduction au Calcul de Structures, aux Éléments Finis et à la Simulation Numérique
22nd Sep 2009
Course
Paris,France

Analisi del comportamento a crash mediante test virtuale
22nd Sep 2009
Seminar
Bologna,Italy

Introduction to FEA Analysis
22nd Sep 2009
Course
Orlando, FL,USA

Recent Advances in the Fatigue Analysis of Welded Structures
7th Oct 2009
Seminar
Gaydon,UK

FEM Basic 1 - Praxisorientierte Strukturmechanik / Festigkeitslehre
19th Oct 2009
Course
Wiesbaden,Germany

Attend Events for Free

NAFEMS Members are entitled to attend a number of seminars and workshops free of charge each year as part of their membership, as well as a library of free publications on joining. Members also receive significant discounts on NAFEMS courses, conferences and publications.

If you are a NAFEMS member, please login above to take advantage of these free places and discounted prices. If you are not a member, click here to read more about the benefits of getting involved.

Welcome and Agenda

Overview of the NAFEMS e-Learning Course

Introduction to Composites FE Analysis

Q and A

Composite FE Analysis

August 25th - September 15th, 2009

Four-Week Training Course

Members Price: £143 | €165| $235
Non-Members Price: £228| €264| $375
Order Ref:el-003
Event Type:Course
Location: E-Learning,Online
Date: August 25, 2009

www.nafems.org/events/nafems/2009/el003/

Overview of Composites e-Learning Class

Composites Analysis

Many designs now use composite structures or components, taking advantage of:

- **increased structural strength and stiffness to weight ratios**
- **simpler manufacturing process**
- **more innovative design capability**

The nature of the composite used can range from:

- **cheap and freely available glass fiber reinforced systems**
  to
- **exotic and specifically tailored carbon, Kevlar or even metal/matrix systems**

Many forms of manufacturing process available.

Overview of Composites e-Learning Class

Composites Analysis

The challenge for the designer and analyst is to determine the resulting stiffness and strength of the design.

Faced with the complexity of real world structural systems the analyst has to make decisions on the FEA analysis :

- **the type of idealization**
- **level of detail required**
- **definition of failure**

The design variations available with a composite material are immense; ply thickness, orientation and property can all be varied to tune the structural response.
A rational approach is needed to predict the strength and stiffness and how to use the FEA data to help design and verify the structure.

Overview of Composites e-Learning Class

Composites Analysis

Your design may include thick composite sections with large numbers of plies, there may be regions of significant ply drop off.

Tee joints may be loaded in tension. In these cases the through thickness effects become very important for strength prediction.

The shape of the structure may imply changes in draping angle or layup thickness and it may be important to model this accurately.

Overview of Composites e-Learning Class

Composites Analysis

There are a wide range of failure theories, together with potentially large amounts of stress or strain data from a multi ply layup.

Due to the nature of the composite the stress components can include many more terms than a conventional metallic material for example.

Whatever the nature of the challenge, this objective of this course is to break down the composite analysis process into clearly defined steps, give an overview of the physics involved and show how to successfully implement practical solutions using Finite Element Analysis.

Overview of Dynamics e-Learning Class

Why an e-learning class?

In the current climate travel and training budgets are tight. To help you still meet your training needs the following e-learning course has been developed to complement the live class.

The e-learning course runs over a four week period with a single two hour session per week.

Bulletin Boards and Email are used to keep in contact between sessions, mentoring homework and allowing interchange between students.

E-learning classes are ideal for companies with a group of engineers requiring training. E-learning classes can be provided to suit your needs and timescale. Contact us to discuss your requirements.

We hope that small companies or individuals can now take part in the training experience.

Introductory Composites FE Analysis Webinar

Agenda

1. What are composites?

Review of different forms and manufacturing processes.

2. How do composites vary from metallic structures?

Overview of material properties and the ABD matrix terms. Hints on practical design methods.

3. How do I set up a composite FEA?

Overview of typical processes. Keeping track of plies, mold lines etc.

Introductory Composites FE Analysis Webinar

Agenda

4. How good is my FEA idealization?

The importance of fiber orientation, draping and thickness effects.

5. How do I know whether the composite has failed?

Basic First Ply failure theories

6. How do I organize my results, where do I start looking?

Failure indices, Strength ratios.

Introductory Composites FE Analysis Webinar

Agenda

7. Through thickness and edge effects such as delamination

Usage of solid and thick shell elements.

8. Advanced failure methods

Progressive ply failure, cohesive elements, fracture mechanics methods (VCCT)

Introductory Composites FE Analysis Webinar

Agenda

1. What are composites?

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2. How do composites vary from metallic structures?

Overview of material properties and the ABD matrix terms. Hints on practical design methods.

3. How do I set up a composite FEA?

Overview of typical processes. Keeping track of plies, mold lines etc.

1. What are composites?

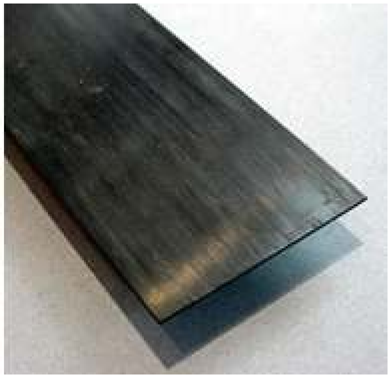
Consider material types:

ISOTROPIC - the same material properties in all directions, steel is a typical example.

Easy to measure properties

ANISOTROPIC - different material properties in all directions, a chunk of volcanic rock is an example.

Tough to measure or predict properties

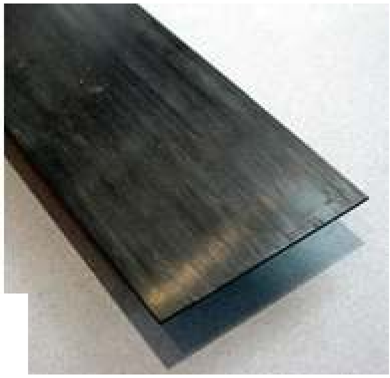


ORTHOTROPIC – special case of anisotropic , clear material directionality in 3 directions –represents a carbon fiber/resin system ,for example, where the along axis, transverse axis and through thickness axis are different.

Measurable and predictable properties – some challenges

1. What are composites?

2D ORTHOTROPIC, A further simplification is where we ignore the through thickness stress. This is the usual starting point for what we call Classical Laminate Theory, the foundation of most FE solutions.



(a) Plane strain - Thick bodies

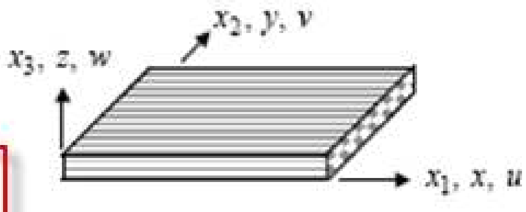
$$\varepsilon_z = \gamma_{xz} = \gamma_{yz} = 0$$

$$\therefore \quad \tau_{xz} = \tau_{yz} = 0$$

(b) Plane Stress - Thin bodies

$$\sigma_z = \tau_{xz} = \tau_{yz} = 0$$

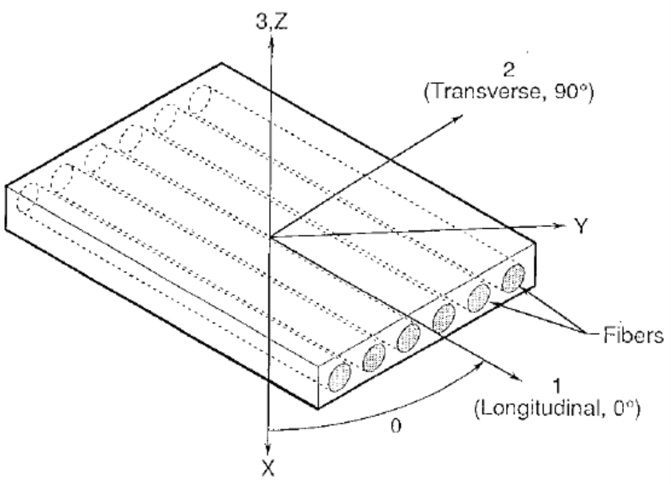
$$\therefore \varepsilon_z = \gamma_{xz} = \gamma_{yz} = 0$$



Direct through stress and shears assumed 0

*** Note the limitations implied here – we will revisit this**

1. What are composites?

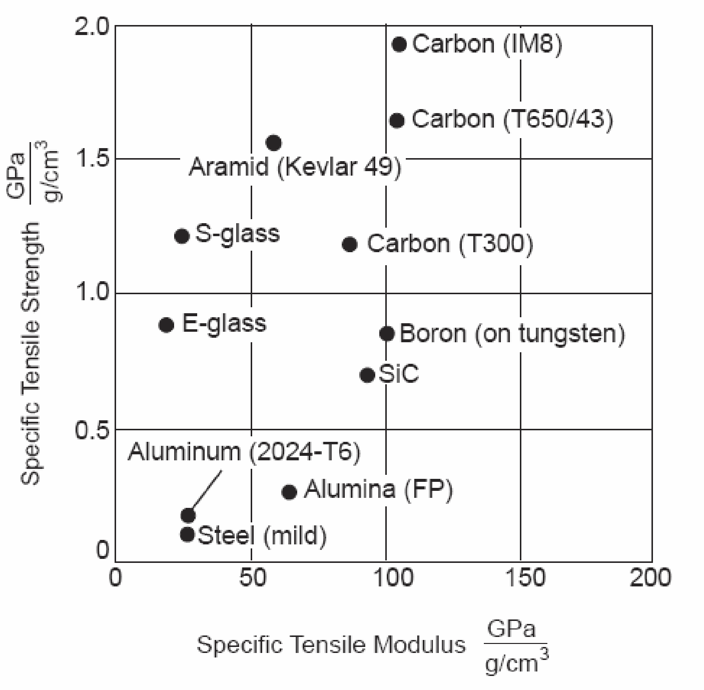


The composite is a system which consists of fibers in a resin or similar medium (usually called the matrix)

The important strength and stiffness characteristics are provided by the high strength fibers

It is important to consider both the fibers and the matrix in the material stiffness and strength considerations

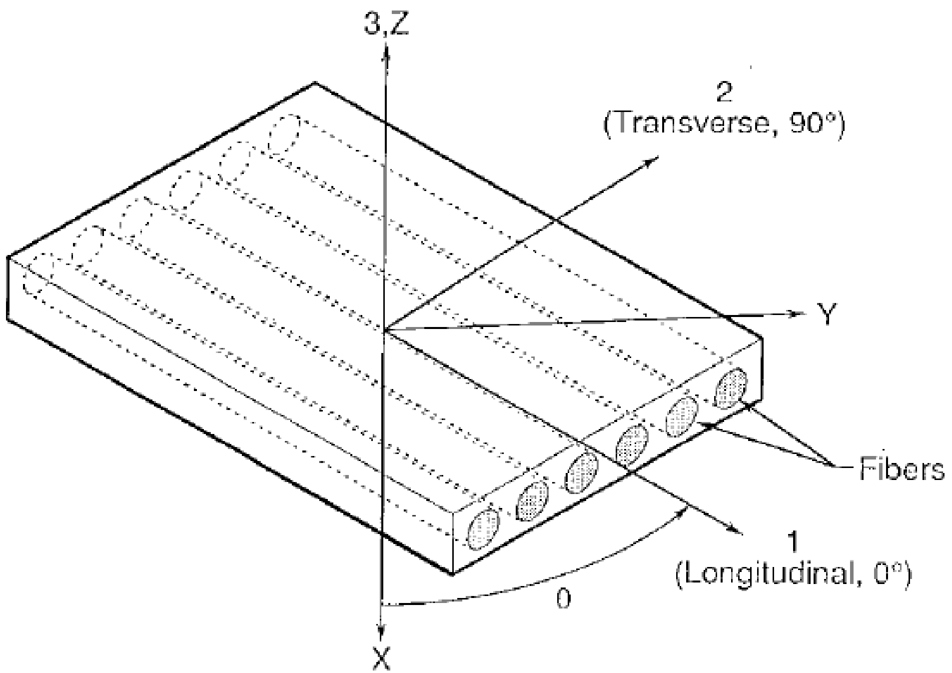
1. What are composites?

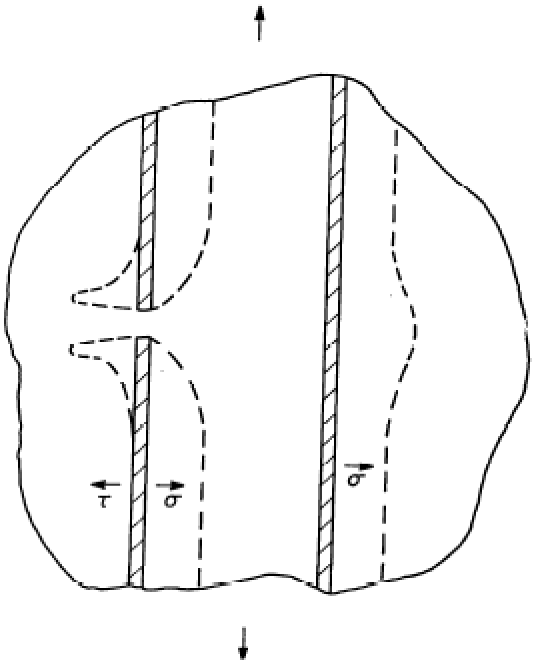


The fibers in isolation in a perfect test setup can have incredibly strong and stiff properties.

However they cannot be used in this form, they need a binding matrix

1. What are composites?



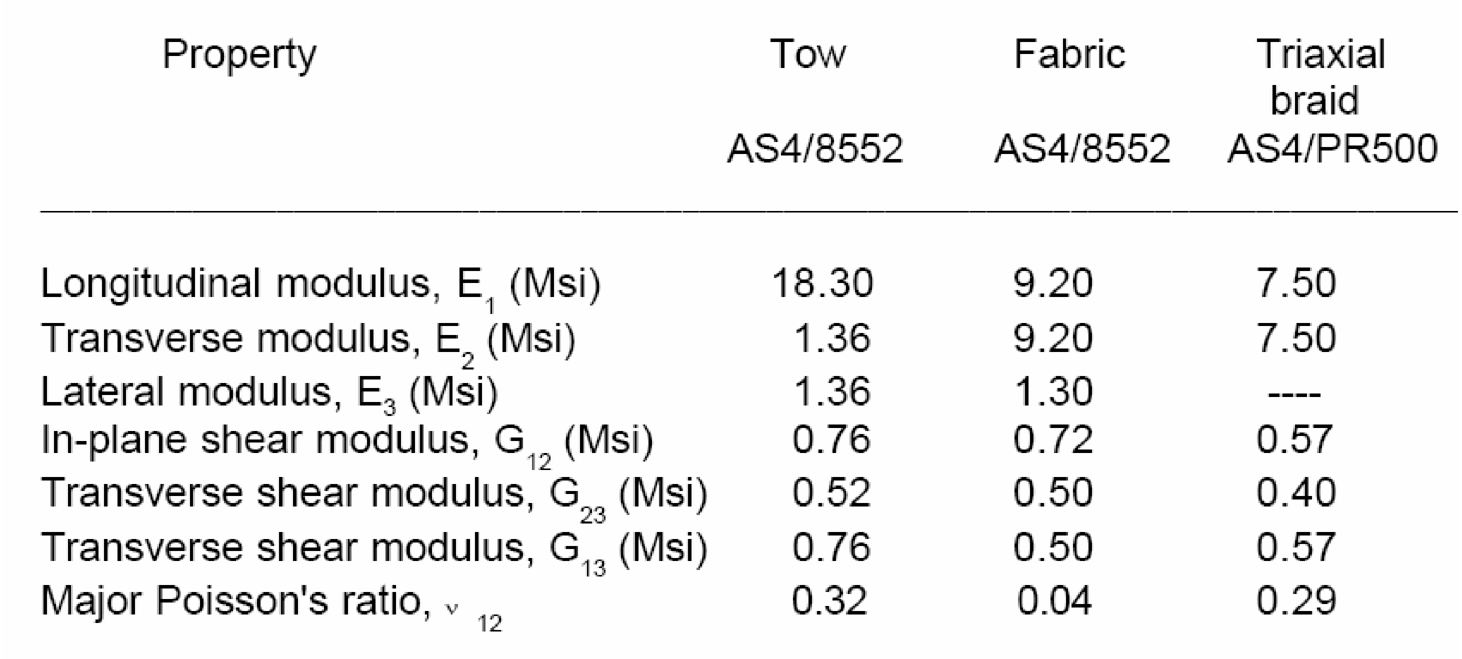


The fibers are not perfect and may have varying levels of microscopic dislocation or cracking

The matrix is relatively weak , but acts to link the fibers together

The strength /stiffness is an aggregate of the two ingredients

1. What are composites?

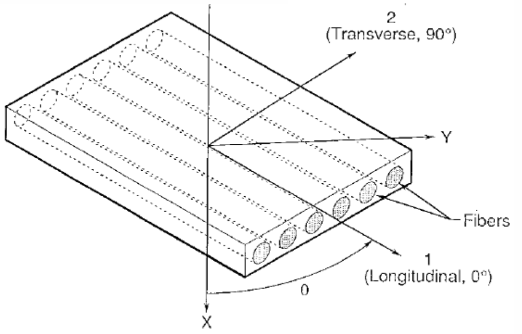


Property	Tow AS4/8552	Fabric AS4/8552	Triaxial braid AS4/PR500
Longitudinal modulus, E_1 (Msi)	18.30	9.20	7.50
Transverse modulus, E_2 (Msi)	1.36	9.20	7.50
Lateral modulus, E_3 (Msi)	1.36	1.30	----
In-plane shear modulus, G_{12} (Msi)	0.76	0.72	0.57
Transverse shear modulus, G_{23} (Msi)	0.52	0.50	0.40
Transverse shear modulus, G_{13} (Msi)	0.76	0.50	0.57
Major Poisson's ratio, ν_{12}	0.32	0.04	0.29

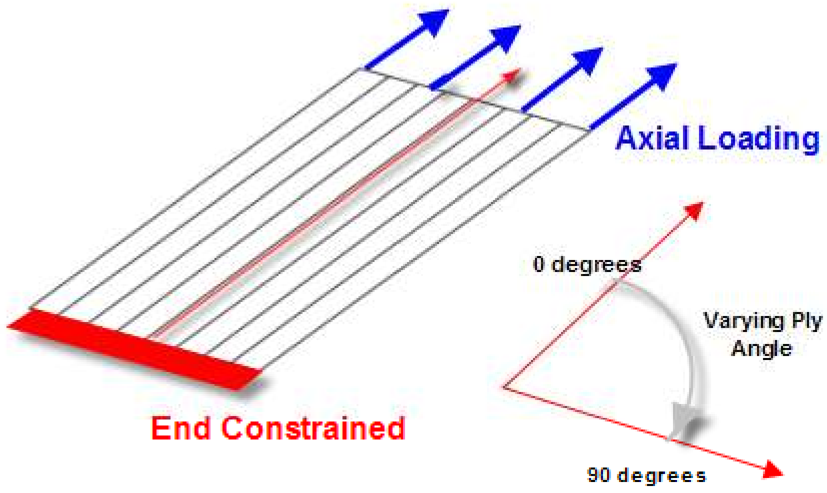
The table shows the stiffness of a group of Graphite – Epoxy systems

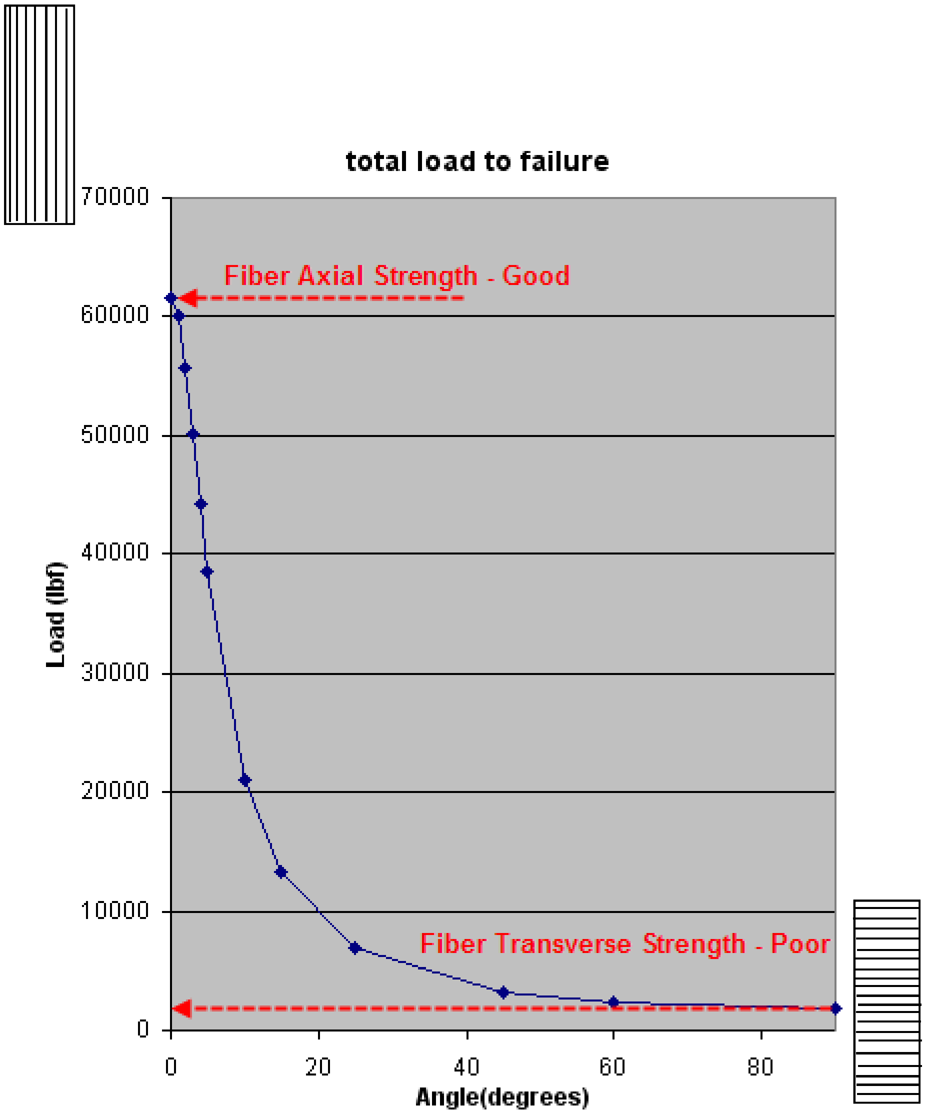
The directionality is clear

Convention is fiber/matrix as the system designation



Single Ply

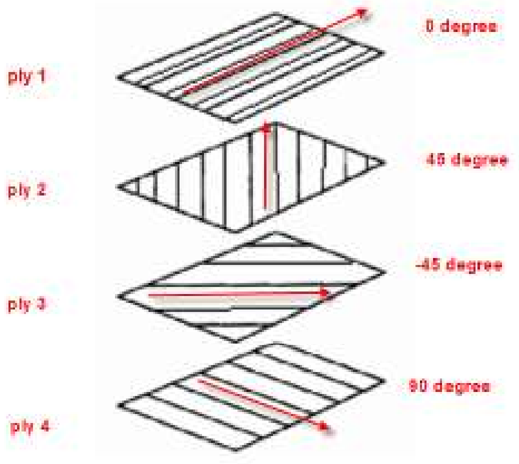




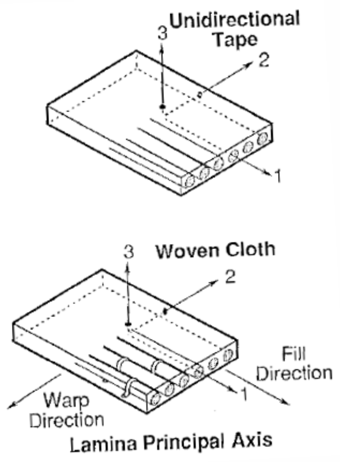
A simple experiment using FEA to predict failing load in a coupon

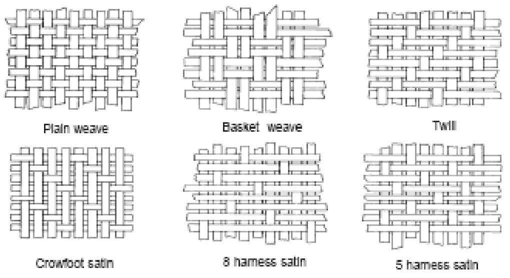
Graph shows effect of ply angle in a single ply layup

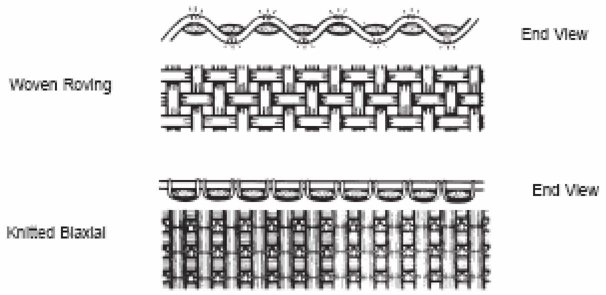
1. What are composites?



A laminate made up of 4 plies







In practice plies are rarely used individually, multiple angles are used to tailor the performance of the composite.

A stack up of plies is formed either by bonding sheets together or by some form of weaving

However the FEA idealization usually assumes a ‘sheet-like’ equivalent

1. What are composites?

‘**Pre-preg layup**‘ is a very common form of assembly where multiple dry unidirectional fiber/matrix sheets (pre-impregnated) are laid up and then wetted with a resin to achieve bonding between the sheets.

Pressure and temperature may be used to achieve good bonding or to achieve more complex shapes

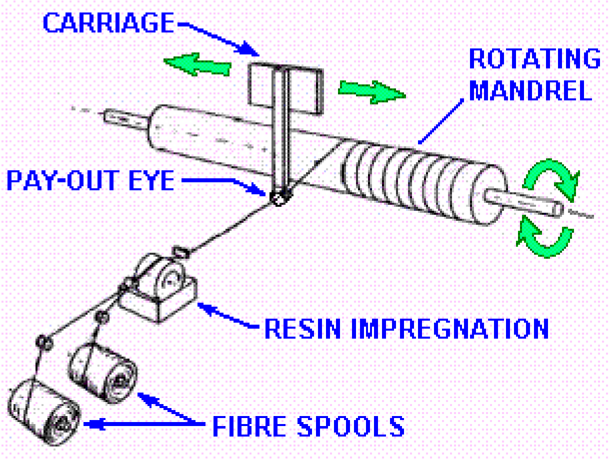
1. What are composites?

Resin Transfer molding (RTM). Cloth systems may be wetted externally and cured, or the system may be augmented by creating a vacuum in the part using a bagging system. Resin is then fed into the system and is absorbed into the composite.

Pressure applied between dies can be used as an alternative to creating a vacuum.

The cure may be at room temperature or elevated temperature dependent on the system

1. What are composites?



Filament winding is used to create tubular based forms. With the use of sophisticated multi axis machines and CNC, spherical, conical and more complex shapes can be formed. It is suitable for very large components such as tunnel liners, rocket fuel tanks etc.

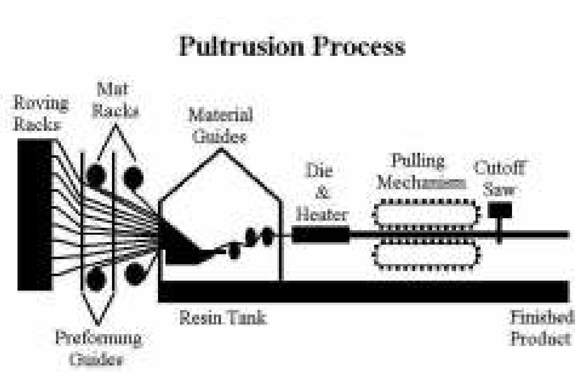
The resin may be added as the filament through a runs a bath, or it may be sprayed or applied later on the mandrel.

The mandrel and component may then be transferred to an oven for curing

1. What are composites?

Other manufacturing processes

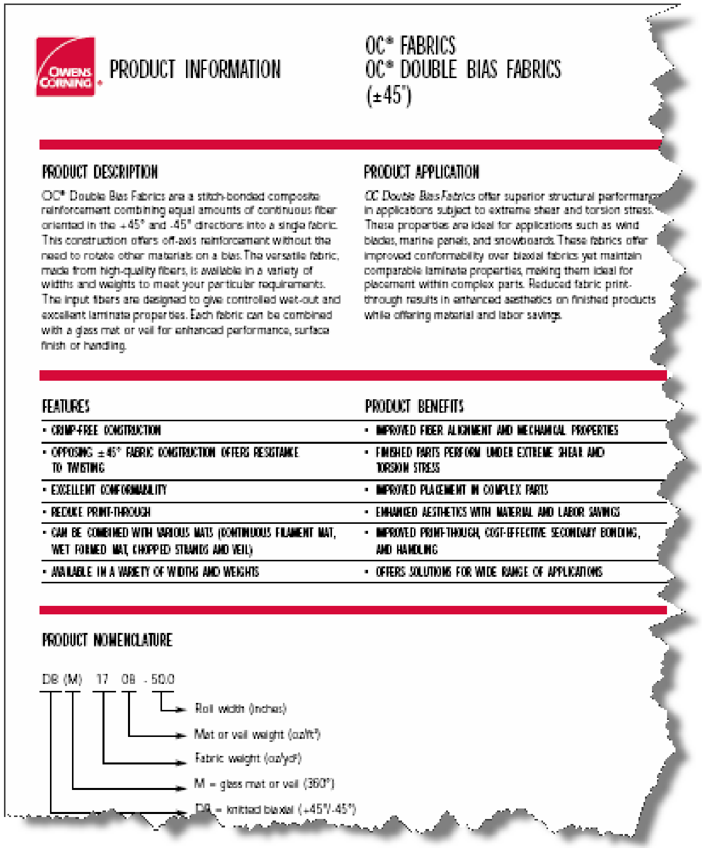
Pultrusion – a sock like woven shape is braided and then pulled through a heated die to form components such as drive shafts, stiffeners, rods etc.



Automated tape placement – a multi axis head under CNC control is able to lay individually programmed paths of tape across a flat bed or die shape. Very sophisticated ply orientations can be designed.

1. What are composites?

Many forms of composites are available, here a fabric is offered in a range of weights which control stiffness and strength



PRODUCT INFORMATION

OC® FABRICS
OC® DOUBLE BIAS FABRICS
(±45°)

PRODUCT DESCRIPTION

OC® Double Bias Fabrics are a stitch-bonded composite reinforcement combining equal amounts of continuous fiber oriented in the +45° and -45° directions into a single fabric. This construction offers off-axis reinforcement without the need to rotate other materials on a bias. The versatile fabric, made from high-quality fibers, is available in a variety of widths and weights to meet your particular requirements. The input fibers are designed to give controlled wet-out and excellent laminate properties. Each fabric can be combined with a glass mat or veil for enhanced performance, surface finish or handling.

PRODUCT APPLICATION

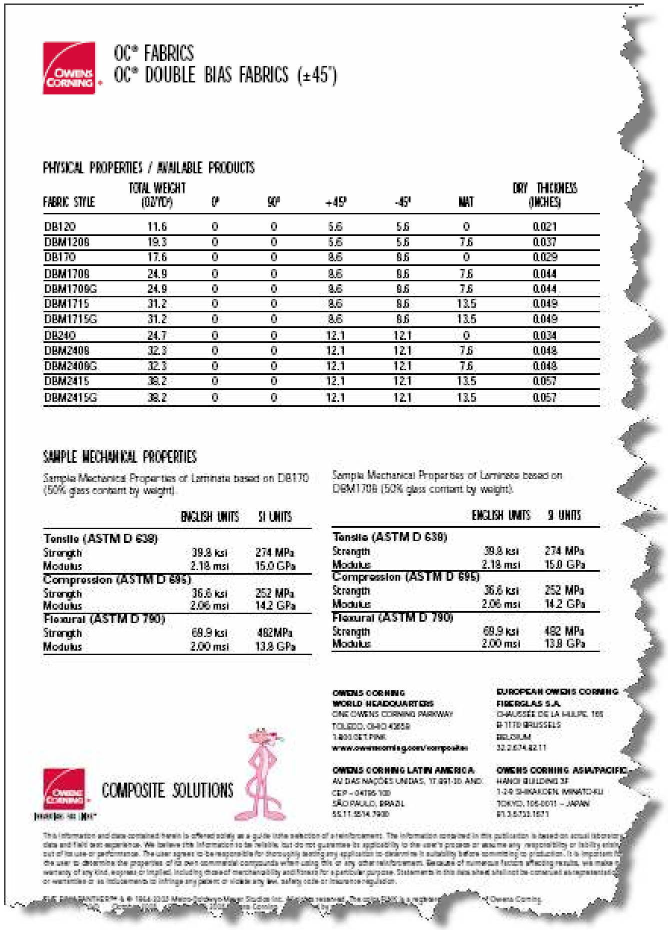
OC Double Bias Fabrics offer superior structural performance in applications subject to extreme shear and torsion stress. These properties are ideal for applications such as wind blades, marine panels, and snowboards. These fabrics offer improved conformability over biaxial fabrics yet maintain comparable laminate properties, making them ideal for placement within complex parts. Reduced fabric print-through results in enhanced aesthetics on finished products while offering material and labor savings.

FEATURES	PRODUCT BENEFITS
• CRIMP-FREE CONSTRUCTION	• IMPROVED FIBER ALIGNMENT AND MECHANICAL PROPERTIES
• OPPOSING ±45° FABRIC CONSTRUCTION OFFERS RESISTANCE TO TWISTING	• FINISHED PARTS PERFORM UNDER EXTREME SHEAR AND TORSION STRESS
• EXCELLENT CONFORMABILITY	• IMPROVED PLACEMENT IN COMPLEX PARTS
• REDUCE PRINT-THROUGH	• ENHANCED AESTHETICS WITH MATERIAL AND LABOR SAVINGS
• CAN BE COMBINED WITH VARIOUS MATS (CONTINUOUS FILAMENT MAT, WET FORMED MAT, CHOPPED STRANDS AND VEIL)	• IMPROVED PRINT-THROUGH, COST-EFFECTIVE SECONDARY BONDING, AND HANDLING
• AVAILABLE IN A VARIETY OF WIDTHS AND WEIGHTS	• OFFERS SOLUTIONS FOR WIDE RANGE OF APPLICATIONS

PRODUCT NOMENCLATURE

DB (M) 17 08 - 50.0

Roll width (inches)
Mat or veil weight (oz/ft²)
Fabric weight (oz/yd²)
M = glass mat or veil (360°)
DB = knitted biaxial (+45°/-45°)



OC® FABRICS
OC® DOUBLE BIAS FABRICS (±45°)

PHYSICAL PROPERTIES / AVAILABLE PRODUCTS

FABRIC STYLE	TOTAL WEIGHT (OZ/YD²)	0°	90°	+45°	-45°	MAT	DRY THICKNESS (INCHES)
DB120	11.6	0	0	5.6	5.6	0	0.021
DBM1208	19.3	0	0	5.6	5.6	7.6	0.037
DB170	17.6	0	0	8.6	8.6	0	0.029
DBM1708	24.9	0	0	8.6	8.6	7.6	0.044
DBM1708G	24.9	0	0	8.6	8.6	7.6	0.044
DBM1715	31.2	0	0	8.6	8.6	13.5	0.049
DBM1715G	31.2	0	0	8.6	8.6	13.5	0.049
DB240	24.7	0	0	12.1	12.1	0	0.034
DBM2408	32.3	0	0	12.1	12.1	7.6	0.048
DBM2408G	32.3	0	0	12.1	12.1	7.6	0.048
DBM2415	38.2	0	0	12.1	12.1	13.5	0.057
DBM2415G	38.2	0	0	12.1	12.1	13.5	0.057

SAMPLE MECHANICAL PROPERTIES

Sample Mechanical Properties of Laminate based on DB170 (50% glass content by weight).

	ENGLISH UNITS	SI UNITS
Tensile (ASTM D 638)		
Strength	39.8 ksi	274 MPa
Modulus	2.18 msi	15.0 GPa
Compression (ASTM D 695)		
Strength	36.6 ksi	252 MPa
Modulus	2.06 msi	14.2 GPa
Flexural (ASTM D 790)		
Strength	69.9 ksi	482MPa
Modulus	2.00 msi	13.8 GPa

Sample Mechanical Properties of Laminate based on DBM1708 (50% glass content by weight).

	ENGLISH UNITS	SI UNITS
Tensile (ASTM D 638)		
Strength	39.8 ksi	274 MPa
Modulus	2.18 msi	15.0 GPa
Compression (ASTM D 695)		
Strength	36.6 ksi	252 MPa
Modulus	2.06 msi	14.2 GPa
Flexural (ASTM D 790)		
Strength	69.9 ksi	482 MPa
Modulus	2.00 msi	13.8 GPa

COMPOSITE SOLUTIONS

OWENS CORNING
WORLD HEADQUARTERS
ONE OWENS CORNING PARKWAY
TOLEDO, OHIO 43659
1.800.GET.PINK
www.owenscorning.com/composites

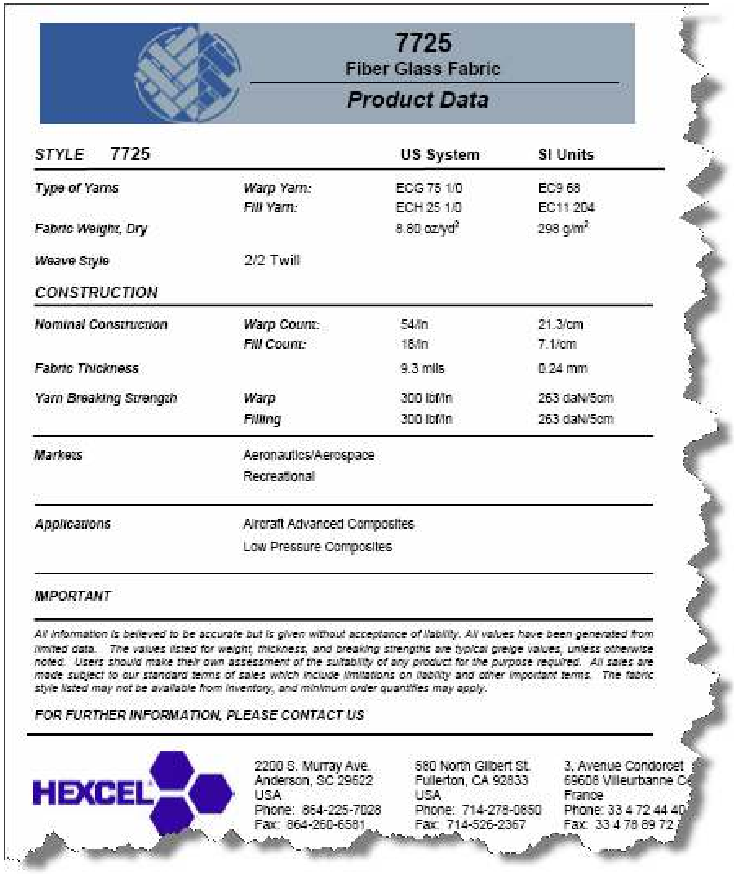
EUROPEAN OWENS CORNING
FIBERGLAS S.A.
CHAUSSÉE DE LA HULPE, 166
B-1170 BRUSSELS
BELGIUM
32.2.674.82.11

OWENS CORNING LATIN AMERICA
AV. DAS NAÇÕES UNIDAS, 17.891-10. AND.
CEP - 04795-100
SÃO PAULO, BRAZIL
55.11.5514.7900

OWENS CORNING ASIA/PACIFIC
HANDI BUILDING 3F
1-2-9 SHIBAKOEN, MINATO-KU
TOKYO, 105-0011 – JAPAN
81.3.5733.1671

1. What are composites?

Here a glass fiber and low strength carbon cloth are offered



7725
Fiber Glass Fabric
Product Data

STYLE 7725		US System	SI Units
Type of Yarns	*Warp Yarn:*	ECG 75 1/0	EC9 68
	Fill Yarn:	ECH 25 1/0	EC11 204
Fabric Weight, Dry		8.80 oz/yd^2	298 g/m^2
Weave Style	2/2 Twill		
CONSTRUCTION			
Nominal Construction	*Warp Count:*	54/in	21.3/cm
	Fill Count:	18/in	7.1/cm
Fabric Thickness		9.3 mils	0.24 mm
Yarn Breaking Strength	*Warp*	300 lbf/in	263 daN/5cm
	Filling	300 lbf/in	263 daN/5cm
Markets	Aeronautics/Aerospace Recreational		
Applications	Aircraft Advanced Composites Low Pressure Composites		

IMPORTANT

All information is believed to be accurate but is given without acceptance of liability. All values have been generated from limited data. The values listed for weight, thickness, and breaking strengths are typical greige values, unless otherwise noted. Users should make their own assessment of the suitability of any product for the purpose required. All sales are made subject to our standard terms of sales which include limitations on liability and other important terms. The fabric style listed may not be available from inventory, and minimum order quantities may apply.

FOR FURTHER INFORMATION, PLEASE CONTACT US

HEXCEL

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Anderson, SC 29622
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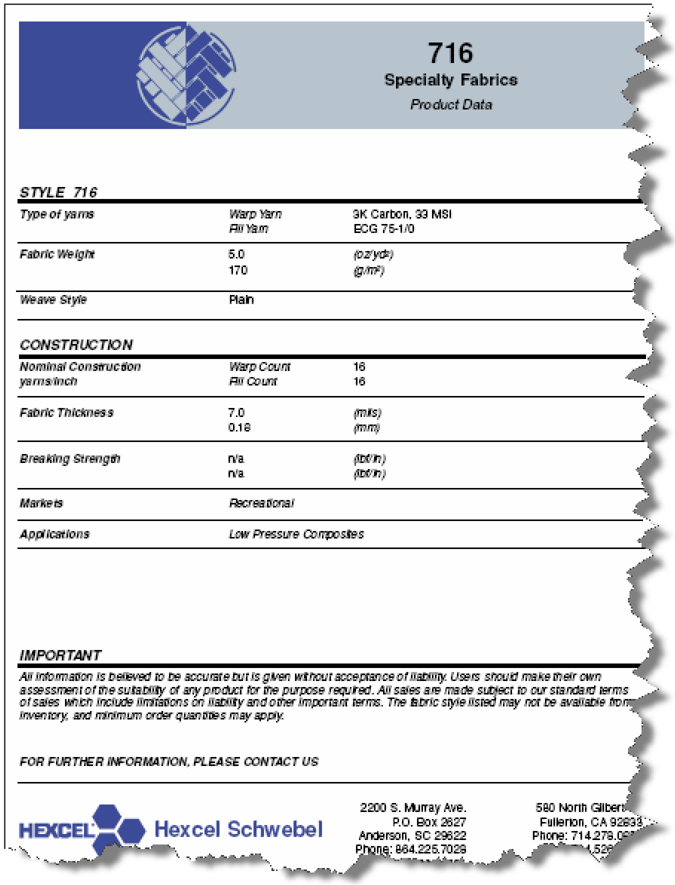
Phone: 864-225-7028
Fax: 864-260-6581

580 North Gilbert St.
Fullerton, CA 92833
USA
Phone: 714-278-0850
Fax: 714-526-2367

3, Avenue Condorcet
69608 Villeurbanne C
France
Phone: 33 4 72 44 40
Fax: 33 4 78 69 72



716
Specialty Fabrics
Product Data

STYLE 716		
Type of yarns	*Warp Yarn* *Fill Yarn*	3K Carbon, 33 MSI ECG 75-1/0
Fabric Weight	5.0 170	*(oz/yd²)* *(g/m²)*
Weave Style	Plain	
CONSTRUCTION		
Nominal Construction yarns/inch	*Warp Count* *Fill Count*	16 16
Fabric Thickness	7.0 0.18	*(mils)* *(mm)*
Breaking Strength	n/a n/a	*(lbf/in)* *(lbf/in)*
Markets	*Recreational*	
Applications	*Low Pressure Composites*	

IMPORTANT

All information is believed to be accurate but is given without acceptance of liability. Users should make their own assessment of the suitability of any product for the purpose required. All sales are made subject to our standard terms of sales which include limitations on liability and other important terms. The fabric style listed may not be available from inventory, and minimum order quantities may apply.

FOR FURTHER INFORMATION, PLEASE CONTACT US

HEXCEL Hexcel Schwebel

2200 S. Murray Ave.
P.O. Box 2627
Anderson, SC 29622
Phone: 864.225.7028

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Fullerton, CA 92833
Phone: 714.278.0

Introductory Composites FE Analysis Webinar

Agenda

1. What are composites?

Review of different forms and manufacturing processes.

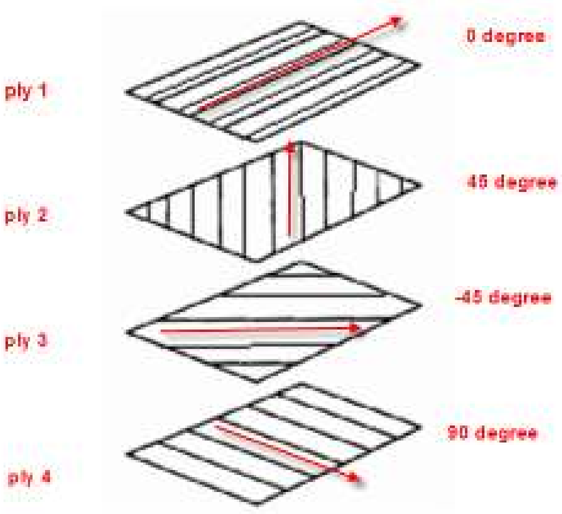
2. How do composites vary from metallic structures?

Overview of material properties and the ABD matrix terms. Hints on practical design methods.

3. How do I set up a composite FEA?

Overview of typical processes. Keeping track of plies, mold lines etc.

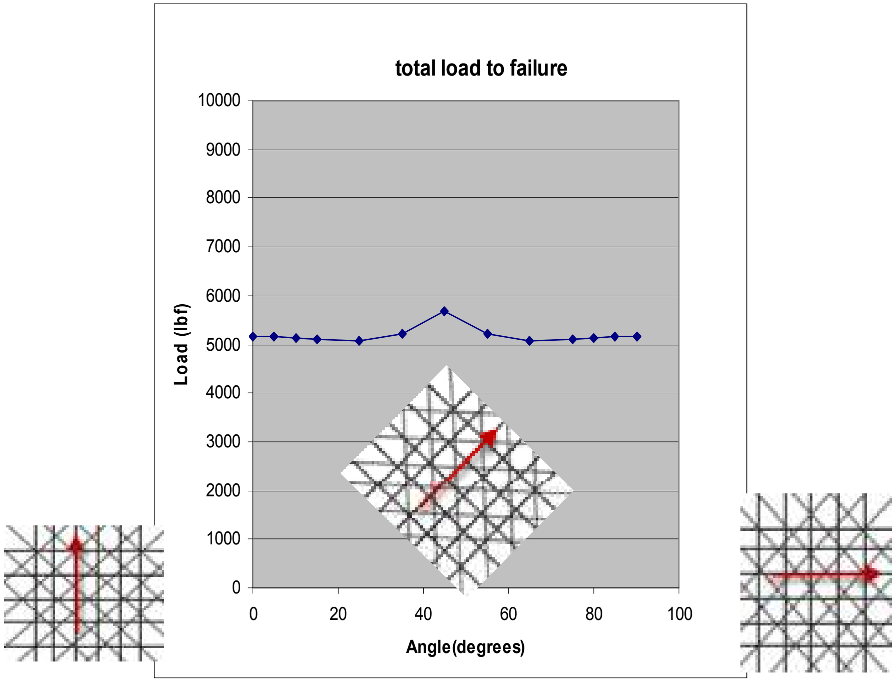
- Single Ply directions exposes weakness
- Ply layups used of multiple orientation

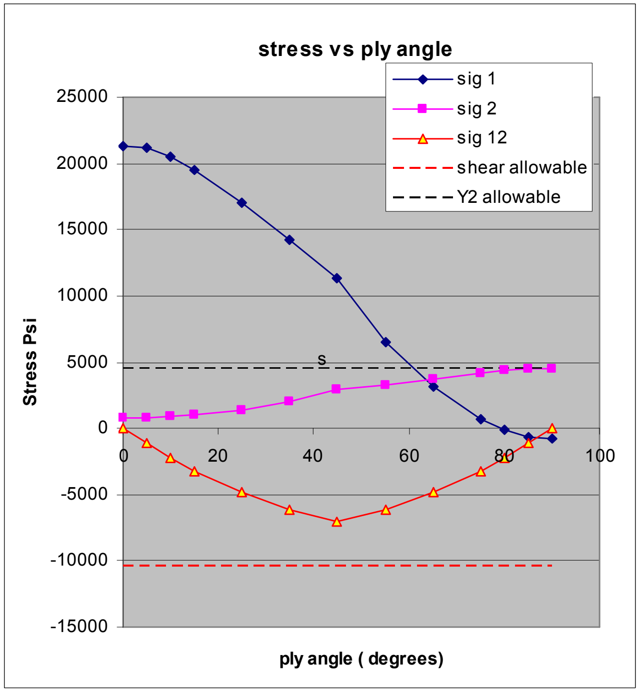


A laminate made up of 4 plies

- Shorthand 0/45/-45/90
- Tuning the layup orientation, thickness and stacking order is key to optimum design

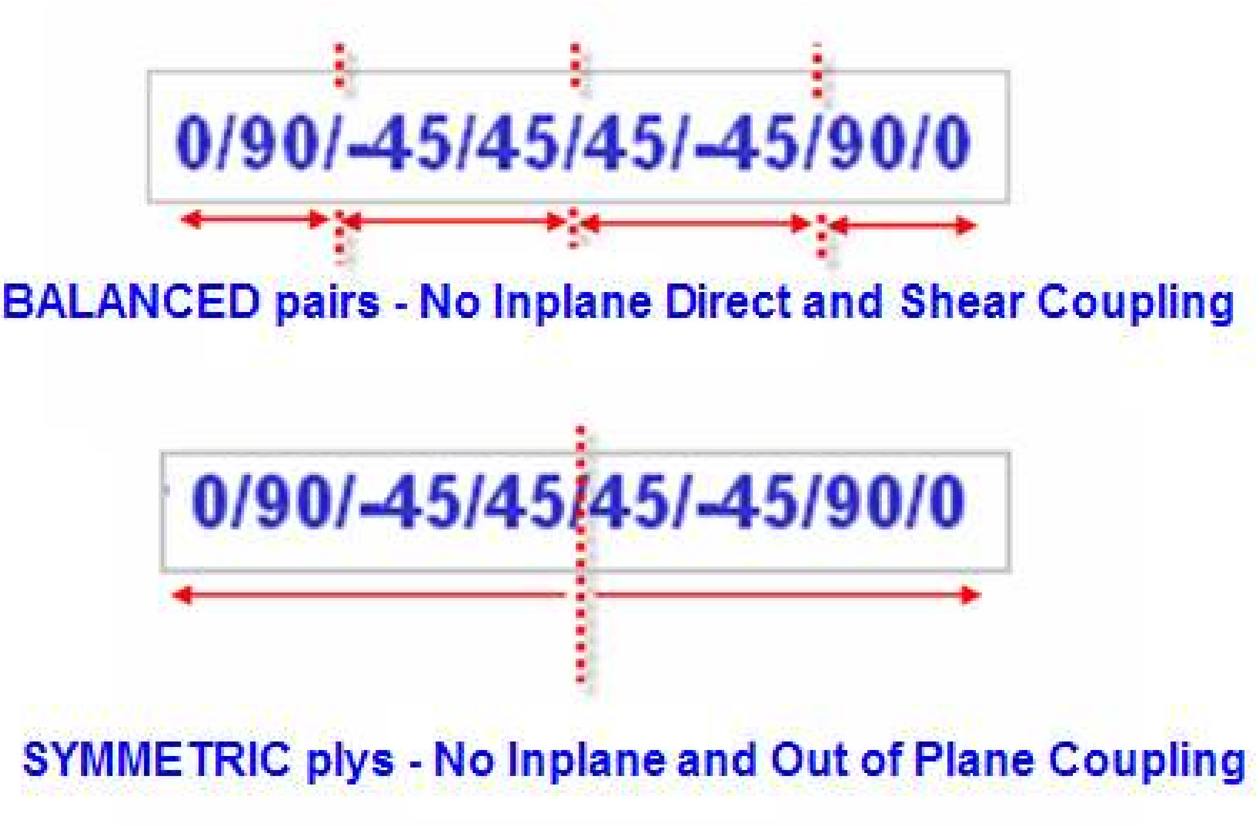
2. How do composites vary from metallic structures?





- Previous Single Ply replaced by 0/90/-45/45/45/-45/90/0
- Maximum Strength is reduced, but now very predictable
- No Optimization! Sometimes called ‘black’ isotropic material ….

❒ why 0/90/-45/45/45/-45/90/0 choice?



0/90/-45/45/45/-45/90/0

BALANCED pairs - No Inplane Direct and Shear Coupling

0/90/-45/45/45/-45/90/0

SYMMETRIC plys - No Inplane and Out of Plane Coupling

2. How do composites vary from metallic structures?

Overview of material properties and the ABD matrix terms. Hints on practical design methods.

$$\begin{Bmatrix} \sigma_1 \\ \sigma_2 \\ \tau_{12} \end{Bmatrix} = \begin{bmatrix} Q_{11} & Q_{12} & 0 \\ Q_{12} & Q_{22} & 0 \\ 0 & 0 & Q_{66} \end{bmatrix} \begin{Bmatrix} \varepsilon_1 \\ \varepsilon_2 \\ \gamma_{12} \end{Bmatrix}$$

$$Q_{11} = \frac{E_1}{(1-\upsilon_{12}\upsilon_{21})}, \quad Q_{12} = \frac{\upsilon_{21} E_1}{(1-\upsilon_{12}\upsilon_{21})} = \frac{\upsilon_{12} E_2}{(1-\upsilon_{12}\upsilon_{21})}$$

$$Q_{22} = \frac{E_2}{(1-\upsilon_{12}\upsilon_{21})}, \quad Q_{66} = G_{12},$$

2. How do composites vary from metallic structures?

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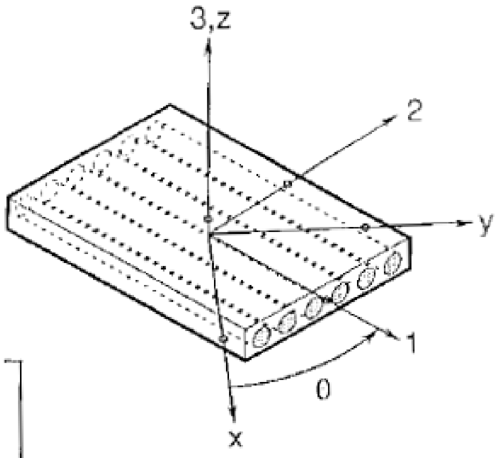
$$\begin{Bmatrix}\sigma_x\\ \sigma_y\\ \tau_{xy}\end{Bmatrix} = [T_1][Q][T_2]^{-1}\begin{Bmatrix}\varepsilon_x\\ \varepsilon_y\\ \gamma_{xy}\end{Bmatrix}$$

or

$$\begin{Bmatrix}\sigma_x\\ \sigma_y\\ \tau_{xy}\end{Bmatrix} = [\bar{Q}]\begin{Bmatrix}\varepsilon_x\\ \varepsilon_y\\ \gamma_{xy}\end{Bmatrix}$$



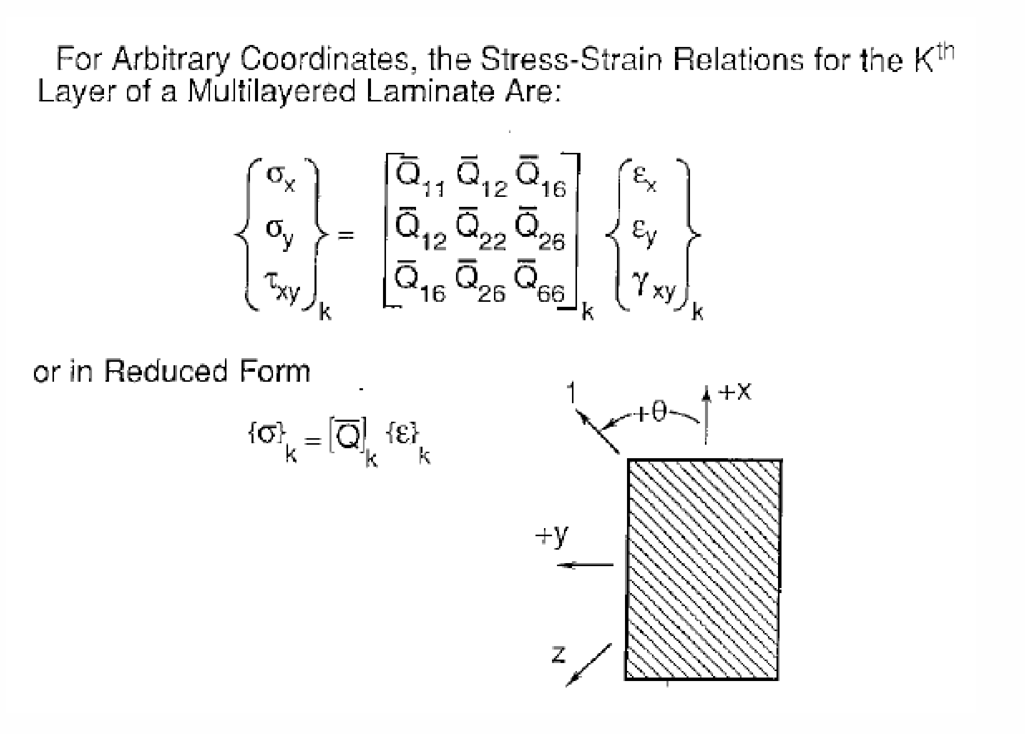
$$[T_1] = \begin{bmatrix}\cos^2\theta & \sin^2\theta & -2\sin\theta\cos\theta\\ \sin^2\theta & \cos^2\theta & 2\sin\theta\cos\theta\\ \sin\theta\cos\theta & -\sin\theta\cos\theta & \cos^2\theta - \sin^2\theta\end{bmatrix}$$

$$[\bar{Q}] = \begin{bmatrix}\bar{Q}_{11} & \bar{Q}_{12} & \bar{Q}_{16}\\ \bar{Q}_{12} & \bar{Q}_{22} & \bar{Q}_{26}\\ \bar{Q}_{16} & \bar{Q}_{26} & \bar{Q}_{66}\end{bmatrix}$$

$$[T_2] = \begin{bmatrix}\cos^2\theta & \sin^2\theta & -\sin\theta\cos\theta\\ \sin^2\theta & \cos^2\theta & \sin\theta\cos\theta\\ 2\sin\theta\cos\theta & -2\sin\theta\cos\theta & \cos^2\theta - \sin^2\theta\end{bmatrix}$$

2. How do composites vary from metallic structures?

Overview of material properties and the ABD matrix terms. Hints on practical design methods.



For Arbitrary Coordinates, the Stress-Strain Relations for the K^{th} Layer of a Multilayered Laminate Are:

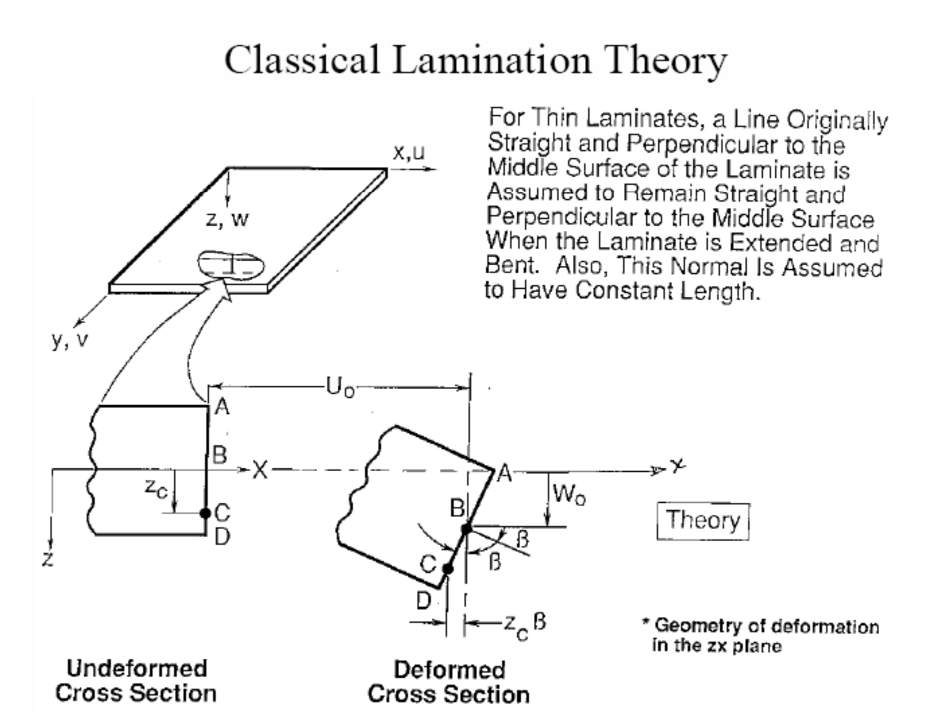
$$\begin{Bmatrix}\sigma_x\\ \sigma_y\\ \tau_{xy}\end{Bmatrix}_k = \begin{bmatrix}\bar{Q}_{11} & \bar{Q}_{12} & \bar{Q}_{16}\\ \bar{Q}_{12} & \bar{Q}_{22} & \bar{Q}_{26}\\ \bar{Q}_{16} & \bar{Q}_{26} & \bar{Q}_{66}\end{bmatrix}_k \begin{Bmatrix}\varepsilon_x\\ \varepsilon_y\\ \gamma_{xy}\end{Bmatrix}_k$$

or in Reduced Form

$$\{\sigma\}_k = [\bar{Q}]_k \{\varepsilon\}_k$$

2. How do composites vary from metallic structures?

Overview of material properties and the ABD matrix terms. Hints on practical design methods.



Classical Lamination Theory

For Thin Laminates, a Line Originally Straight and Perpendicular to the Middle Surface of the Laminate is Assumed to Remain Straight and Perpendicular to the Middle Surface When the Laminate is Extended and Bent. Also, This Normal Is Assumed to Have Constant Length.

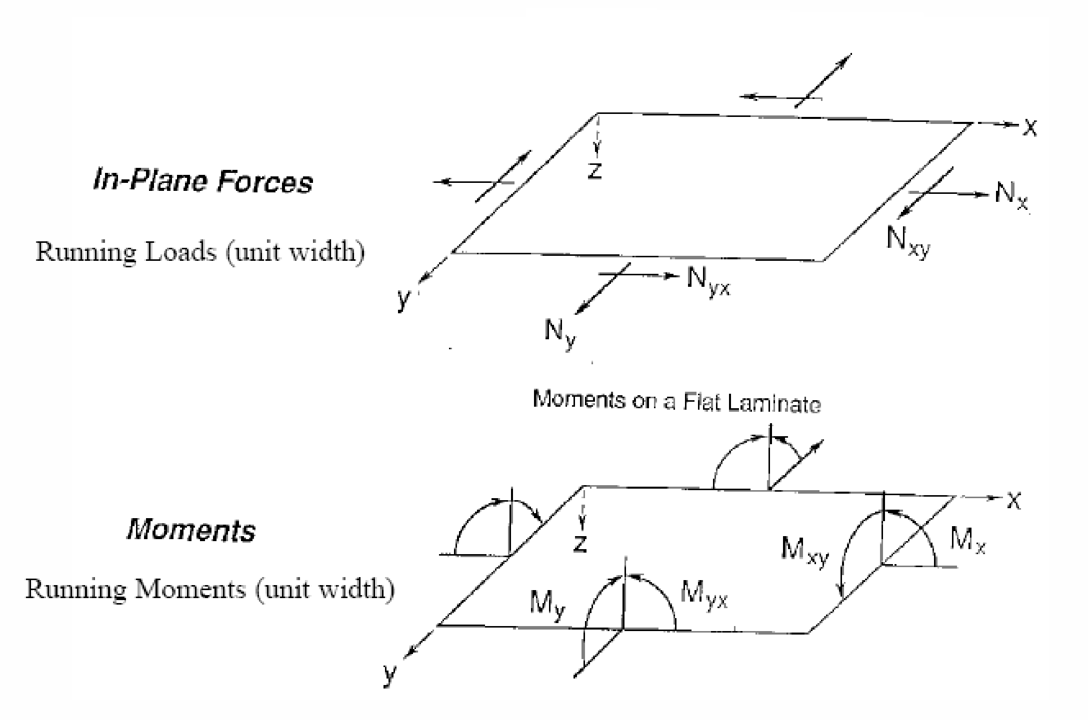
2. How do composites vary from metallic structures?

Overview of material properties and the ABD matrix terms. Hints on practical design methods.

$$\begin{Bmatrix} \sigma_x \\ \sigma_y \\ \tau_{xy} \end{Bmatrix}_k = \begin{bmatrix} \bar{Q}_{11} & \bar{Q}_{12} & \bar{Q}_{16} \\ \bar{Q}_{12} & \bar{Q}_{22} & \bar{Q}_{26} \\ \bar{Q}_{16} & \bar{Q}_{26} & \bar{Q}_{66} \end{bmatrix}_k \left\{ \begin{Bmatrix} \varepsilon_x^\circ \\ \varepsilon_y^\circ \\ \gamma_{xy}^\circ \end{Bmatrix} + Z \begin{Bmatrix} K_x \\ K_y \\ K_{xy} \end{Bmatrix} \right\}$$

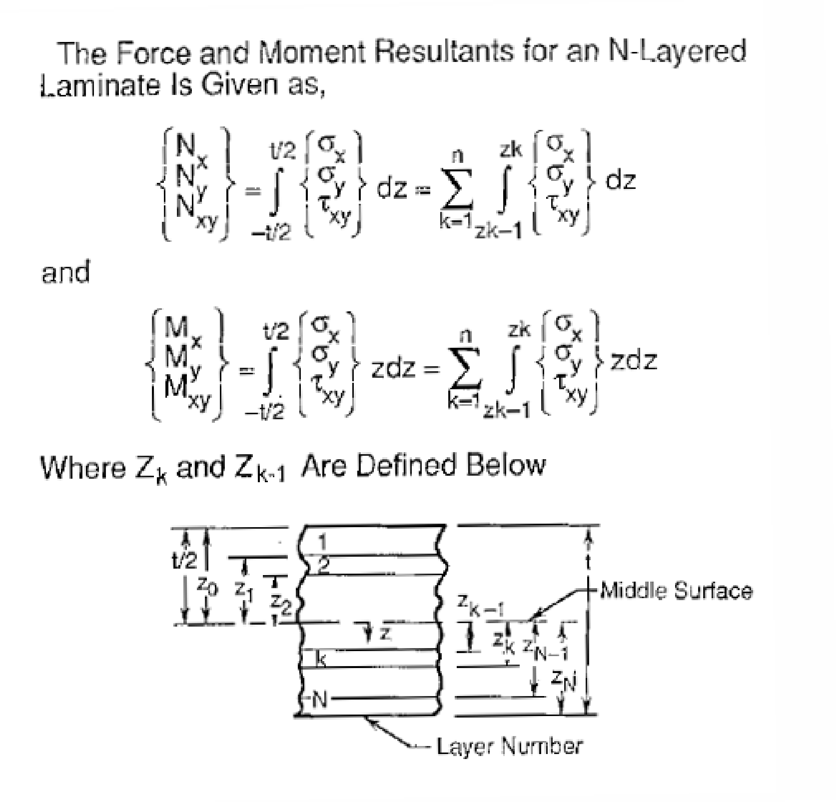
2. How do composites vary from metallic structures?

Overview of material properties and the ABD matrix terms. Hints on practical design methods.



2. How do composites vary from metallic structures?

Overview of material properties and the ABD matrix terms. Hints on practical design methods.



The Force and Moment Resultants for an N-Layered Laminate Is Given as,

$$\begin{Bmatrix} N_x \\ N_y \\ N_{xy} \end{Bmatrix} = \int_{-t/2}^{t/2} \begin{Bmatrix} \sigma_x \\ \sigma_y \\ \tau_{xy} \end{Bmatrix} dz = \sum_{k=1}^{n} \int_{zk-1}^{zk} \begin{Bmatrix} \sigma_x \\ \sigma_y \\ \tau_{xy} \end{Bmatrix} dz$$

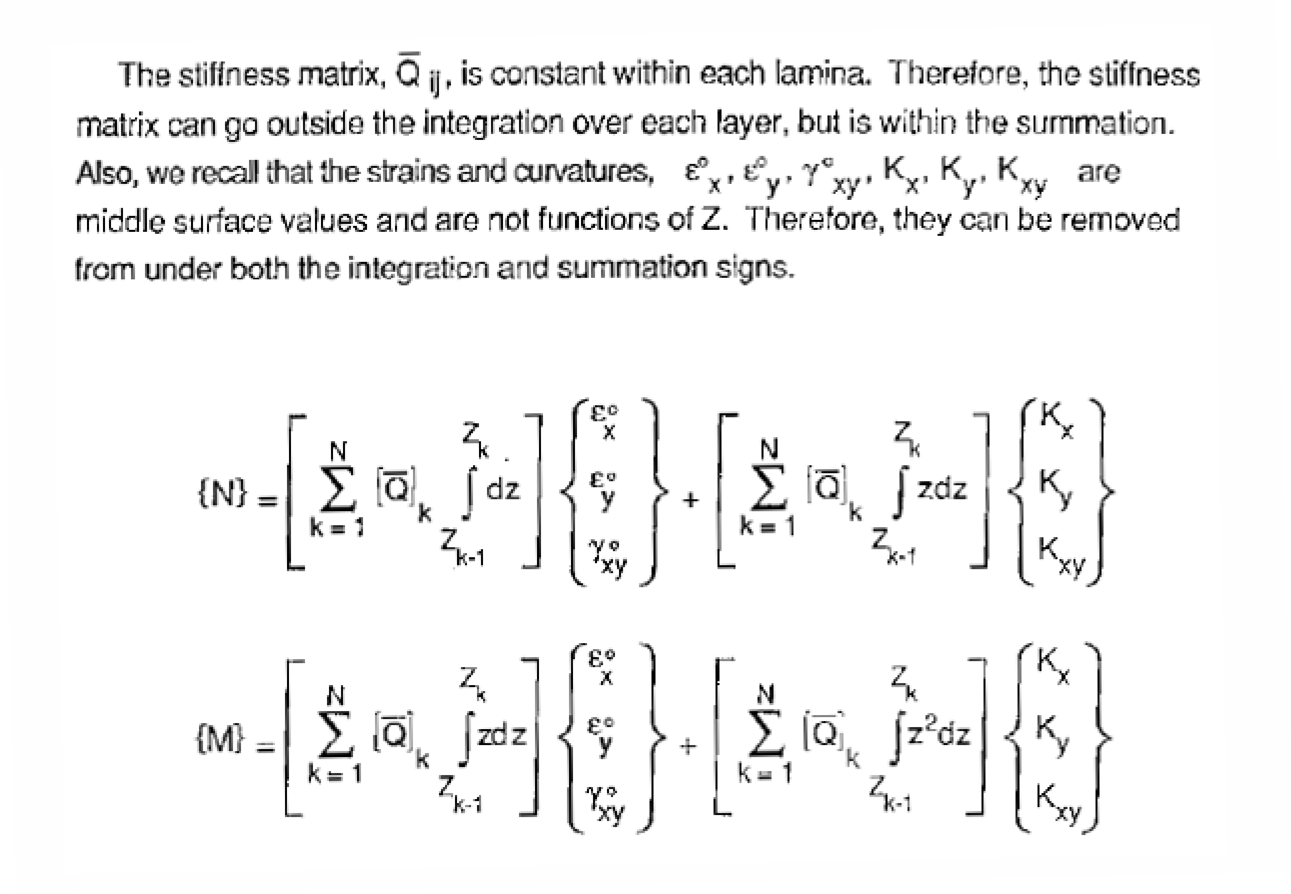
and

$$\begin{Bmatrix} M_x \\ M_y \\ M_{xy} \end{Bmatrix} = \int_{-t/2}^{t/2} \begin{Bmatrix} \sigma_x \\ \sigma_y \\ \tau_{xy} \end{Bmatrix} zdz = \sum_{k=1}^{n} \int_{zk-1}^{zk} \begin{Bmatrix} \sigma_x \\ \sigma_y \\ \tau_{xy} \end{Bmatrix} zdz$$

Where Z_k and Z_{k-1} Are Defined Below

2. How do composites vary from metallic structures?

Overview of material properties and the ABD matrix terms. Hints on pra



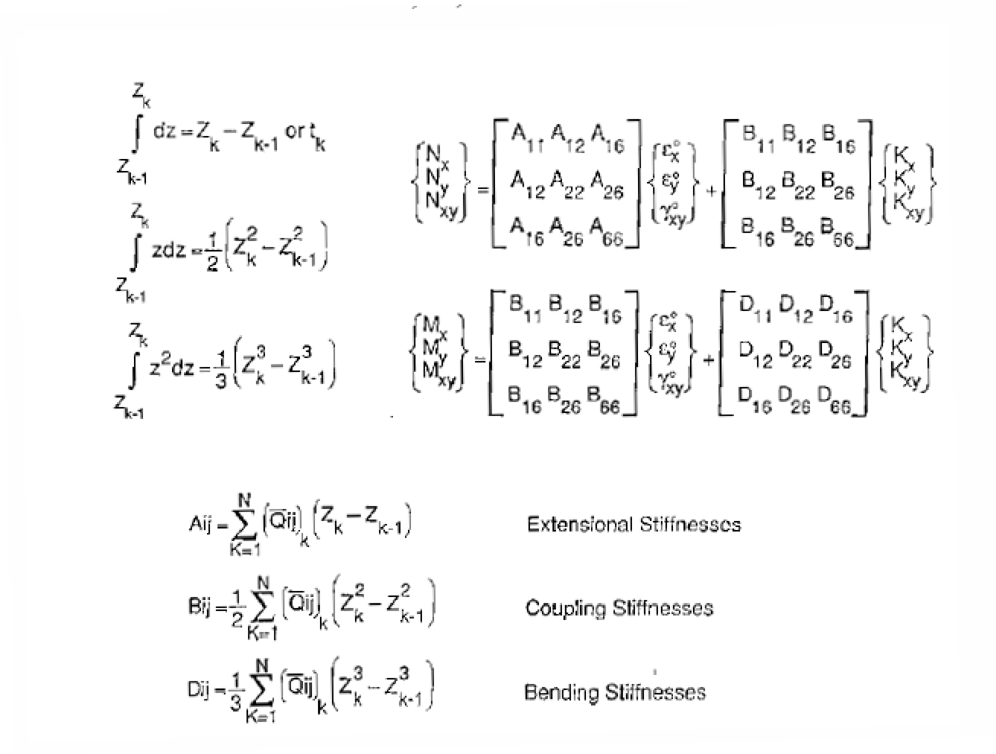
The stiffness matrix, $\bar{Q}_{ij}$, is constant within each lamina. Therefore, the stiffness matrix can go outside the integration over each layer, but is within the summation. Also, we recall that the strains and curvatures, $\varepsilon^\circ_x, \varepsilon^\circ_y, \gamma^\circ_{xy}, K_x, K_y, K_{xy}$ are middle surface values and are not functions of Z. Therefore, they can be removed from under both the integration and summation signs.

$$\{N\} = \left[\sum_{k=1}^{N} [\bar{Q}]_k \int_{Z_{k-1}}^{Z_k} dz\right] \begin{Bmatrix}\varepsilon^\circ_x\\ \varepsilon^\circ_y\\ \gamma^\circ_{xy}\end{Bmatrix} + \left[\sum_{k=1}^{N} [\bar{Q}]_k \int_{Z_{k-1}}^{Z_k} z\,dz\right] \begin{Bmatrix}K_x\\ K_y\\ K_{xy}\end{Bmatrix}$$

$$\{M\} = \left[\sum_{k=1}^{N} [\bar{Q}]_k \int_{Z_{k-1}}^{Z_k} z\,dz\right] \begin{Bmatrix}\varepsilon^\circ_x\\ \varepsilon^\circ_y\\ \gamma^\circ_{xy}\end{Bmatrix} + \left[\sum_{k=1}^{N} [\bar{Q}]_k \int_{Z_{k-1}}^{Z_k} z^2\,dz\right] \begin{Bmatrix}K_x\\ K_y\\ K_{xy}\end{Bmatrix}$$

2. How do composites vary from metallic structures?

Overview of material properties and the ABD matrix terms. Hints on practical design methods.



$$\int_{Z_{k-1}}^{Z_k} dz = Z_k - Z_{k-1} \text{ or } t_k$$

$$\int_{Z_{k-1}}^{Z_k} z\,dz = \frac{1}{2}\left(Z_k^2 - Z_{k-1}^2\right)$$

$$\int_{Z_{k-1}}^{Z_k} z^2 dz = \frac{1}{3}\left(Z_k^3 - Z_{k-1}^3\right)$$

$$\begin{Bmatrix} N_x \\ N_y \\ N_{xy} \end{Bmatrix} = \begin{bmatrix} A_{11} & A_{12} & A_{16} \\ A_{12} & A_{22} & A_{26} \\ A_{16} & A_{26} & A_{66} \end{bmatrix} \begin{Bmatrix} \varepsilon_x^o \\ \varepsilon_y^o \\ \gamma_{xy}^o \end{Bmatrix} + \begin{bmatrix} B_{11} & B_{12} & B_{16} \\ B_{12} & B_{22} & B_{26} \\ B_{16} & B_{26} & B_{66} \end{bmatrix} \begin{Bmatrix} K_x \\ K_y \\ K_{xy} \end{Bmatrix}$$

$$\begin{Bmatrix} M_x \\ M_y \\ M_{xy} \end{Bmatrix} = \begin{bmatrix} B_{11} & B_{12} & B_{16} \\ B_{12} & B_{22} & B_{26} \\ B_{16} & B_{26} & B_{66} \end{bmatrix} \begin{Bmatrix} \varepsilon_x^o \\ \varepsilon_y^o \\ \gamma_{xy}^o \end{Bmatrix} + \begin{bmatrix} D_{11} & D_{12} & D_{16} \\ D_{12} & D_{22} & D_{26} \\ D_{16} & D_{26} & D_{66} \end{bmatrix} \begin{Bmatrix} K_x \\ K_y \\ K_{xy} \end{Bmatrix}$$

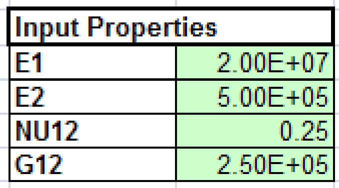
$A_{ij} = \sum_{K=1}^{N} \left(\overline{Q}_{ij}\right)_k \left(Z_k - Z_{k-1}\right)$ Extensional Stiffnesses

$B_{ij} = \frac{1}{2} \sum_{K=1}^{N} \left(\overline{Q}_{ij}\right)_k \left(Z_k^2 - Z_{k-1}^2\right)$ Coupling Stiffnesses

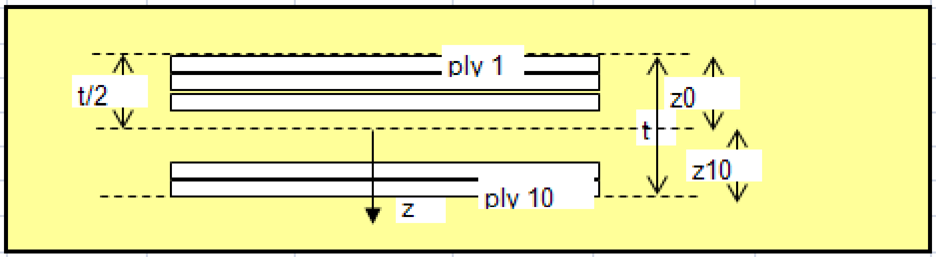
$D_{ij} = \frac{1}{3} \sum_{K=1}^{N} \left(\overline{Q}_{ij}\right)_k \left(Z_k^3 - Z_{k-1}^3\right)$ Bending Stiffnesses

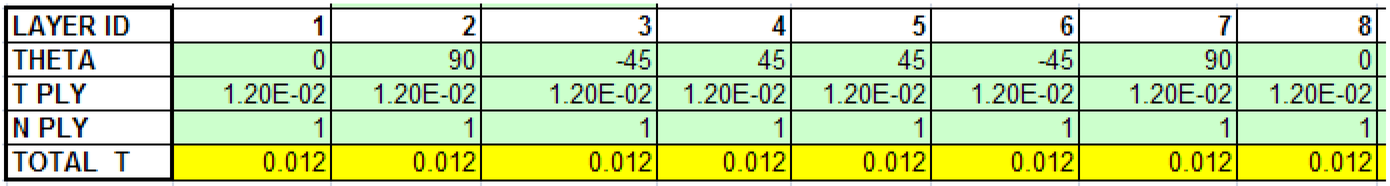
2. How do composites vary from metallic structures?

Overview of material properties and the ABD matrix terms. Hints on practical design methods.



Input Properties	
E1	2.00E+07
E2	5.00E+05
NU12	0.25
G12	2.50E+05

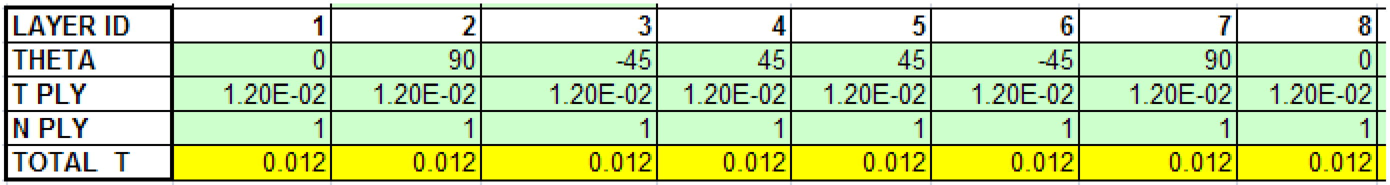




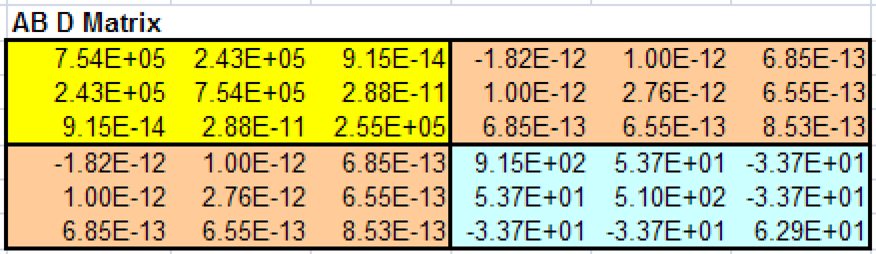
LAYER ID	1	2	3	4	5	6	7	8
THETA	0	90	-45	45	45	-45	90	0
T PLY	1.20E-02	1.20E-02	1.20E-02	1.20E-02	1.20E-02	1.20E-02	1.20E-02	1.20E-02
N PLY	1	1	1	1	1	1	1	1
TOTAL T	0.012	0.012	0.012	0.012	0.012	0.012	0.012	0.012

2. How do composites vary from metallic structures?

Overview of material properties and the ABD matrix terms. Hints on practical design methods.



LAYER ID	1	2	3	4	5	6	7	8
THETA	0	90	-45	45	45	-45	90	0
T PLY	1.20E-02	1.20E-02	1.20E-02	1.20E-02	1.20E-02	1.20E-02	1.20E-02	1.20E-02
N PLY	1	1	1	1	1	1	1	1
TOTAL T	0.012	0.012	0.012	0.012	0.012	0.012	0.012	0.012

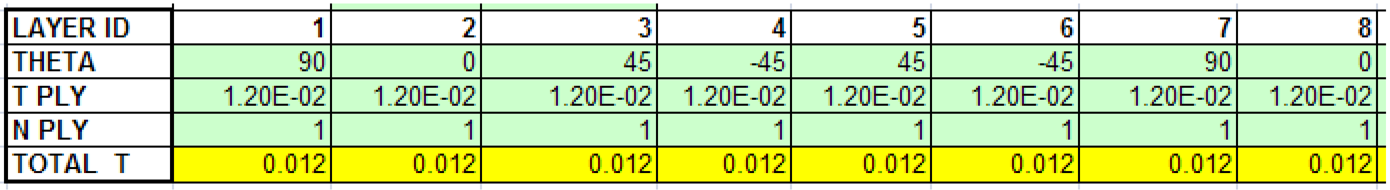


AB D Matrix

7.54E+05	2.43E+05	9.15E-14	-1.82E-12	1.00E-12	6.85E-13
2.43E+05	7.54E+05	2.88E-11	1.00E-12	2.76E-12	6.55E-13
9.15E-14	2.88E-11	2.55E+05	6.85E-13	6.55E-13	8.53E-13
-1.82E-12	1.00E-12	6.85E-13	9.15E+02	5.37E+01	-3.37E+01
1.00E-12	2.76E-12	6.55E-13	5.37E+01	5.10E+02	-3.37E+01
6.85E-13	6.55E-13	8.53E-13	-3.37E+01	-3.37E+01	6.29E+01

2. How do composites vary from metallic structures?

Overview of material properties and the ABD matrix terms. Hints on practical design methods.



LAYER ID	1	2	3	4	5	6	7	8
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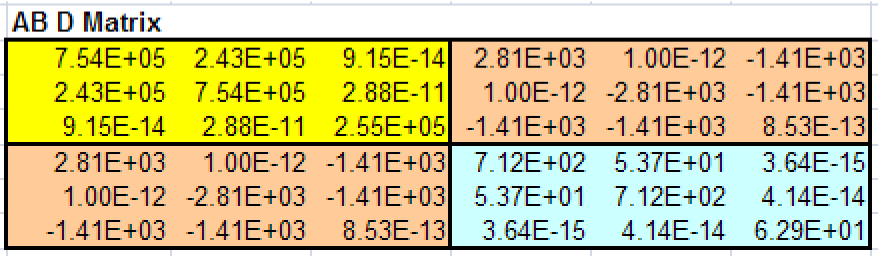
T PLY	1.20E-02	1.20E-02	1.20E-02	1.20E-02	1.20E-02	1.20E-02	1.20E-02	1.20E-02
N PLY	1	1	1	1	1	1	1	1
TOTAL T	0.012	0.012	0.012	0.012	0.012	0.012	0.012	0.012



AB D Matrix

7.54E+05	2.43E+05	9.15E-14	2.81E+03	1.00E-12	-1.41E+03
2.43E+05	7.54E+05	2.88E-11	1.00E-12	-2.81E+03	-1.41E+03
9.15E-14	2.88E-11	2.55E+05	-1.41E+03	-1.41E+03	8.53E-13
2.81E+03	1.00E-12	-1.41E+03	7.12E+02	5.37E+01	3.64E-15
1.00E-12	-2.81E+03	-1.41E+03	5.37E+01	7.12E+02	4.14E-14
-1.41E+03	-1.41E+03	8.53E-13	3.64E-15	4.14E-14	6.29E+01

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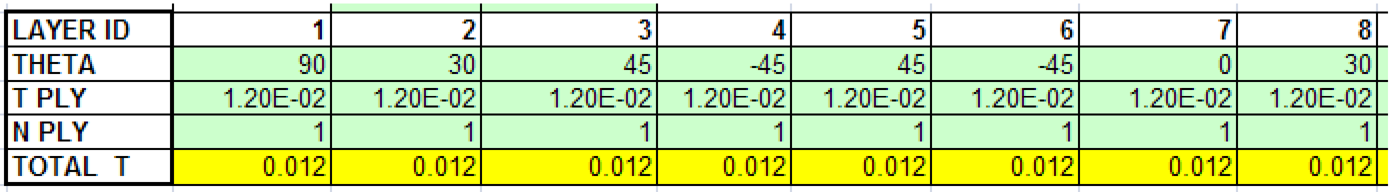
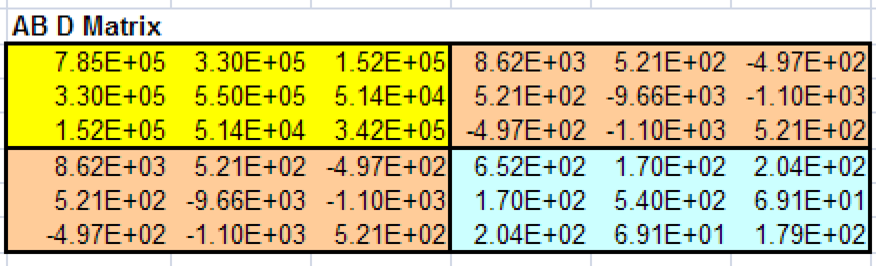
Overview of material properties and the ABD matrix terms. Hints on practical design methods.



LAYER ID	1	2	3	4	5	6	7	8
THETA	90	30	45	-45	45	-45	0	30
T PLY	1.20E-02	1.20E-02	1.20E-02	1.20E-02	1.20E-02	1.20E-02	1.20E-02	1.20E-02
N PLY	1	1	1	1	1	1	1	1
TOTAL T	0.012	0.012	0.012	0.012	0.012	0.012	0.012	0.012



AB D Matrix

7.85E+05	3.30E+05	1.52E+05	8.62E+03	5.21E+02	-4.97E+02
3.30E+05	5.50E+05	5.14E+04	5.21E+02	-9.66E+03	-1.10E+03
1.52E+05	5.14E+04	3.42E+05	-4.97E+02	-1.10E+03	5.21E+02
8.62E+03	5.21E+02	-4.97E+02	6.52E+02	1.70E+02	2.04E+02
5.21E+02	-9.66E+03	-1.10E+03	1.70E+02	5.40E+02	6.91E+01
-4.97E+02	-1.10E+03	5.21E+02	2.04E+02	6.91E+01	1.79E+02

Introductory Composites FE Analysis Webinar

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Review of different forms and manufacturing processes.

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3. How do I set up a composite FEA?

Overview of typical processes. Keeping track of plies, mold lines etc.

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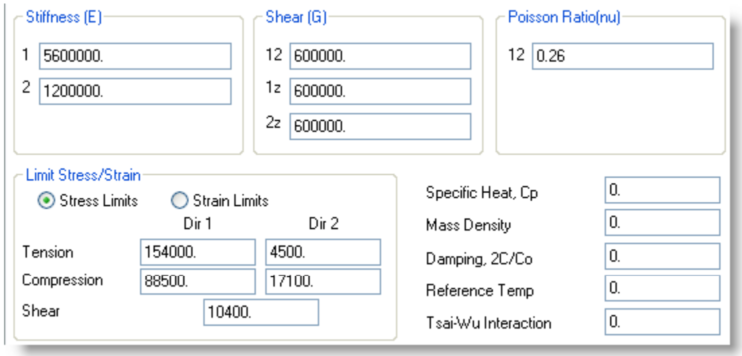
Overview of typical processes. Keeping track of plies, mold lines etc.

material property definitions



Stiffness (E)
1 5600000.
2 1200000.

Shear (G)
12 600000.
1z 600000.
2z 600000.

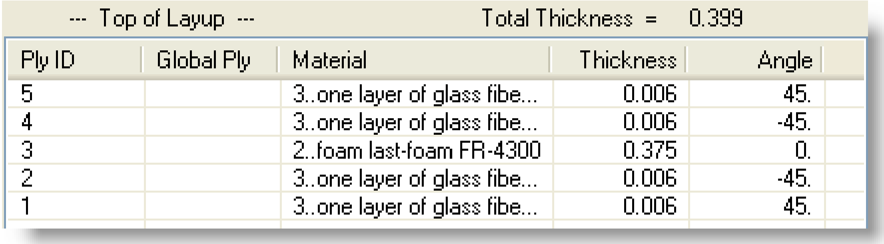
Poisson Ratio(nu)
12 0.26

Limit Stress/Strain
● Stress Limits ○ Strain Limits

	Dir 1	Dir 2
Tension	154000.	4500.
Compression	88500.	17100.
Shear	10400.	

Specific Heat, Cp	0.
Mass Density	0.
Damping, 2C/Co	0.
Reference Temp	0.
Tsai-Wu Interaction	0.

Component ply definitions

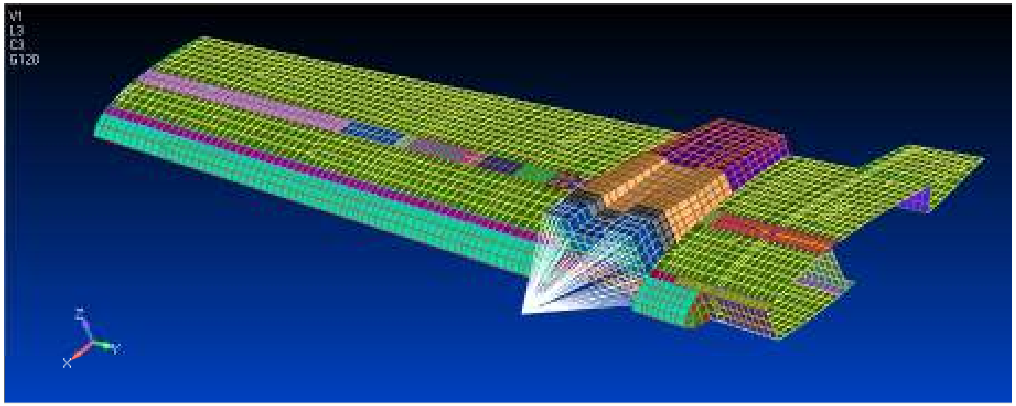


--- Top of Layup --- Total Thickness = 0.399

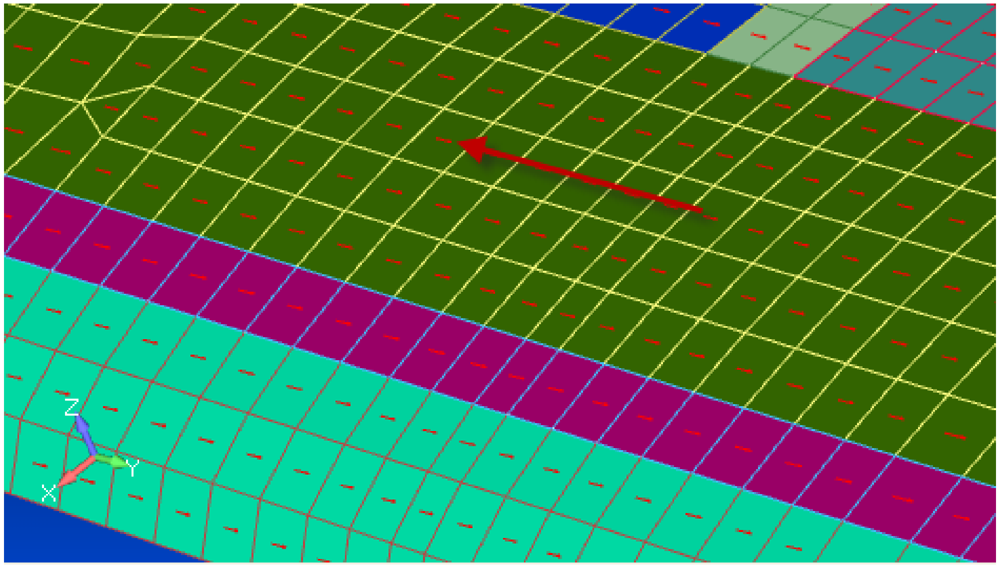
Ply ID	Global Ply	Material	Thickness	Angle
5		3..one layer of glass fibe...	0.006	45.
4		3..one layer of glass fibe...	0.006	-45.
3		2..foam last-foam FR-4300	0.375	0.
2		3..one layer of glass fibe...	0.006	-45.
1		3..one layer of glass fibe...	0.006	45.

3. How do I set up a composite FEA?

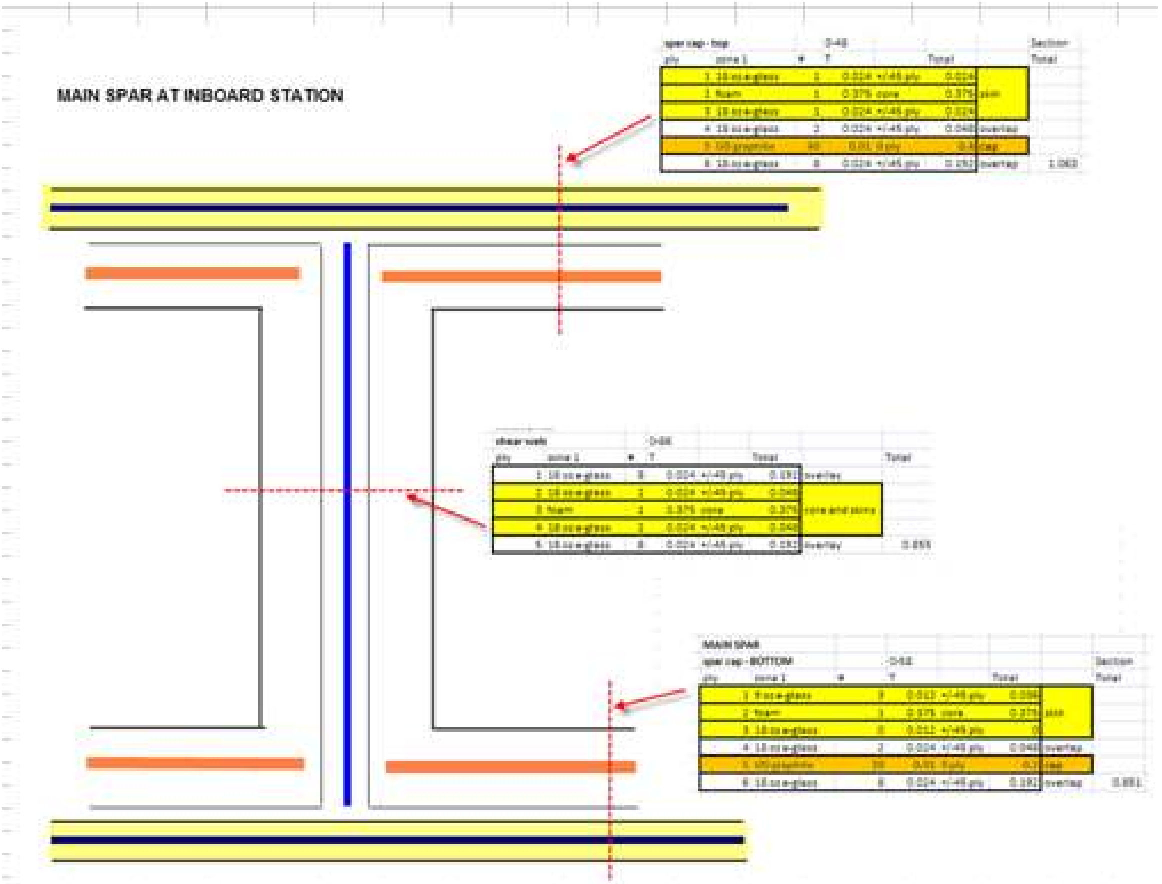
Meshing



Setting up ply orientation

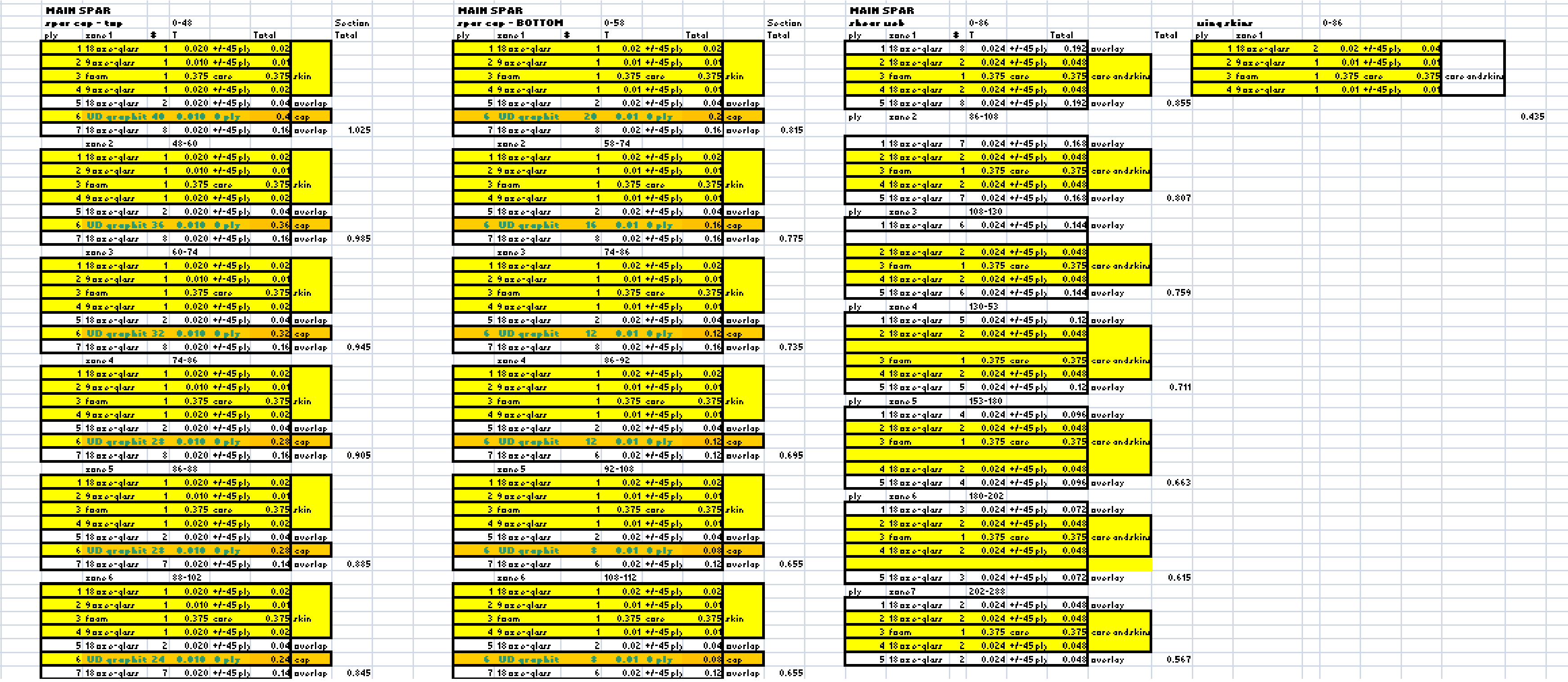


3. How do I set up a composite FEA?



Clear strategy needed to control order of ply layup to represent manufacturing process

3. How do I set up a composite FEA?



MAIN SPAR							
spar cap - top			0-48				Section
ply	zone 1	#	T		Total		Total
1	18 oz e-glass	1	0.020	+/-45 ply	0.02	skin	
2	9 oz e-glass	1	0.010	+/-45 ply	0.01		
3	foam	1	0.375	core	0.375		
4	9 oz e-glass	1	0.020	+/-45 ply	0.02		
5	18 oz e-glass	2	0.020	+/-45 ply	0.04	overlap	
6	UD graphit 40		0.010	0 ply	0.4	cap	
7	18 oz e-glass	8	0.020	+/-45 ply	0.16	overlap	1.025
	zone 2		48-60				
1	18 oz e-glass	1	0.020	+/-45 ply	0.02	skin	
2	9 oz e-glass	1	0.010	+/-45 ply	0.01		
3	foam	1	0.375	core	0.375		
4	9 oz e-glass	1	0.020	+/-45 ply	0.02		
5	18 oz e-glass	2	0.020	+/-45 ply	0.04	overlap	
6	UD graphit 36		0.010	0 ply	0.36	cap	
7	18 oz e-glass	8	0.020	+/-45 ply	0.16	overlap	0.985
	zone 3		60-74				
1	18 oz e-glass	1	0.020	+/-45 ply	0.02	skin	
2	9 oz e-glass	1	0.010	+/-45 ply	0.01		
3	foam	1	0.375	core	0.375		
4	9 oz e-glass	1	0.020	+/-45 ply	0.02		
5	18 oz e-glass	2	0.020	+/-45 ply	0.04	overlap	
6	UD graphit 32		0.010	0 ply	0.32	cap	
7	18 oz e-glass	8	0.020	+/-45 ply	0.16	overlap	0.945
	zone 4		74-86				
1	18 oz e-glass	1	0.020	+/-45 ply	0.02	skin	
2	9 oz e-glass	1	0.010	+/-45 ply	0.01		
3	foam	1	0.375	core	0.375		
4	9 oz e-glass	1	0.020	+/-45 ply	0.02		
5	18 oz e-glass	2	0.020	+/-45 ply	0.04	overlap	
6	UD graphit 28		0.010	0 ply	0.28	cap	
7	18 oz e-glass	8	0.020	+/-45 ply	0.16	overlap	0.905
	zone 5		86-88				
1	18 oz e-glass	1	0.020	+/-45 ply	0.02	skin	
2	9 oz e-glass	1	0.010	+/-45 ply	0.01		
3	foam	1	0.375	core	0.375		
4	9 oz e-glass	1	0.020	+/-45 ply	0.02		
5	18 oz e-glass	2	0.020	+/-45 ply	0.04	overlap	
6	UD graphit 28		0.010	0 ply	0.28	cap	
7	18 oz e-glass	7	0.020	+/-45 ply	0.14	overlap	0.885
	zone 6		88-102				
1	18 oz e-glass	1	0.020	+/-45 ply	0.02	skin	
2	9 oz e-glass	1	0.010	+/-45 ply	0.01		
3	foam	1	0.375	core	0.375		
4	9 oz e-glass	1	0.020	+/-45 ply	0.02		
5	18 oz e-glass	2	0.020	+/-45 ply	0.04	overlap	
6	UD graphit 24		0.010	0 ply	0.24	cap	
7	18 oz e-glass	7	0.020	+/-45 ply	0.14	overlap	0.845

MAIN SPAR							
spar cap - BOTTOM			0-58				Section
ply	zone 1	#	T		Total		Total
1	18 oz e-glass	1	0.02	+/-45 ply	0.02	skin	
2	9 oz e-glass	1	0.01	+/-45 ply	0.01		
3	foam	1	0.375	core	0.375		
4	9 oz e-glass	1	0.01	+/-45 ply	0.01		
5	18 oz e-glass	2	0.02	+/-45 ply	0.04	overlap	
6	UD graphit	20	0.01	0 ply	0.2	cap	
7	18 oz e-glass	8	0.02	+/-45 ply	0.16	overlap	0.815
	zone 2		58-74				
1	18 oz e-glass	1	0.02	+/-45 ply	0.02	skin	
2	9 oz e-glass	1	0.01	+/-45 ply	0.01		
3	foam	1	0.375	core	0.375		
4	9 oz e-glass	1	0.01	+/-45 ply	0.01		
5	18 oz e-glass	2	0.02	+/-45 ply	0.04	overlap	
6	UD graphit	16	0.01	0 ply	0.16	cap	
7	18 oz e-glass	8	0.02	+/-45 ply	0.16	overlap	0.775
	zone 3		74-86				
1	18 oz e-glass	1	0.02	+/-45 ply	0.02	skin	
2	9 oz e-glass	1	0.01	+/-45 ply	0.01		
3	foam	1	0.375	core	0.375		
4	9 oz e-glass	1	0.01	+/-45 ply	0.01		
5	18 oz e-glass	2	0.02	+/-45 ply	0.04	overlap	
6	UD graphit	12	0.01	0 ply	0.12	cap	
7	18 oz e-glass	8	0.02	+/-45 ply	0.16	overlap	0.735
	zone 4		86-92				
1	18 oz e-glass	1	0.02	+/-45 ply	0.02	skin	
2	9 oz e-glass	1	0.01	+/-45 ply	0.01		
3	foam	1	0.375	core	0.375		
4	9 oz e-glass	1	0.01	+/-45 ply	0.01		
5	18 oz e-glass	2	0.02	+/-45 ply	0.04	overlap	
6	UD graphit	12	0.01	0 ply	0.12	cap	
7	18 oz e-glass	6	0.02	+/-45 ply	0.12	overlap	0.695
	zone 5		92-108				
1	18 oz e-glass	1	0.02	+/-45 ply	0.02	skin	
2	9 oz e-glass	1	0.01	+/-45 ply	0.01		
3	foam	1	0.375	core	0.375		
4	9 oz e-glass	1	0.01	+/-45 ply	0.01		
5	18 oz e-glass	2	0.02	+/-45 ply	0.04	overlap	
6	UD graphit	8	0.01	0 ply	0.08	cap	
7	18 oz e-glass	6	0.02	+/-45 ply	0.12	overlap	0.655
	zone 6		108-112				
1	18 oz e-glass	1	0.02	+/-45 ply	0.02	skin	
2	9 oz e-glass	1	0.01	+/-45 ply	0.01		
3	foam	1	0.375	core	0.375		
4	9 oz e-glass	1	0.01	+/-45 ply	0.01		
5	18 oz e-glass	2	0.02	+/-45 ply	0.04	overlap	
6	UD graphit	8	0.01	0 ply	0.08	cap	
7	18 oz e-glass	6	0.02	+/-45 ply	0.12	overlap	0.655

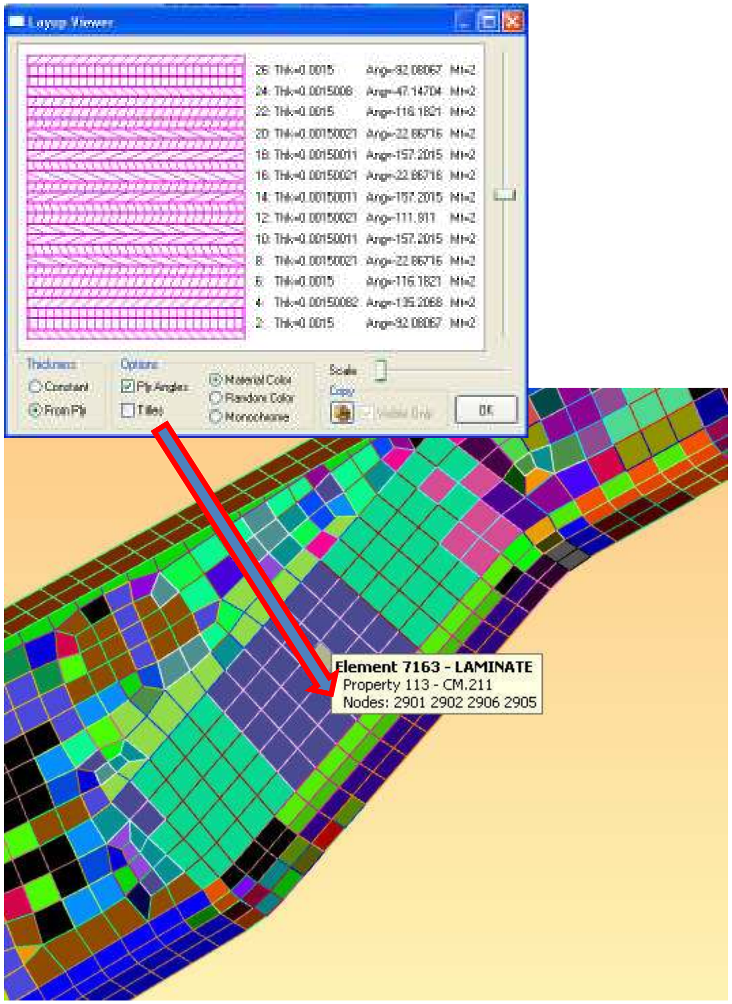
MAIN SPAR							
shear web			0-86				
ply	zone 1	#	T		Total		Total
1	18 oz e-glass	8	0.024	+/-45 ply	0.192	overlay	
2	18 oz e-glass	2	0.024	+/-45 ply	0.048	core and skin	
3	foam	1	0.375	core	0.375		
4	18 oz e-glass	2	0.024	+/-45 ply	0.048		
5	18 oz e-glass	8	0.024	+/-45 ply	0.192	overlay	0.855
ply	zone 2		86-108				
1	18 oz e-glass	7	0.024	+/-45 ply	0.168	overlay	
2	18 oz e-glass	2	0.024	+/-45 ply	0.048	core and skin	
3	foam	1	0.375	core	0.375		
4	18 oz e-glass	2	0.024	+/-45 ply	0.048		
5	18 oz e-glass	7	0.024	+/-45 ply	0.168	overlay	0.807
ply	zone 3		108-130				
1	18 oz e-glass	6	0.024	+/-45 ply	0.144	overlay	
2	18 oz e-glass	2	0.024	+/-45 ply	0.048	core and skin	
3	foam	1	0.375	core	0.375		
4	18 oz e-glass	2	0.024	+/-45 ply	0.048		
5	18 oz e-glass	6	0.024	+/-45 ply	0.144	overlay	0.759
ply	zone 4		130-53				
1	18 oz e-glass	5	0.024	+/-45 ply	0.12	overlay	
2	18 oz e-glass	2	0.024	+/-45 ply	0.048	core and skin	
3	foam	1	0.375	core	0.375		
4	18 oz e-glass	2	0.024	+/-45 ply	0.048		
5	18 oz e-glass	5	0.024	+/-45 ply	0.12	overlay	0.711
ply	zone 5		153-180				
1	18 oz e-glass	4	0.024	+/-45 ply	0.096	overlay	
2	18 oz e-glass	2	0.024	+/-45 ply	0.048	core and skin	
3	foam	1	0.375	core	0.375		
4	18 oz e-glass	2	0.024	+/-45 ply	0.048		
5	18 oz e-glass	4	0.024	+/-45 ply	0.096	overlay	0.663
ply	zone 6		180-202				
1	18 oz e-glass	3	0.024	+/-45 ply	0.072	overlay	
2	18 oz e-glass	2	0.024	+/-45 ply	0.048	core and skin	
3	foam	1	0.375	core	0.375		
4	18 oz e-glass	2	0.024	+/-45 ply	0.048		
5	18 oz e-glass	3	0.024	+/-45 ply	0.072	overlay	0.615
ply	zone 7		202-288				
1	18 oz e-glass	2	0.024	+/-45 ply	0.048	overlay	
2	18 oz e-glass	2	0.024	+/-45 ply	0.048	core and skin	
3	foam	1	0.375	core	0.375		
4	18 oz e-glass	2	0.024	+/-45 ply	0.048		
5	18 oz e-glass	2	0.024	+/-45 ply	0.048	overlay	0.567

wing skins			0-86				
ply	zone 1						
1	18 oz e-glass	2	0.02	+/-45 ply	0.04	core and skin	
2	9 oz e-glass	1	0.01	+/-45 ply	0.01		
3	foam	1	0.375	core	0.375		
4	9 oz e-glass	1	0.01	+/-45 ply	0.01		
							0.435

Good book keeping is essential !

- Either via spreadsheet
- Or FE software tools

4. How good is my FEA idealization?



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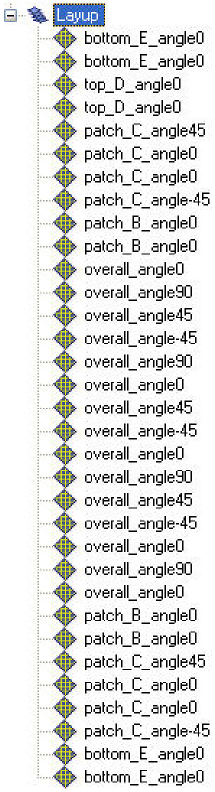
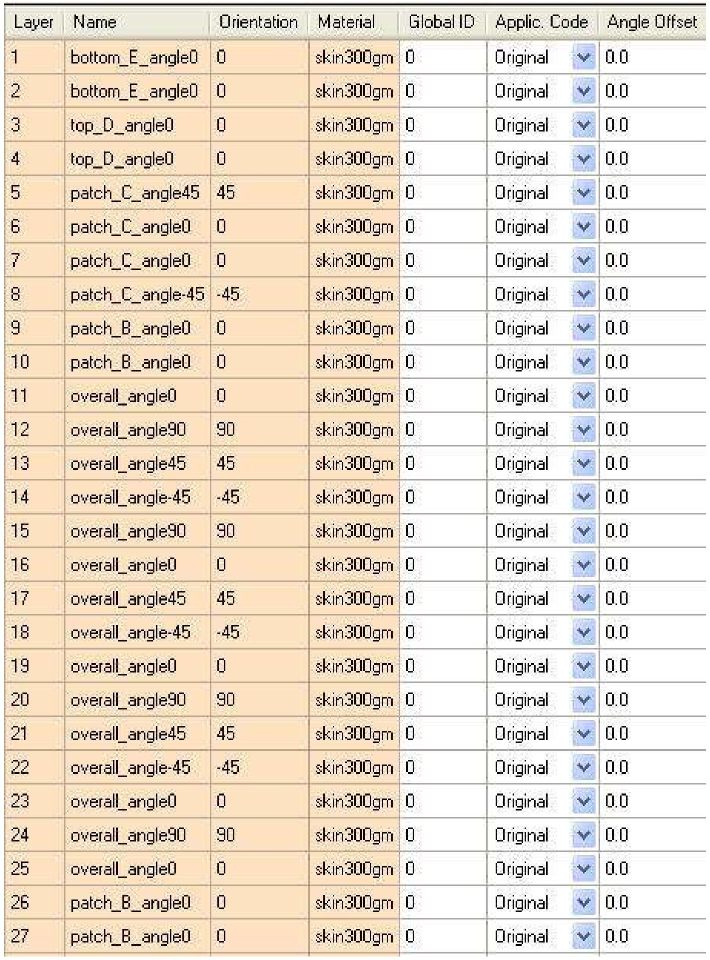
26: Thk=0.0015        Ang=92.08067  Mt=2
24: Thk=0.0015008     Ang=47.14704  Mt=2
22: Thk=0.0015        Ang=116.1821  Mt=2
20: Thk=0.00150021    Ang=22.86716  Mt=2
18: Thk=0.00150011    Ang=157.2015  Mt=2
16: Thk=0.00150021    Ang=22.86716  Mt=2
14: Thk=0.00150011    Ang=157.2015  Mt=2
12: Thk=0.00150021    Ang=111.911   Mt=2
10: Thk=0.00150011    Ang=157.2015  Mt=2
8:  Thk=0.00150021    Ang=22.86716  Mt=2
6:  Thk=0.0015        Ang=116.1821  Mt=2
4:  Thk=0.00150082    Ang=135.2068  Mt=2
2:  Thk=0.0015        Ang=92.08067  Mt=2
```

Element 7163 - LAMINATE
Property 113 - CM.211
Nodes: 2901 2902 2906 2905

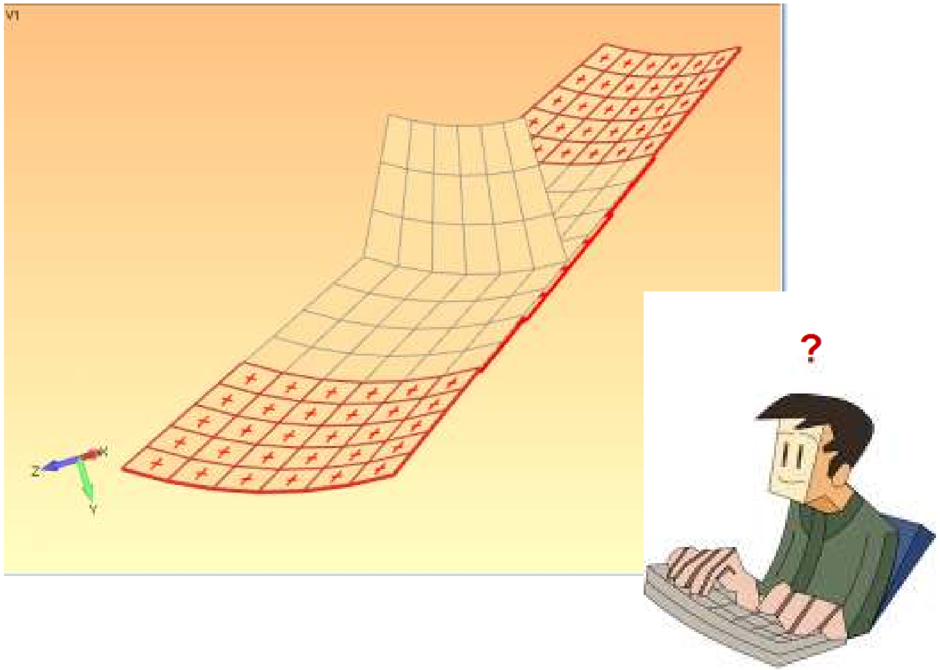


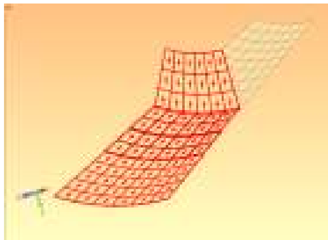
- Layup
  - bottom_E_angle0
  - bottom_E_angle0
  - top_D_angle0
  - top_D_angle0
  - patch_C_angle45
  - patch_C_angle0
  - patch_C_angle0
  - patch_C_angle-45
  - patch_B_angle0
  - patch_B_angle0
  - overall_angle0
  - overall_angle90
  - overall_angle45
  - overall_angle-45
  - overall_angle90
  - overall_angle0
  - overall_angle45
  - overall_angle-45
  - overall_angle0
  - overall_angle90
  - overall_angle45
  - overall_angle-45
  - overall_angle0
  - overall_angle90
  - overall_angle0
  - patch_B_angle0
  - patch_B_angle0
  - patch_C_angle45
  - patch_C_angle0
  - patch_C_angle0
  - patch_C_angle-45
  - bottom_E_angle0
  - bottom_E_angle0

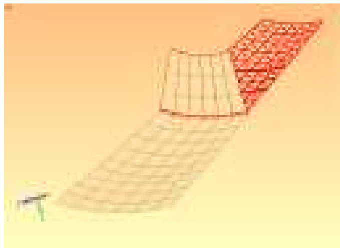


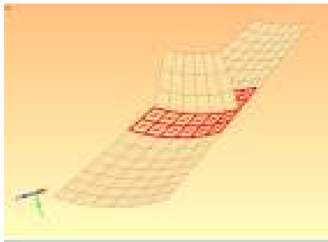
Layer	Name	Orientation	Material	Global ID	Applic. Code	Angle Offset
1	bottom_E_angle0	0	skin300gm	0	Original	0.0
2	bottom_E_angle0	0	skin300gm	0	Original	0.0
3	top_D_angle0	0	skin300gm	0	Original	0.0
4	top_D_angle0	0	skin300gm	0	Original	0.0
5	patch_C_angle45	45	skin300gm	0	Original	0.0
6	patch_C_angle0	0	skin300gm	0	Original	0.0
7	patch_C_angle0	0	skin300gm	0	Original	0.0
8	patch_C_angle-45	-45	skin300gm	0	Original	0.0
9	patch_B_angle0	0	skin300gm	0	Original	0.0
10	patch_B_angle0	0	skin300gm	0	Original	0.0
11	overall_angle0	0	skin300gm	0	Original	0.0
12	overall_angle90	90	skin300gm	0	Original	0.0
13	overall_angle45	45	skin300gm	0	Original	0.0
14	overall_angle-45	-45	skin300gm	0	Original	0.0
15	overall_angle90	90	skin300gm	0	Original	0.0
16	overall_angle0	0	skin300gm	0	Original	0.0
17	overall_angle45	45	skin300gm	0	Original	0.0
18	overall_angle-45	-45	skin300gm	0	Original	0.0
19	overall_angle0	0	skin300gm	0	Original	0.0
20	overall_angle90	90	skin300gm	0	Original	0.0
21	overall_angle45	45	skin300gm	0	Original	0.0
22	overall_angle-45	-45	skin300gm	0	Original	0.0
23	overall_angle0	0	skin300gm	0	Original	0.0
24	overall_angle90	90	skin300gm	0	Original	0.0
25	overall_angle0	0	skin300gm	0	Original	0.0
26	patch_B_angle0	0	skin300gm	0	Original	0.0
27	patch_B_angle0	0	skin300gm	0	Original	0.0

3. How do I set up a composite FEA?

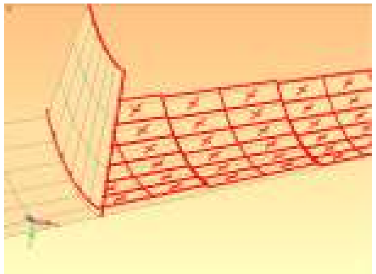




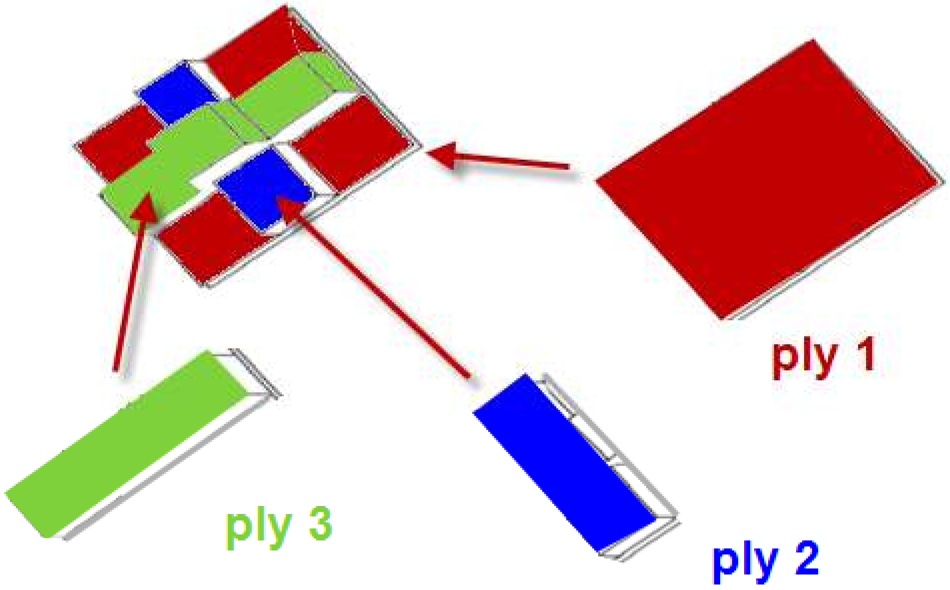


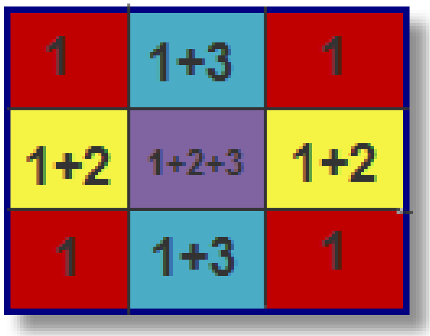


Visualization and control of components and layups is essential in the pre-processor



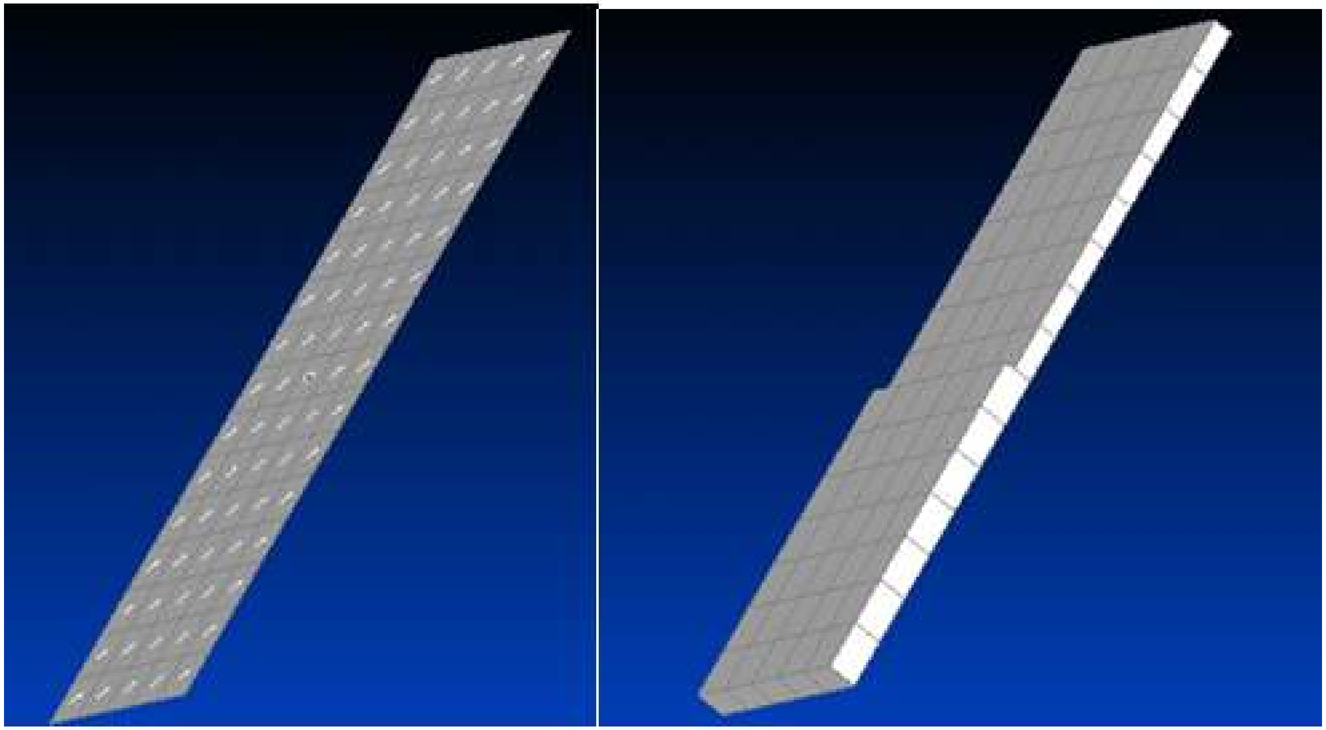
3. How do I set up a composite FEA?





1	1+3	1
1+2	1+2+3	1+2
1	1+3	1

FEA Process for Composite Structure Analysis

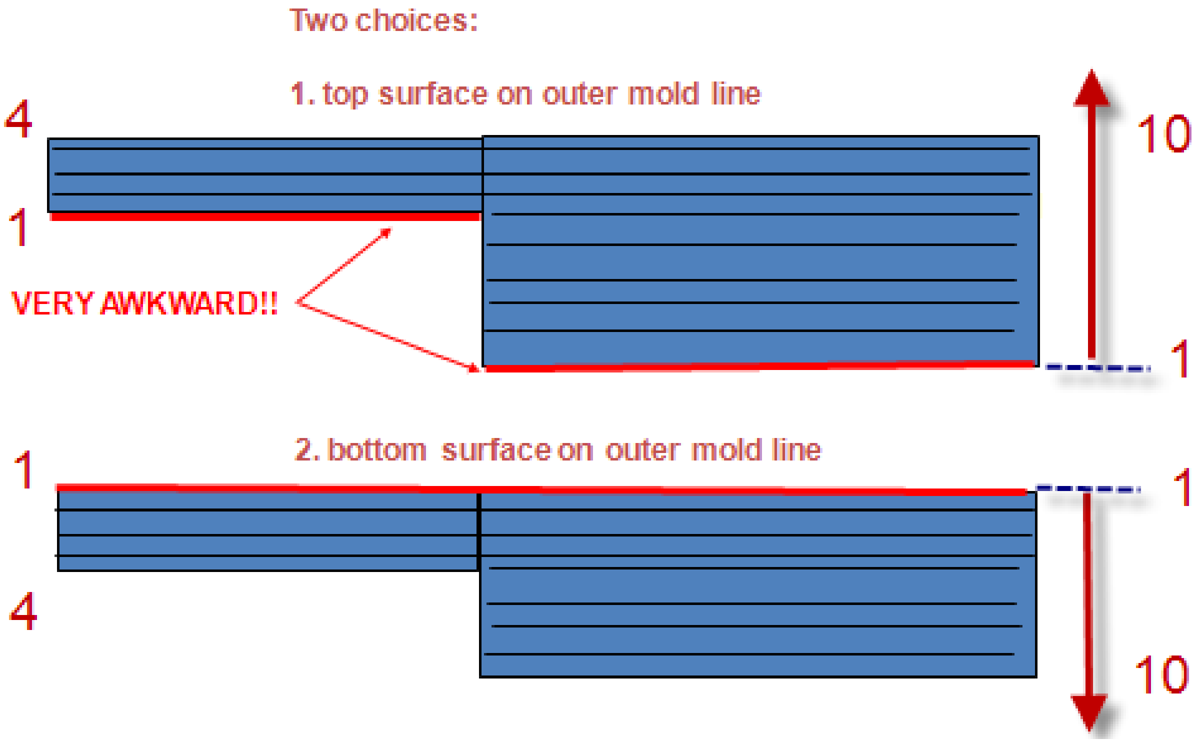


planar view

3D view

Account for Outer/Inner Mold Line continuity

3. How do I set up a composite FEA?



Account for Outer/Inner Mold Line continuity

- **Orientate element normal**
- **Use global ply ids if available**

Introductory Composites FE Analysis Webinar

Agenda

4. How good is my FEA idealization?

The importance of fiber orientation, draping and thickness effects.

5. How do I know whether the composite has failed?

Basic First Ply failure theories

6. How do I organize my results, where do I start looking?

Failure indices, Strength ratios.

4. How good is my FEA idealization?

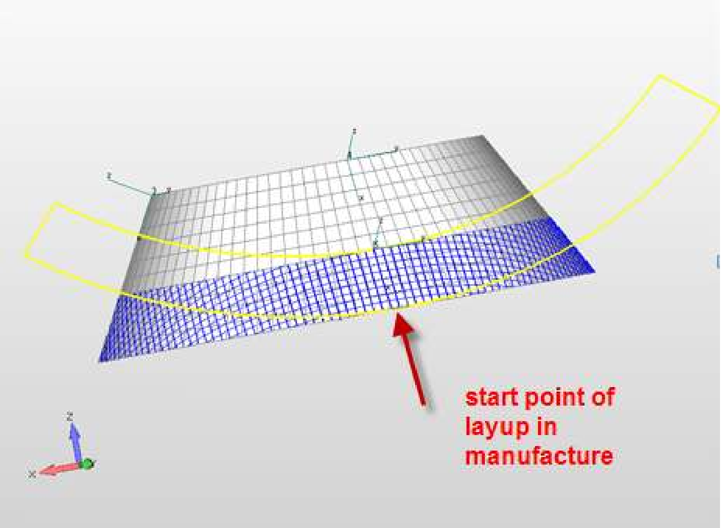
The importance of fiber orientation, draping and thickness effects.

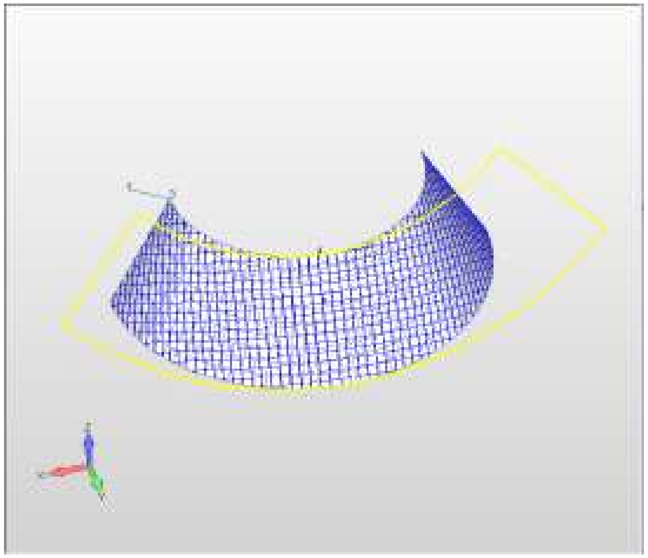
As a ply is draped over a mold it will align the fibers into a net like pattern.

Each fiber would ideally like to form an minimum energy path, rather like a great circle on a globe.

The presence of the adjacent fibers, both in the same ply and throughout the lay up will inhibit this action.

- Two sections of cloth are shown draped over a conical shape.
- The flat pattern required to achieve the lay up geometry is shown





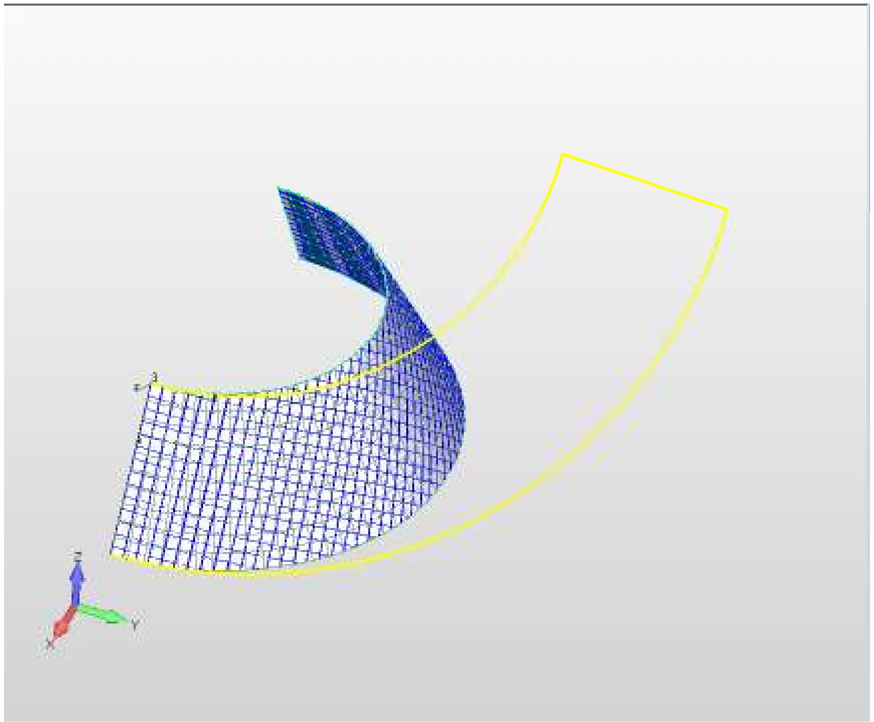
4. How good is my FEA idealization?

The importance of fiber orientation, draping and thickness effects.

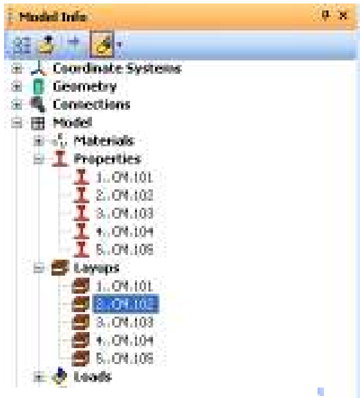
Here is an alternative manufacturing solution

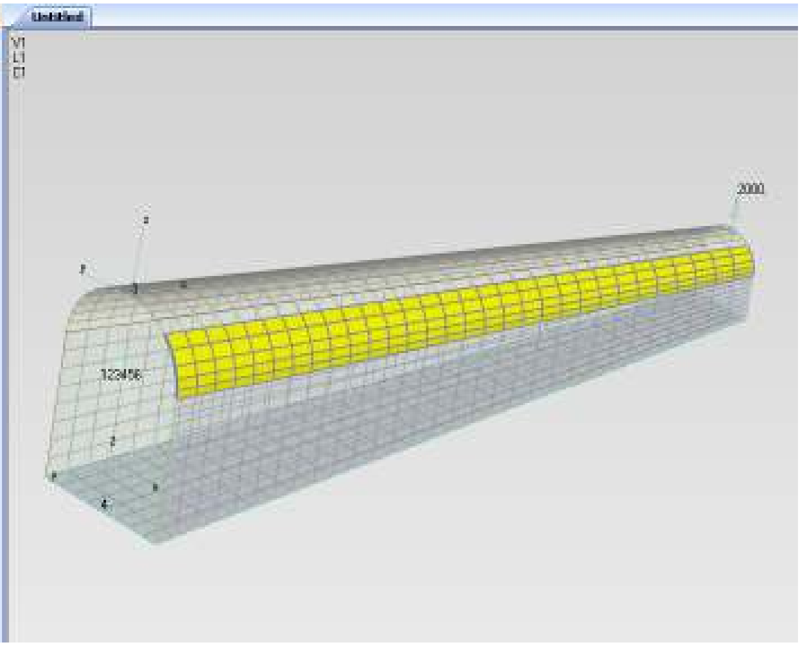
- The layup is orientated to run along one edge
- The flat pattern adjusts to suit
- Notice the drift in angle as we go round the cone

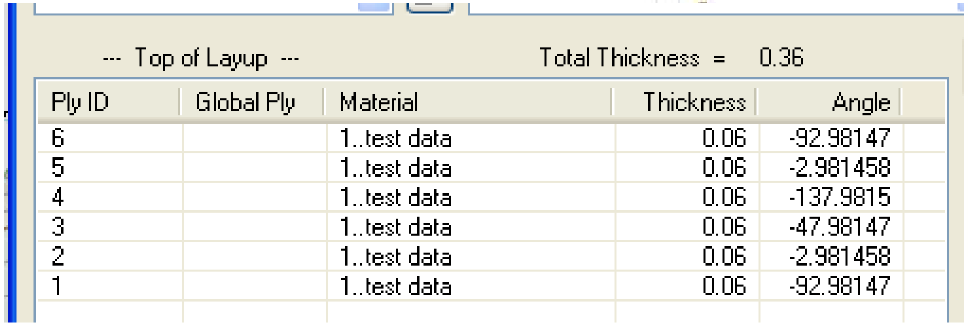
Mapping the change in fiber orientation onto the FE mesh is important



4. How good is my FEA idealization?







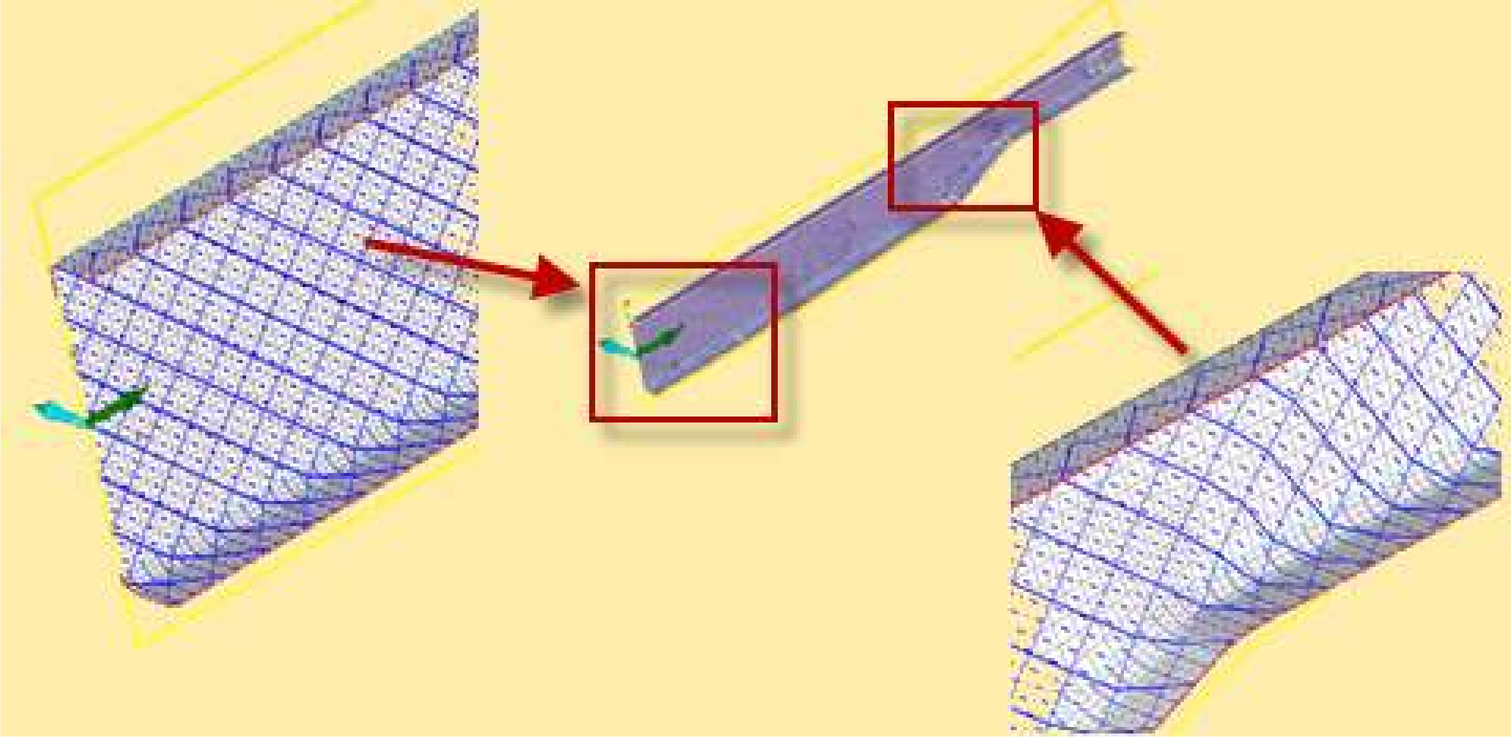
--- Top of Layup ---   Total Thickness = 0.36

Ply ID	Global Ply	Material	Thickness	Angle
6		1..test data	0.06	-92.98147
5		1..test data	0.06	-2.981458
4		1..test data	0.06	-137.9815
3		1..test data	0.06	-47.98147
2		1..test data	0.06	-2.981458
1		1..test data	0.06	-92.98147

Here is a section of an aircraft fuselage

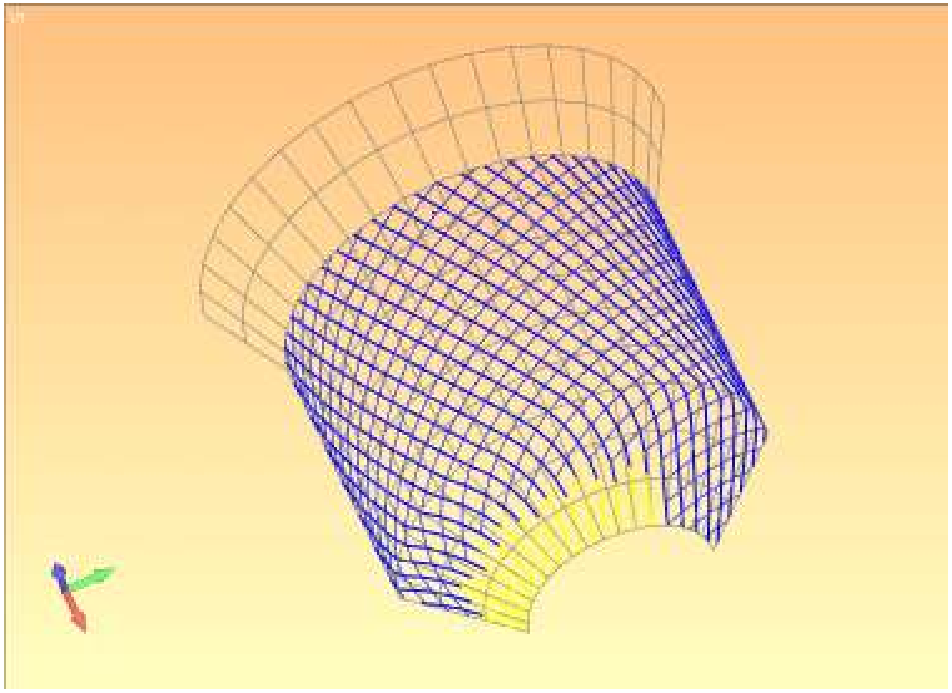
The drift in the ply angles can be seen in the lay up data table

4. How good is my FEA idealization?



The draping around the neck of this component can be clearly seen

4. How good is my FEA idealization?

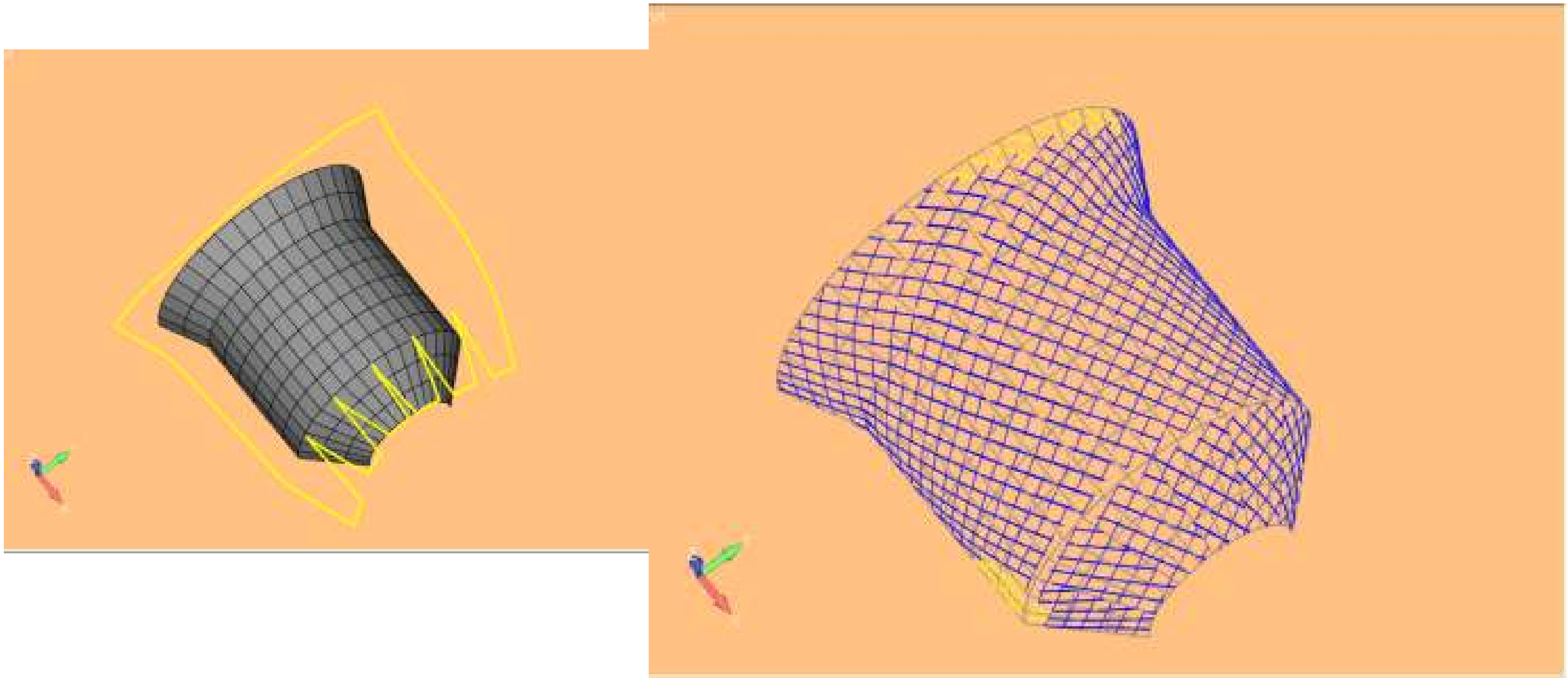


Feasible draping angles depend on the shearing stiffness of the ply as it is laid over the mold.

This will depend on the type of ply – pre-preg or cloth and its mechanical characteristics

Most draping tools will allow the visualization of regions where the shearing action of the layup process is reaching practical limits , or where it is infeasible.

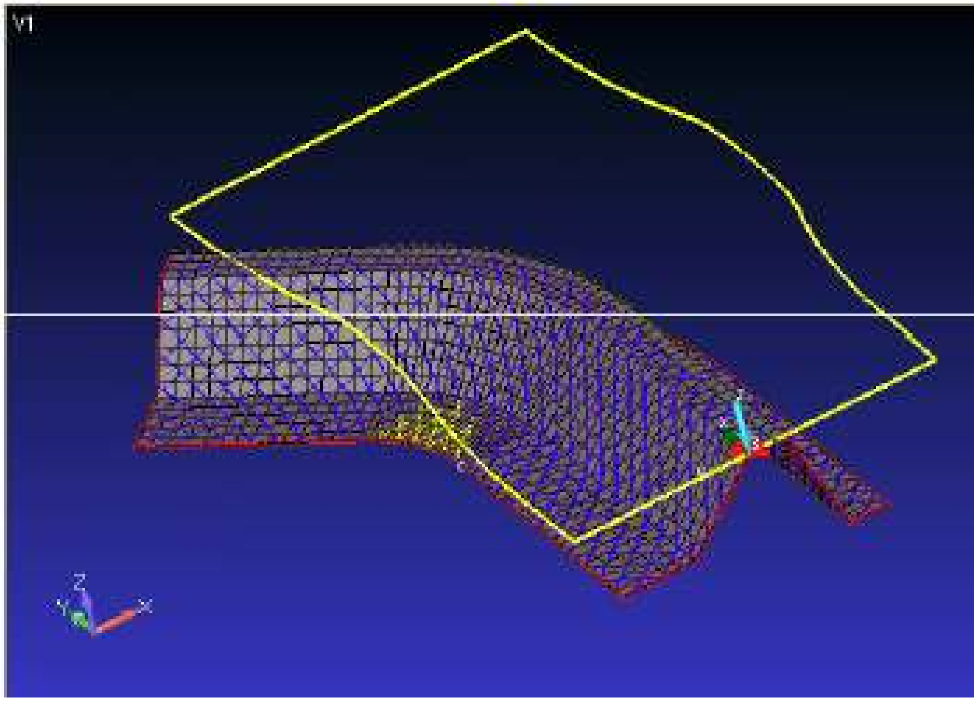
4. How good is my FEA idealization?



Here the analyst has introduced darts into the draping pattern to reduce the shearing action

Note the discontinuous plies

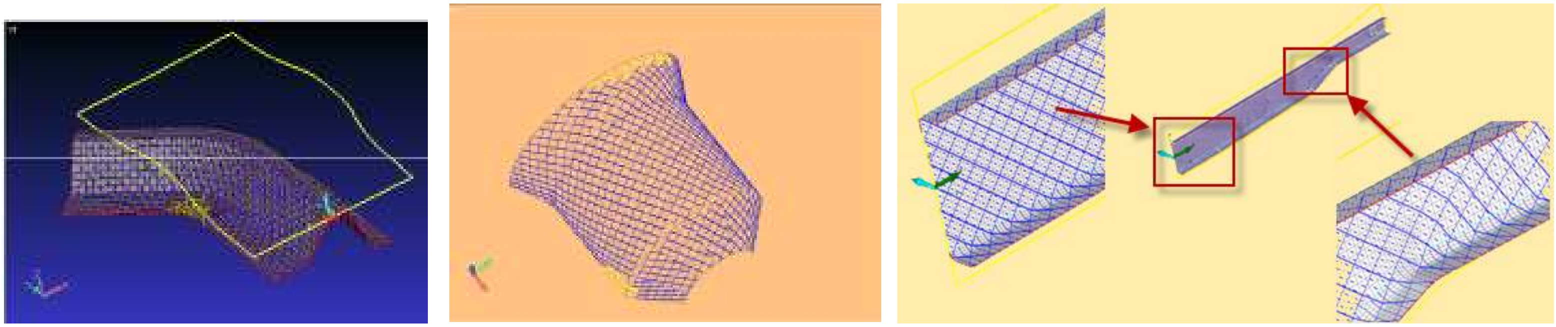
4. How good is my FEA idealization?

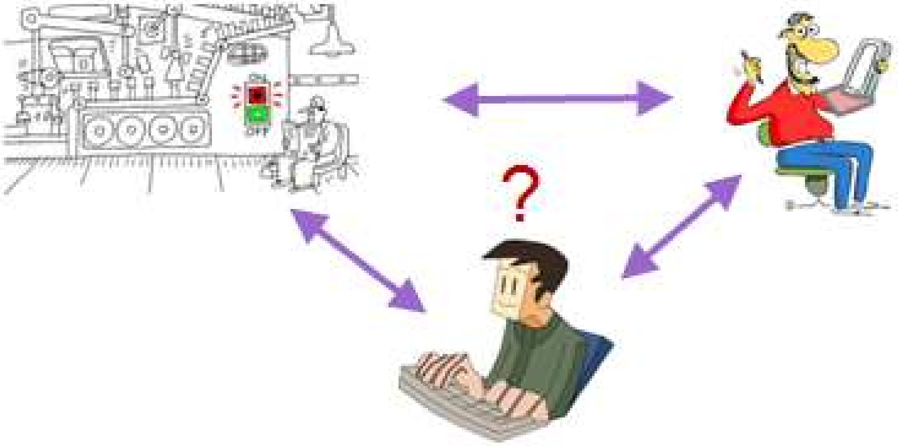


Here the analyst has forced the fibers to follow the flat cap of the stiffener

This could be a specific design intent, or it may follow the known pattern of a pultrusion or molding

4. How good is my FEA idealization?





For this type of FE software to be effective

- the analyst must be in the loop!

Introductory Composites FE Analysis Webinar

Agenda

4. How good is my FEA idealization?

The importance of fiber orientation, draping and thickness effects.

5. How do I know whether the composite has failed?

Basic First Ply failure theories

6. How do I organize my results, where do I start looking?

Failure indices, Strength ratios.

5. How do I know whether the composite has failed?

Basic First Ply failure theories

The initial approach we take in FEA analysis is to assume that as soon as a composite strength value is exceeded by the stress level present, then the structure has failed, or at least is not fit for further service.

This approach is called **First Ply Failure** mode .

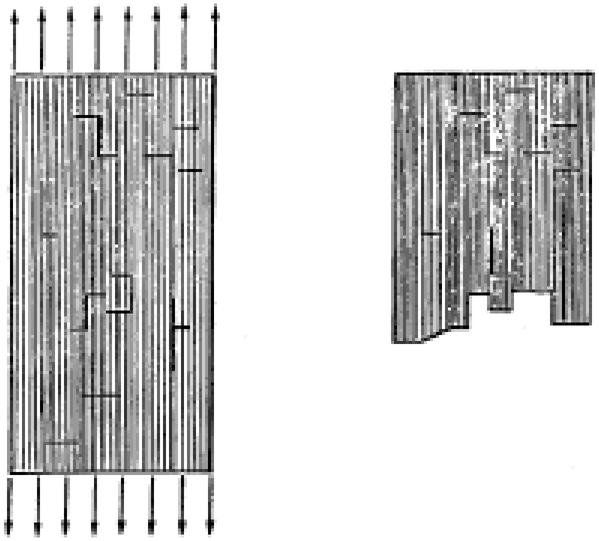
However, it is a far from trivial task to establish the strength of a composite material.

Unlike isotropic materials the strength is dependent on the directional properties of an individual ply, which can vary longitudinally, transversely and in shear, and may well be different between tension and compression.

In addition the ply layup will also control the strength

A great deal of research has been carried out to try to understand the failure mechanisms of a ply

5. How do I know whether the composite has failed?

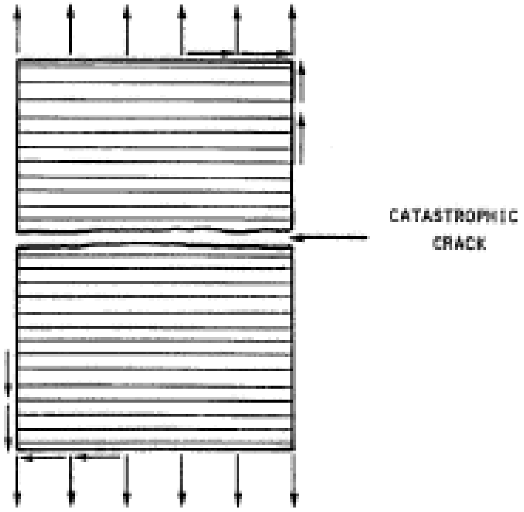


Probably the most intuitive ply failure mode is in tension. The sketch shows a typical failure appearance.

At the microscopic level there is a lot happening, with fiber pull out, fiber breaking and matrix cracking.

However the strength under this loading condition is repeatable for a given as supplied condition within a statistical variation.

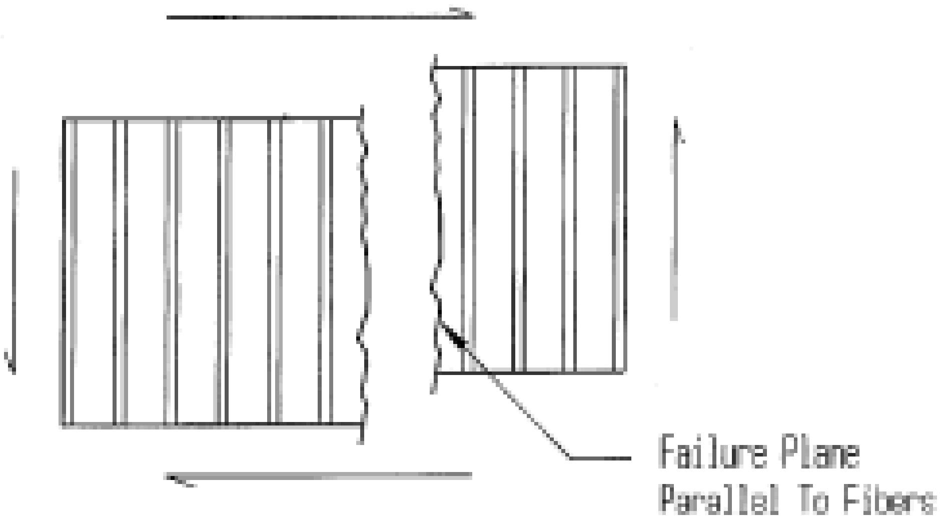
5. How do I know whether the composite has failed?



Similarly the transverse tension is dominated by the strength of the matrix

The microscopic level sees a surprisingly complex behavior with the fibers acting as stress raisers in the matrix stress field.

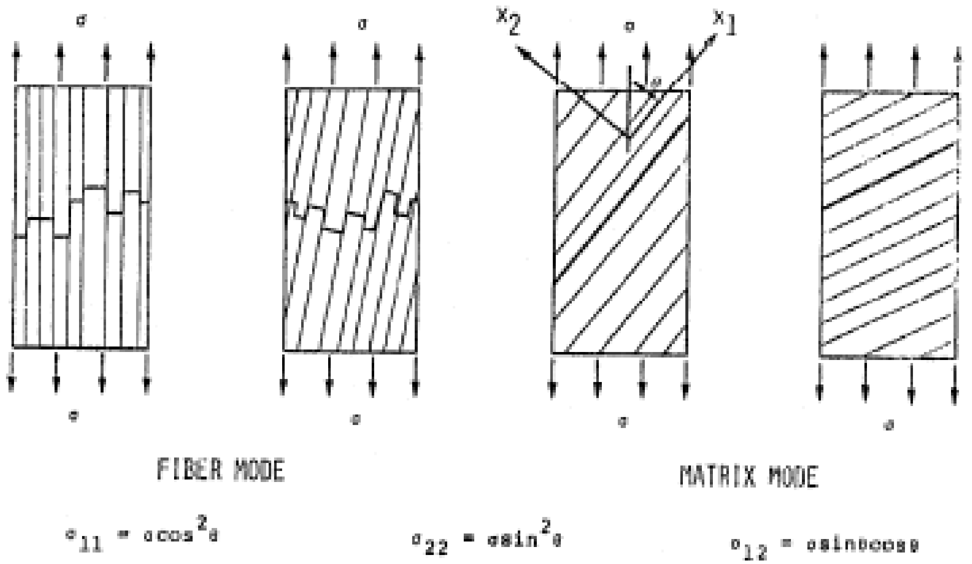
5. How do I know whether the composite has failed?



The failure under in plane shear can be assumed as a shear line failure along the matrix

Again at a microscopic level the behavior is more complex with local cracking behavior building to a total failure

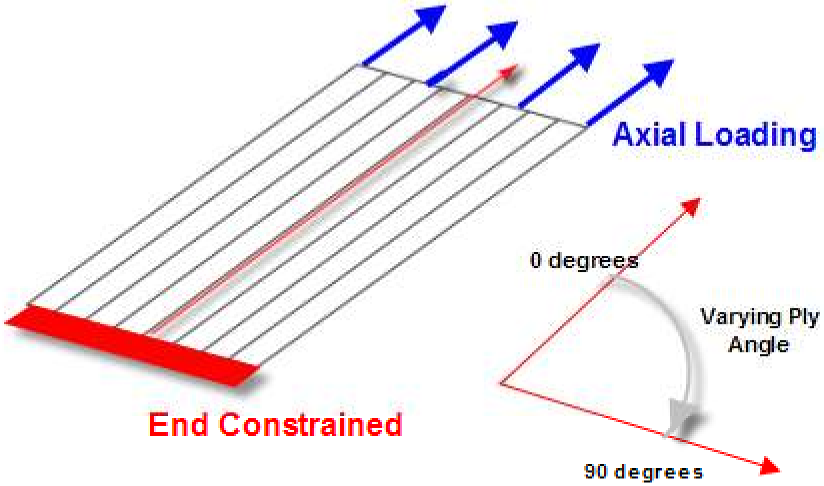
5. How do I know whether the composite has failed?

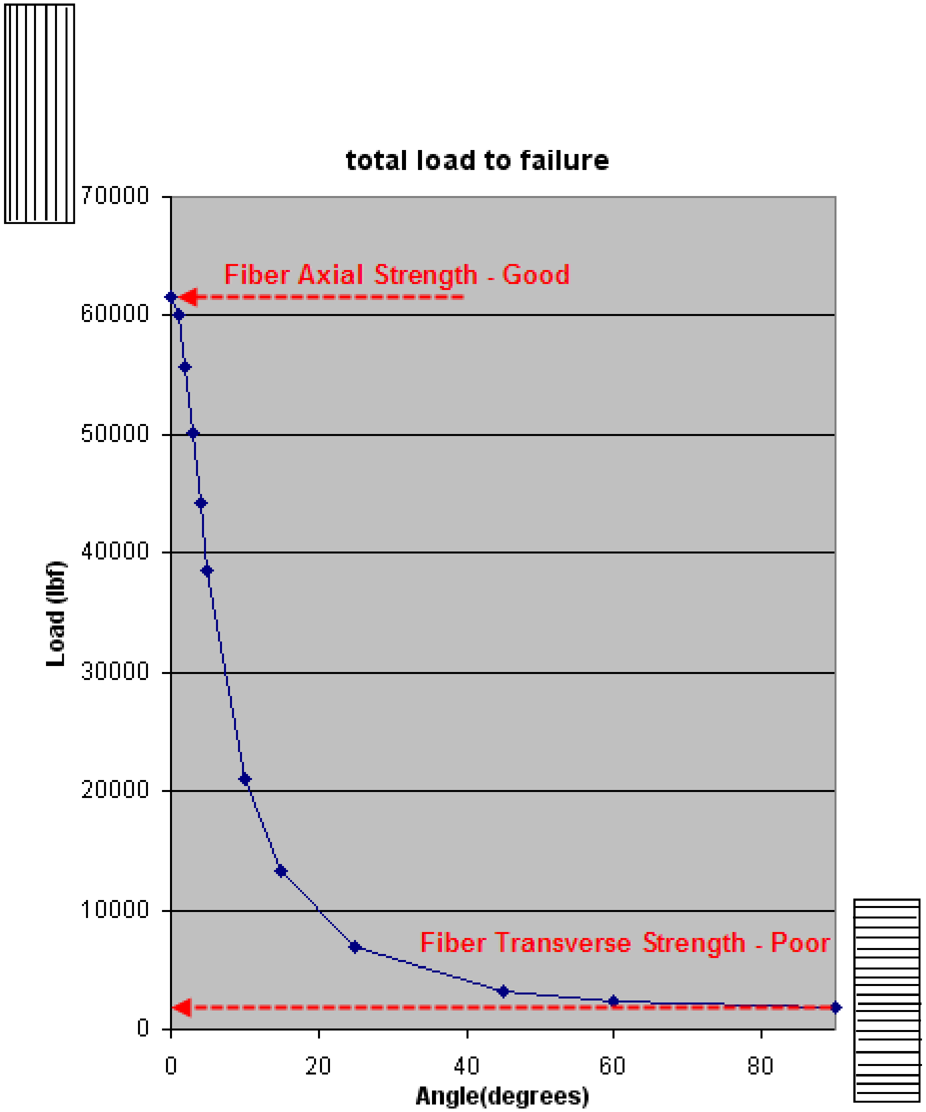


However, in general in the tensile loading quadrant defined by both longitudinal and transverse tension is relatively straight forward

Shear will play a strong role for all off axis loading directions.

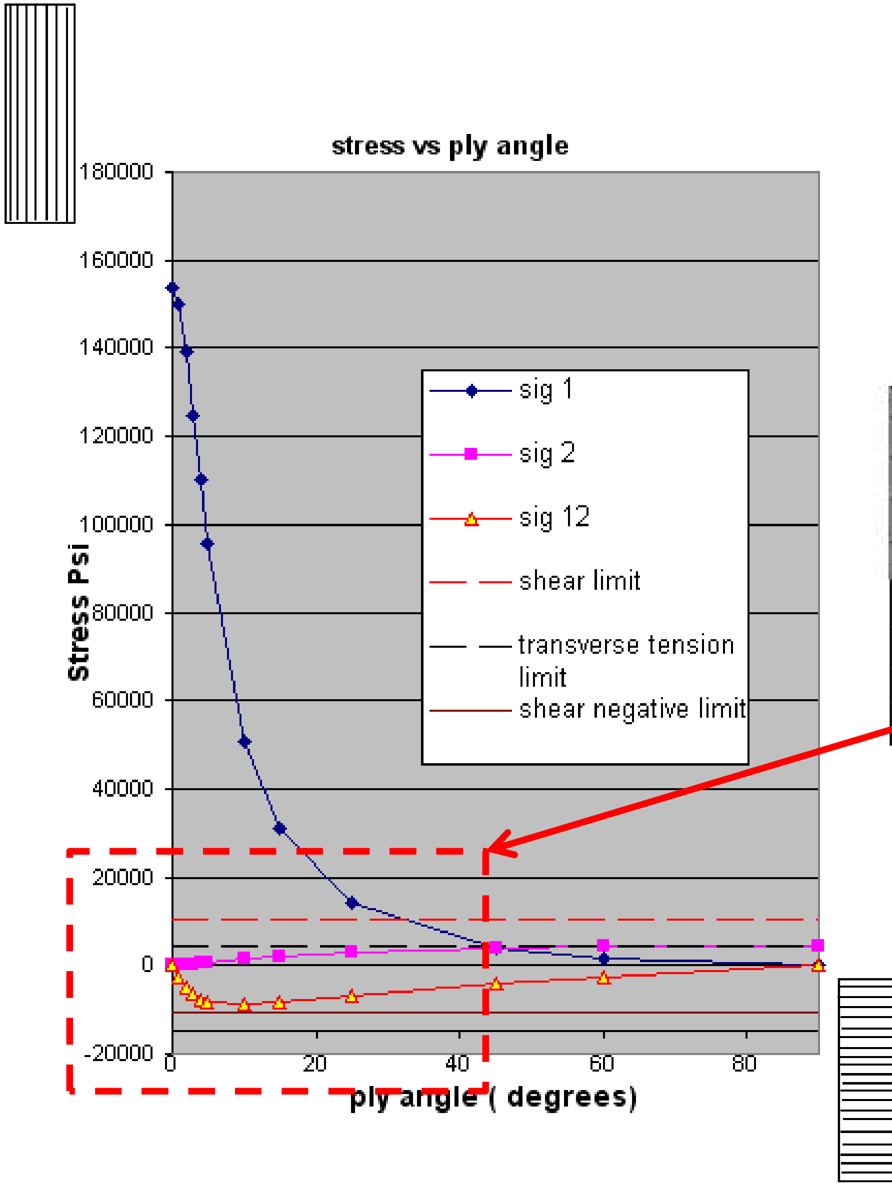
Single Ply

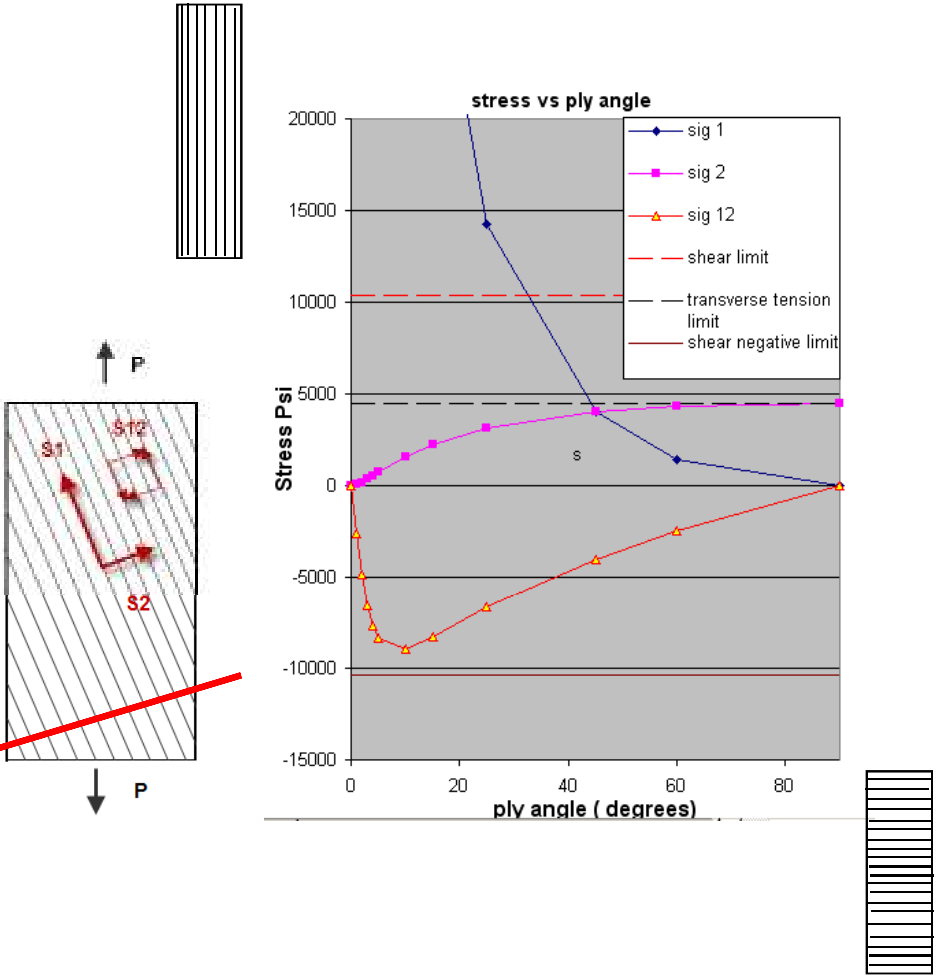




A simple experiment using FEA to predict failing load in a coupon

Graph shows effect of ply angle in a single ply layup

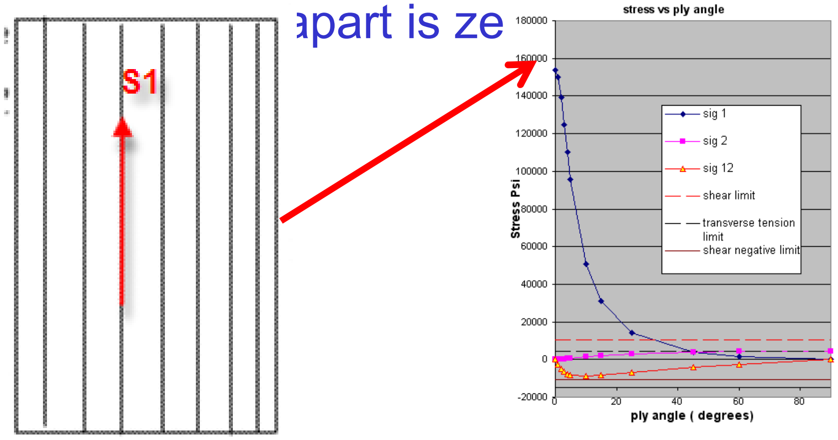




Now the stress components are examined

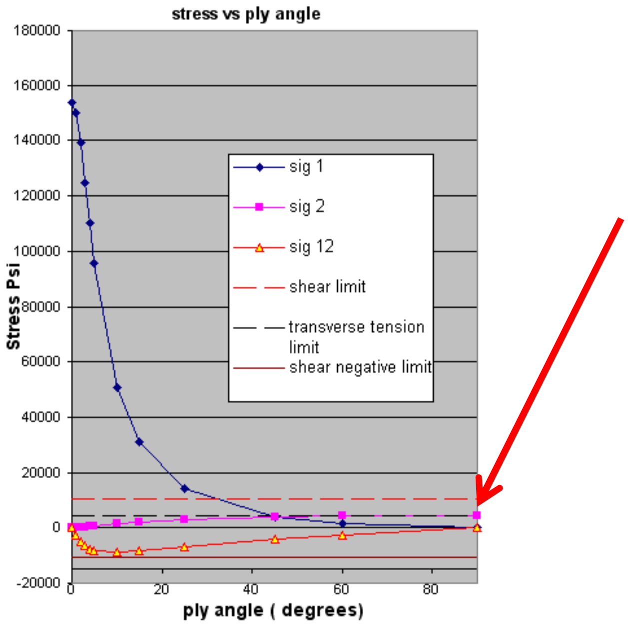
Ply Angle 0 degrees

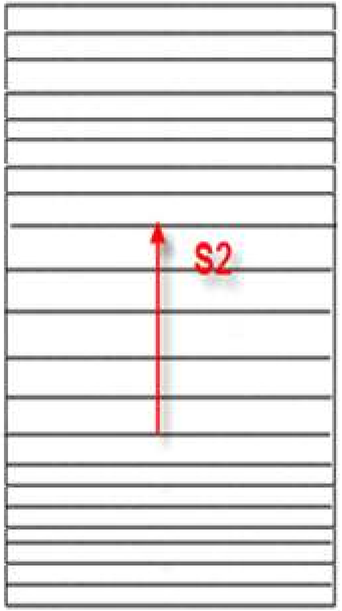
- full allowable axial **fiber/matrix** stress is obtained, 154,000 psi (1063MPa)
- This is a **fiber** failure mode
- fibers are carrying the load in the most favorable, axial direction
- resin is acting to stabilize the fibers, and not carrying any significant load (although resin does provide bridging mechanism for fiber gaps or breaks)
- transverse stress that will tend to apart is ze
- shear stress is zero



Ply Angle 90 degrees

- transverse properties of the material resisting the load
- transverse tension allowable is only 4,500 PSI (31 MPa) , based mainly on matrix tensile strength – **matrix failure**
- (interestingly the fibers act as stress raisers in practice in the resin, so tensile strength is less than matrix alone)





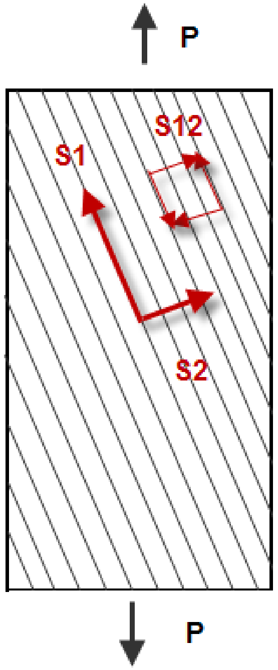
0 -> degrees

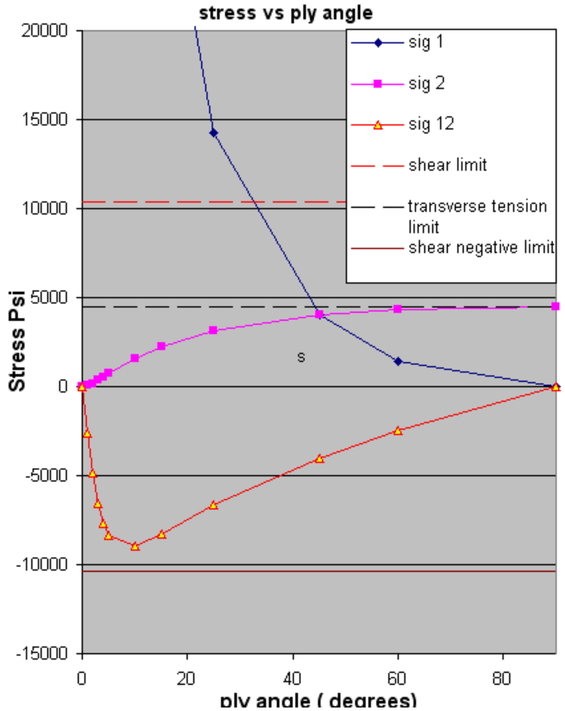
- even a few degrees from zero strength drops off rapidly
- At 10 degrees the stress at failure is down to just over 40,000 psi (276 MPa)
- fibers are now subjected to transverse stresses,
  fibers and the resin have to balance the applied stress state
- weaker transverse strength of the resin reduces the strength.
  longitudinal, transverse and shear stresses present
- 5-20 degrees shear dominates
- 30-90 degrees transverse dominates

$$\sigma_1 = \frac{P}{A}\cos^2\theta$$

$$\sigma_2 = \frac{P}{A}\sin^2\theta$$

$$\tau_{12} = -\frac{P}{A}\sin\theta\cos\theta$$





How did we predict the strength of the single ply?

- A **failure theory** analogous to Von Mises stresses for Isotropic materials is used to predict failure

- Many failure theories exist, just using one here:

Tsai-Wu Failure Theory

$$\left(\frac{1}{X_t}-\frac{1}{X_c}\right)\sigma_1+\left(\frac{1}{Y_t}-\frac{1}{Y_c}\right)\sigma_2+\frac{\sigma_1^2}{X_tX_c}+\frac{\sigma_2^2}{Y_tY_c}+\frac{\tau_{12}^2}{S^2}+2F_{12}\sigma_1\sigma_2=\mathrm{F.I.}$$

Xt	**tension limit, along fiber**
Xc	**compression limit, along fiber**
Yt	**tension limit, transverse fiber**
Yc	**compression limit, transverse fiber**
S	**shear limit**
F12	**interaction term**

Failure Index > 1.0 is bad news

Other Failure Theories

No account of tension/compression

Hill: $$\frac{\sigma_1^2}{x^2} - \frac{\sigma_1\sigma_2}{x^2} + \frac{\sigma_2^2}{y^2} + \frac{\tau_{12}^2}{s^2} = \text{F.I.}$$

Hoffman: $$\left(\frac{1}{x_t} - \frac{1}{x_c}\right)\sigma_1 + \left(\frac{1}{y_t} - \frac{1}{y_c}\right)\sigma_2 + \frac{\sigma_1^2}{x_t x_c} + \frac{\sigma_2^2}{y_t y_c} + \frac{\tau_{12}^2}{s^2} - \boxed{\frac{\sigma_1\sigma_2}{x_t x_c}} = \text{F.I.}$$

Interaction term

Max stress: $$\text{Max}\left[\left(\frac{\sigma_1}{X_t}\right), \left(\frac{\sigma_2}{Y_t}\right), \left(\frac{|\tau_{12}|}{S}\right)\right]$$

Max strain: $$\text{Max}\left[\left(\frac{\varepsilon_1}{X_t}\right), \left(\frac{\varepsilon_2}{Y_t}\right), \left(\frac{|\gamma_{12}|}{S}\right)\right]$$

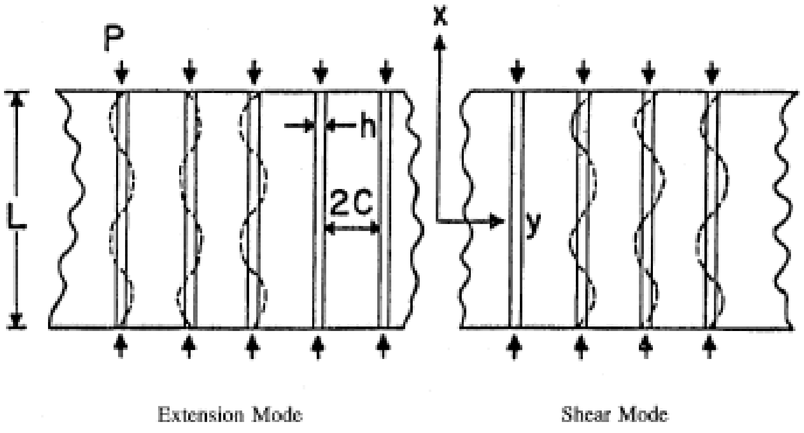
No interaction between stresses

Other failure modes

As soon as the composite is put into compression then a rather different type of behavior occurs.

For **longitudinal compression** various local buckling and shear models have been suggested. The relative stiffness of the fiber and matrix is important as well as the spacing of the fibers and geometry within the matrix.

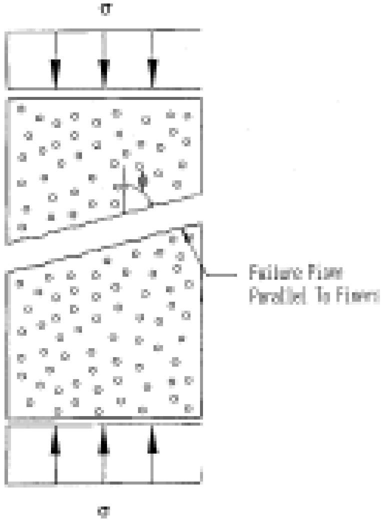
The sketch shows two local buckling forms and the photo shows evidence in practice



5. How do I know whether the composite has failed?

Transverse compression is interesting because the strength is generally higher the transverse tension. The matrix tends to act to stabilize the fibers until some form of shear cracking occurs.

This behavior is not well understood in general and is the subject of much manipulation of the failure theories. As is shown on the next few slides the behavior is broken out as a separate phenomenon in some theories.



Tsai – Wu Explored

$$\left(\frac{1}{x_t}-\frac{1}{x_c}\right)\sigma_1+\left(\frac{1}{y_t}-\frac{1}{y_c}\right)\sigma_2+\frac{\sigma_1^2}{x_t x_c}+\frac{\sigma_2^2}{y_t y_c}+\frac{\tau_{12}^2}{s^2}+2F_{12}\sigma_1\sigma_2=\text{F.I.}$$

Consider stress state with no axial (with fiber) stress

Establish locus of failure stress

$$\sigma_1=0.0$$

$$\text{F.I.}=1.0$$

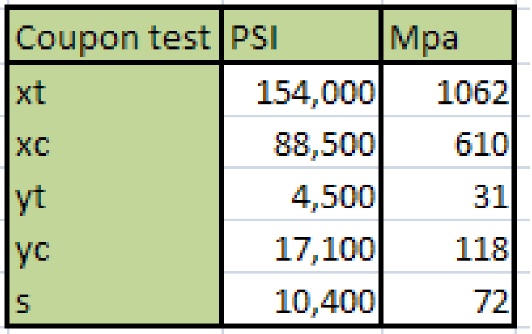
$$\text{F}_{12}=0.0$$

$$\left(\frac{1}{y_t}-\frac{1}{y_c}\right)\sigma_2+\frac{\sigma_2{}^2}{y_t y_c}+\frac{\tau_{12}{}^2}{s^2}=1.0$$

$$\frac{\tau_{12}{}^2}{s^2}=1.0-\left(\frac{1}{y_t}-\frac{1}{y_c}\right)\sigma_2-\frac{\sigma_2{}^2}{y_t y_c}$$

$$\tau_{12}{}^2=s^2\left\{1.0-\left(\frac{1}{y_t}-\frac{1}{y_c}\right)\sigma_2-\frac{\sigma_2{}^2}{y_t y_c}\right\}$$

Strengths



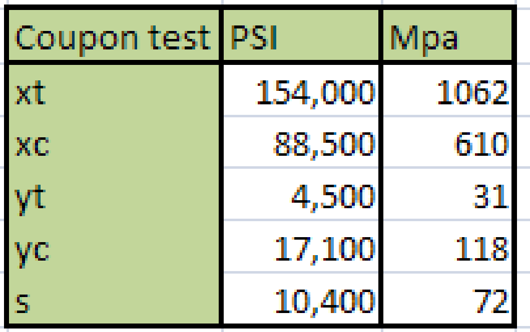
Coupon test	PSI	Mpa
xt	154,000	1062
xc	88,500	610
yt	4,500	31
yc	17,100	118
s	10,400	72

Hill Explored

$$\frac{\sigma_1^2}{x^2} - \frac{\sigma_1\sigma_2}{x^2} + \frac{\sigma_2^2}{y^2} + \frac{\tau_{12}^2}{s^2} = \text{F.I.}$$

Consider stress state with no axial (with fiber) stress

Strengths



Coupon test	PSI	Mpa
xt	154,000	1062
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yc	17,100	118
s	10,400	72

Establish locus of failure stress

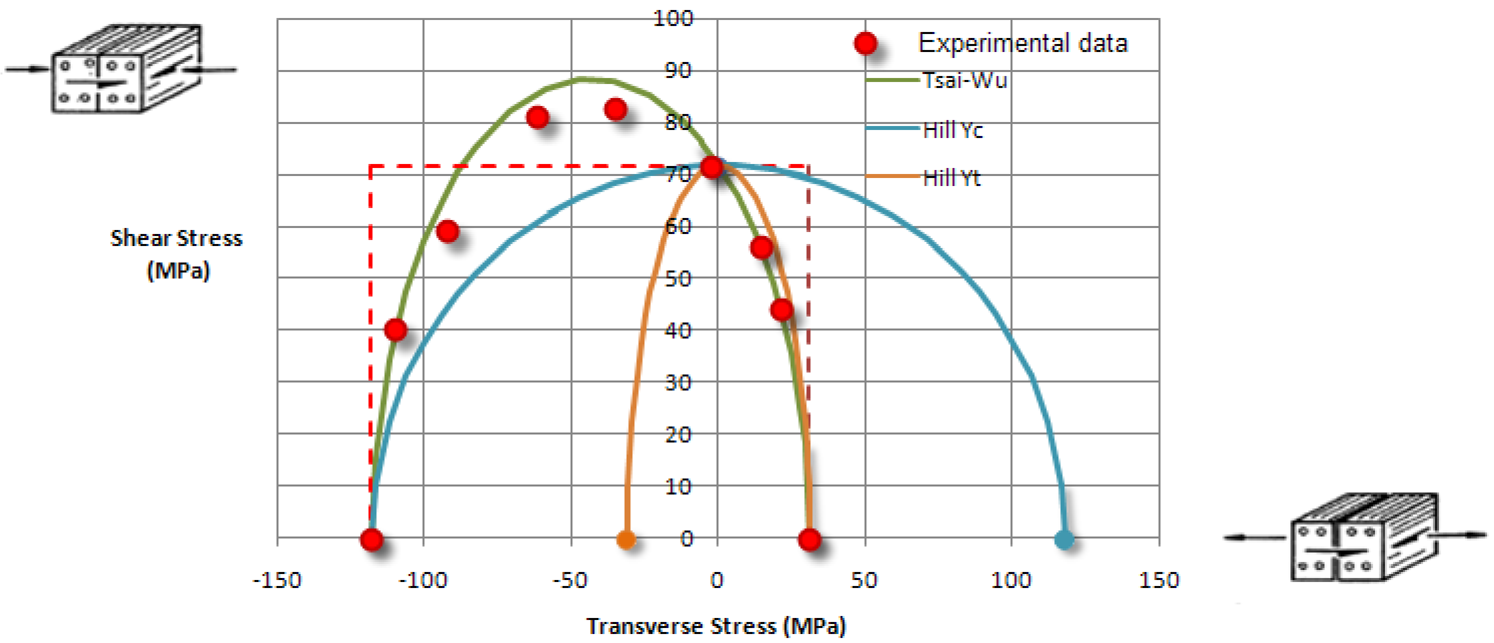
$$\sigma_1 = 0.0$$

$$\text{F.I.} = 1.0$$

$$\frac{\sigma_2{}^2}{y^2} + \frac{\tau_{12}{}^2}{s^2} = 1.0$$

$$\frac{\tau_{12}{}^2}{s^2} = 1.0 - \frac{\sigma_2{}^2}{y^2}$$

$$\tau_{12}{}^2 = s^2\left\{1.0 - \frac{\sigma_2{}^2}{y^2}\right\}$$

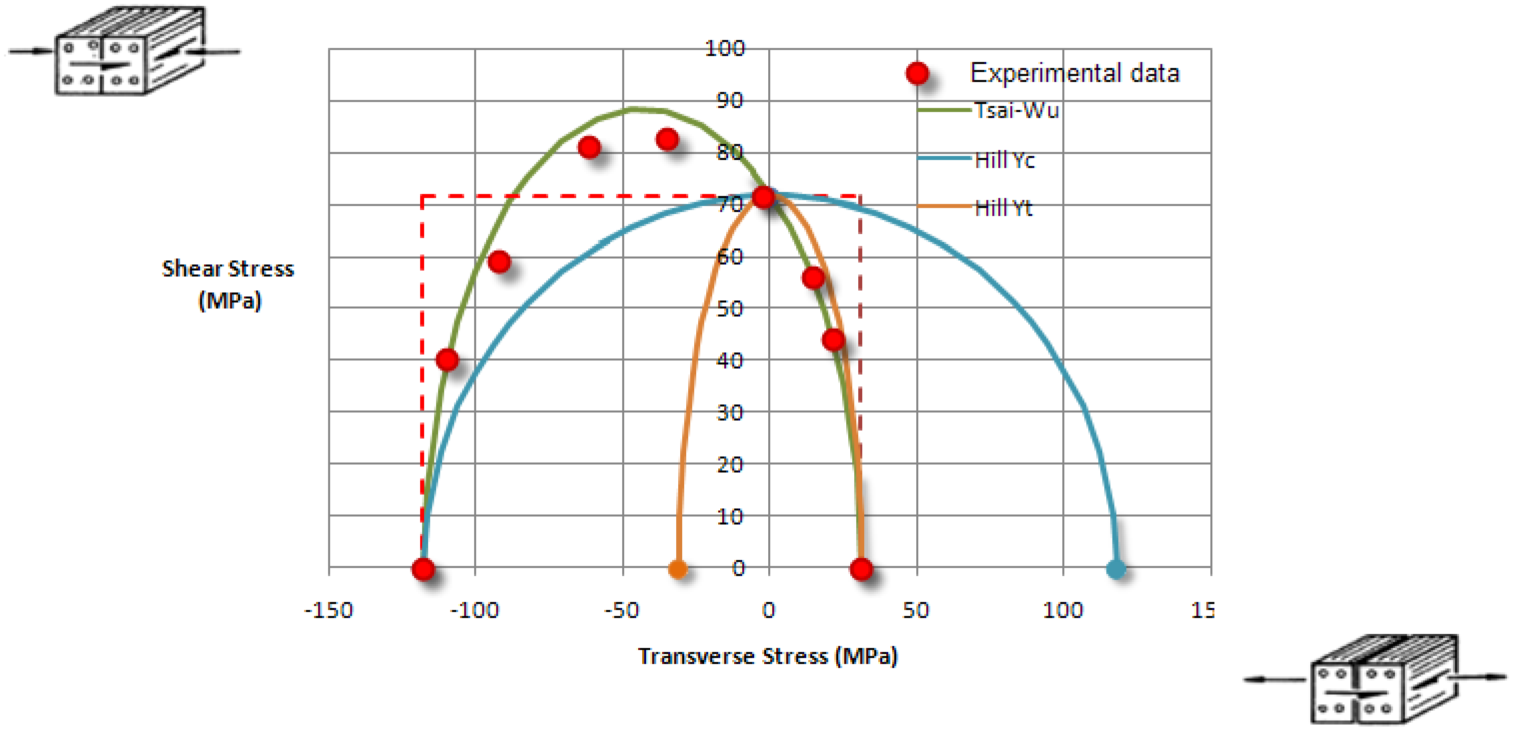


This stress state zone is of great interest as it involves complex failure modes

On our test with 5 degrees off axis and higher $\sigma_1 \to 0.0$ MPa

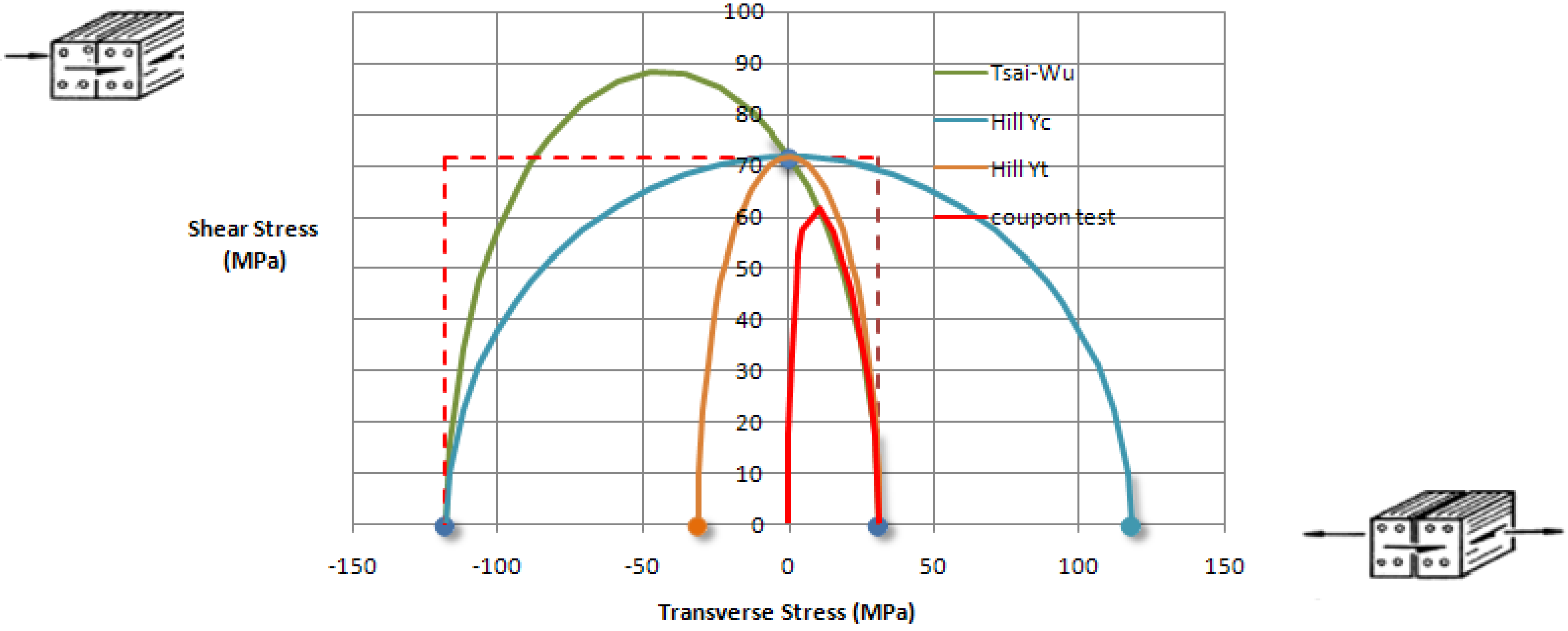
- Tsai-Wu shows the effect of interaction when shear and compressive transverse stresses combine.

- Experimental evidence tends to confirm that Tsai-Wu predictions modify the simple stress limit values



Hill shows the limitation when the same strengths are used in Tension and Compression for transverse strength

Experimental evidence shows a clear bias in the strength allowables and this affects the interaction



For the FEA results using the Tsai-Wu criteria we can see the results for ply off axis > 10 degrees fit well into this reduced envelope as the axial stresses tend to zero.

In this case either Hill using transverse tension allowable, or Tsai-Wu would give the same results, which is intuitively correct.

We need to be aware that more complex loading states will not ‘fit’ Hill well.

Advanced failure modes:

The Tsai-Wu , Hill and Hoffman failure theories are just one of many that were developed using know failure points and then interpolating in stress space using quadratic relationships

One of the limitations of this approach is that all failure is based on a full and continuous interaction between stress states.

It has been found experimentally that failure modes tend to be dominated by either fiber failure modes or matrix failure. There may be little interaction between them.

The continuous quadratic family of theories do not differentiate between these fundamental failure modes.

A class of failure theories has evolved which are sometimes described as ‘phenomenological’ to indicate the nature of the failure is implicit in the theory

Advanced failure modes:

Hashin-Rotem failure criteria breaks up the assessment of failure into several sub criterion:

Tensile Fiber Failure:

$$\left(\frac{\sigma_1}{X_t}\right)^2 + \left(\frac{\tau_{12}}{S_{12}}\right)^2 = 1$$

Compressive fiber failure:

$$\left(\frac{\sigma_1}{X_c}\right)^2 = 1$$

Tensile Matrix failure:

$$\left(\frac{\sigma_2}{Y_t}\right)^2 + \left(\frac{\tau_{12}}{S_{12}}\right)^2 = 1$$

Compressive Matrix failure

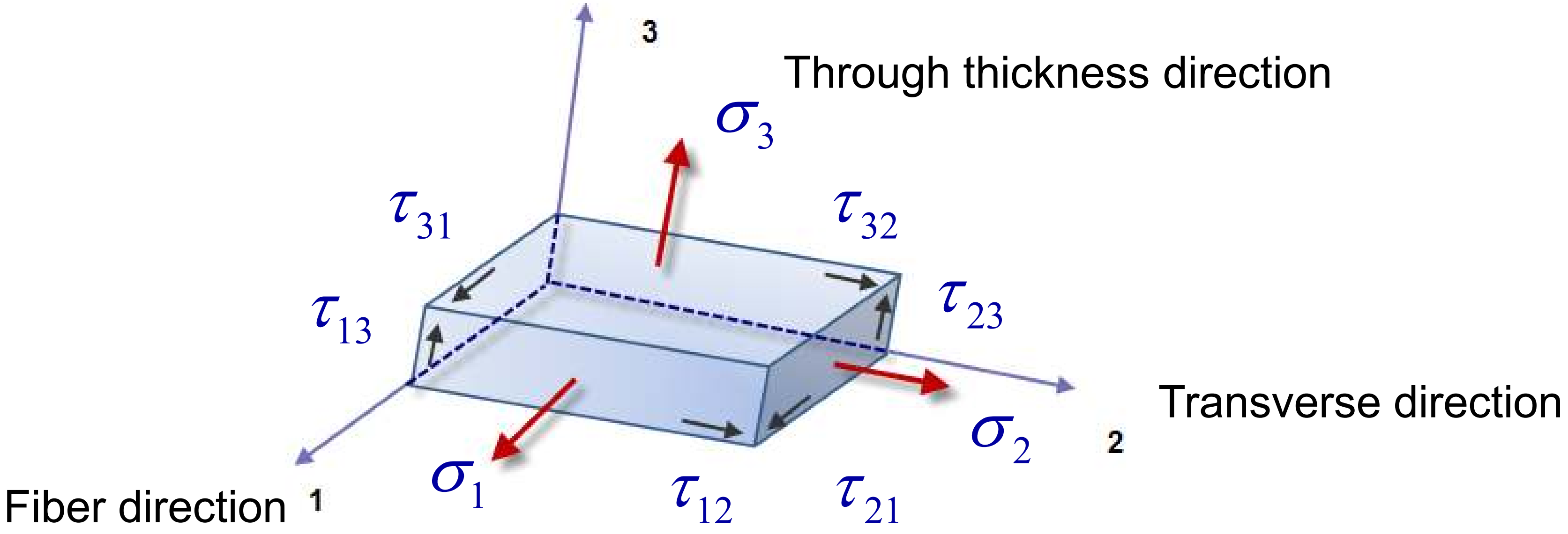
$$\left(\frac{\sigma_2}{2S_{23}}\right)^2 + \left[\left(\frac{Y_c}{2S_{23}}\right)^2 - 1\right]\frac{\sigma_2}{Y_c} + \left(\frac{\tau_{12}}{S_{12}}\right)^2 = 1$$

Advanced failure modes:

Through thickness failure

$$\left(\frac{\sigma_3}{Z_t}\right)^2+\left(\frac{\tau_{23}}{S_{23}}\right)^2+\left(\frac{\tau_{31}}{S_{31}}\right)^2=1$$

Note that additional Stress and Strength definitions are made:



Advanced failure modes:

Each mode is assessed to see which has the highest failure index above 1.0, and hence which prompts the failure

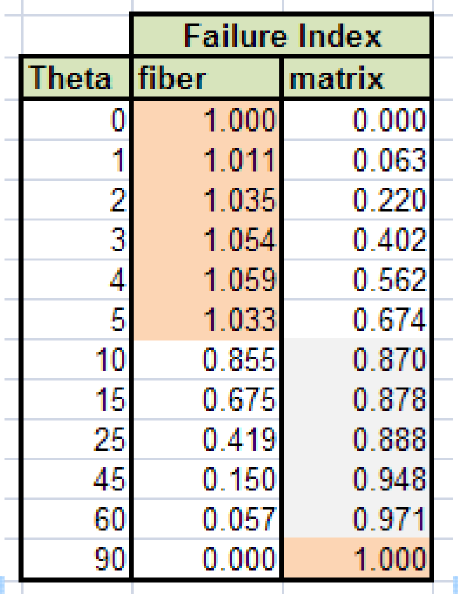
For our coupon we will ignore through thickness failure and assume inter laminar shear strength equals in plane shear strength:

$$S_{23} = S_{12}$$

Only tensile axial stresses are present, and only tensile transverse stresses so only those two terms are considered

The results show that the two failure modes are clearly defined

- Fiber failure occurs up to at least 5 degrees off axis
- Matrix failure occurs somewhere before 10 degrees and continues to 90
- There is a very small reduction in failing load for the matrix failure



	Failure Index	
Theta	fiber	matrix
0	1.000	0.000
1	1.011	0.063
2	1.035	0.220
3	1.054	0.402
4	1.059	0.562
5	1.033	0.674
10	0.855	0.870
15	0.675	0.878
25	0.419	0.888
45	0.150	0.948
60	0.057	0.971
90	0.000	1.000

It is interesting to compare the terms of the two failure criterion when the axial stresses are low and matrix failure is assumed to dominate

Hashin

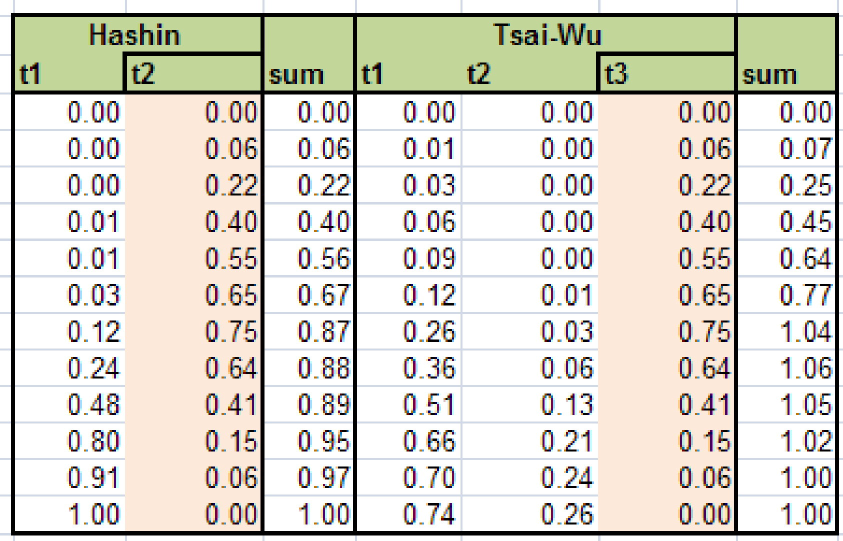
$$\left(\frac{\sigma_2}{Y_t}\right)^2+\left(\frac{\tau_{12}}{S_{12}}\right)^2=1.0$$

Tsai-Wu

$$\left(\frac{1}{y_t}-\frac{1}{y_c}\right)\sigma_2+\frac{\sigma_2{}^2}{y_t y_c}+\frac{\tau_{12}{}^2}{S_{12}{}^2}=1.0$$

Each equation has an identical shear term

However the Tsai-Wu first two terms appear to over contribute to a small extent and the compressive transverse strength is intuitively not likely to provide meaningful contribution



Hashin			Tsai-Wu			
t1	t2	sum	t1	t2	t3	sum
0.00	0.00	0.00	0.00	0.00	0.00	0.00
0.00	0.06	0.06	0.01	0.00	0.06	0.07
0.00	0.22	0.22	0.03	0.00	0.22	0.25
0.01	0.40	0.40	0.06	0.00	0.40	0.45
0.01	0.55	0.56	0.09	0.00	0.55	0.64
0.03	0.65	0.67	0.12	0.01	0.65	0.77
0.12	0.75	0.87	0.26	0.03	0.75	1.04
0.24	0.64	0.88	0.36	0.06	0.64	1.06
0.48	0.41	0.89	0.51	0.13	0.41	1.05
0.80	0.15	0.95	0.66	0.21	0.15	1.02
0.91	0.06	0.97	0.70	0.24	0.06	1.00
1.00	0.00	1.00	0.74	0.26	0.00	1.00

Advanced failure modes:

Similarly if the transverse stress is ignored in the fiber failure region the equations can be compared

Hashin

$$\left(\frac{\sigma_1}{X_t}\right)^2+\left(\frac{\tau_{12}}{S_{12}}\right)^2=1$$

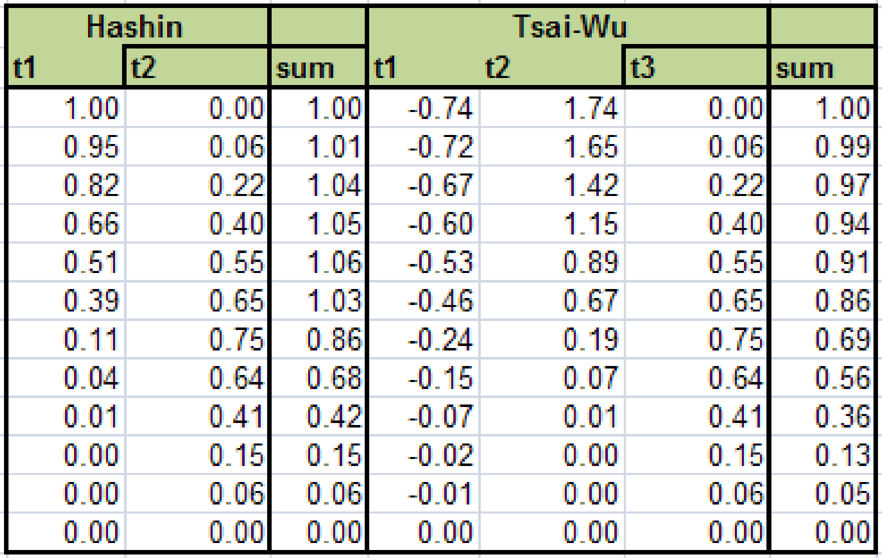
Tsai-Wu

$$\left(\frac{1}{X_t}-\frac{1}{X_c}\right)\sigma_1+\frac{\sigma_1{}^2}{X_tX_c}+\frac{\tau_{12}{}^2}{S_{12}{}^2}=1.0$$

Note the Hashin method re-uses the shear term. The Tsai –Wu method cannot do this as it is continuous with no distinction. It is added here for comparison.

The Tsai-Wu direct terms are a balancing act and again there is no intuitive feel for their individual contributions

In both methods the shear term extends the domain of the pure fiber failure mode



Hashin			Tsai-Wu			
t1	t2	sum	t1	t2	t3	sum
1.00	0.00	1.00	-0.74	1.74	0.00	1.00
0.95	0.06	1.01	-0.72	1.65	0.06	0.99
0.82	0.22	1.04	-0.67	1.42	0.22	0.97
0.66	0.40	1.05	-0.60	1.15	0.40	0.94
0.51	0.55	1.06	-0.53	0.89	0.55	0.91
0.39	0.65	1.03	-0.46	0.67	0.65	0.86
0.11	0.75	0.86	-0.24	0.19	0.75	0.69
0.04	0.64	0.68	-0.15	0.07	0.64	0.56
0.01	0.41	0.42	-0.07	0.01	0.41	0.36
0.00	0.15	0.15	-0.02	0.00	0.15	0.13
0.00	0.06	0.06	-0.01	0.00	0.06	0.05
0.00	0.00	0.00	0.00	0.00	0.00	0.00

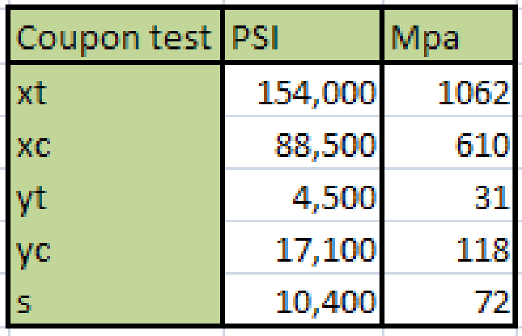
Hashin Explored

Compressive Matrix failure

Consider stress state with no axial (with fiber) stress (implicit in matrix compressive term)

Establish locus of failure stress

Strengths



Coupon test	PSI	Mpa
xt	154,000	1062
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F.I. $= 1.0$

$$\left(\frac{\sigma_2}{2S_{23}}\right)^2 + \left[\left(\frac{Y_c}{2S_{23}}\right)^2 - 1\right]\frac{\sigma_2}{Y_c} + \left(\frac{\tau_{12}}{S_{12}}\right)^2 = 1$$

$$\left(\frac{\tau_{12}}{S_{12}}\right)^2 = 1 - \left(\frac{\sigma_2}{2S_{23}}\right)^2 - \left[\left(\frac{Y_c}{2S_{23}}\right)^2 - 1\right]\frac{\sigma_2}{Y_c}$$

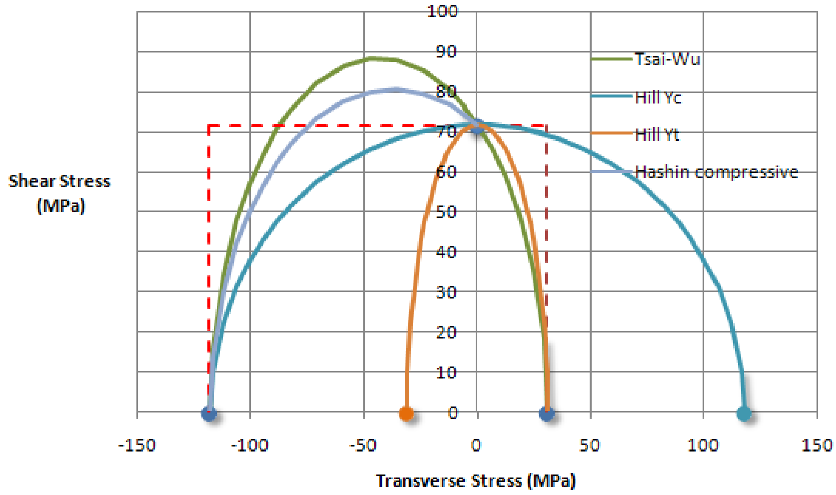
$$\tau_{12}{}^2 = S_{12}{}^2 - \left(\frac{S_{12}\sigma_2}{2S_{23}}\right)^2 - \left[\left(\frac{Y_c}{2S_{23}}\right)^2 - 1\right]\frac{\sigma_2 S_{12}{}^2}{Y_c}$$

Advanced failure modes:

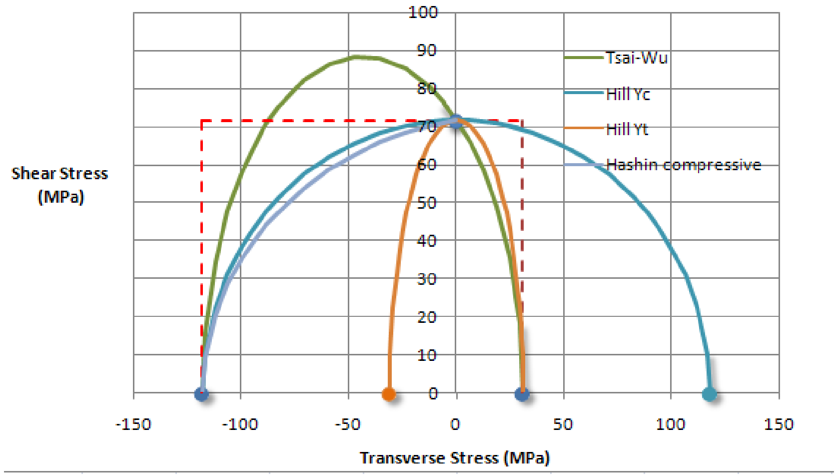
To better understand the failure Hashin failure mode in compression the failure locus under transverse stress and shear stress has been added to the Hill and Tsai-Wu curves

$$\tau_{12}^2 = S_{12}^2 - \left(\frac{S_{12}\sigma_2}{2S_{23}}\right)^2 - \left[\left(\frac{Y_c}{2S_{23}}\right)^2 - 1\right]\frac{\sigma_2 S_{12}^2}{Y_c}$$

S23 = 0.5 s12



S23 = s12



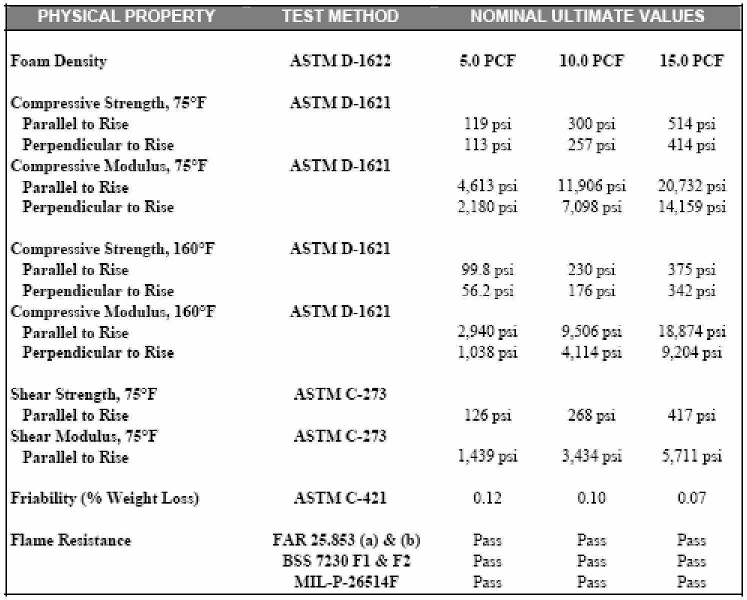
Test data

Strength data is available from suppliers, but needs to treated with caution

Test data costs a lot of resource to compile and is not widely available in industry

Academic papers and text books tend to contain useful data

Test if you can afford it, but is a complex process



PHYSICAL PROPERTY	TEST METHOD	NOMINAL ULTIMATE VALUES		
Foam Density	ASTM D-1622	5.0 PCF	10.0 PCF	15.0 PCF
Compressive Strength, 75°F	ASTM D-1621			
Parallel to Rise		119 psi	300 psi	514 psi
Perpendicular to Rise		113 psi	257 psi	414 psi
Compressive Modulus, 75°F	ASTM D-1621			
Parallel to Rise		4,613 psi	11,906 psi	20,732 psi
Perpendicular to Rise		2,180 psi	7,098 psi	14,159 psi
Compressive Strength, 160°F	ASTM D-1621			
Parallel to Rise		99.8 psi	230 psi	375 psi
Perpendicular to Rise		56.2 psi	176 psi	342 psi
Compressive Modulus, 160°F	ASTM D-1621			
Parallel to Rise		2,940 psi	9,506 psi	18,874 psi
Perpendicular to Rise		1,038 psi	4,114 psi	9,204 psi
Shear Strength, 75°F	ASTM C-273			
Parallel to Rise		126 psi	268 psi	417 psi
Shear Modulus, 75°F	ASTM C-273			
Parallel to Rise		1,439 psi	3,434 psi	5,711 psi
Friability (% Weight Loss)	ASTM C-421	0.12	0.10	0.07
Flame Resistance	FAR 25.853 (a) & (b)	Pass	Pass	Pass
	BSS 7230 F1 & F2	Pass	Pass	Pass
	MIL-P-26514F	Pass	Pass	Pass

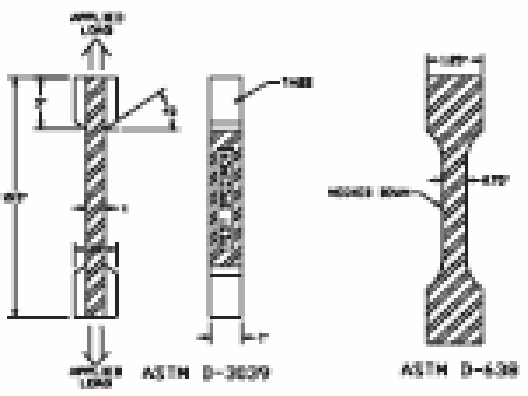


Figure 4-10 Test Specimen Configuration for ASTM D-3039 and D-638 Tensile Tests (Structural Composites, Inc.)

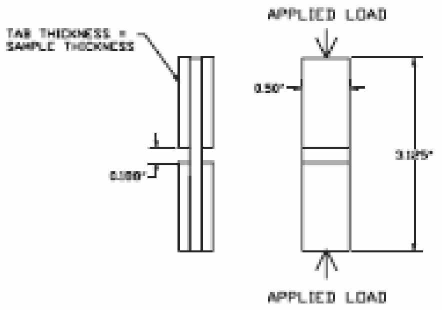


Figure 4-12 Test Specimen Configuration for SACMA SRM-1 Compression Test

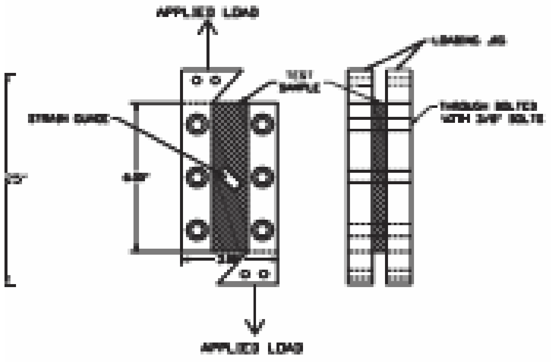


Figure 4-17 Test Specimen Configuration for ASTM D-4255 Rail Shear Test, Method A

Some of the test methods referenced

5. How do I know whether the composite has failed?

R0031

Basic First Ply failure theories

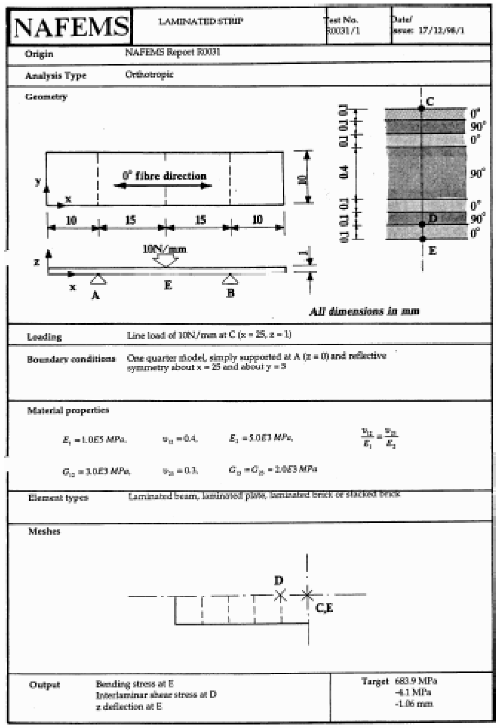
NAFEMS COMPOSITE BENCHMARKS Issue 2

TEST 1 - Laminated strip under three-point bending;
TEST 2 - Wrapped thick cylinder under pressure and thermal loading;
TEST 3 - Three-layer sandwich shell under normal pressure loading.

The purpose of these tests is to demonstrate that the program can carry out an effective
composite analysis and :-
a) accurately predict displacements;
b) recover meaningful direct stresses;
c) recover meaningful interlaminar shear stresses;
using flat laminated plate, brick, curved shell and thick sandwich shell elements

5. How do I know whether the composite has failed?

Basic First Ply failure theories



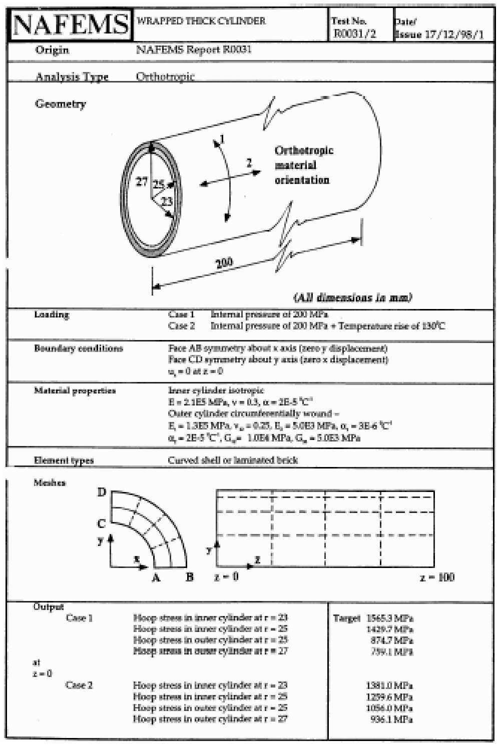
NAFEMS LAMINATED STRIP	Test No. R0031/1 — Date/ Issue: 17/12/98/1
Origin	NAFEMS Report R0031
Analysis Type	Orthotropic
Loading	Line load of 10N/mm at C (x = 25, z = 1)
Boundary conditions	One quarter model, simply supported at A (z = 0) and reflective symmetry about x = 25 and about y = 5
Element types	Laminated beam, laminated plate, laminated brick or stacked brick

Material properties

$E_1 = 1.0E5$ MPa, $\nu_{12} = 0.4$, $E_2 = 5.0E3$ MPa, $\frac{\nu_{12}}{E_1} = \frac{\nu_{21}}{E_2}$

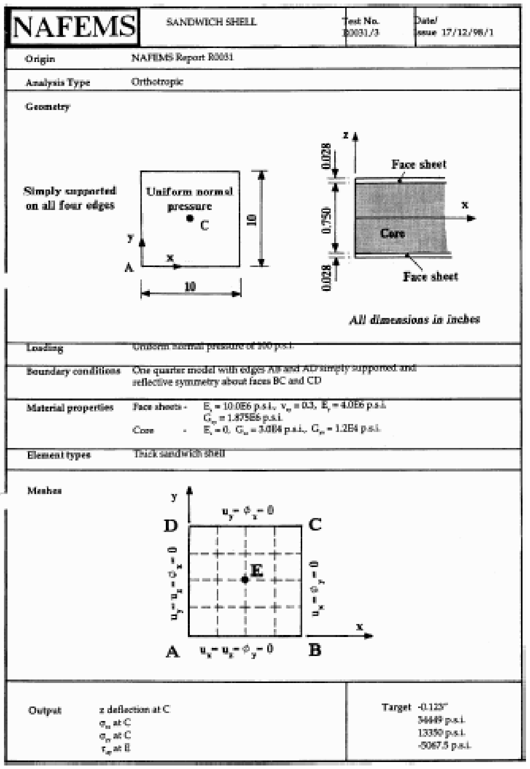
$G_{12} = 3.0E3$ MPa, $\nu_{23} = 0.3$, $G_{13} = G_{23} = 2.0E3$ MPa

Output	Bending stress at E Interlaminar shear stress at D z deflection at E	Target 683.9 MPa -4.1 MPa -1.06 mm



NAFEMS WRAPPED THICK CYLINDER	Test No. R0031/2 — Date/ Issue 17/12/98/1
Origin	NAFEMS Report R0031
Analysis Type	Orthotropic
Loading	Case 1 Internal pressure of 200 MPa Case 2 Internal pressure of 200 MPa + Temperature rise of 130°C
Boundary conditions	Face AB symmetry about x axis (zero y displacement) Face CD symmetry about y axis (zero x displacement) $u_z = 0$ at $z = 0$
Material properties	Inner cylinder isotropic $E = 2.1E5$ MPa, $\nu = 0.3$, $\alpha = 2E\text{-}5\ {}^{\circ}C^{-1}$ Outer cylinder circumferentially wound - $E_1 = 1.3E5$ MPa, $\nu_{12} = 0.25$, $E_2 = 5.0E3$ MPa, $\alpha_1 = 3E\text{-}6\ {}^{\circ}C^{-1}$ $\alpha_2 = 2E\text{-}5\ {}^{\circ}C^{-1}$, $G_{12} = 1.0E4$ MPa, $G_{23} = 5.0E3$ MPa
Element types	Curved shell or laminated brick

Output at z = 0	Case 1	Hoop stress in inner cylinder at r = 23 Hoop stress in inner cylinder at r = 25 Hoop stress in outer cylinder at r = 25 Hoop stress in outer cylinder at r = 27	Target 1565.3 MPa 1429.7 MPa 874.7 MPa 759.1 MPa
	Case 2	Hoop stress in inner cylinder at r = 23 Hoop stress in inner cylinder at r = 25 Hoop stress in outer cylinder at r = 25 Hoop stress in outer cylinder at r = 27	1381.0 MPa 1259.6 MPa 1056.0 MPa 936.1 MPa



NAFEMS SANDWICH SHELL	Test No. R0031/3 — Date/ Issue 17/12/98/1
Origin	NAFEMS Report R0031
Analysis Type	Orthotropic
Loading	Uniform normal pressure of 100 p.s.i.
Boundary conditions	One quarter model with edges AB and AD simply supported and reflective symmetry about faces BC and CD
Material properties	Face sheets - $E_x = 10.0E6$ p.s.i., $\nu_{xy} = 0.3$, $E_y = 4.0E6$ p.s.i. $G_{xy} = 1.875E6$ p.s.i. Core - $E_z = 0$, $G_{xz} = 3.0E4$ p.s.i., $G_{yz} = 1.2E4$ p.s.i.
Element types	Thick sandwich shell

Output	z deflection at C σ_{xx} at C σ_{yy} at C τ_{xy} at E	Target -0.123" 34449 p.s.i. 13350 p.s.i. -5067.5 p.s.i.

Introductory Composites FE Analysis Webinar

Agenda

4. How good is my FEA idealization?

The importance of fiber orientation, draping and thickness effects.

5. How do I know whether the composite has failed?

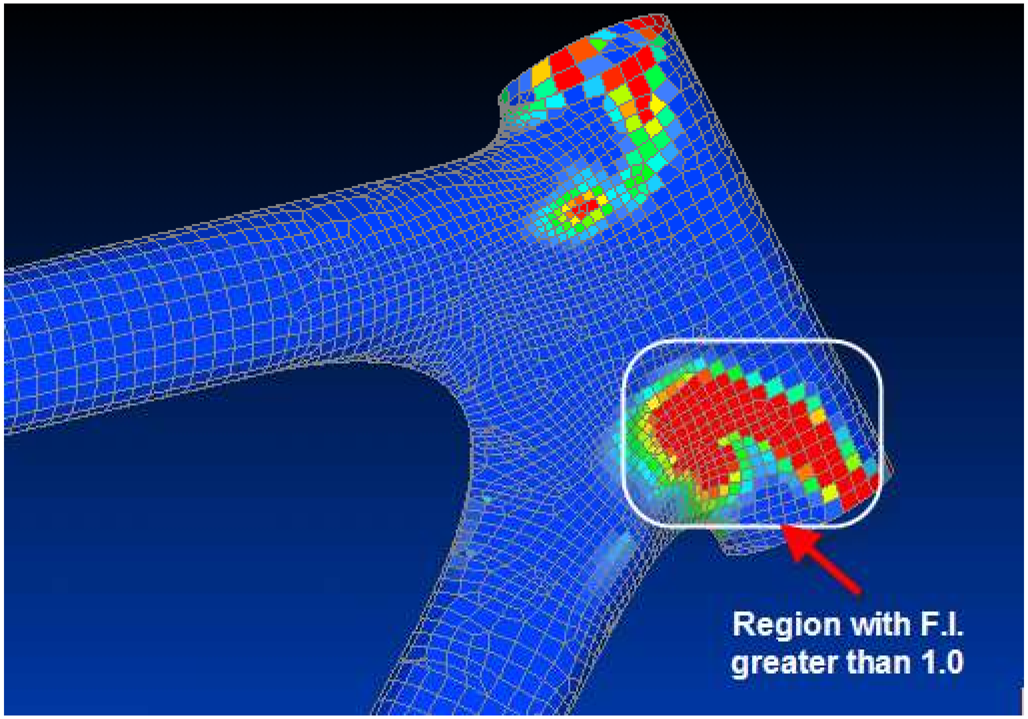
Basic First Ply failure theories

6. How do I organize my results, where do I start looking?

Failure indices, Strength ratios.

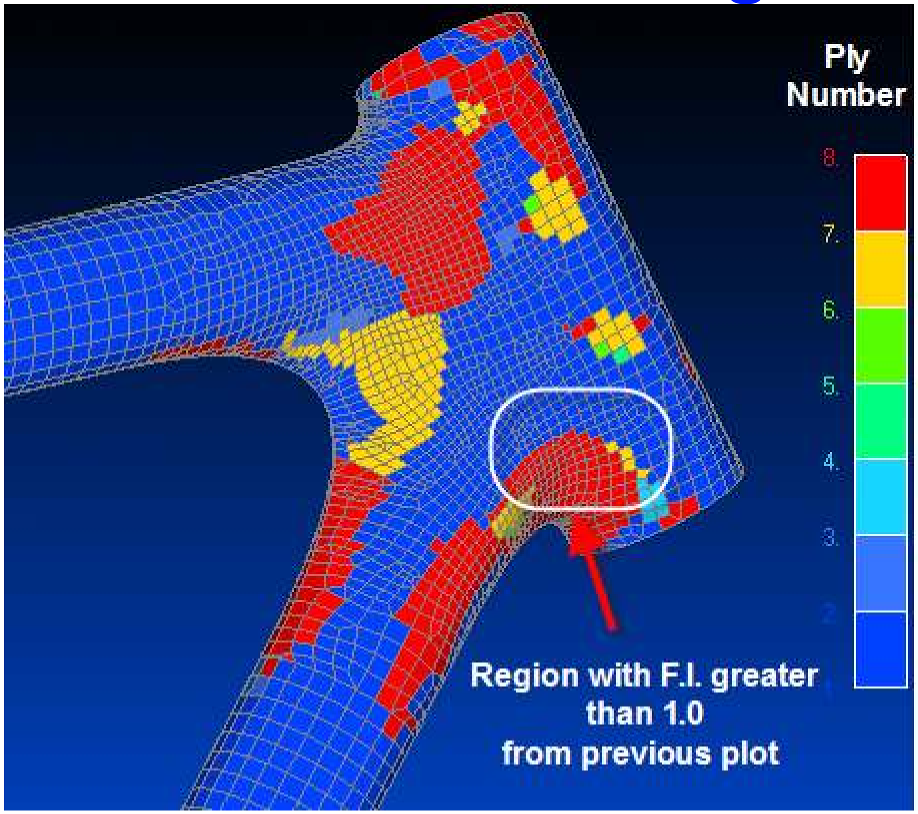
6. How do I organize my results, where do I start looking?

Failure indices, Strength ratios.



- **Identify regions where F.I. shows failure in the layup**

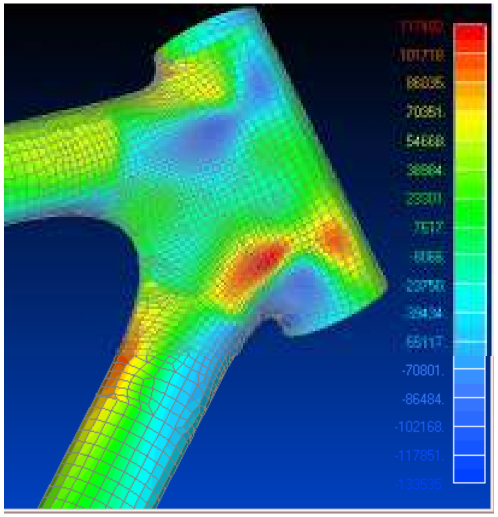
6. How do I organize my results, where do I start looking?

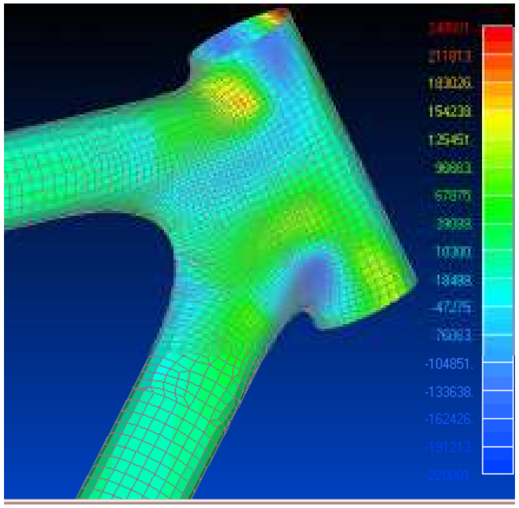


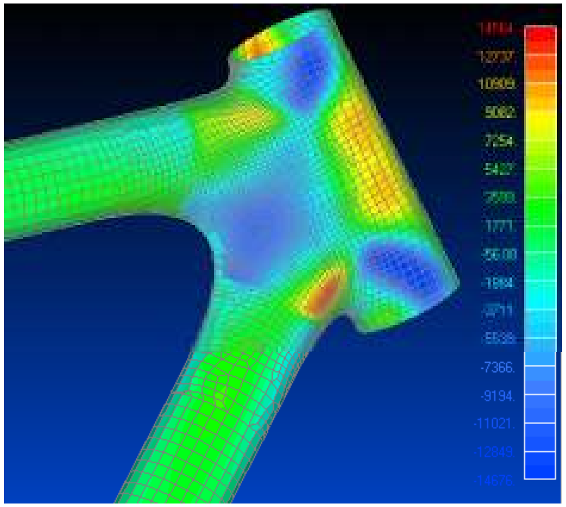
Region with F.I. greater than 1.0 from previous plot

- Identify which plies are failing in the layup in that region

6. How do I organize my results, where do I start looking?







- Review Direct X, Direct Y and Shear XY ply stresses in the individual ply
- Assess major mode of failure
- Assess coupling through plies
- Redesign if required

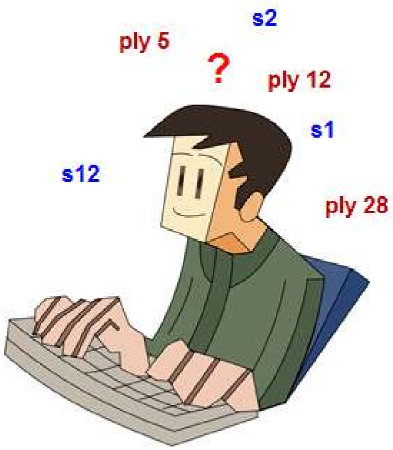
6. How do I organize my results, where do I start looking?

Various stress sorting and filtering schemes are available dependent on solver and post processor used

It is important to get familiar with these

Use contour plots and any specific ply mapping tools

The quantity of data can be immense



6. How do I organize my results, where do I start looking?

The Failure Index is a quadratic term, it does not scale linearly with stress level

Failure is when F.I >= 1.0

$$\left(\frac{1}{x_t}-\frac{1}{x_c}\right)\sigma_1+\left(\frac{1}{y_t}-\frac{1}{y_c}\right)\sigma_2+\frac{\sigma_1^2}{x_t x_c}+\frac{\sigma_2^2}{y_t y_c}+\frac{\tau_{12}^2}{s^2}+2F_{12}\sigma_1\sigma_2=\text{F.I.}$$

For most Failure Criteria the F.I equation can be recast as a Strength Ratio of actual stress/allowable stress with F.I. set to 1.0

Now Strength Ratio scales linearly with stress

Failure is when S.R < 1.0

Acts like a Reserve Factor as used in Europe   MS = RF-1.

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Agenda

7. Through thickness and edge effects such as delamination

Usage of solid and thick shell elements.

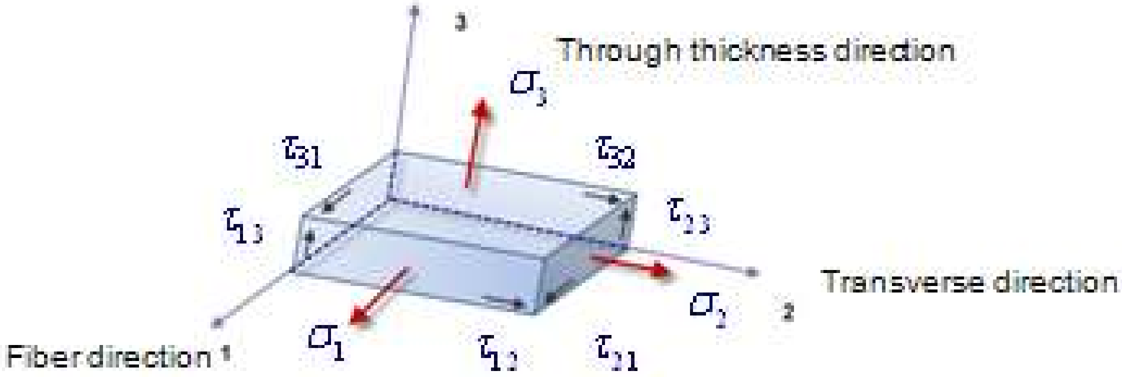
8. Advanced failure methods

Progressive ply failure, cohesive elements, fracture mechanics methods (VCCT)

7. Through thickness and edge effects such as delamination

Usage of solid and thick shell elements

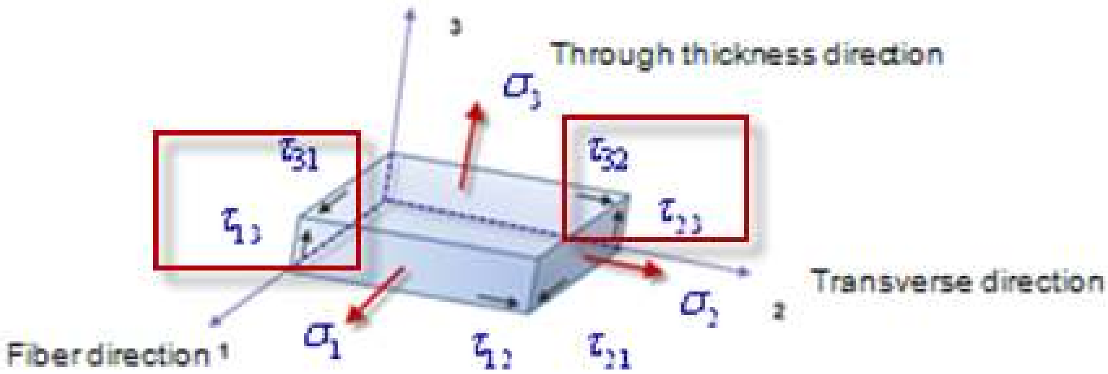
We have not discussed the through thickness terms which include the interlaminar shears and the direct through thickness stress



7. Through thickness and edge effects such as delamination

For thin shells in bending the interlaminar shears created by relative stretching between plies are approximated by assuming a simple through shear distribution analogous to classical shear solutions in solid isotropic sections

Hence interlaminar shear stresses and strength assessment under simple bending is quite acceptable



7. Through thickness and edge effects such as delamination

However, the thin shell theory assumes that the stresses are continuous within a ply and takes no account of any possible free edge effect where stresses go to zero

In cases where this may be important it may be necessary to use thick shell or solid elements that can cater for this or to use a micro level element mesh where each ply is modeled with thick shells or solids

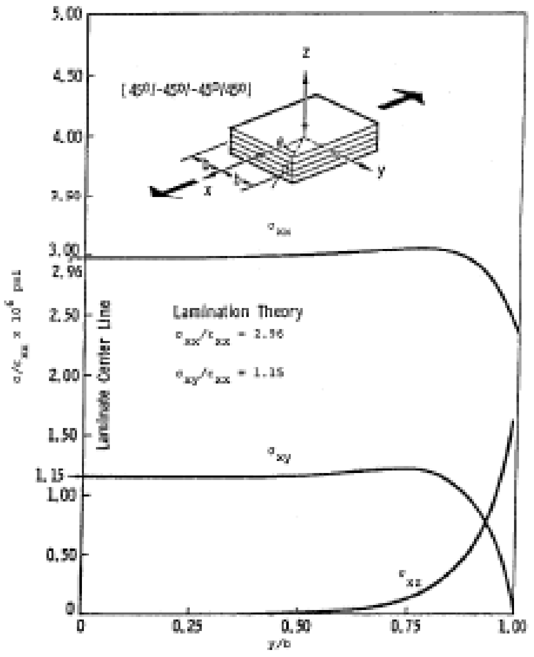
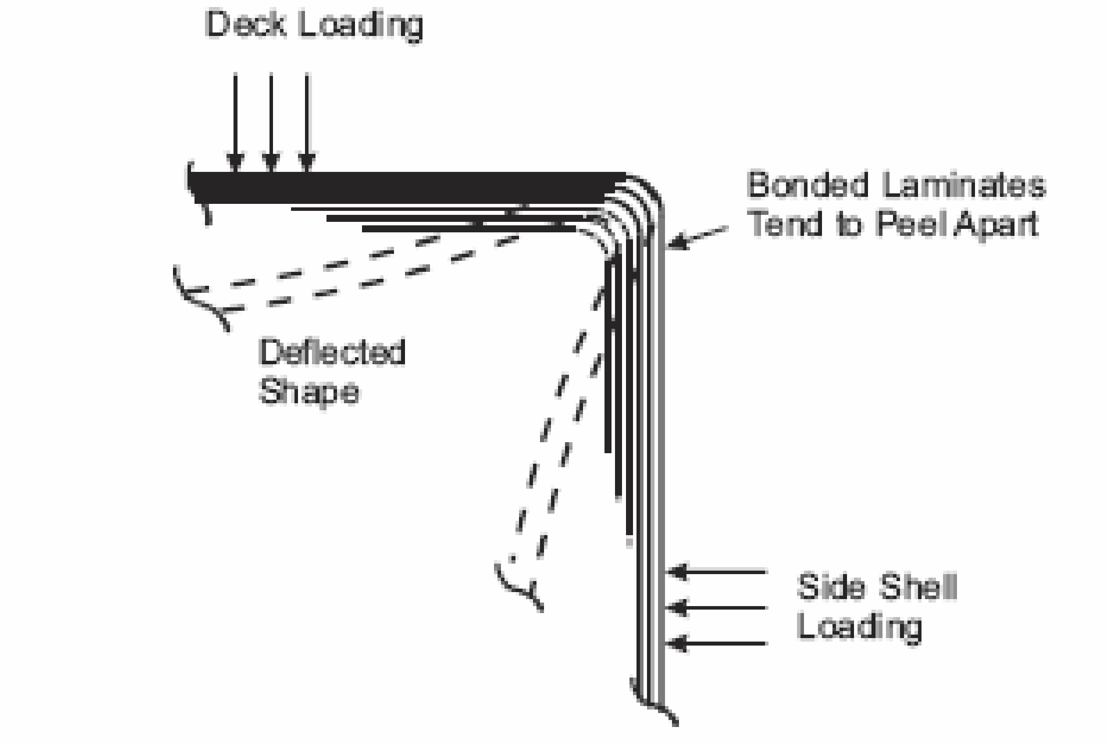


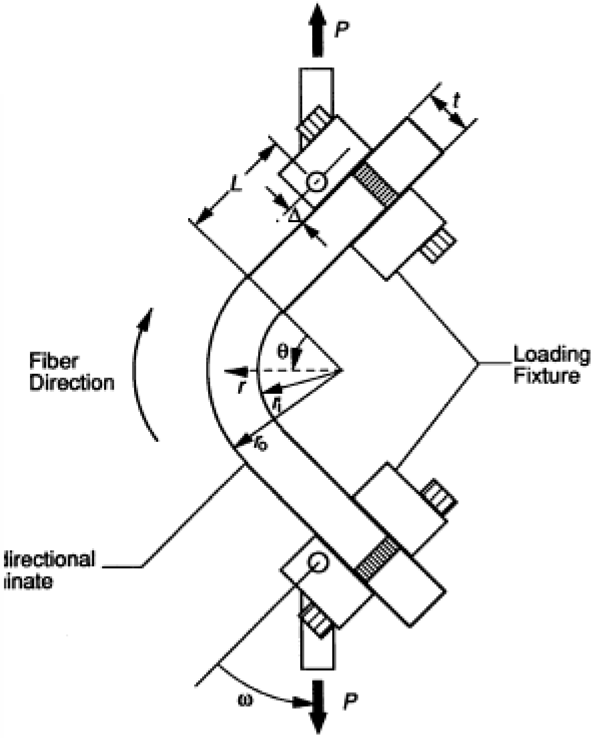
FIGURE 4.22. INTERLAMINAR STRESS NORMALIZED WITH RESPECT TO THE APPLIED STRAIN [4.4]

7. Through thickness and edge effects such as delamination

Bending effect such as shown here will promote interlaminar shears and also direct through stresses

Solids or thick shells are required

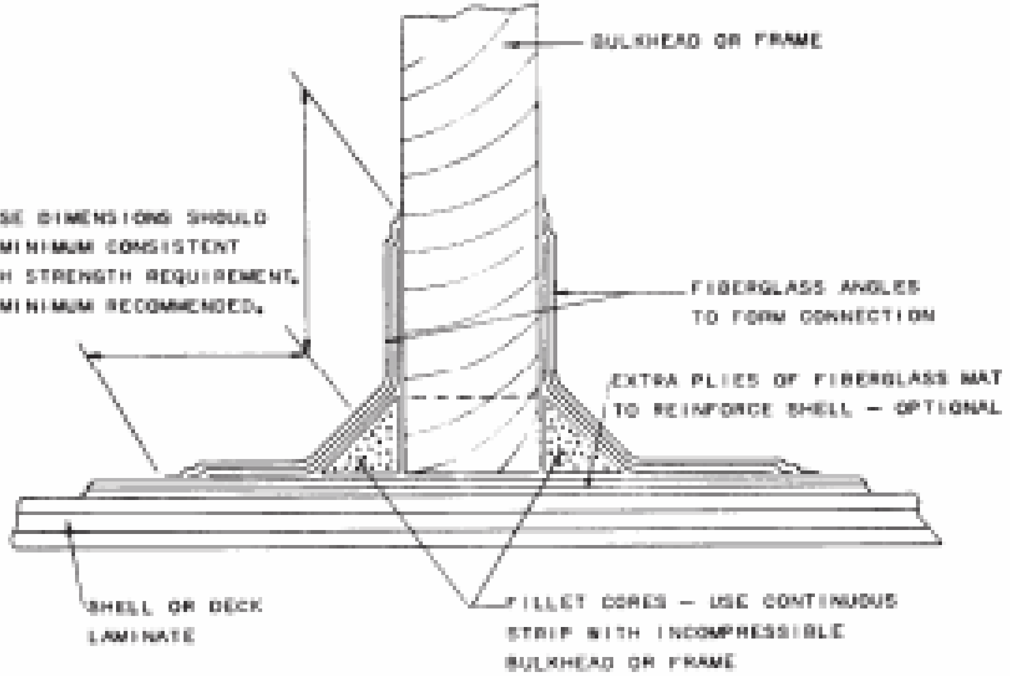




7. Through thickness and edge effects such as delamination

This fitting will exhibit peel stresses, through thickness stresses and other stress patterns tending to act in a 3D sense through thickness

For heavy fittings, plane strain may be a useful analysis method



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Agenda

7. Through thickness and edge effects such as delamination

Usage of solid and thick shell elements.

8. Advanced failure methods

Progressive ply failure, cohesive elements, fracture mechanics methods (VCCT)

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Progressive ply failure, cohesive elements, fracture mechanics methods (VCCT)

We saw the definition of First Ply Failure in a previous section

Progressive Ply Failure takes this further by assuming that the stiffness of the failed ply can be reduced in some manner and the analysis continues

Ply failures can continue to occur with subsequent reduction in stiffness

A PPF strategy requires:

A failure criteria which can identify the mode of failure (such as Hashin, Puck, LARC02)

A rational strategy for reducing element stiffness based on the mode of failure seen

8. Advanced failure methods

Progressive Ply Failure is sensitive to how the stiffness is reduced at each non-linear load step.

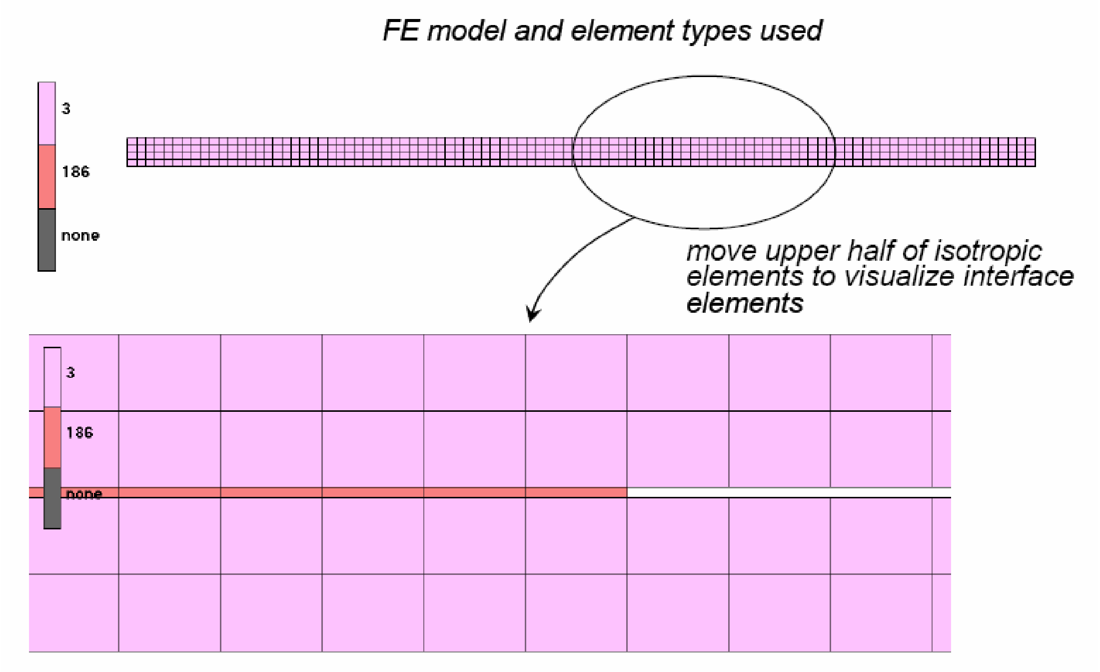
If the drop is too great then instabilities can occur, so usually a maximum percentage of stiffness in a particular orthotropic direction is used.

The user can elect to modify this to simulate more ductile composite materials.

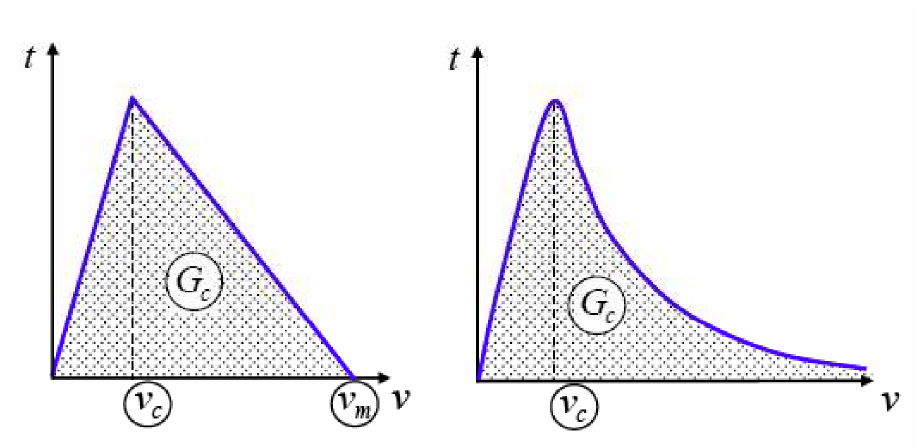
Certain failure modes such as longitudinal tension may be classified as final failure in their own right.

8. Advanced failure methods

Cohesive Element methods aim to model specific debonding or delamination situations by inserting a layer of special elements between the plies or materials



8. Advanced failure methods



The behavior of the crack or delamination front is controlled by an energy rate law to allow tuning for different types material (e.g. brittle or ductile)

The actual failure method is still using a stress based approach

8. Advanced failure methods

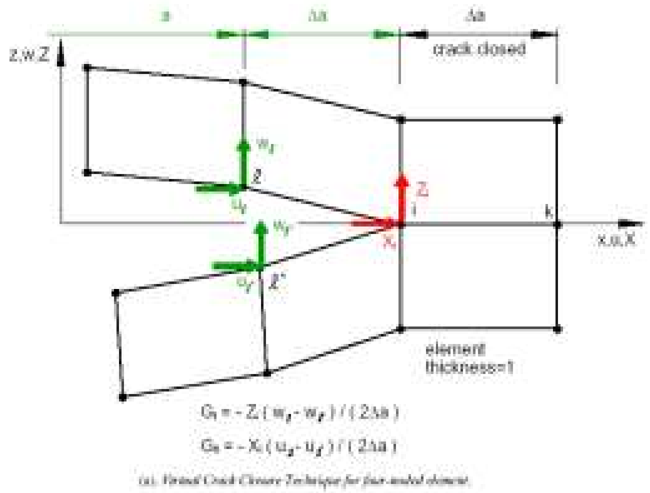
Virtual Crack Closure Technique or VCCT use a fracture mechanics approach to delamination

Originally the method was used for cracks in isotropic materials and it has had great success

More recently it has been used to model delamination

The sketch at right shows a pair of nodes spanning a delamination that has just ooccured.

The displacements of the nodes are known and the force required to oppose the opening action and close the crack back up can be deduced from the stress state



$G_I = -Z_i\,(w_\ell - w_{\ell^*}) / (2\Delta a)$

$G_{II} = -X_i\,(u_\ell - u_{\ell^*}) / (2\Delta a)$

(a). Virtual Crack Closure Technique for four-noded element.

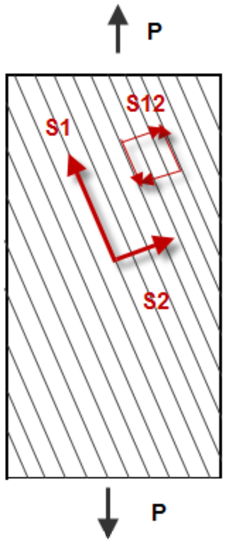
8. Advanced failure methods

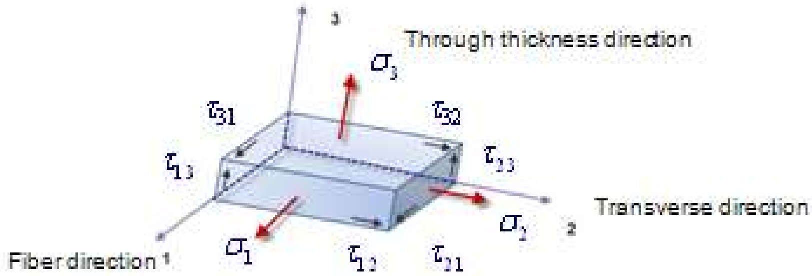
The force and displacements are known , so the energy required to close the crack is known.

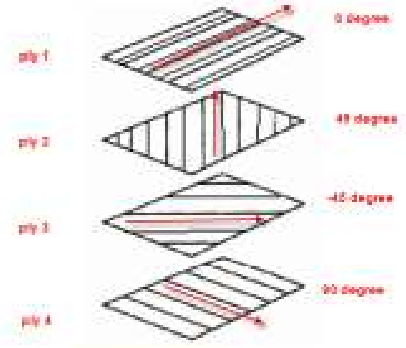
This is equal to the energy required to produce the delamination.

The rate of change of energy with respect to the crack growth rate is analogous to the Stress Intensity Factor in isotropic materials.

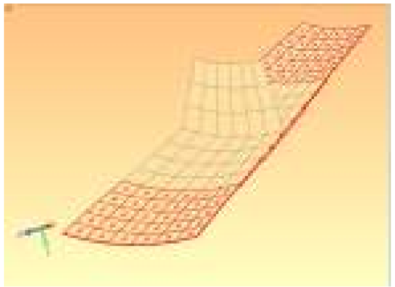
The strain energy release rate can be compared to the fracture toughness of the material to establish whether a crack will propagate or not.







A laminate made up of 4 plies



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Thank you!

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