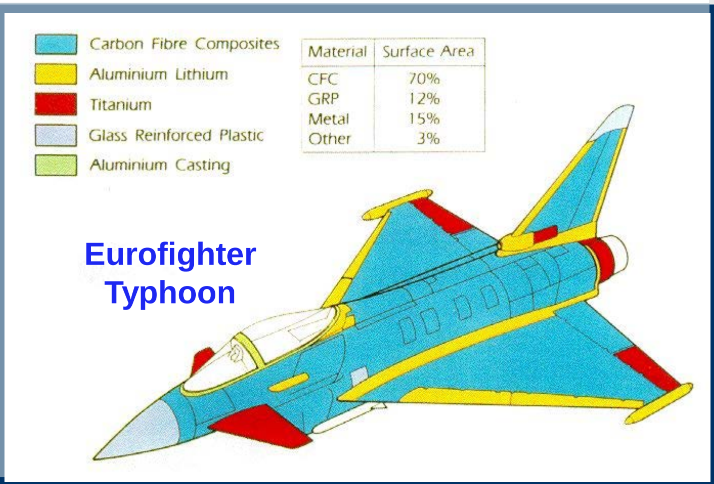
Lecture Note #2

IIT Kharagpur

2019 1st Semester

Short Course

Design of Aircraft Components using Composite Materials



Indian Institute of Technology-Kharagpur
Department of Aerospace Engineering
Professor Changduk Kong

Lecture Note

An Introduction to Composite Materials (Part 1)

Indian Institute of Technology-Kharagpur
Department of Aerospace Engineering
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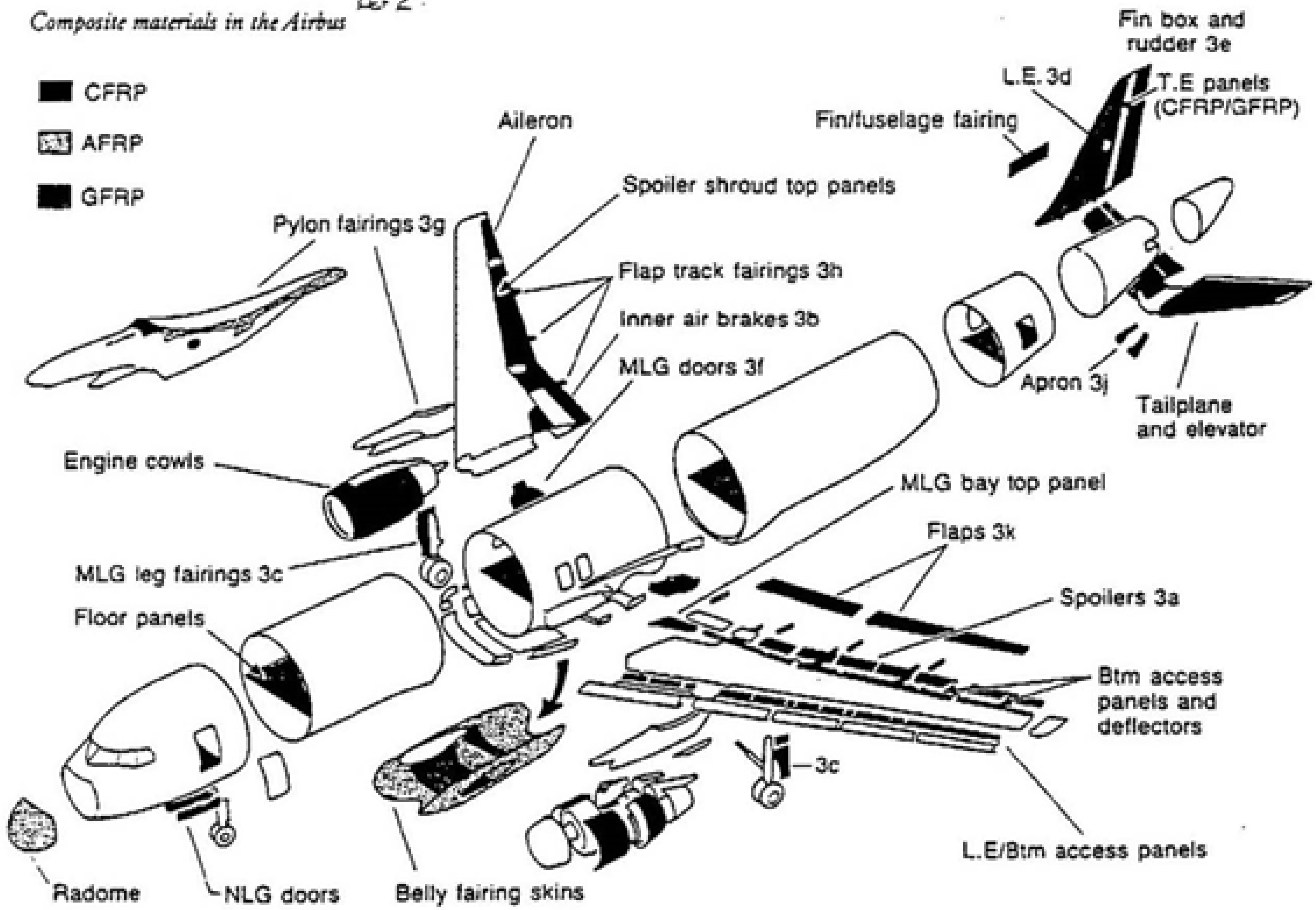
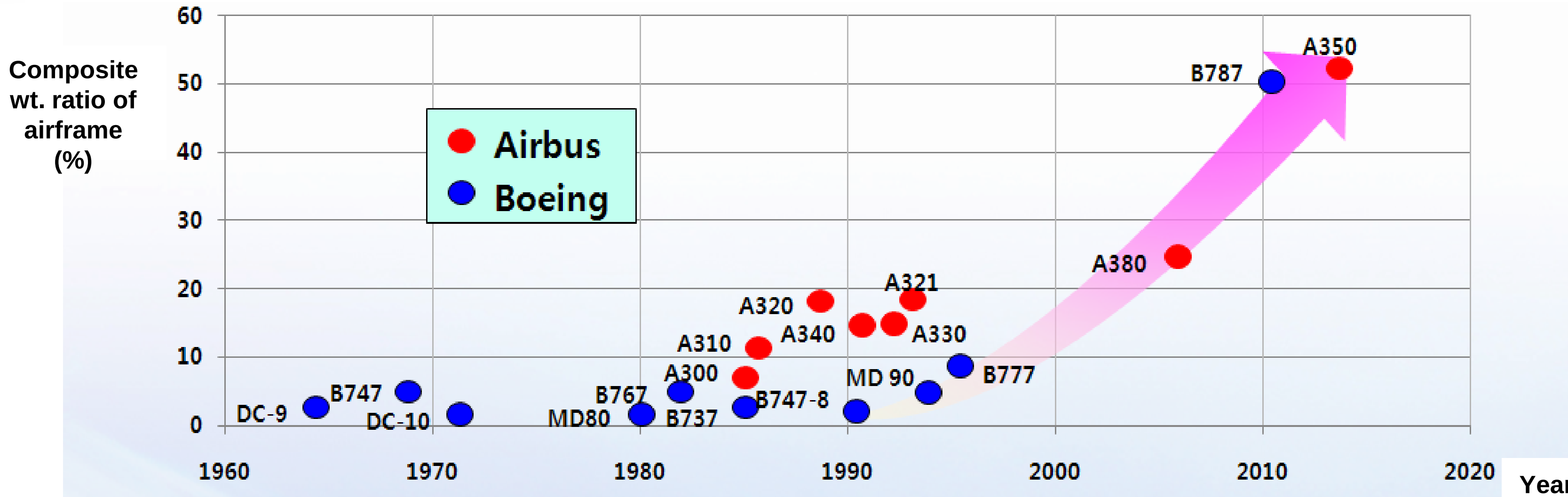
Composite materials in the Airbus

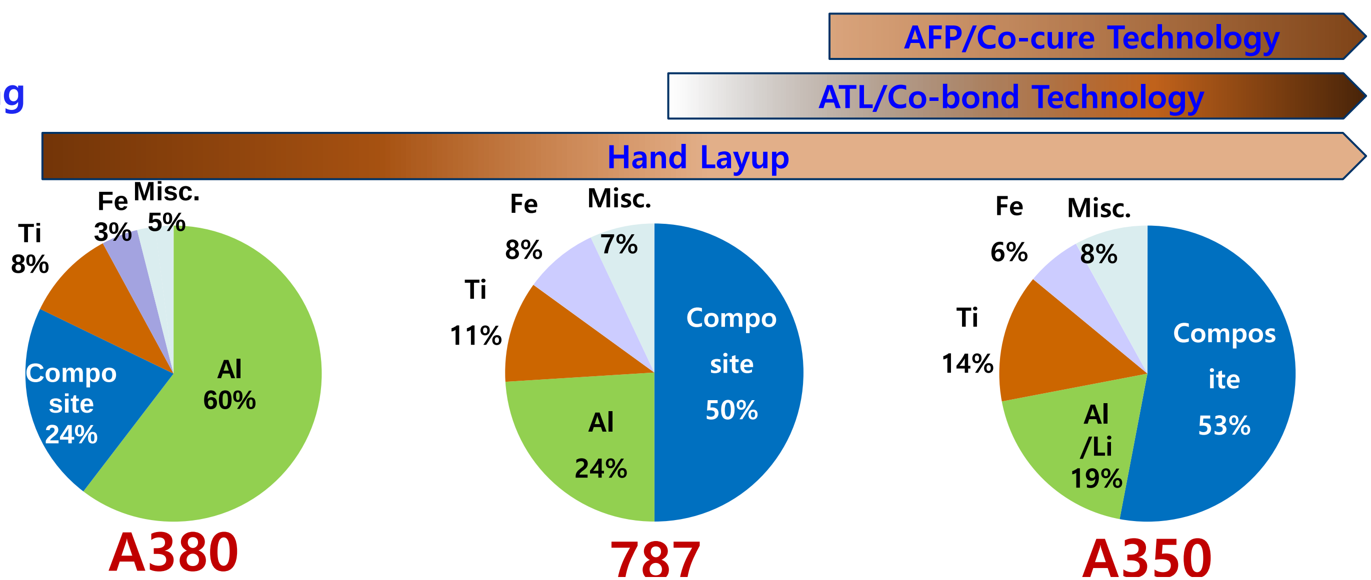


A320 composite structures

Application of Composites to Civil Aircrafts



Composite manufacturing Technology

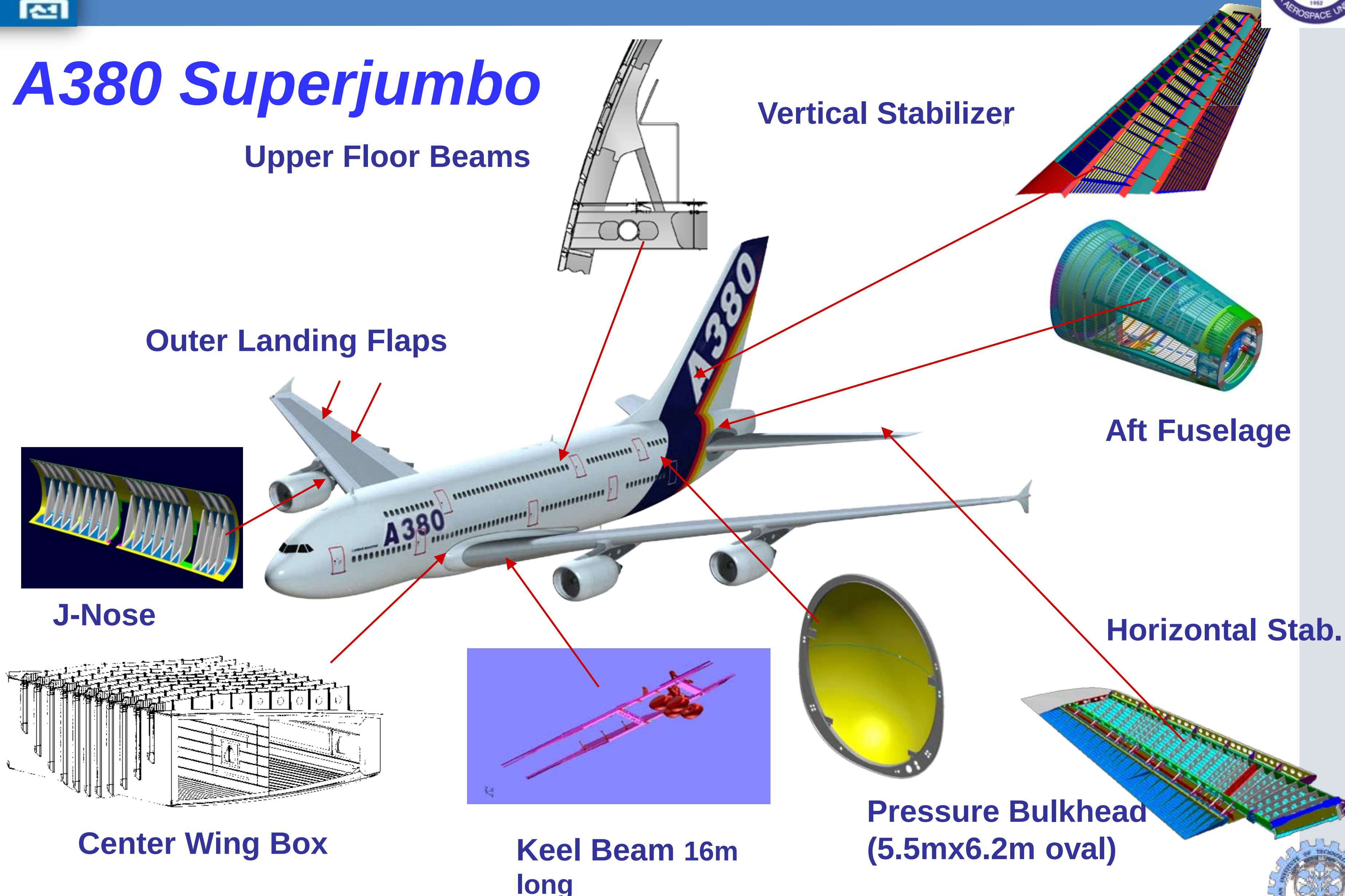


Airbus A340 - 18% CFRP

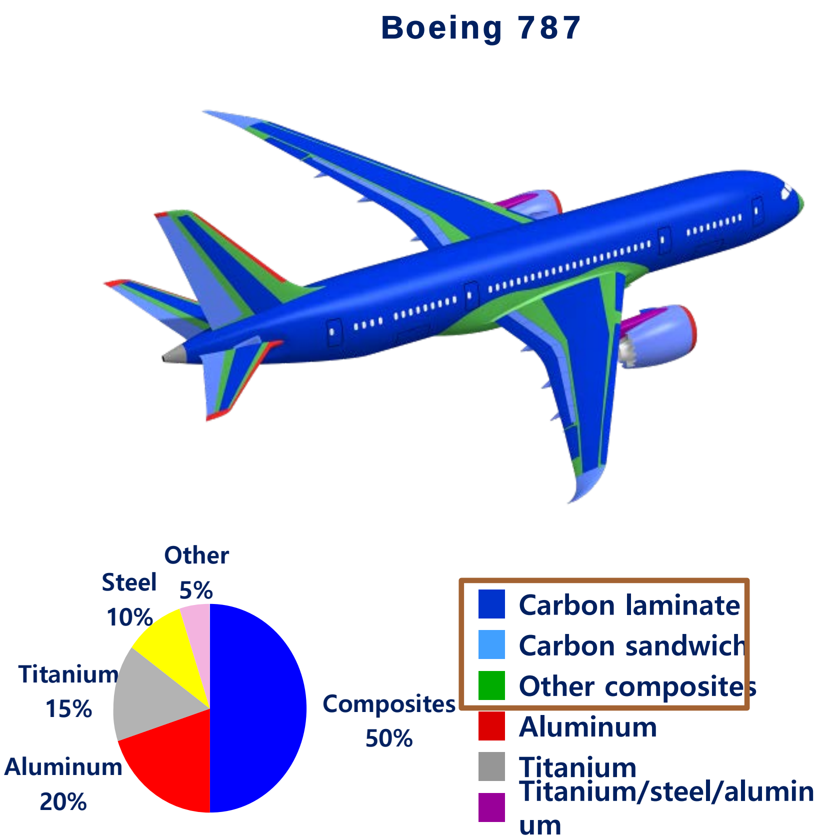
Ultra-long-range aircraft

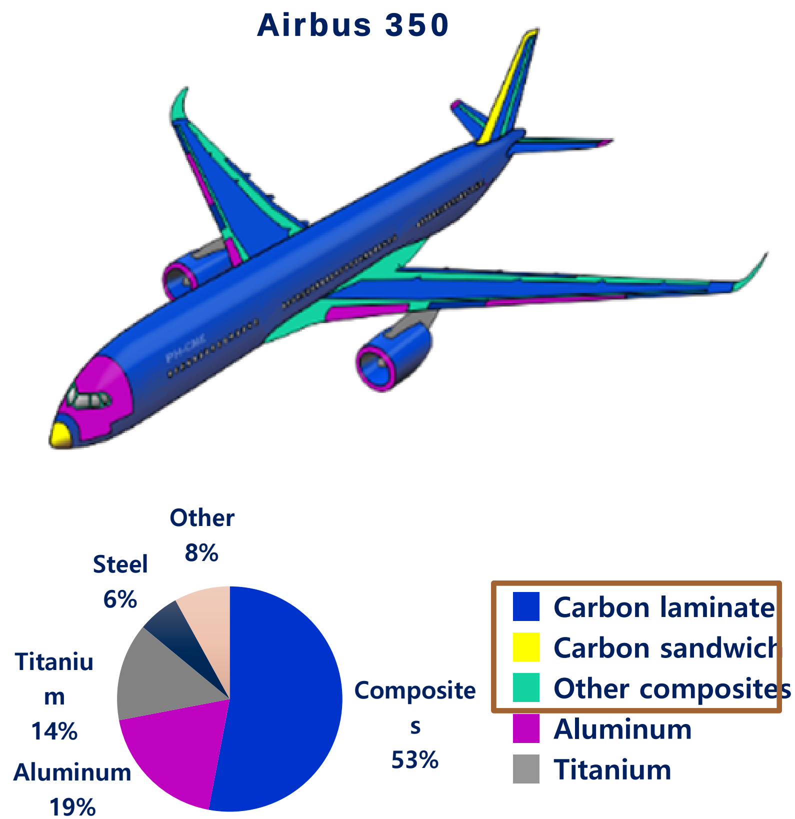
Composite keel beams and wing leading edges

A380 Superjumbo



Application of Composites to Civil Aircrafts





Composite Wt. ratio of B787 Airframe : 50% (Volume ratio: 80%)

➡ **Wt. Reduction : 10~ 20% (12~ 24 ton)**

Contents

- MATIERIALS
- MANUFACTURE
- BEHAVIOUR (Static, Long Term, Impact)
- INSPECTION, REPAIR, JOINTS

MATERIALS

Classification

Composite Materials

- Two or more constituents
- Combined for better properties

Examples:

Natural: wood, bone

Micro: alloys, mixed plastics

Macro: distinct particle or fibre reinforced Plastics/metals/carbon/ceramic

New technology of "macro-composites" as Advanced Composite Materials

The Incentives:

Lighter, Stronger, Fatigue resistant, Corrosion resistant, Fatigue resistant, Optimised directional properties Fewer components, Fewer processes …

Basic Components of macro composites: **Reinforcement, Matrix, Interface**

Matrix/Reinforcement/Interface

★
Matrix:

- **Continuous medium providing:**

- Binding
- Load transfer
- Surface Protection
- Toughness
- Wear resistance
- Chemical resistance

Types of matrixes:

- Plastics (or Resins) Thermoset: Epoxies, Polyester (Operating temperature: up to $150°C$)
  Thermoplastic: PEEK, PP (Operating temperature: up to $300°C$)
- Metals Al Alloy (Operating temperature: up to $400°C$)
- Ceramics (Operating temperature: up to $1000°C$)

$$Al_2O_3, SiC, MgO, Si_3N_4 \ ..$$

Thermoset plastics

- **Cross-linked**
- **Can not be re-softened-one shot cure**
- **Cure = polymerization by:**

Catalyst

Heat ($125,175°C$)

Pressure $100\ psi\ \ (Typically\ 0.75 \sim 1.5\,MPa\ )$

Chemical reaction!

- exotherm
- volatiles

control!

- Shelf life 1 year @ $-18°C$

Examples of thermoset plastics used of composite matrixes

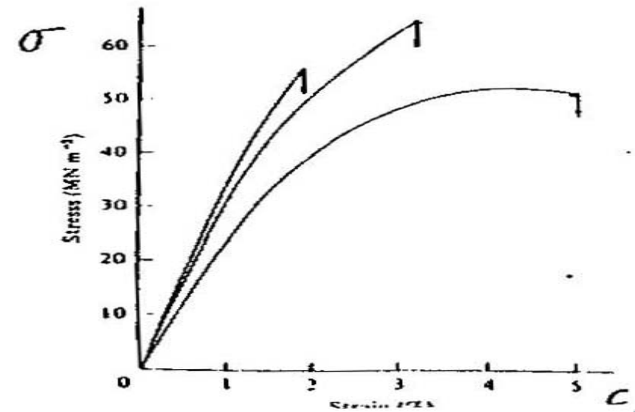
Polyesters	Operating temperature limit	$150°C$
Epoxies	Operating temperature limit	$150{,}250°C$
Polyimides	Operating temperature limit	$300°C$
Phenolics	Operating temperature limit	$300°C$

Limiting Performance

- Toughness (epoxies tend to e relatively brittle)
- Moisture (epoxies can absorb up to 2% weight of water which tends to plasticize i.e. soften and degrade the material)
- Temperature (typically up to a maximum temperature of 150 DegC)

Stress Strain behaviour

(tends to be non-linear with low strengths and failure strains from 1 to 4%)



Strain (%)

ε

Thermoplastic

- Not cross-linked
  Only entanglements + Van der Wals

- Can be re-softened
  Multi-stage processing

- No chemical reaction
  just melt + solidify
  cooling rate → crystallinity

- Healing

- Re-use of scrap

- Infinite shelf life

- High processing temperatures and pressures: 300-400 °C
  200-400 psi
  ancillary materials!

Examples of thermoplastics used for composite matrixes

Operating temperature

- PAK's PEEK (ICI, APC2) T_g 145 °C
- PAS's: PPS (Phillips, Avtel) T_g 90 °C
- Polysulphone (Amoco, Radel) T_g 215 °C
- PEI's: Polyetherimide (GE, Ultem) T_g 216 °C

(T_g =Glass Transition Temperature $\rightarrow T_{operation}$)

Properties

- General mechanical properties → lower than Thermosets
- Tougher → Impact resistance
- Negligible moisture < 0.2% → Moisture resistance
- Good outgassing, radiation, cryogenic performance → Space applications

Limiting Performance:

- Difficult to process (high temperatures and pressures)
- Expensive material and ancillaries

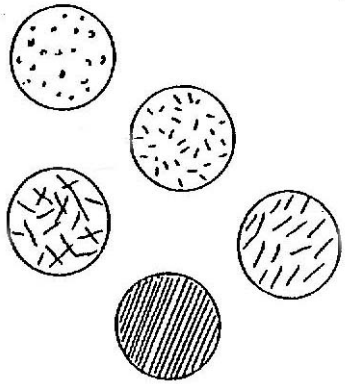
Reinforcement

Discontinuous medium providing:

- High strength
- High stiffness

Types of reinforcements:

- Particulate
- Fibrous
  - Short
  - Long
  - Continuous
    - Random
    - Aligned



E.g.:

Carbon HS, HM
Glass E, S
Polymeric Kevlar
Hybrids
Metal
Boron
Ceramic SiC

Common Fibres

❖ **Graphite** : heat treated more than 1700°C
Carbon : heat treated less than 1700°C

Carbon (Graphite) → Strength & stiffness
High strength
High modulus
Intermediate modulus

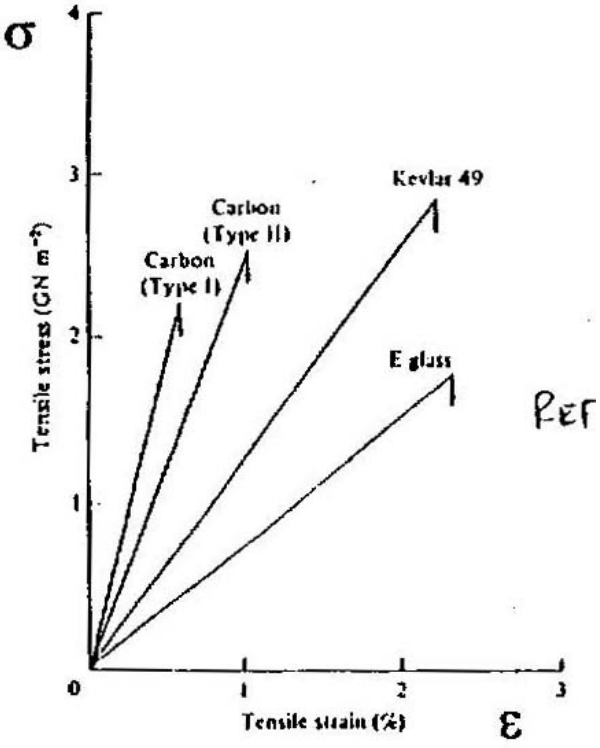
Kevlar → Strength & toughness
Glass → Cheap all-rounder
Hybrids

C

F S.

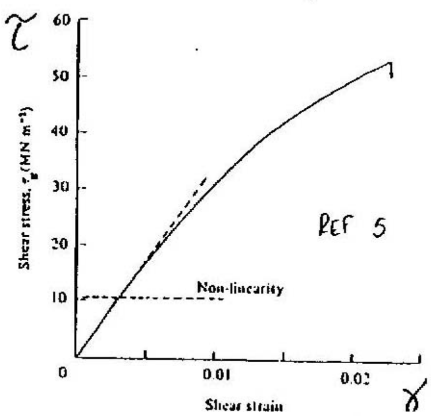
K

G



Interface

Fibre-matrix

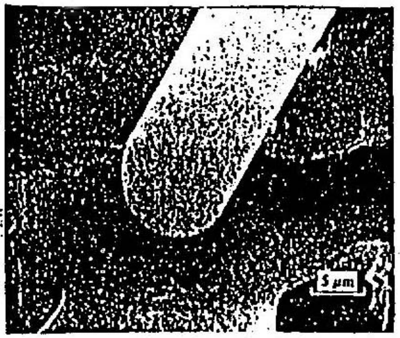


CRITICAL

Non-linear response

Bond Strength
→ Shear
→ Toughness

Sizing agent
→ Fibre protection
→ Wetting
→ act as Coupling Agent



untreated

Treated by sizing agent

Terminology

Materials

Fibres Tows e.g. 5 or 10k filaments

Matrix

Interface

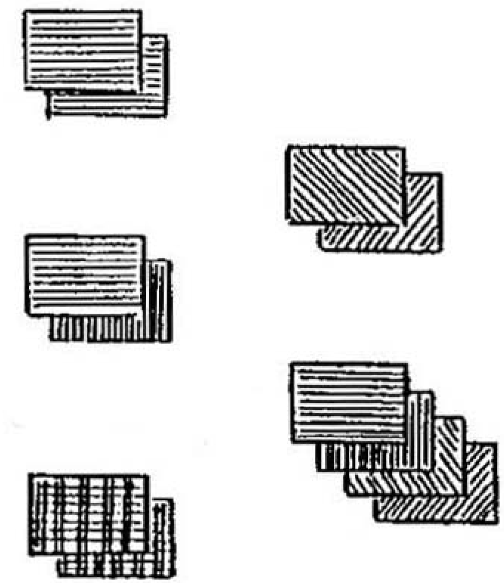
Cores Honeycomb, foam, syntactic

Lamina Ply or Layer
UD tape or Woven fabric

Laminate

UD: all 0° layers
Angle-ply: ±θ, usually ±45°
Cross-ply: 0, 90°
Quasi-isotropic: 0, ±45, 90°

Woven weave and weft patterns



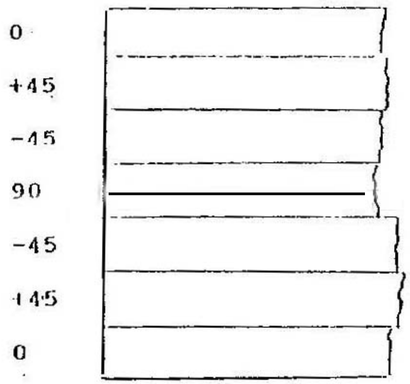
Material Properties

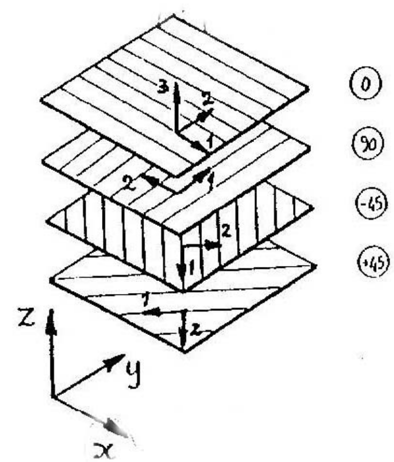
$$\left.\begin{array}{l} Homogenous \\ \\ isotopic \end{array}\right\} Metals$$

$$\left.\begin{array}{l} Heterogeneous \\ \\ Anisotropic \\ \\ Orthotropic \end{array}\right\} Composites$$

Laminate Codes

e.g. $| 0, \pm 45, 90^\circ |_s$



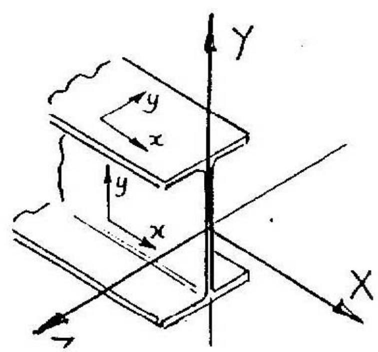


Axes Systems

Lamina Material Axes 1 2 3

Laminate Plate axes x y z

Component Structural axes X, Y, Z



Fibre Weight Fraction

$$w_f = \frac{W_f}{W_f + W_m}$$

Fibre Volume Fraction

$$v_f = \frac{V_f}{V_f + V_m}$$

Voidage (Porosity)

$$v_c = v_f + v_m + v_v = 1$$

$$v_V = 1 - (v_f + v_m)$$

$$(V_f = W_f/\rho_f \ \& \ V_m = W_m/\rho_m)$$

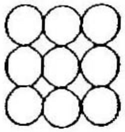
$$(Approximately : v_m = 1 - v_f)$$

Fibre Packing

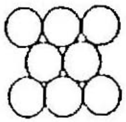
E.g. for UD lamina

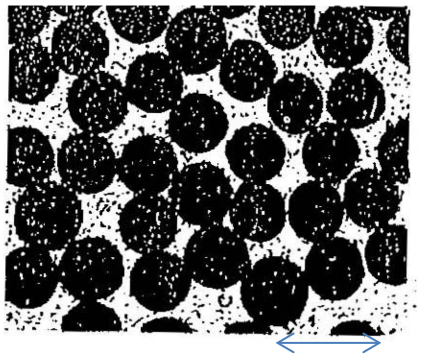
Theoretical

Square array:
75% fibre vol. fraction



Hexagonal array:
90% fibre Vol. fraction





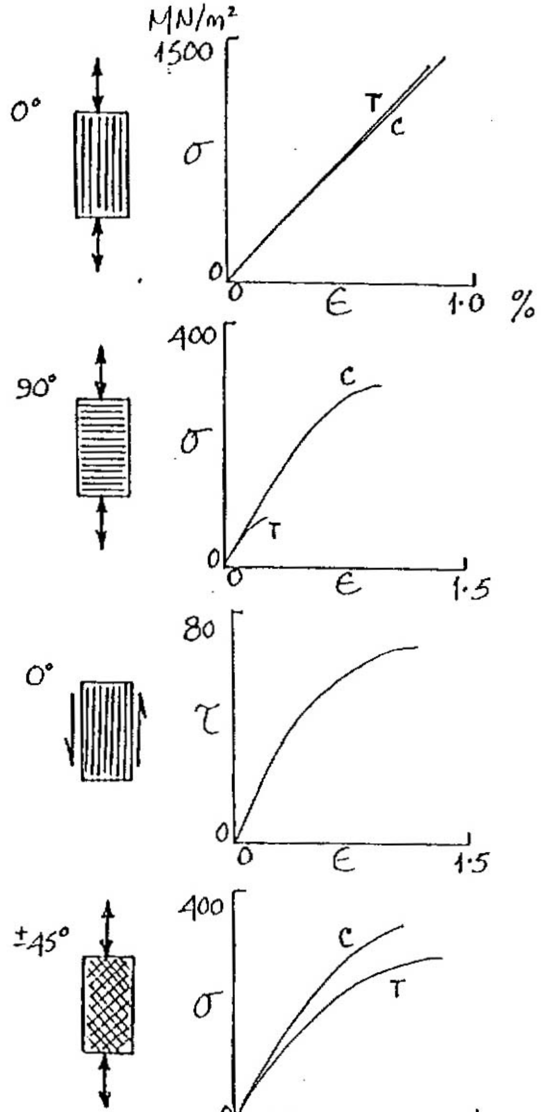
Actual vol. fraction = 60-70%

Overall response

Fibre dominated – linear

Matrix dominated – non linear

Interface dominated – non linear



COMPARISON OF PROPERTIES

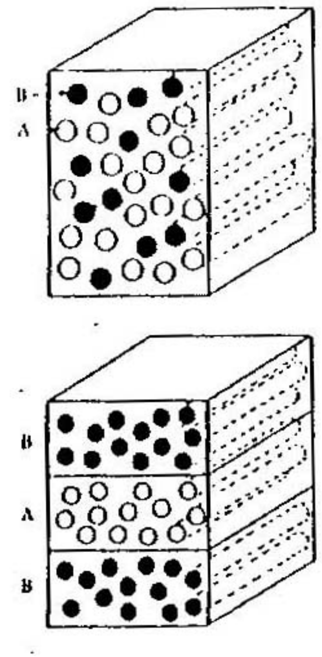
PROPERTY	UNIT	FIBRES		RESINS	COMPOSITES (UD)	METALS
		C(HM) C(HS) G(E)	K(49)	Epoxy Pestr	Vf 0.6 C/Ep G/Ep K/Ep (HM) (E) (49)	Steel Ally low
E_0 E_{90}	GN/m2 GN/m2	390 250 76 12 20	125	4.5 3	140 40 84 9 8 6	200 72
G	GN/m2				4.8 4 2.1	
TS_0 TS_{90}	MN/m2 MN/m2	2200 2800 2000	3000	50 50	1400 800 1450 40 36 39	1500 530
CS_0 CS_{90}	MN/m2 MN/m2			150 150	-1250 -600 -300 -200 -200 -150	
EL_0 EL_{90}	% %	0.5 1.0 2.0	2.5	1 1.5	0.8 1.8 2.1 0.6 0.45 0.6	20 11
Density	Mg/m3	1.95 1.75 2.56	1.45	1.3 1.3	1.6 2.0 1.4	7.8 2.8

Composite Material Types and Forms

Material Type

- *Epoxy*
- *Polyimide*
- *PEEK*

+ C, K, G / hybrid



Hybrid with A and B fibers

Material Form

1. Wet lay-up form

"Dry" tows, UD/stitched or Woven fabrics
Separate "wet" matrix resin

Thermoset resin poured/squeezed into fibre forms

(Thermoplastic resin melted in)

2.Prepreg form

Pre-impregnated tows, UD tapes or woven fabrics
i.e. with matrix resin already included

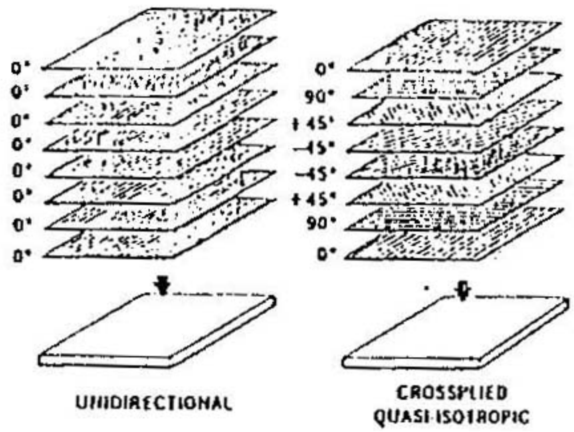
Laminates

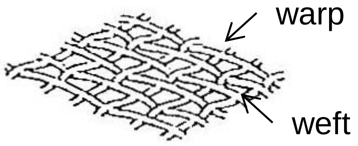
Stacked Orientated Layers

1. Made from UD tapes

- Controlled directional props
- Unkinked fibres
- High strength, stiffness
- Compression stability
- Flat, single curvature i.e. “low drape”
- Highly loaded

Applications: spars, wing skins, stiffeners



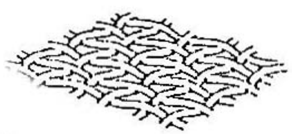


Twill weave

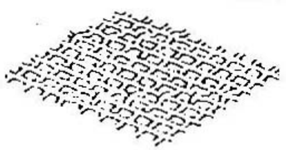
2. Made from Woven fabrics (or stitched tapes)

- Weave, weft control
- Kinked fibres
- Lower strength, stiffness – esp. compression
- Good impact resistance
- Complex curvature i.e. drapable
- Moderately loaded

Applications: cowlings, nacelles, flaps, leading edges



Plain weave

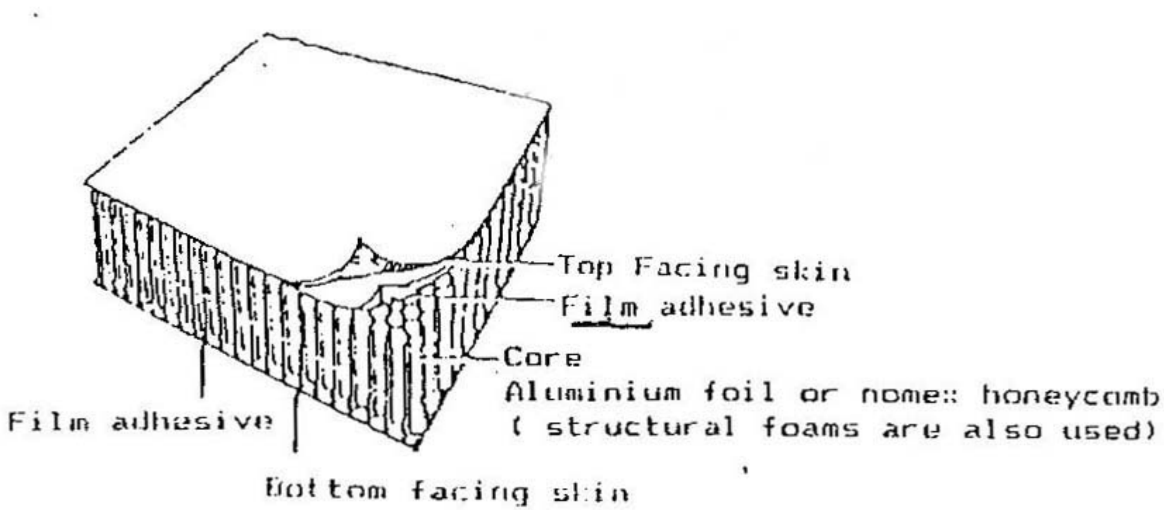


Satin weave

Sandwich Panels

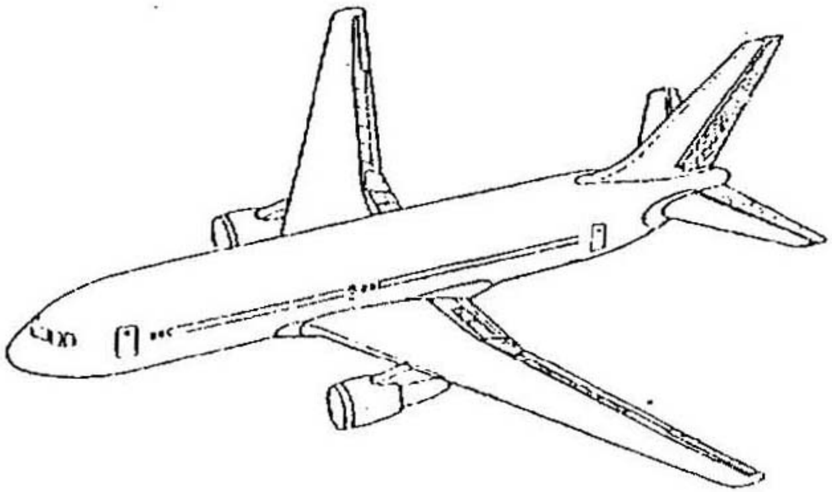
- Using cores of metal or Nomex honeycomb or foams
- High bending stiffness
- Lightweight
- Moisture ingress!

E.g. Applications: flaps, flooring, stiffeners



Choice of Material Type and Form

- Type of Structure
- Operational loading
- Operational temperature



Type of Structure

· **Panels**

Flat

Single curvature

Double curvature

Stiffened

Discrete

e.g. stringers

Sandwich

e.g. honeycomb or foam

· **Continuous sections**

Open L T Z U

Closed O

Complex #

Operational Loading

High

Moderate

Low

Direct Loads - 0° fibres
Shear - ±45° fibres
Transverse - 90° fibres

Operational Temperature

Airframe

Subsonic: $-30 \sim 60^o\ C$

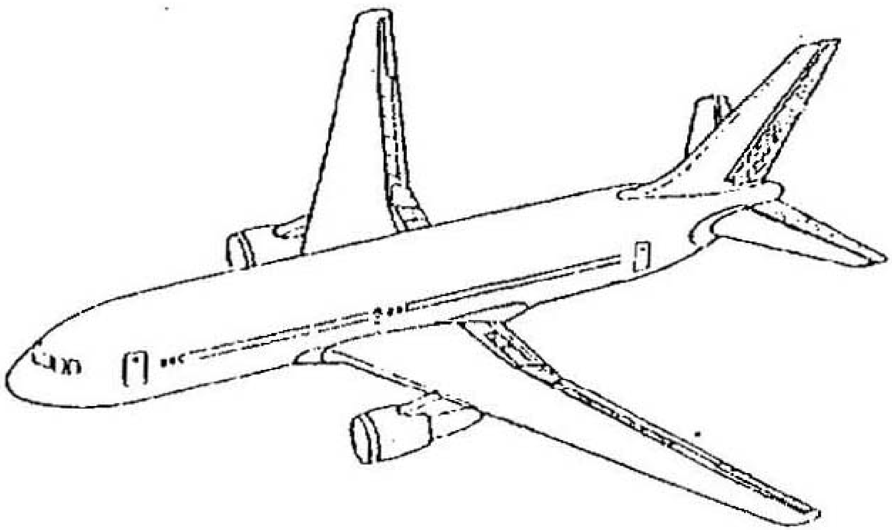
Supersonic: $-30 \sim 150^o\ C$

Engine

Outer

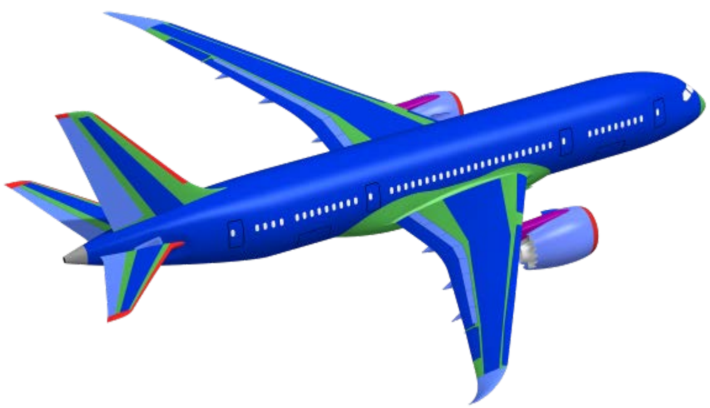
Inner: $300 \sim 400^o\ C$

Hot: $1000^o\ C$

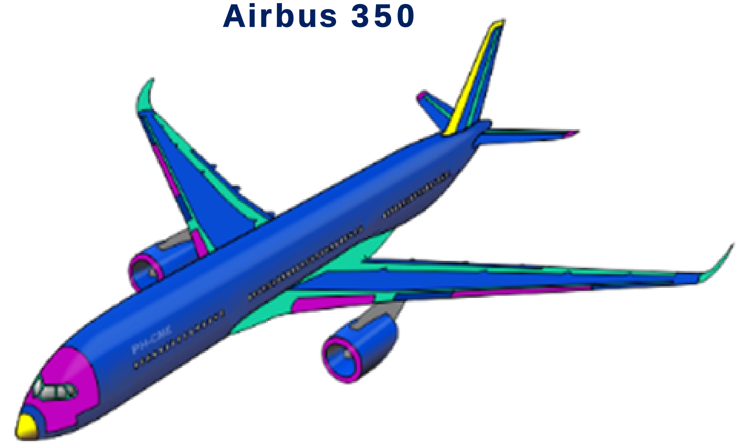


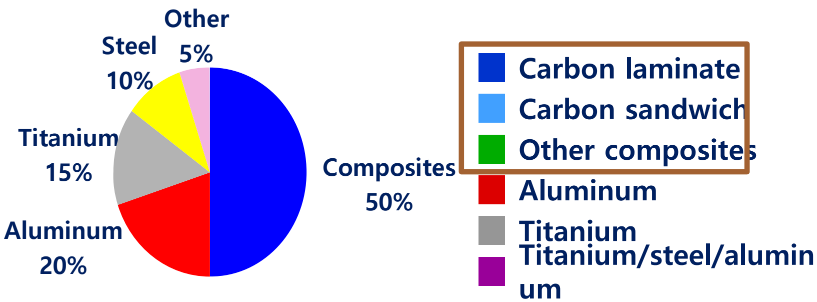
Application of Composites to Civil Aircrafts

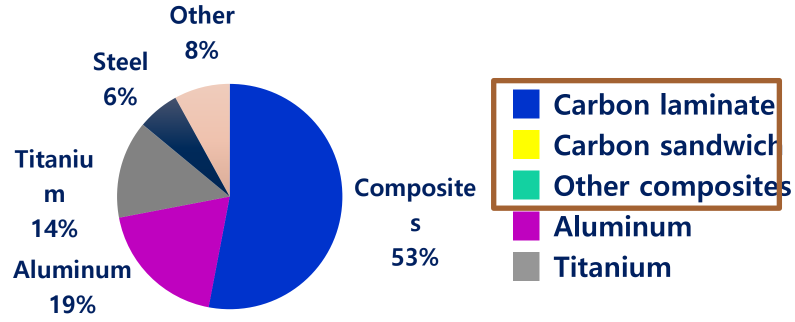
Boeing 787



Airbus 350







Composite Wt. ratio of B787 Airframe : 50% (Volume ratio: 80%)

Wt. Reduction : 10~ 20% (12~ 24 ton)

MANUFACTURE

Many parameters → product quality

Basic Processes:

Lay-up

Heat and Pressure

- Thermosets : heat-reaction-cure cycle → typically 4~8 hours ‘one shot’
- Thermoplastics : heat-melt-solidify cycle

Impregnation

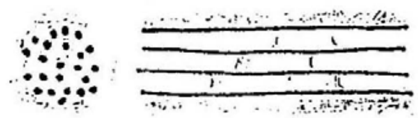
- Matrix → fibre tows

Consolidation

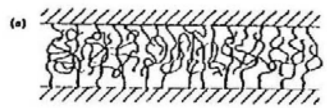
- Lamina : lamina
- Laminate compaction

Mechanisms of bonding

- Absorption and wetting

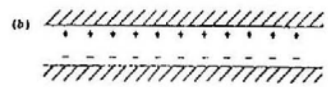


- Inter-diffusion (Autohesion)

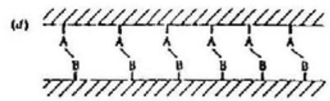


- Thermoplastic resin

- Electrostatic attraction

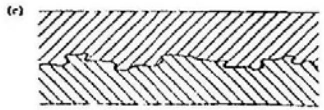


- Chemical bonding



- Thermoset resin

- Mechanical adhesion

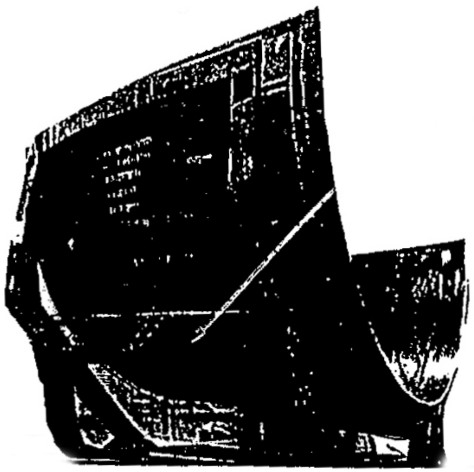


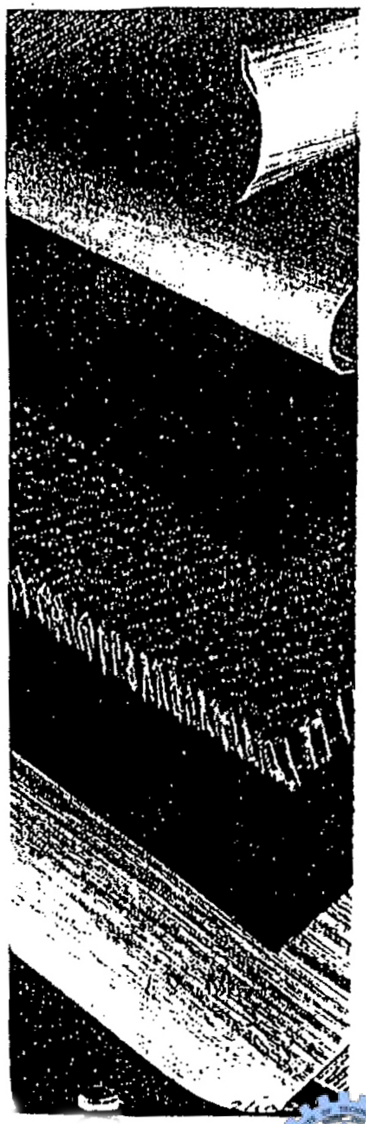
Manufacturing Techniques

Material form! / Structural form!

Hand lay-up

Labour intensive





Tooling

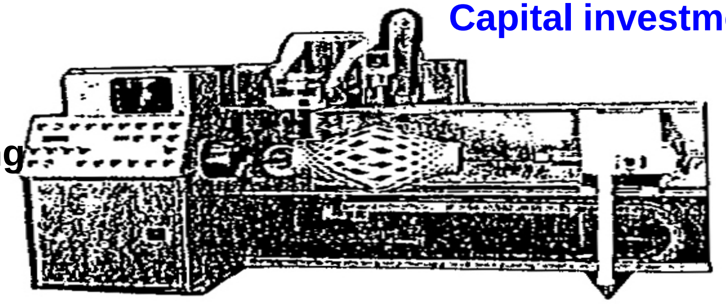
For low CTE
(Coeff. of Thermal Expansion)

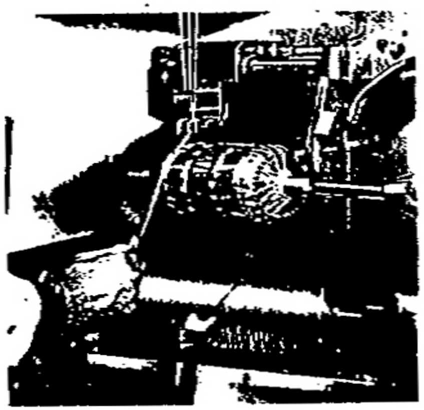
CFRP tooling

Auto-layup

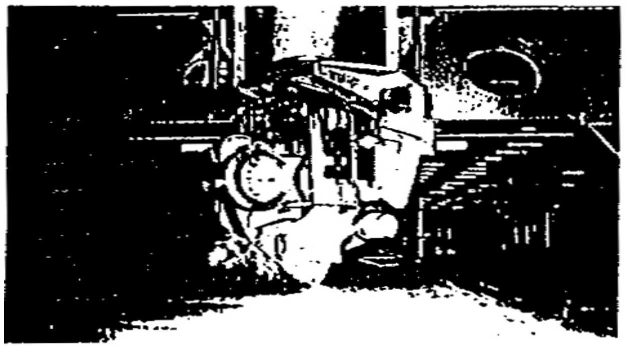
Capital investment!

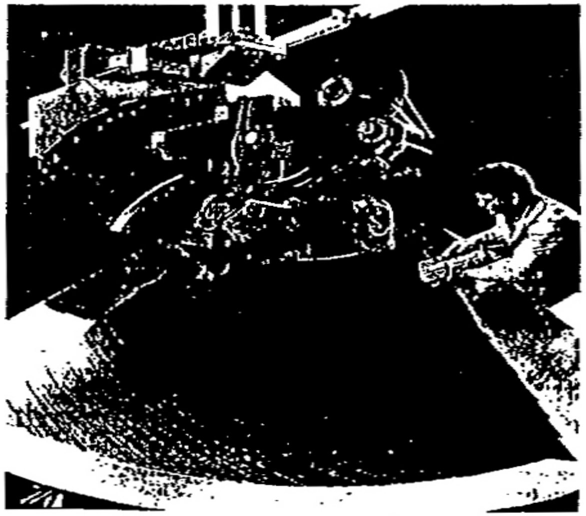
- **Filament Winding**





- **Tape Laying**





By courtesy of The LTV Corp

(b) Curved contoured surface

Fig. 4.3.3 A Contoured Tape Laying Machines (cont'd)

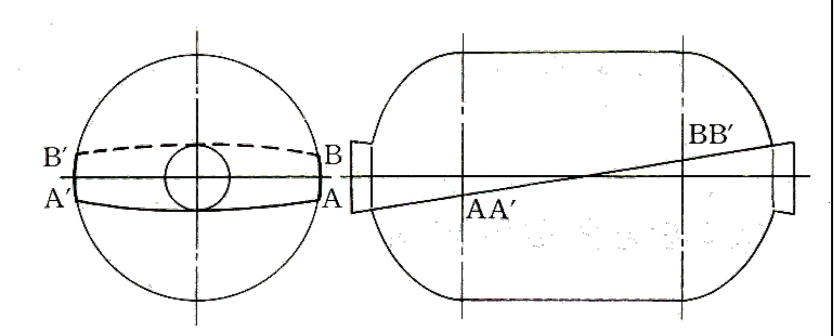


그림 5-9 폴라 와인딩 패턴

Polar winding pattern

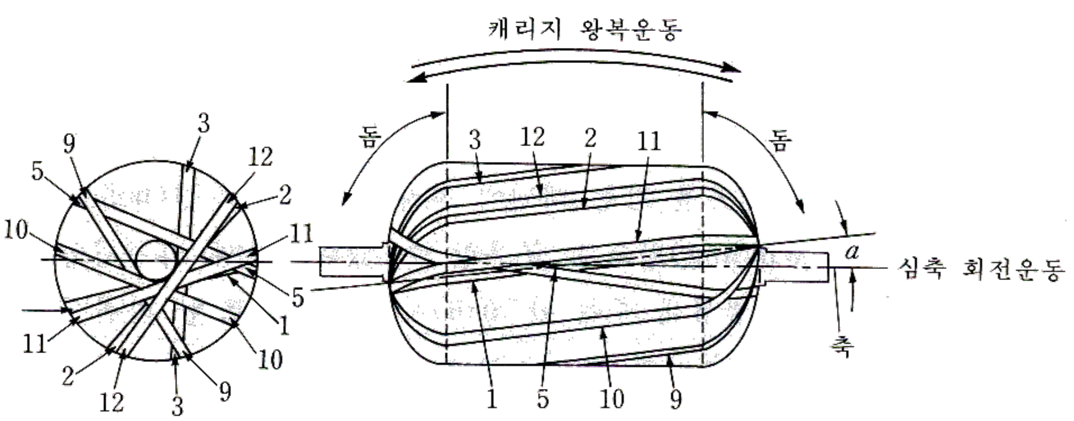
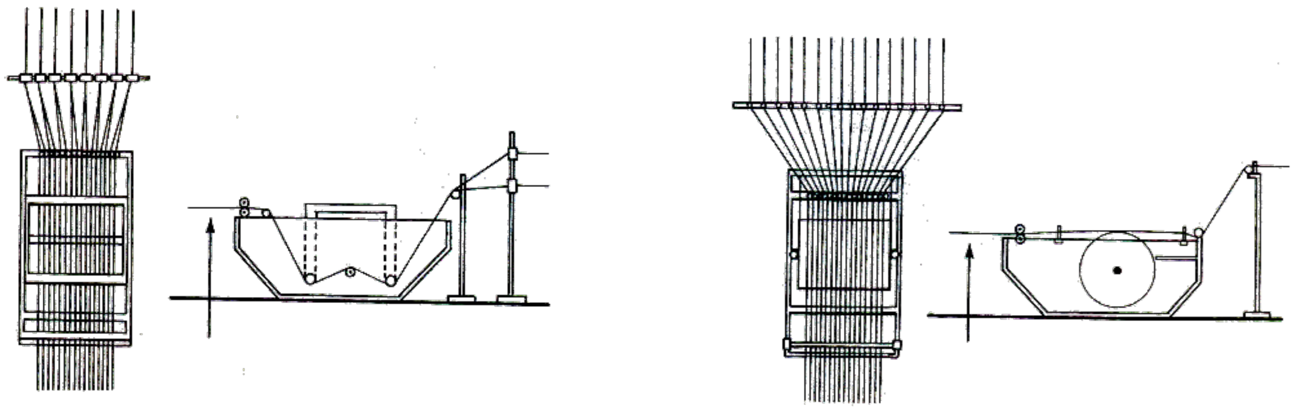


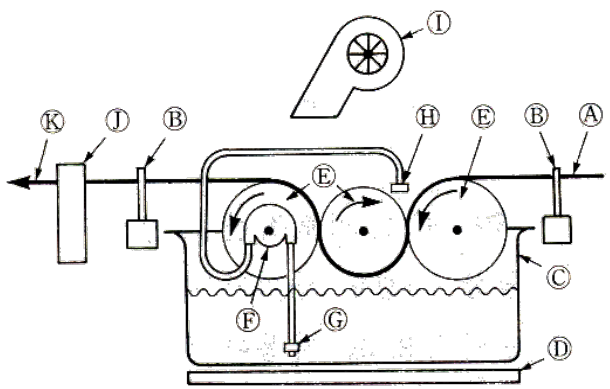
그림 5-10 헬리컬 와인딩 패턴

Helical winding pattern



(a) (b)

그림 5-12 수지함침 용기 **Resin tank**



Ⓐ 필라멘트 보빈(bobbin)에서 나온 건조 로빙띠
Ⓑ 필라멘트를 정렬하기 위한 빗살 구조
Ⓒ 수지용기
Ⓓ 용기 가열기—전열판
Ⓔ 함침롤러(3)
Ⓕ 수지공급펌프
Ⓖ 계량기가 달린 펌프입구
Ⓗ 수지를 투입하는 출구
Ⓘ 더운 공기 송풍장치
Ⓙ 필라멘트를 정렬하기 위한 와인딩 고리
Ⓚ 심축에 와인딩되는 습식 필라멘트띠

그림 5-13 특수구조의 수지함침 용기

Special equipped resin tank

Winding tension is controlled by band holder

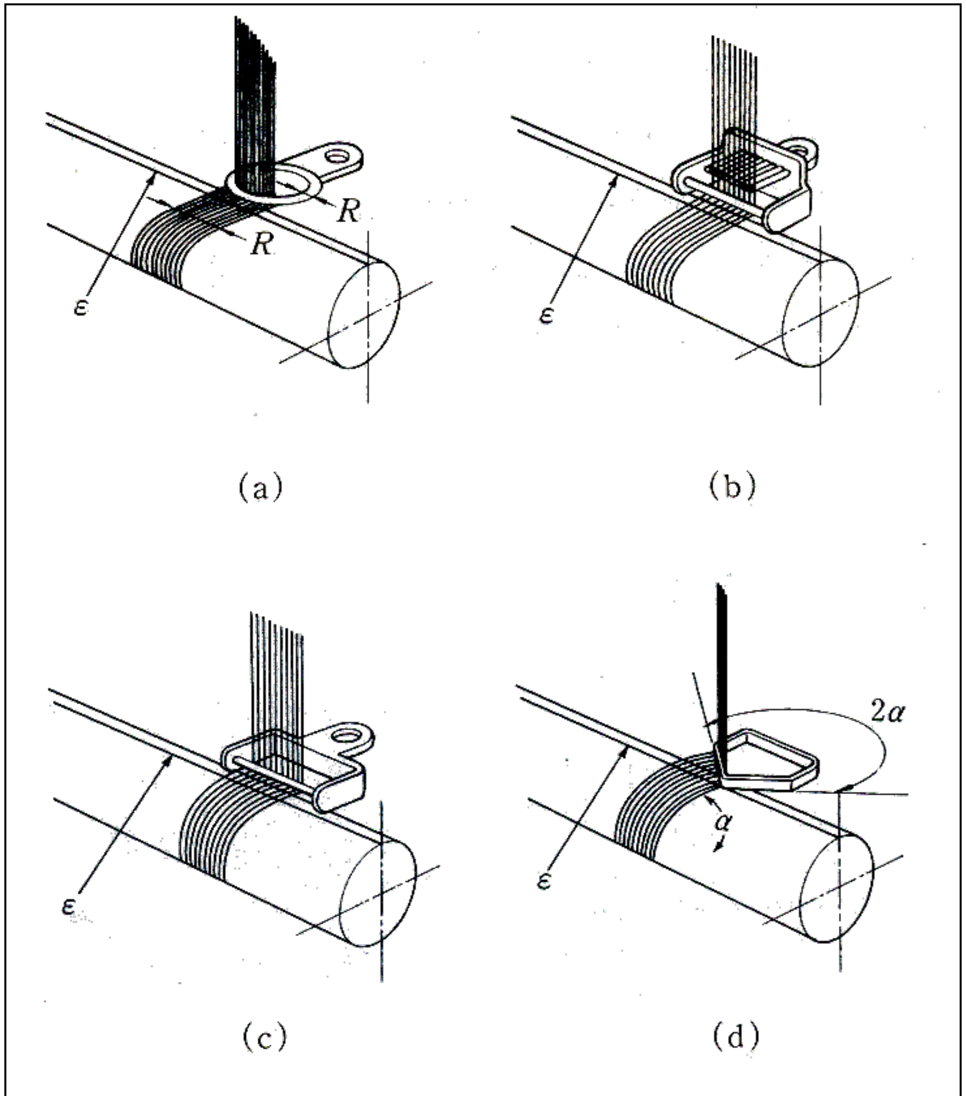


그림 5-14 대표적인 밴드고리 형상

Filament winding stress analysis

1) Netting (or Helical) Winding Analysis Method

〔기　　호〕

F	섬유방향 하중	Fiber directional load	Y	원주방향 길이	Circumferential length
F_h	원주방향 하중	Circumferential load	S	섬유방향 응력	Fiber directional stress
F_l	축방향 하중	axial directional load	S_h	원주방향 응력	Circumferential stress
α	와인딩 각도	Winding angle	S_l	축방향 응력	axial directional stress
t	두께	thickness	S_f	섬유가 받는 응력	Fiber stress
X	축방향 길이	axial directional load	W	밴드폭	Band width

$$F_h = 2F\sin\alpha$$

$$S_h = \frac{F_h}{Xt} = \frac{2F}{Xt}\sin\alpha$$

여기서, $S_f = \frac{2F}{Wt}$, $W = X\sin\alpha$ 이므로

$$S_h = S_f \sin^2\alpha$$

이다. 또한, $F_l = 2F\cos\alpha$ 이므로

$$S_l = \frac{F_l}{Yt} = \frac{2F}{Yt}\cos\alpha$$

이다. 여기서, $W = Y\cos\alpha$, $S_f = \frac{2F}{Wt}$ 이므로 다음과 같이 된다.

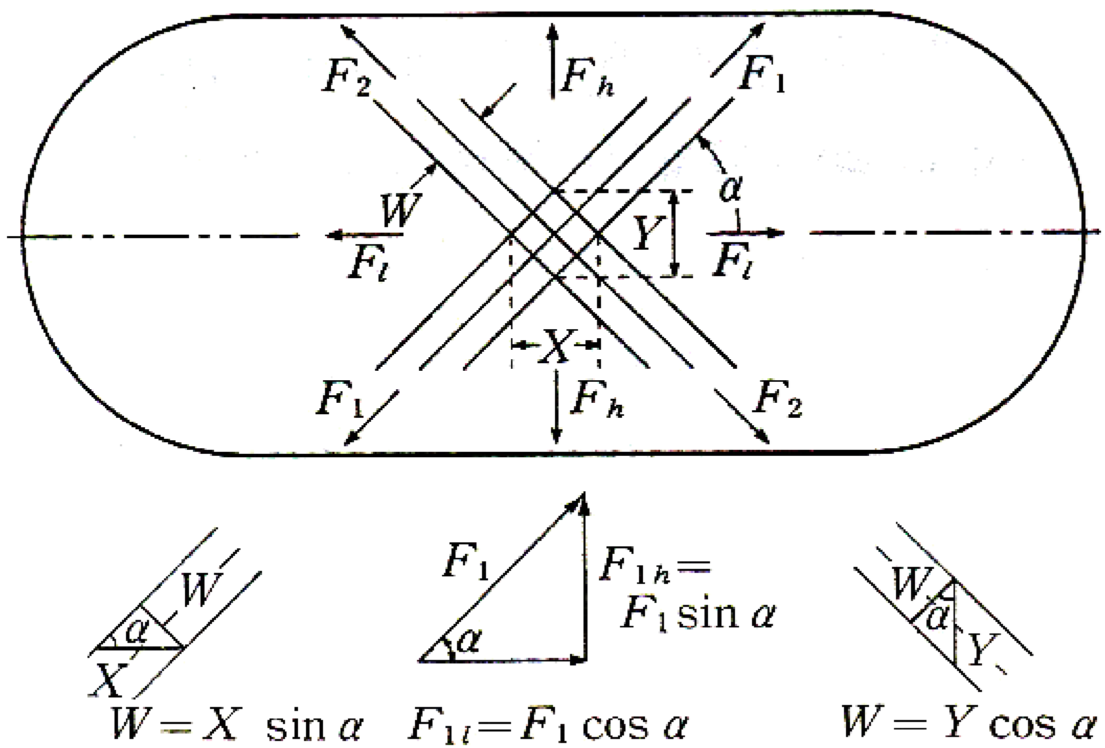


그림 5-16 망상구조 응력해석도

$$S_l = S_f \cos^2 \alpha$$

Hoop stress S_h = 2x axial stress S_l

$$\frac{S_h}{S_l} = \frac{S_f \sin^2 \alpha}{S_f \cos^2 \alpha} = \tan^2 \alpha = 2$$

$$\tan \alpha = \sqrt{2}$$

$$\alpha = 54.75^\circ$$

· This is optimal winding angle for long pipe loaded by internal pressure!

2) Compound Winding Stress Analysis Method

[Assumption]

i) All fibers have same tension force
ii) No bending load

iii) Pressure vessel: axisymmetric

Ⅳ) Relatively thin to diameter

Ⅴ) No shear stress between fibers

Ⅵ) Compound winding using hoop, helical and axial winding

Ⅶ) All fiber is continuous

If all fibers have same stress S;

$$S_h = \frac{PD}{2t} \qquad \text{(a)}$$

$$S_l = \frac{PD}{4t} \qquad \text{(b)}$$

$$S_h = \frac{S_h' t_h}{t} + \frac{S_{h\theta} t_\theta}{t} \qquad \text{(c)}$$

$$S_l = \frac{S_l' t_l}{t} + \frac{S_{l\theta} t_\theta}{t} \qquad \text{(d)}$$

t: Total winding thickness

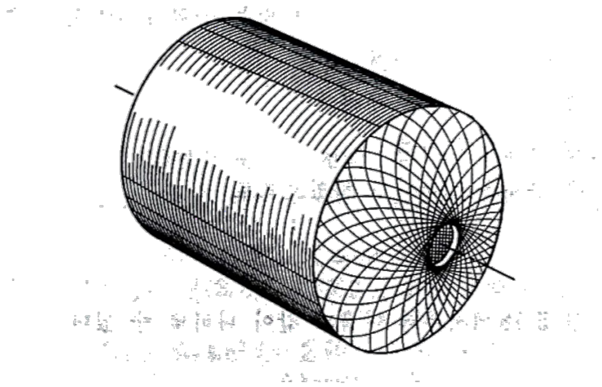


그림 5-17 원주방향, 헬리컬, 축방향 와인딩이 복합된 압력용기 구조

S_h' : hoop stress of hoop winding fiber

S_1' : axial stress of axial winding

$S_{h\theta}$: hoop stress of helical winding fiber

$S_{1\theta}$: axial stress of helical winding fiber

t_h : hoop winding fiber thickness

t_θ : helical winding fiber thickness

t_1 : axial winding fiber thickness

$$S_h' = S_l' = S \quad \text{(e)}$$

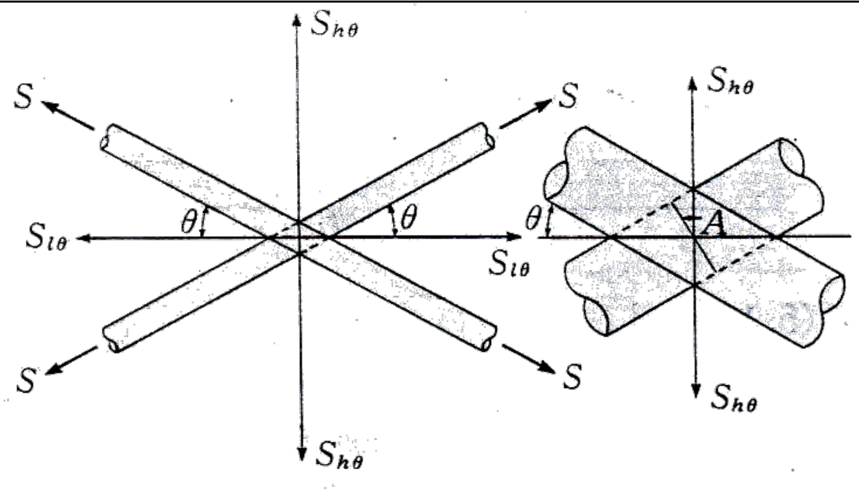


그림 5-18 헬리컬 와인딩 섬유의 응력구조도

$$S_{h\theta} = S \sin^2\theta \quad \text{(f)}$$

$$S_{l\theta} = S \cos^2\theta \quad \text{(g)}$$

*using netting winding results

Substitute (f, g) for (c, d)

$$S_h = \frac{S \cdot t_h}{t} + \frac{S \sin^2\theta t_\theta}{t} \quad \text{(h)}$$

$$S_l = \frac{S \cdot t_l}{t} + \frac{S \cos^2\theta t_\theta}{t} \quad \text{(i)}$$

$$S_h + S_l = S \quad \text{(j)}$$

$$S_h = 2 S_l = \frac{PD}{2t}$$

from (j), (k)

$$t = \frac{3PD}{4S} \quad \text{(l)}$$

***Total thickness is independent of winding angle and only function of pressure, diameter and fiber's strength!**

from (h), (i)

$$St_h + St_\theta \sin^2\theta = 2St_l + 2St_\theta \cos^2\theta$$

$$t_\theta = \frac{2t_l - t_h}{\sin^2\theta - 2\cos^2\theta}$$

$$= \frac{2t_l - t_h}{1 - 3\cos^2\theta} \qquad \text{(m)}$$

$$t = t_\theta + t_h + t_l \qquad \text{(n)}$$

from (m), (n)

$$t_l = \frac{t}{3} - t_\theta \cos^2\theta$$

$$t_h = \frac{2t}{3} - t_\theta \sin^2\theta$$

*** If helical winding thickness and angle are given, hoop and axial thicknesses can be decided! .**

Auto-layup

★ Pultrusion

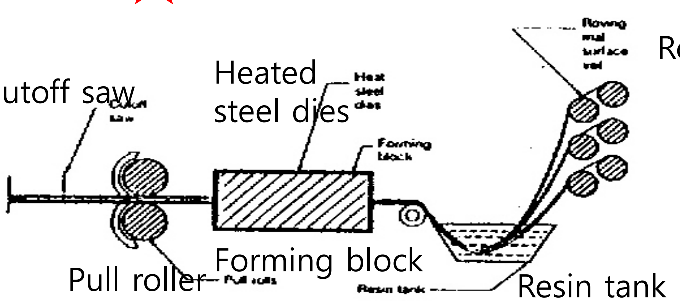


Fig 4.8.1 Typical Pultrusion Process

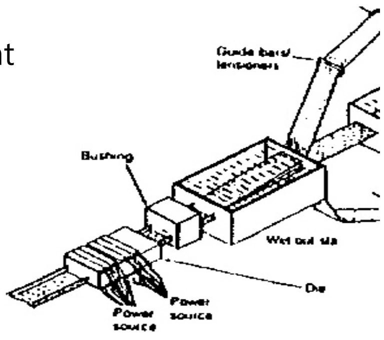
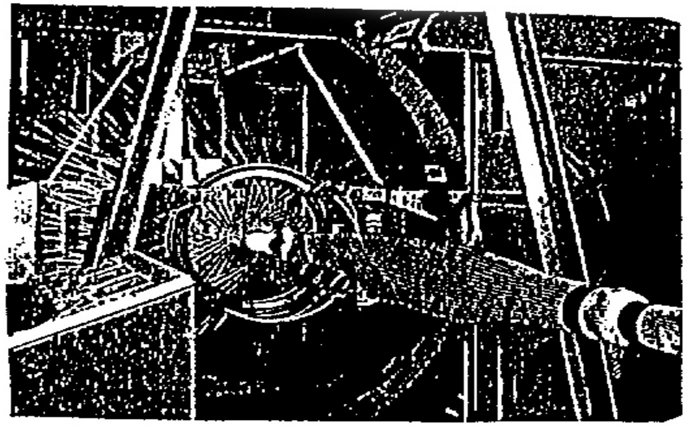
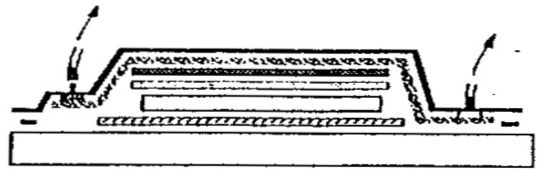


Fig 4.8.2. Pultrusion Method Utilizi

★ Braiding

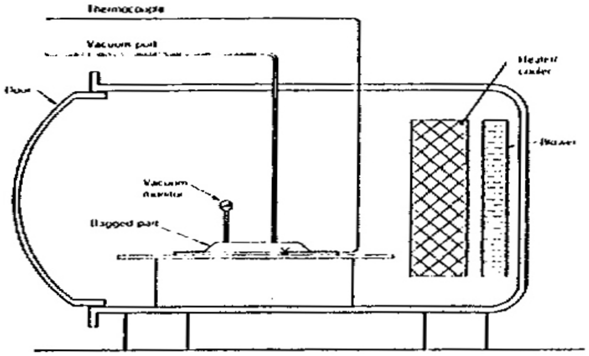


Vacuum bag

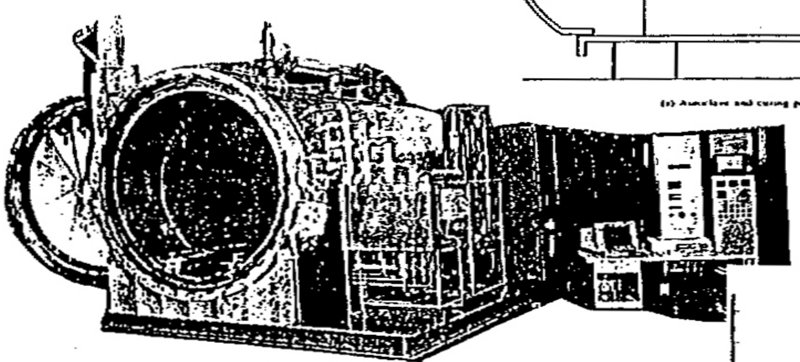


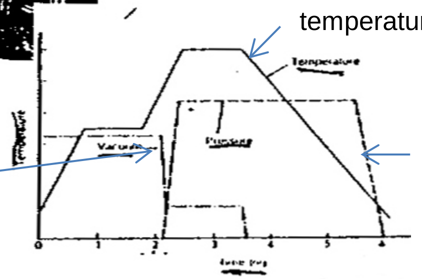
By courtesy of Ren plastics

Autoclave



(a) Autoclave and curing part arrangement





Typical autoclave cycle

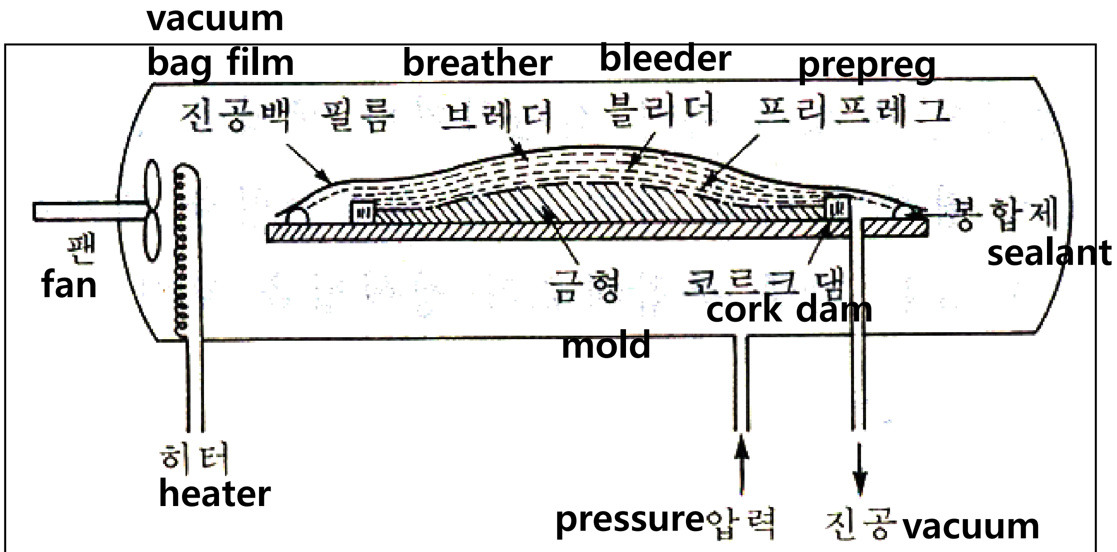


그림 5-1 오토클레이브 성형구조

Autoclave System

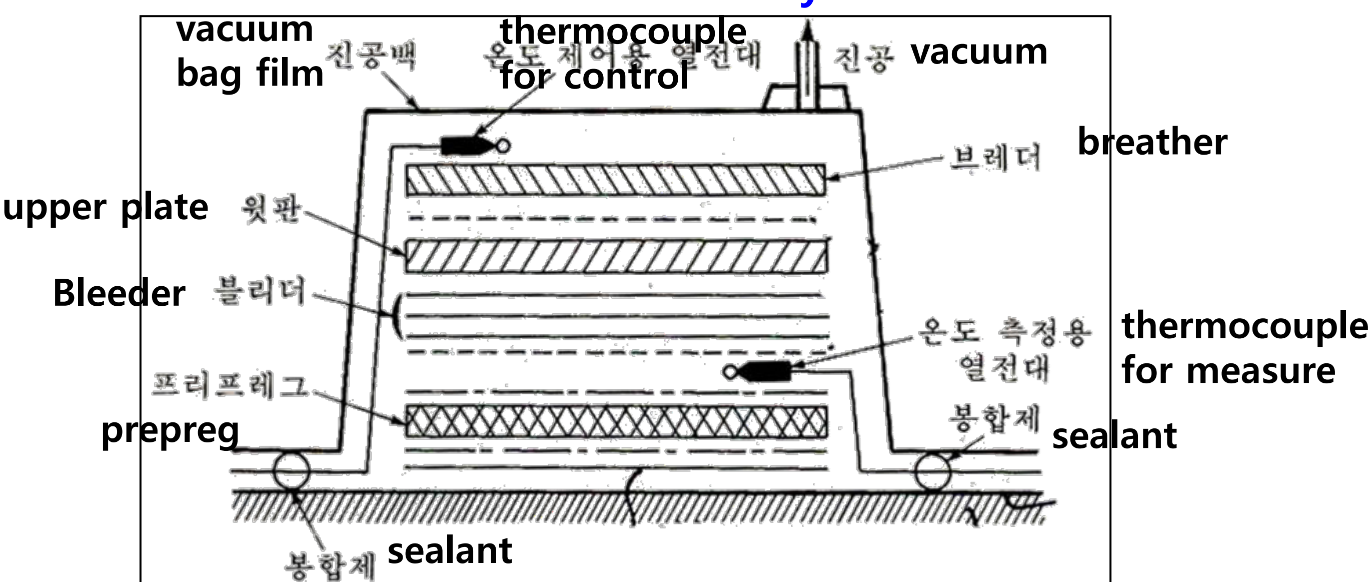


그림 5-4 오토클레이브 성형 적층구조

Autoclave laying-up structure

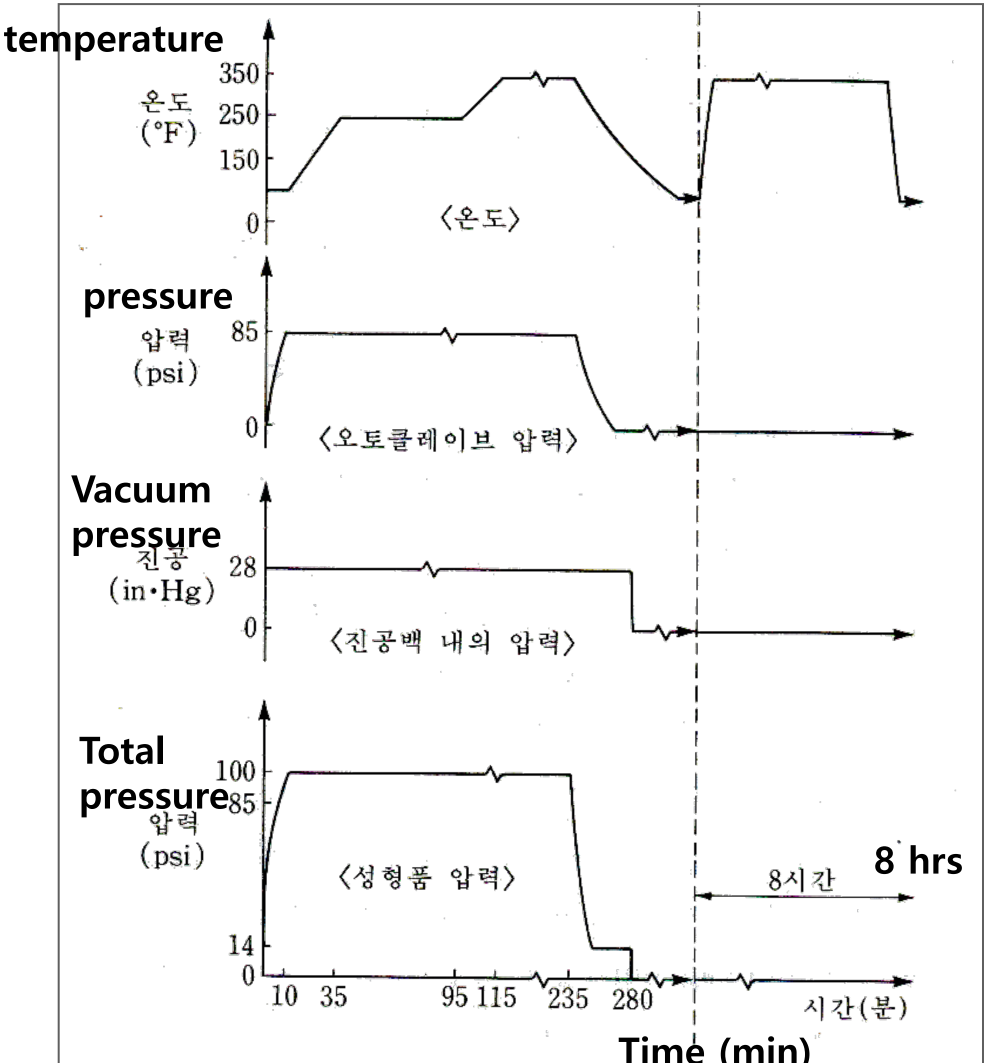


그림 5-5 AS4/3501-6(탄소/에폭시) 프리프레그의 경화 사이클

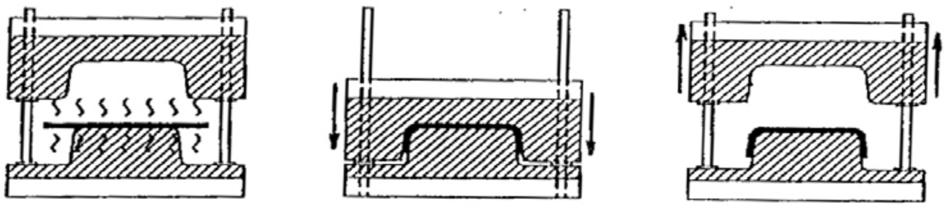
AS4/3501-6 C/EP Prepreg Curing Cycle

Curing Cycle:

- **Thermoset:** **125-175 C, 0.5-1Mpa, 4-8 hrs**
- **Thermoplastic:** **300-400 C, 1-4 MPa, 1-30 min**

Closed Mould

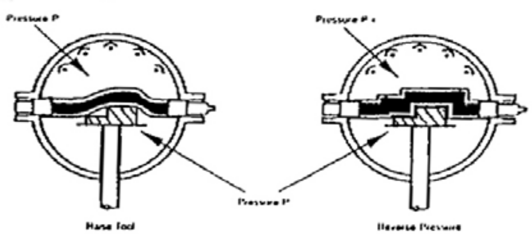
- Male + female matched metal molds
- Or one metal mold+ one flexible membrane mold
- Heat and pressure
- Use mainly for thermoplastic composites



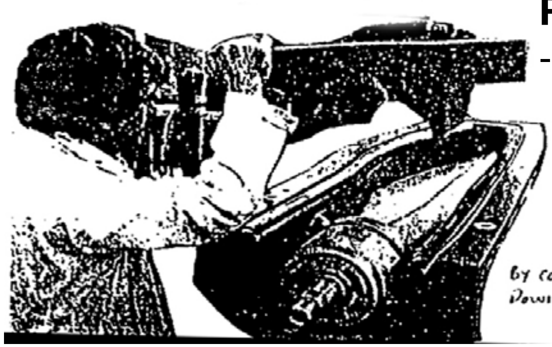
Diaphragm Forming

- Superplastic deformation by Al alloy and polyimide film diaphragm under temp. and press. to form composites onto a mold tool
- Thermoplastic prepreg composites

Diaphragm forming



Resin injection



By courtesy of Dowty Rotol

Resin Transfer Forming

- enclosing pre-shaped fabric (preform) within a mold tool and then transferring resin into the mold with heat and press. to consolidate and cure.

Comparison of Thermoset and Thermoplastic Manufacture

Thermoset composites:

lay-up then cure/consolidate

E.g.:

- Hand or “auto” lay-up
- vac bag/ autoclave cure

Thermoplastic composites:

Lay-up and melt/consolidate in one

E.g.:

- Dynamic “auto” process techniques
  But yet to be developed for production

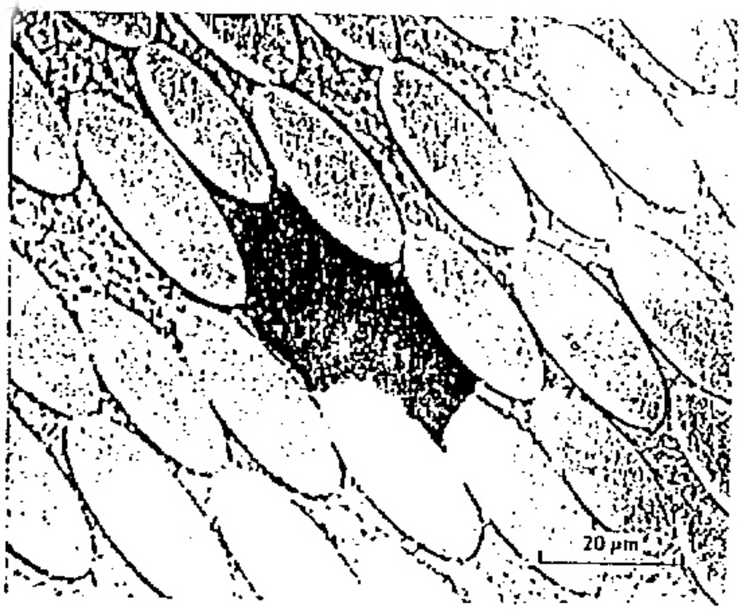
Manufacturing Defects

Voids (Porosity)

- Along fibres or between layers
- Incomplete wet-out
- Volatiles

Gross defects

- Delamination between layers
- Surface contamination
- Poor consolidation



Clean room conditions!

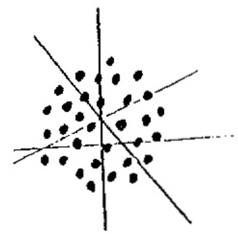
Operator training!

Non-destructive:

- Shop floor inspection → Surface flaws, shape, thickness
- Ultrasonic scanning → Porosity, delamination
- X-ray → Fibre alignment, cracks
  Radio opaque dyes!
- Thermal imaging → Flaws, damage, honeycomb
  disbonds (glass OK, carbon not OK)
- Vibrational response → Stiffness, layup
- Acoustic emission / proof test → Screening

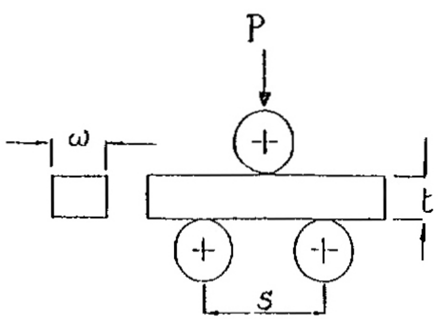
Destructive:

Microscopic → Voids, delamination, fibre content



Density, Acid digest, burn-off → Fibre/ resin content

Short beam shear → Inter-laminar strength



$$\tau = \frac{3P}{4tw} \qquad \sigma = \frac{3Ps}{2t^2 w}$$

Low s/t → shear failure

Tensile, compressive, shear, flexure → Strength, stiffness, fail modes

STATIC BEHAVIOUR

At microscopic scale   Fibre, matrix, interface constituent
Properties are considered

"Micromechanics"

At Macroscopic scale   Lamina and laminate properties are considered

"Macromechanics"

Differential properties:

	Fibre dominated	**Matrix dominated**
Mechanical	high stiffness	low stiffness
Thermal	small-ve α	larger+ve α
Moisture	no charge	Swelling and shrinkage

Micro-Mechanics

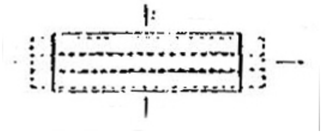
Lamina Stress Strain Behaviour

Assumptions:

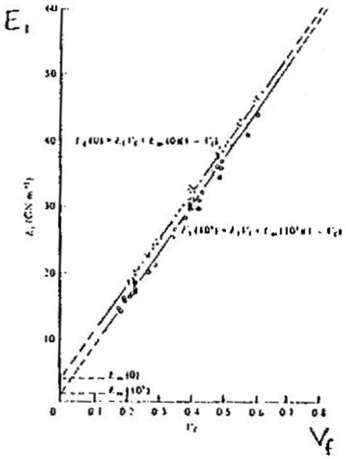
- Linear elastic response
- Perfect fibre-matrix bonding
- Neglecting Poisson strain

"Rules of Mixtures" Prediction of lamina elastic properties from fibre and matrix properties

Longitudinal modulus E_1

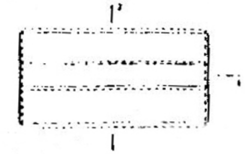


Parallel model



R.o.M: $E_1 = E_f V_f + E_m V_m$

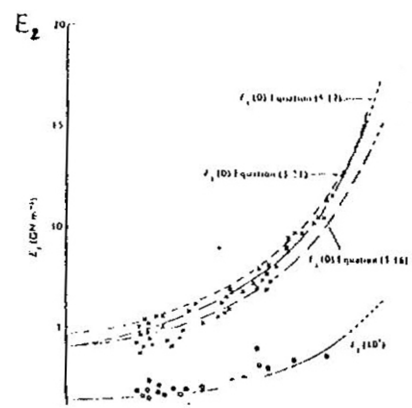
Transverse Modulus E_2



Series model

R.o.M:

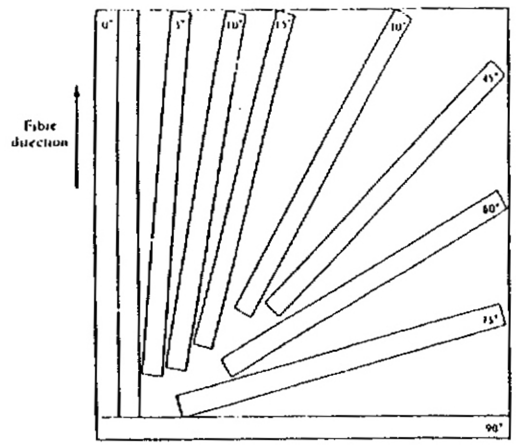
$$E_2 = \frac{E_f E_m}{E_f V_m + E_m V_f}$$

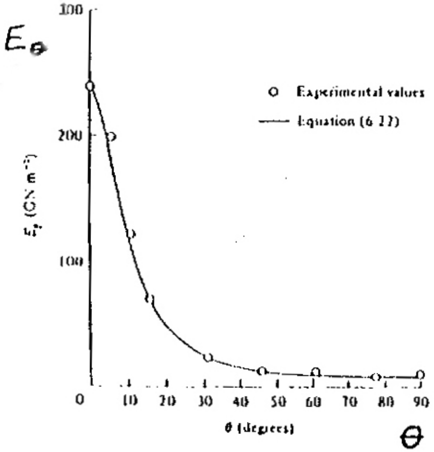


Use effective matrix modulus to account for poisson:

$$E_m' = \frac{E_m}{1 - \nu_m^2}$$

Orientation dependence of Modulus





Thermal Expansion coefficients:

Longitudinal

"R.o.M" $$\alpha_1 = \frac{E_f \alpha_f V_f + E_m \alpha_m V_m}{E_f V_f + E_m V_m}$$

Transverse

"R.o.M" $$\alpha_2 = (1 + \nu_m)\alpha_m V_m + (1 + \nu_f)\alpha_m V_f - \alpha_1(\nu_f V_f + \nu_m V_m)$$

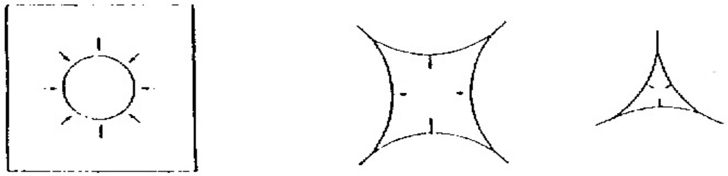
Note:

Matrix large + ve : **Typically +60 μ strain**

→ significant residual strains + stresses

Fibre small (-ve) : **Typically -0.5 μ strain**

Across fibre : **Matrix contraction**

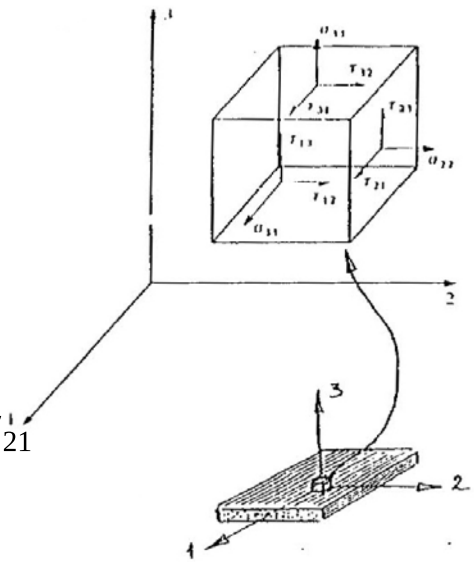


Macromechanics

Lamina Stress strain Behaviour

Assumptions:

- Average apparent properties
- Linear elastic response:



9 stress-strain components:

$$\sigma_1 \quad \sigma_2 \quad \sigma_3 \quad \tau_{23} \quad \tau_{32} \quad \tau_{31} \quad \tau_{13} \quad \tau_{12} \quad \tau_{21}$$

similarly for strains ε and γ

Linear elastic stress-strain relations:

$$\sigma = E\varepsilon, \ \tau = G\gamma$$

Matrix Algebra

$$\{\sigma_1\} = [C]\{\varepsilon_1\}$$

$$\{\varepsilon_1\} = [S]\{\sigma_1\}$$

Where C = stiffness matrix!
S = Compliance matrix! = $[C]^{-1}$

Constitutive Stress-Strain relations

Fully anisotropic

$$\begin{Bmatrix} \sigma_1 \\ \sigma_2 \\ \sigma_3 \\ \tau_{23} \\ \tau_{31} \\ \tau_{12} \end{Bmatrix} = \begin{bmatrix} C_{11} & C_{12} & C_{13} & C_{14} & C_{15} & C_{16} \\ C_{21} & C_{22} & C_{23} & C_{24} & C_{25} & C_{26} \\ C_{31} & C_{32} & C_{33} & C_{34} & C_{35} & C_{36} \\ C_{41} & C_{42} & C_{43} & C_{44} & C_{45} & C_{46} \\ C_{51} & C_{52} & C_{53} & C_{54} & C_{55} & C_{56} \\ C_{61} & C_{62} & C_{63} & C_{64} & C_{65} & C_{66} \end{bmatrix} \begin{Bmatrix} \varepsilon_1 \\ \varepsilon_2 \\ \varepsilon_3 \\ \gamma_{23} \\ \gamma_{13} \\ \gamma_{12} \end{Bmatrix}_{36imc}$$

independent material constalants

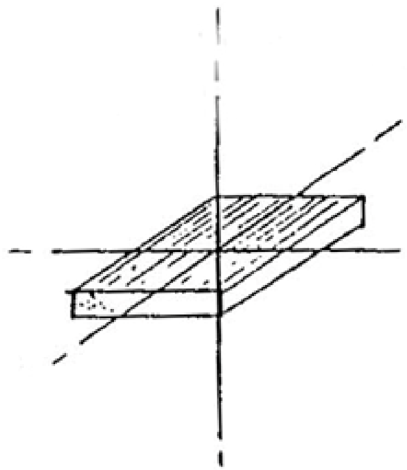
Independant of the order of loading (reciprocal behaviour)

$$\begin{Bmatrix} \sigma_1 \\ \sigma_2 \\ \sigma_3 \\ \tau_{23} \\ \tau_{31} \\ \tau_{12} \end{Bmatrix} = \begin{bmatrix} C_{11} & C_{12} & C_{13} & C_{14} & C_{15} & C_{16} \\ C_{21} & C_{22} & C_{23} & C_{24} & C_{25} & C_{26} \\ C_{31} & C_{32} & C_{33} & C_{34} & C_{35} & C_{36} \\ C_{41} & C_{42} & C_{43} & C_{44} & C_{45} & C_{46} \\ C_{51} & C_{52} & C_{53} & C_{54} & C_{55} & C_{56} \\ C_{61} & C_{62} & C_{63} & C_{64} & C_{65} & C_{66} \end{bmatrix} \begin{Bmatrix} \varepsilon_1 \\ \varepsilon_2 \\ \varepsilon_3 \\ \gamma_{23} \\ \gamma_{13} \\ \gamma_{12} \end{Bmatrix}_{21imc}$$

$$C_{ij} = C_{ji}$$

3mutually perpendicular planes of symmetry (orthotropic)

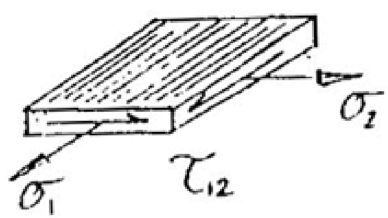
$$\begin{Bmatrix} \sigma_1 \\ \sigma_2 \\ \sigma_3 \\ \tau_{23} \\ \tau_{31} \\ \tau_{12} \end{Bmatrix} = \begin{bmatrix} C_{11} & C_{12} & C_{13} & 0 & 0 & 0 \\ C_{21} & C_{22} & C_{23} & 0 & 0 & 0 \\ C_{31} & C_{32} & C_{33} & 0 & 0 & 0 \\ 0 & 0 & 0 & C_{44} & 0 & 0 \\ 0 & 0 & 0 & 0 & C_{55} & 0 \\ 0 & 0 & 0 & 0 & 0 & C_{66} \end{bmatrix} \begin{Bmatrix} \varepsilon_1 \\ \varepsilon_2 \\ \varepsilon_3 \\ \gamma_{23} \\ \gamma_{13} \\ \gamma_{12} \end{Bmatrix}_{9\,imc}$$



Plane stress

$$\begin{Bmatrix} \sigma_1 \\ \sigma_2 \\ \tau_{12} \end{Bmatrix} = \begin{bmatrix} C_{11} & C_{12} & 0 \\ C_{21} & C_{22} & 0 \\ 0 & 0 & C_{66} \end{bmatrix} \begin{Bmatrix} \varepsilon_1 \\ \varepsilon_2 \\ \gamma_{12} \end{Bmatrix}_{4\,imc}$$

Reduced stiffness matrix



$$\begin{Bmatrix} \sigma_1 \\ \sigma_2 \\ \tau_{12} \end{Bmatrix} = \begin{bmatrix} Q_{11} & Q_{12} & 0 \\ Q_{12} & Q_{22} & 0 \\ 0 & 0 & Q_{66} \end{bmatrix} \begin{Bmatrix} \varepsilon_1 \\ \varepsilon_2 \\ \gamma_{12} \end{Bmatrix}_{4\,imc}$$

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

“Generally orthotropic”

Generally orthotropic constitutive relation

Starting with:

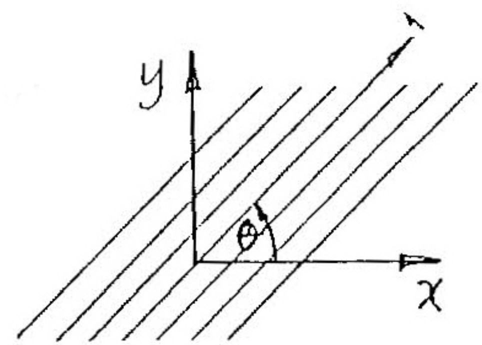
$$\begin{Bmatrix}\sigma_1\\ \sigma_2\\ \tau_{12}\end{Bmatrix}=\begin{bmatrix}Q_{11} & Q_{12} & 0\\ Q_{12} & Q_{22} & 0\\ 0 & 0 & Q_{66}\end{bmatrix}\begin{Bmatrix}\varepsilon_1\\ \varepsilon_2\\ \gamma_{12}\end{Bmatrix}$$

Where $[Q]$ =**Reduced stiffness matrix in 1-2 material axes**

Transform by trigonometric transformation
to produce general structural axis relations at angle θ

i.e.:

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



Where $\left[\overline{Q}\right]$ is the “**Transformed reduced stiffness matrix**”

In general x-y plate axes

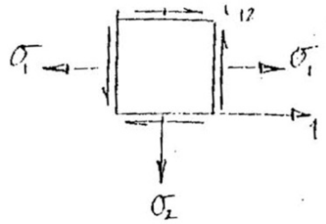
Calculated from $\left[\overline{Q}\right] = [T]^{-1}[Q][T]^{-T}$

$$where\ [T] = \begin{bmatrix} m^2 & n^2 & 2mn \\ n^2 & m^2 & -2mn \\ -mn & mn & m^2 - n^2 \end{bmatrix} \qquad And\ \ m = \cos\theta \qquad n = \sin\theta$$

i.e. simply geometric transformation

transformed, e.g. to the general x-y plate axes. This results in **a fully populated matrix and shear coupling** so that the matrix becomes fully populated, usually written as . However, there are still only 4 independent material constants required to describe the stress strain behaviour.

under a 2D plane stress system



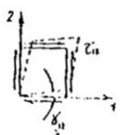
Strains:

$$\varepsilon_1 = \frac{\sigma_1}{E_1} + \frac{-\nu_{21}\sigma_2}{E_2} + 0$$

$$\varepsilon_2 = \frac{-\nu_{12}\sigma_1}{E_1} + \frac{\sigma_2}{E_2} + 0$$

Consider Stress components separately

$$\gamma_{12} = 0 \quad + 0 \quad + \frac{\tau_{12}}{G_{12}}$$



★ I.e.: **compliance relationship**:

$$\begin{Bmatrix} \varepsilon_1 \\ \varepsilon_2 \\ \gamma_{12} \end{Bmatrix} = \begin{bmatrix} \frac{1}{E_1} & \frac{-V_{21}}{E_2} & 0 \\ \frac{-V_{12}}{E_1} & \frac{1}{E_2} & 0 \\ 0 & 0 & \frac{1}{G_{12}} \end{bmatrix} \begin{Bmatrix} \sigma_1 \\ \sigma_2 \\ \tau_{12} \end{Bmatrix}$$

Stresses

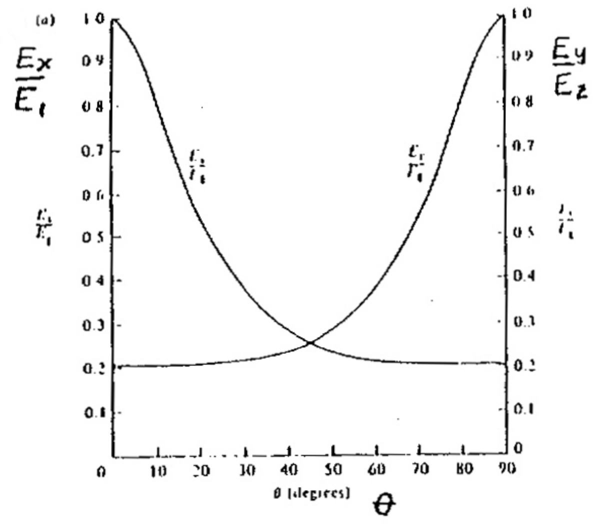
★ **Constitutive relationship**:

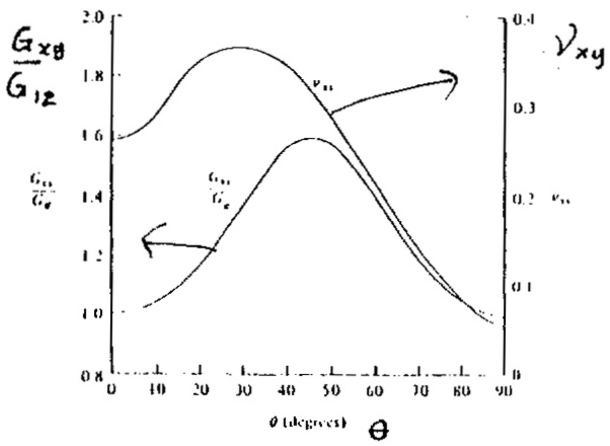
$$\begin{Bmatrix} \sigma_1 \\ \sigma_2 \\ \tau_{12} \end{Bmatrix} = \begin{bmatrix} \frac{E_1}{(1-\nu_{12}\nu_{21})} & \frac{\nu_{21}E_1}{(1-V_{12}V_{21})} & 0 \\ \frac{\nu_{12}E_2}{(1-\nu_{12}\nu_{21})} & \frac{E_2}{(1-V_{12}V_{21})} & 0 \\ 0 & 0 & G_{12} \end{bmatrix} \begin{Bmatrix} \varepsilon_1 \\ \varepsilon_2 \\ \gamma_{12} \end{Bmatrix}$$

Note: 4 imc: $E_1 \quad E_2 \quad \nu_{12} \quad G_{12}$

Note $\nu_{21} \neq \nu_{12}$ **But reciprocal relation** $\rightarrow \nu_{12}E_2 = \nu_{21}E_1$

Orientation Dependence of Moduli





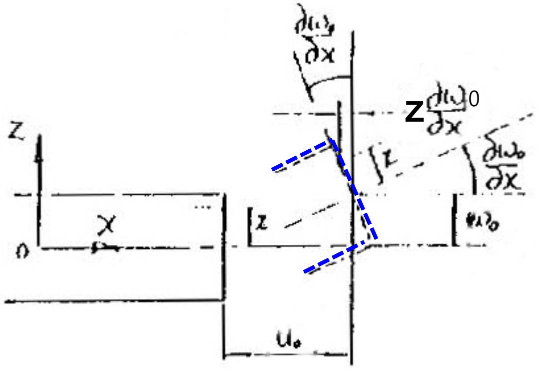
Laminate Stress-Strain behaviour

Macromechanic scale

Assumptions:

- Perfect lamina bonding
- Infinitely thin bond
- Thin laminate

Resulting deformation



Laminate stiffness or compliance matrix

- created from summation of transformed lamina matrixes

★ **Laminate Constitutive Relation**

$$\begin{Bmatrix} N_x \\ N_y \\ N_{xy} \end{Bmatrix} = \begin{bmatrix} A_{11} & A_{12} & A_{16} \\ A_{12} & A_{22} & A_{26} \\ A_{16} & A_{26} & A_{66} \end{bmatrix} \begin{Bmatrix} \varepsilon^\circ{}_x \\ \varepsilon^\circ{}_y \\ \gamma^\circ{}_{xy} \end{Bmatrix} + \begin{bmatrix} B_{11} & B_{12} & B_{16} \\ B_{12} & B_{22} & B_{26} \\ B_{16} & B_{26} & B_{66} \end{bmatrix} \begin{Bmatrix} \kappa_x \\ \kappa_y \\ \kappa_{xy} \end{Bmatrix}$$

$$\begin{Bmatrix} M_x \\ M_y \\ M_{xy} \end{Bmatrix} = \begin{bmatrix} B_{11} & B_{12} & B_{16} \\ B_{12} & B_{22} & B_{26} \\ B_{16} & B_{26} & B_{66} \end{bmatrix} \begin{Bmatrix} \varepsilon^\circ{}_x \\ \varepsilon^\circ{}_y \\ \gamma^\circ{}_{xy} \end{Bmatrix} + \begin{bmatrix} D_{11} & D_{12} & D_{16} \\ D_{12} & D_{22} & D_{26} \\ D_{16} & D_{26} & D_{66} \end{bmatrix} \begin{Bmatrix} \kappa_x \\ \kappa_y \\ \kappa_{xy} \end{Bmatrix}$$

***Details refer to Lecture Note “CLT”**

I.e.:

$$\begin{Bmatrix} N \\ M \end{Bmatrix} = \begin{bmatrix} A & B \\ B & D \end{bmatrix} \begin{Bmatrix} \varepsilon^{\circ} \\ \kappa \end{Bmatrix}$$ x,y plate axes

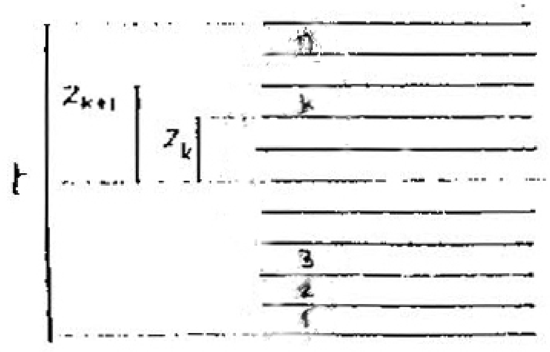
"Laminate stiffnesses"

Where:

$$A = \sum \left[\overline{Q}\right]_k (Z_{k+1} - Z_k)$$

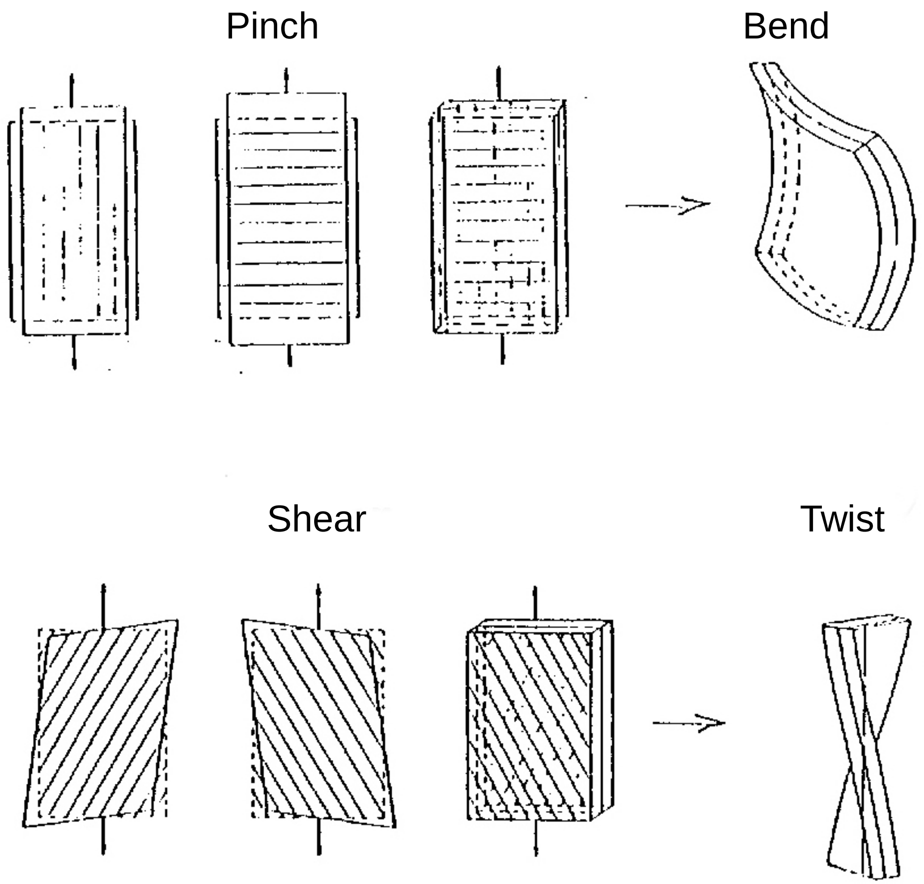
$$B = \frac{1}{2} \sum \left[\overline{Q}\right]_k (Z_{k+1}^2 - Z_k^2)$$

$$D = \frac{1}{3} \sum \left[\overline{Q}\right]_k (Z_{k+1}^3 - Z_k^3)$$



Coupling between lamina

Lamina @ different orientations → Out of plane coupling stresses

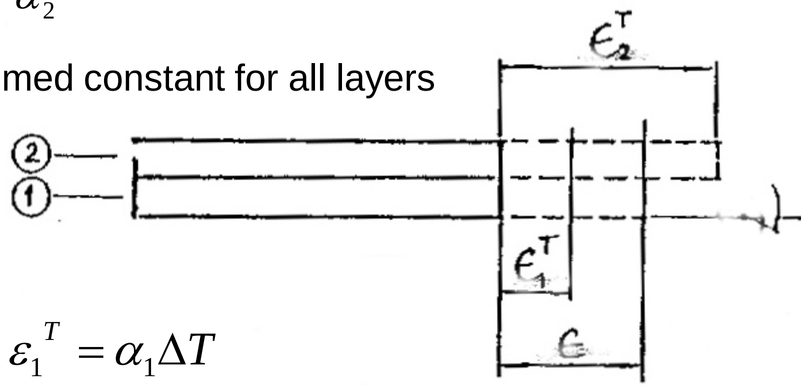


✓ **In order to remove coupling effect → Need symmetric, balanced laminates!**

Laminate residual stresses and strains

Layer 1, expansion coefficient α_1
Layer 2, expansion coefficient α_2

Temperature change ΔT assumed constant for all layers



Free thermal strains: Layer 1: $\varepsilon_1^{\ T} = \alpha_1 \Delta T$

Layer 2: $\varepsilon_2^{\ T} = \alpha_2 \Delta T$

Residual strains: Layer 1: $\varepsilon_1^{\ R} = \varepsilon - \varepsilon_1^{\ T}$

Layer 2: $\varepsilon_2^{\ R} = \varepsilon - \varepsilon_2^{\ T}$

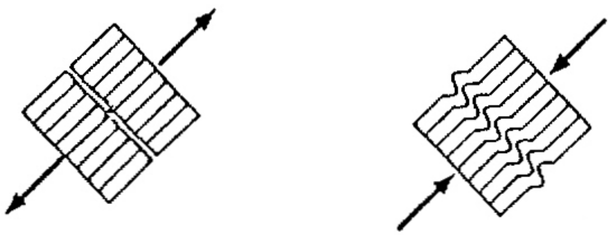
Where ε =Laminate "**common strain**"

I.e.:
Layer residual strain = laminate common strain less layer free thermal strain

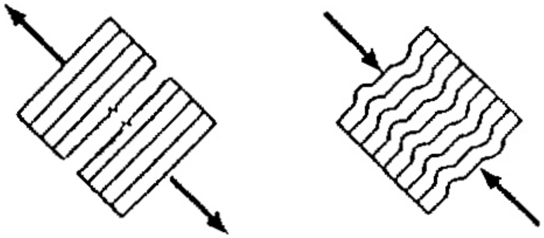
Residual stresses $\{\sigma\} = [Q]\{\varepsilon^R\}$

★5 main intra-laminar failure modes:

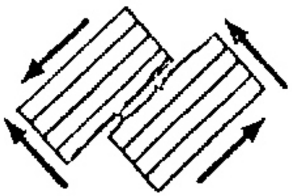
- **Longitudinal tension** and **compression failure**



- **Transverse tension** and **compression failure**

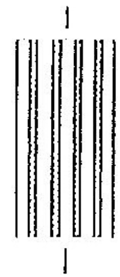


- **In-plane shear failure**



- **Mixed modes and Delamination (inter-laminar failure)** : **Not included in CLT**

Longitudinal Tensile Strength

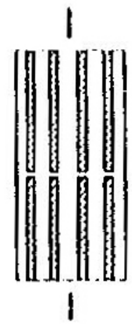


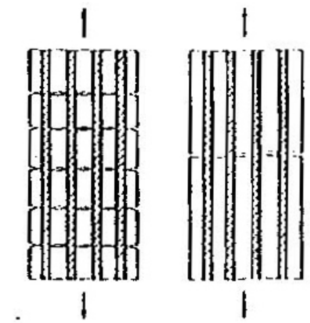
<u>Assuming uniform strength fibres</u>

“R.o.M” $\sigma_1^* = \sigma_f * V_f + \sigma_m * V_m$

High V_f (structural composites)

$\rightarrow \sigma_1^* \approx \sigma_f * V_f$ **“Fibre dominated”**

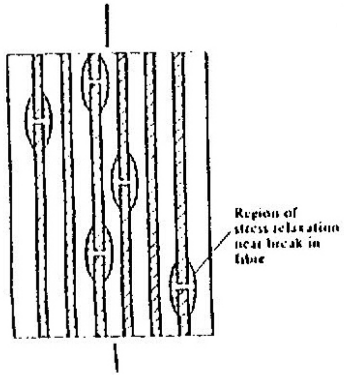


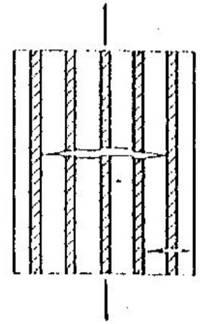


<u>For variable strength fibres</u>:

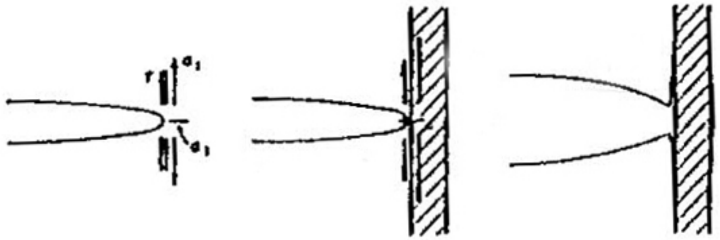
Strength variation: Along fibres and from fibre to fibre

$\rightarrow$ **Progressive failure**

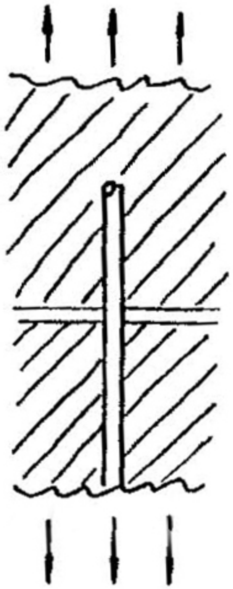




Matrix crack at fibre interface



Fibre pull-out (Toughness)



Fibre fracture energy W_f

Matrix fracture energy W_m

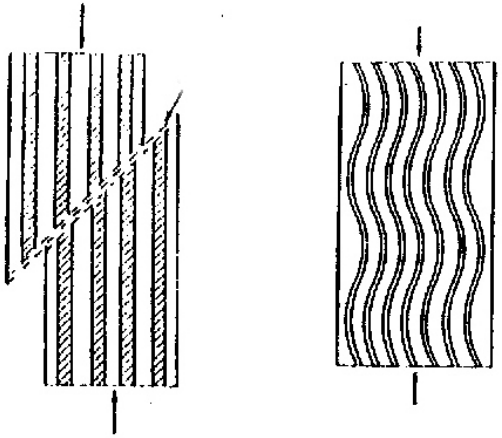
Fibre debond energy W_d

Fibre pull-out energy W_p

$$W_p >> W_d >> W_m >> W_f$$

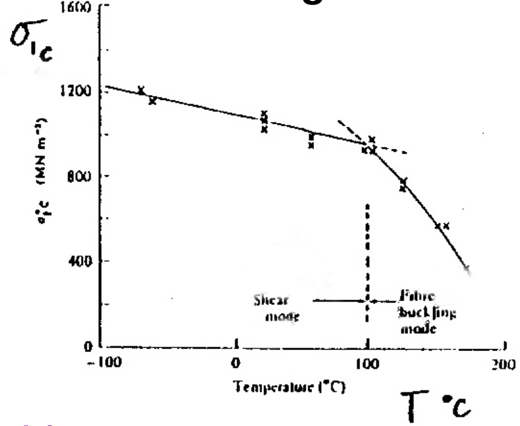
Pull-out before fracture →energy absorption →toughness

Longitudinal compressive Strength



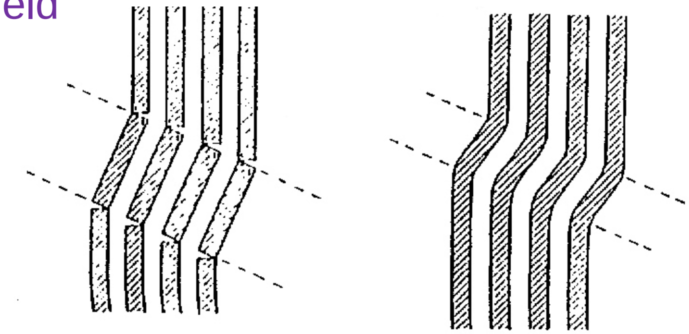
For high V_f

Compression -> **shear failure or in phase fibre buckling**

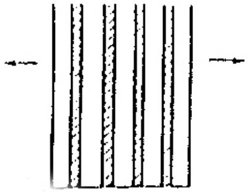


Matrix support is critical

Carbon tends to shear, Kevlar tends to yield



Transverse tensile strength



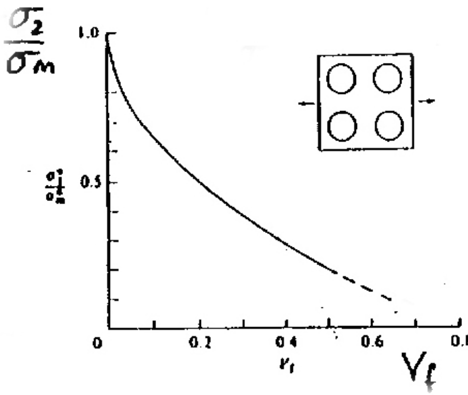
Fibre -> “-ve reinforcement

Cross-section reduction

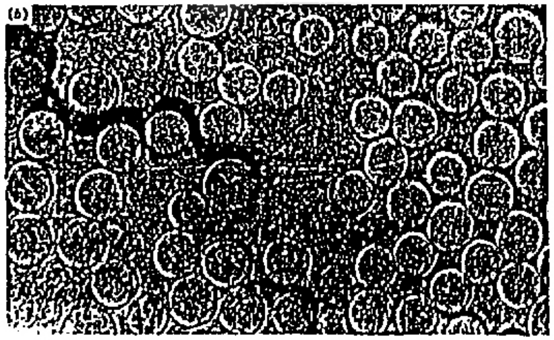
Stress concentration

Depending on:

- matrix stress-strain response
- fibre interface bond strength
- V_f , fibre packing voids etc.

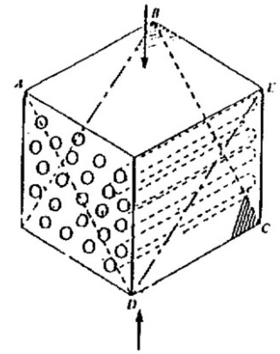


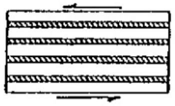
strain magnification between fibres!



Transverse Compressive Strength

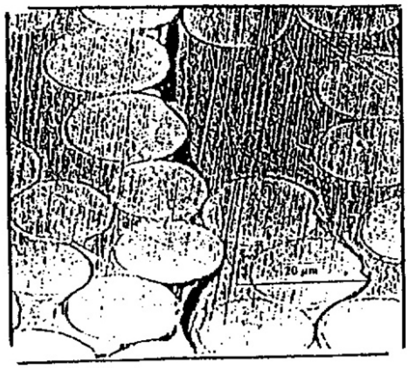
- Shear failure across fibres





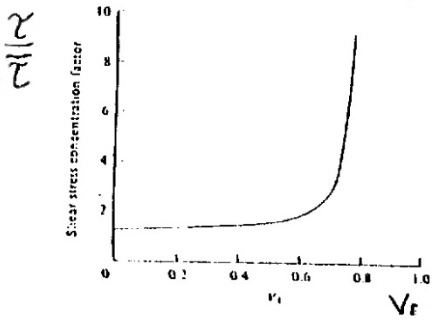
In-plane Shear Strength

- Matrix or interface dominated



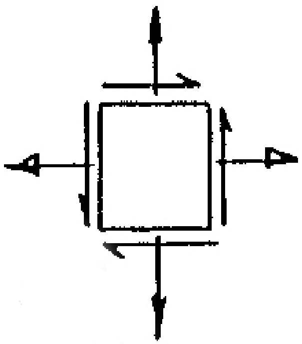
Depending on :

- Matrix stress-strain response
- Fibre interface bond strength
- V_f , fibre packing , void, etc.



Multi-axial loading

- still 5 basic failure modes of intra-lamina failure
- but note in practice : mixed intra-laminar modes and delaminate inter-laminar mode



★

Intra-laminar Failure Criteria :

Maximum Stress Theory

Failure occurs when: $\sigma_1 = \sigma_1^*, \sigma_2 = \sigma_2^*, \sigma_{12} = \sigma_{12}^*$

Maximum Strain Theory

Failure occurs when: $\varepsilon_1 = \varepsilon_1^*, \varepsilon_2 = \varepsilon_2^*, \gamma_{12} = \gamma_{12}^*$

Maximum work theory Tsai-Hill(Von-Mises)

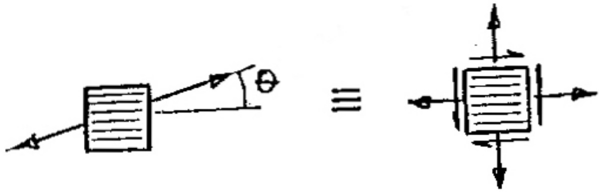
Failure occurs when :
$$\left(\frac{\sigma_1}{\sigma_1^*}\right)^2 + \left(\frac{\sigma_1\sigma_2}{\sigma_1^{*2}}\right) + \left(\frac{\sigma_2}{\sigma_2^*}\right)^2 + \left(\frac{\tau_{12}}{\tau_{12}^*}\right)^2 = 1$$

E.g for off-axis loading

Resolve stress into 1,2 material axis components :

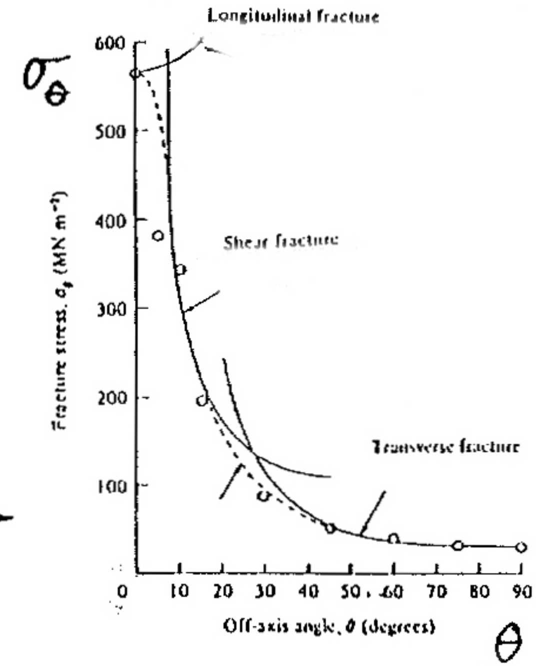
$$\sigma_1, \sigma_2, \tau_{12}$$

and check intra-lamina strengths:



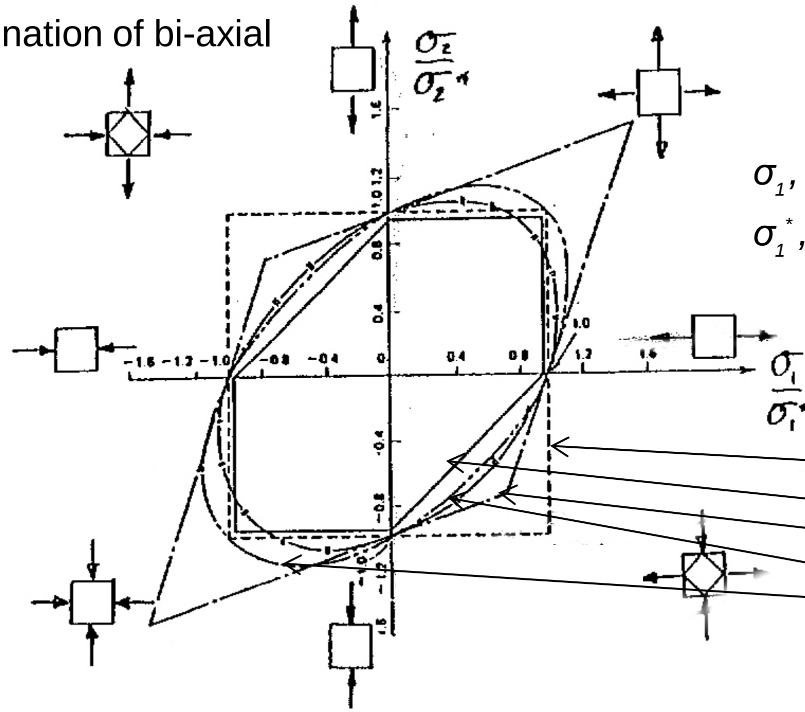
$$\sigma_1^*, \sigma_2^*, \tau_{12}^*$$

according to failure criteria



Failure Envelopes

- Locus of combination of bi-axial stress

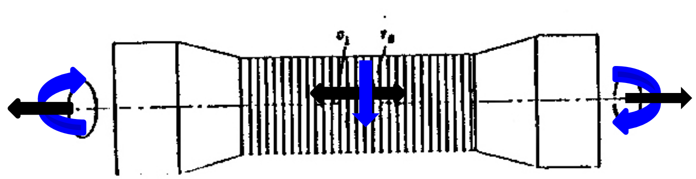


σ_1, σ_2 : applied stresses in 1,2 directions

σ_1^*, σ_2^* : strengths in 1,2 directions

Hoop specimen multi-axial test

- To validate failure criteria



Laminate strength

Definition of laminate failure :

1st ply failure(≈yield failure) (FPF)

last ply failure = ultimate failure(LPF)

Iterative Method

- For applied loading on laminate
- Laminate theory

$$\rightarrow \ \sigma_1, \sigma_2, \tau_{12} \,]_k$$

- Lamina failure criteria, e.g:

$$\sigma_1 \geq \sigma_1{*},\ \sigma_2 \geq \sigma_2{*}, \tau_{12} \geq \tau_{12}{*} \ \Big]_k$$

failed lamina properties : E_1 ,E_2 $G_{12} \rightarrow 0$

- Repeat until all lamina failed

Assumptions!

Laminate analysis theory

Linear elasticity $\left(\sigma_2 ! \ \tau_{12} !\right) \Rightarrow$ *highly nonlinear*

$\rightarrow$ *give rise to errors in laminar stress calculation*

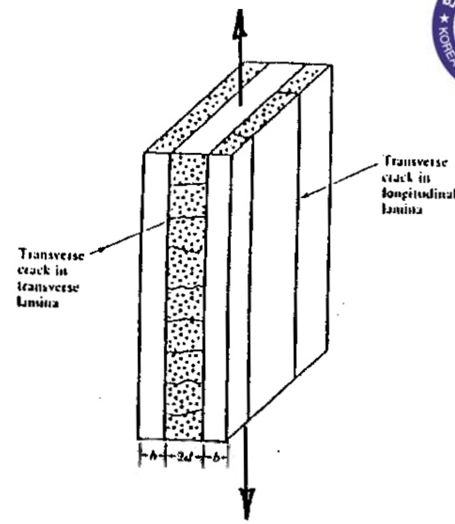
Cross-ply Laminates

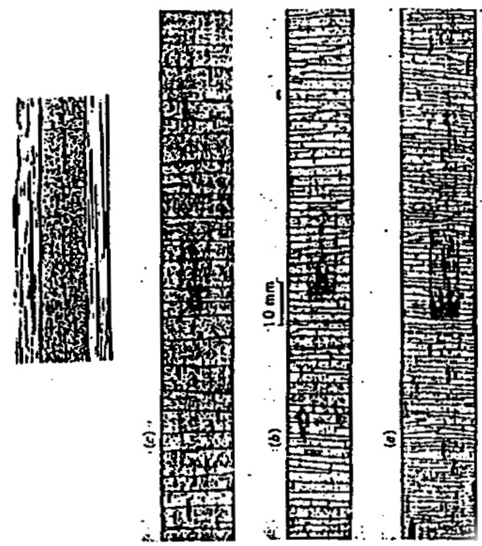
Transverse ply failure ≠ final failure

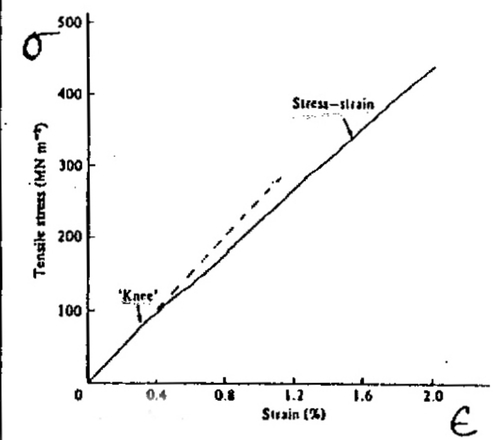
Multiple transverse cracking

Exponential load transfer

E.g. **Cross-ply laminate modulus** : E_{CP}







Before transverse cracking "R.o.M" $E_{CP1} = \dfrac{E_1 b}{b+d} + \dfrac{E_2 d}{b+d}$

After transverse cracking and total disband
Note Poisson constraint effects → longitudinal lamina cracking

Angle-ply Laminates

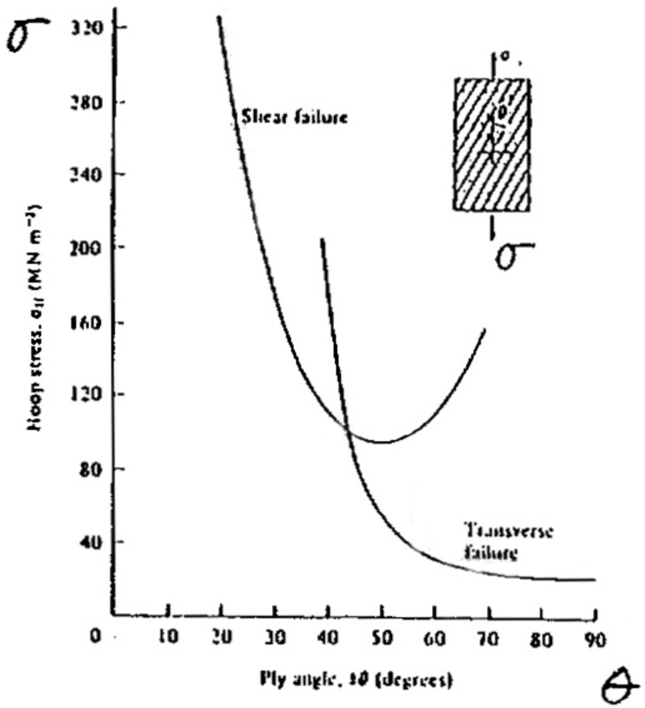
Failure modes:

$\theta < 45^\circ$ → **Shear**

mixed mode around 45°

$\theta \geq 45^\circ$ → **Transverse**

Non-linear response
Edge effects !



Through-thickness Edge Stresses

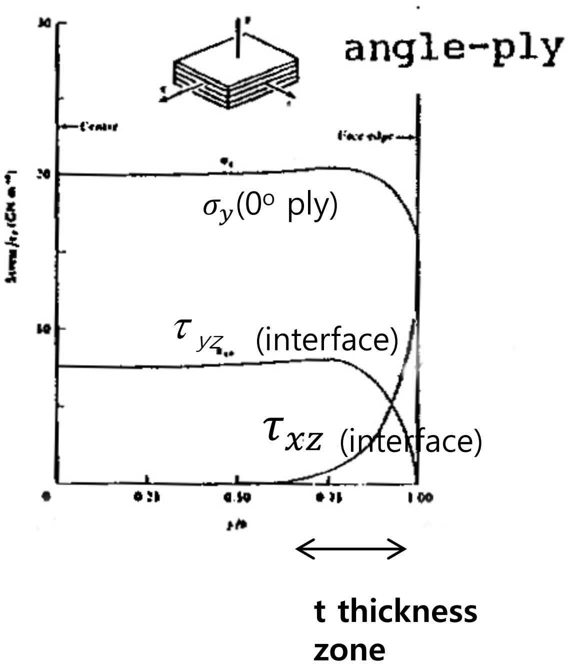
Classical laminate theory $\rightarrow "\sigma_z, \tau_{xz}, \tau_{yz} = 0"$

Only away from laminate edge > $t_{laminate}$ from edge

Reality $\rightarrow "\sigma_z, \tau_{xz}, \tau_{yz} \neq 0"$

Very high near laminate edge (< $t_{laminate}$ from edge)

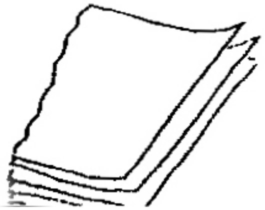
: σ_z , τ_{xz}



Layup stacking sequence!

Effect failure processes!

Edge peel!



Notch Stresses

Cross-section reduction

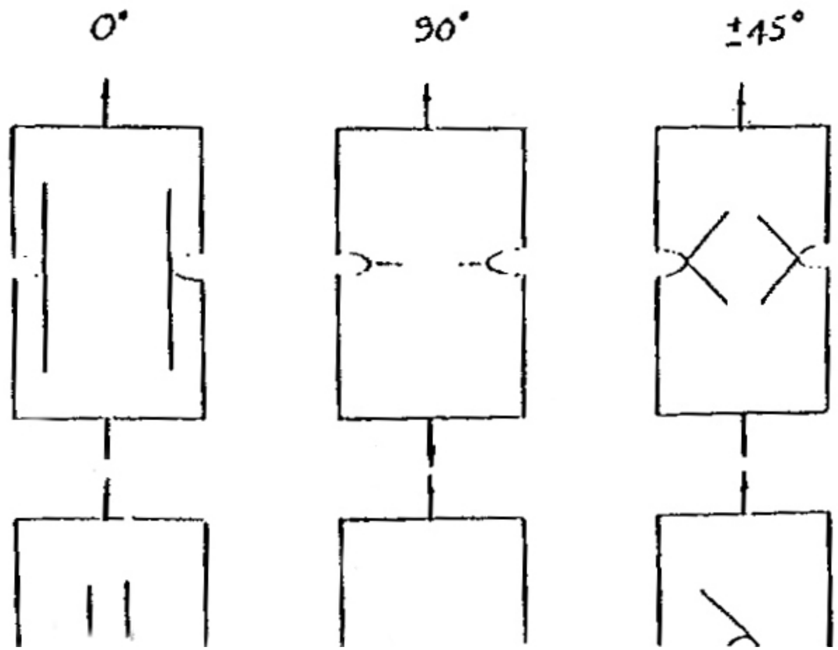
Stress concentration

Free-edge peel stress

Static concentration factors → **x10!**

Depending on :

- Notch size
- Layup orientations
- Stacking sequence
- Fibre-matrix bond strength



Measurement of design data

Standards : ASTM(America Society for Testing Materials) , CRAG (Composite Research Advisory Group)

Elastic constants , failure stresses and failure strains

- UD laminate $E_1\ E_2\ G_{12}\ \nu_{12}$

$$\sigma_1{}^*\sigma_2{}^*\tau_{12}{}^*\varepsilon_1{}^*\varepsilon_2{}^*\gamma_{12}{}^*$$

- Laminate $E_x\ E_y\ G_{xy}\ \nu_{xy}$

$$\sigma_x{}^*\sigma_y{}^*\tau_{xy}{}^*\varepsilon_x{}^*\varepsilon_y{}^*\gamma_{xy}{}^*$$

Failure stresses and strains are required for both tension & compression

Specimen:

Elastic /anisotropic response → Local stress dissipation problems

- Tabs
- Long gauge lengths
- Parallel sides
- Careful alignment

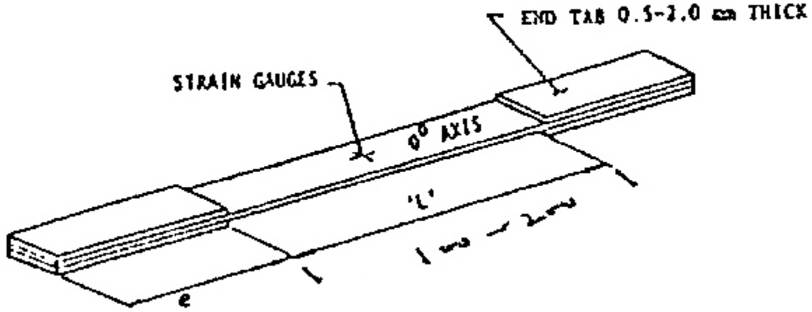
Specimen Size: thickness (8~16 ply) x
width (10~30mm) x
length (100~200 mm)

Variability

- Consider lowest values – Extreme value statistics(Weibull) : generally 20~30 specimens

Tension

Flat parallel sided specimen



Lamina properties

Longitudinal, all 0°

Transverse, all 90°

Shear ±45°

Laminate properties

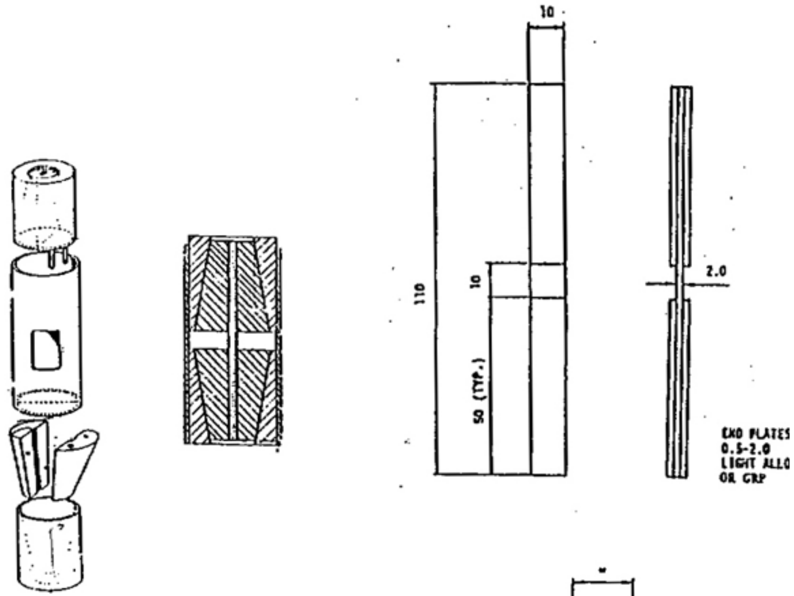
Plane

Notched

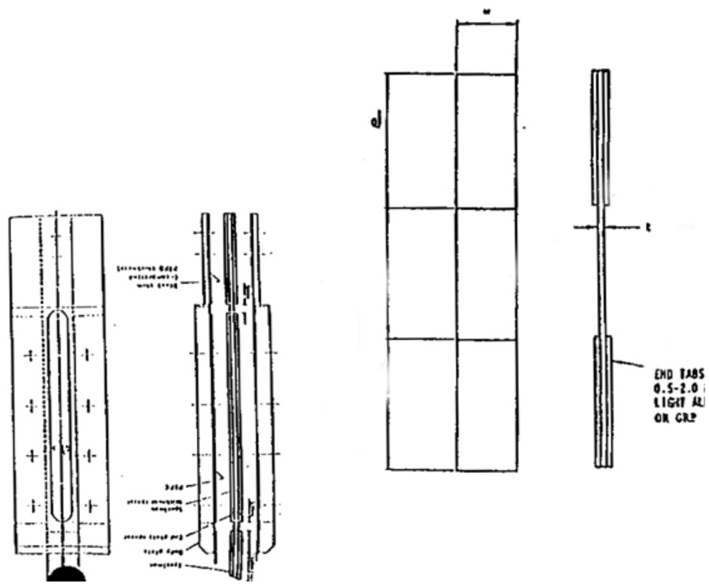
Compression

Avoid buckling

Short gauge length

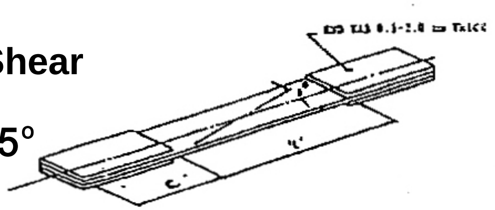


Long gauge length +anti-buckling guide

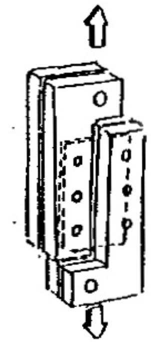


In-plane Shear

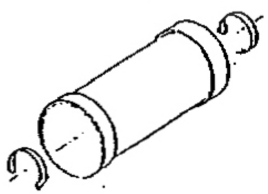
± 45°



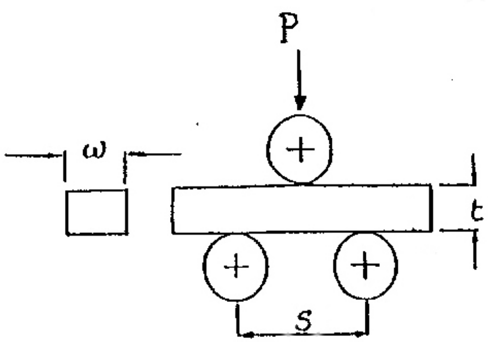
Rail Shear

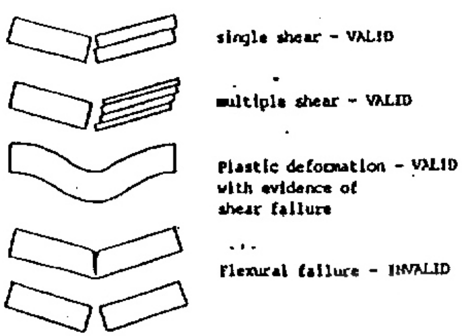


Torsion Shear



Inter-laminar Shear





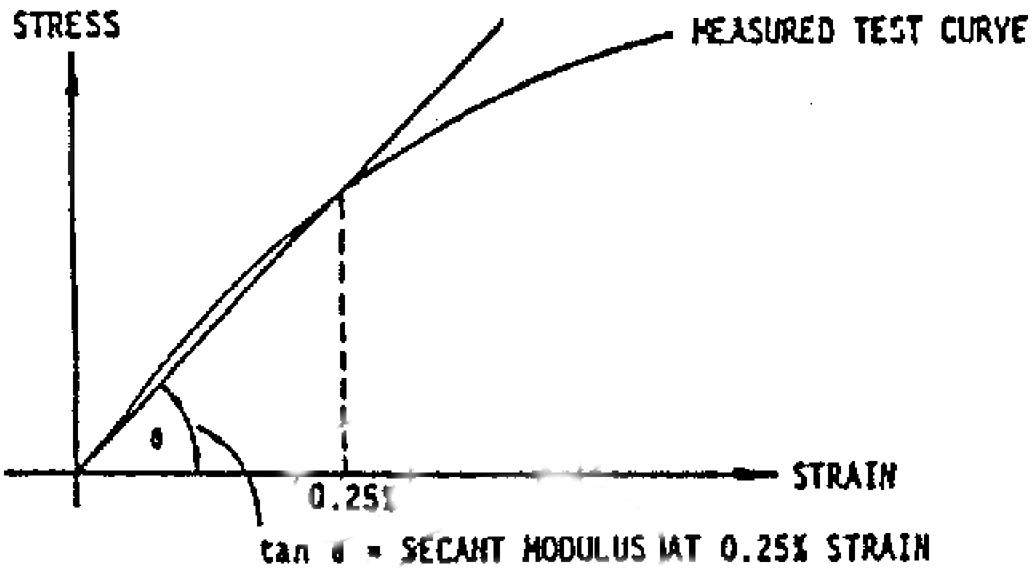
Specimen size: 1cm w x 2cm s x 16 plies t

Normalization

- to chosen value of fibre content for consistency
- according to micromechanics “ R.o.M”

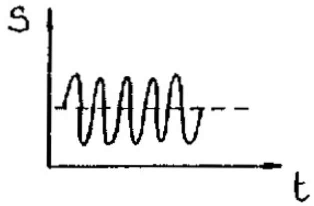
Measurement of elastic modulus

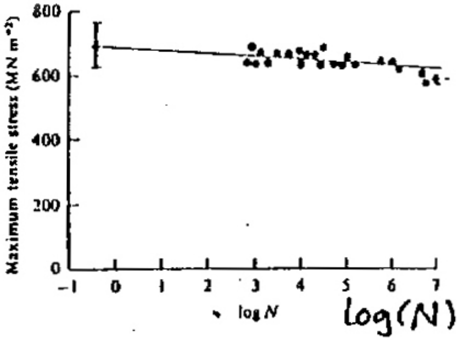
Tangent or secant



Fatigue ★

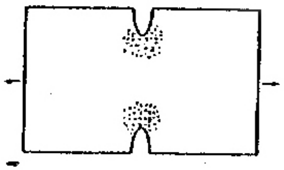
-Insensitive in tension



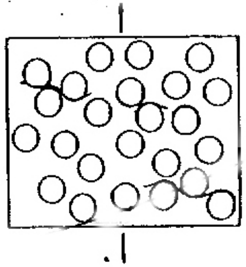


*** More sensitive in compression!**

-Notch concentration reduction!



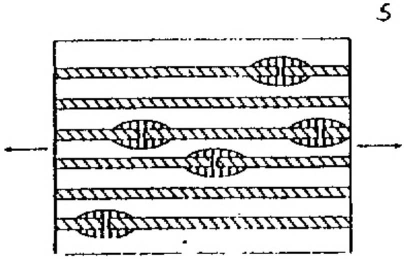
Strain magnification



Global damage accumulation

→ some stiffness degradation : matrix, interface

→ stiffness critical designs!



Hygrothermal effects

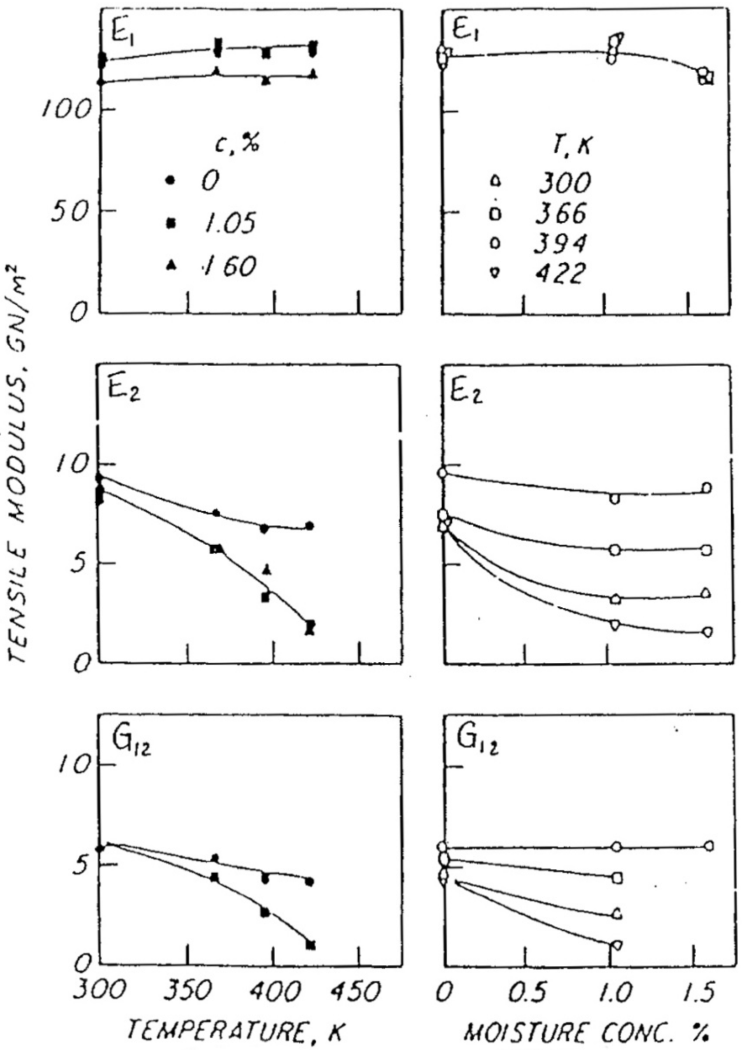
- Temperature and moisture
- Prolonged or cycling application
- Flexibilizing and swelling

→ some stiffness degradation : matrix, interface

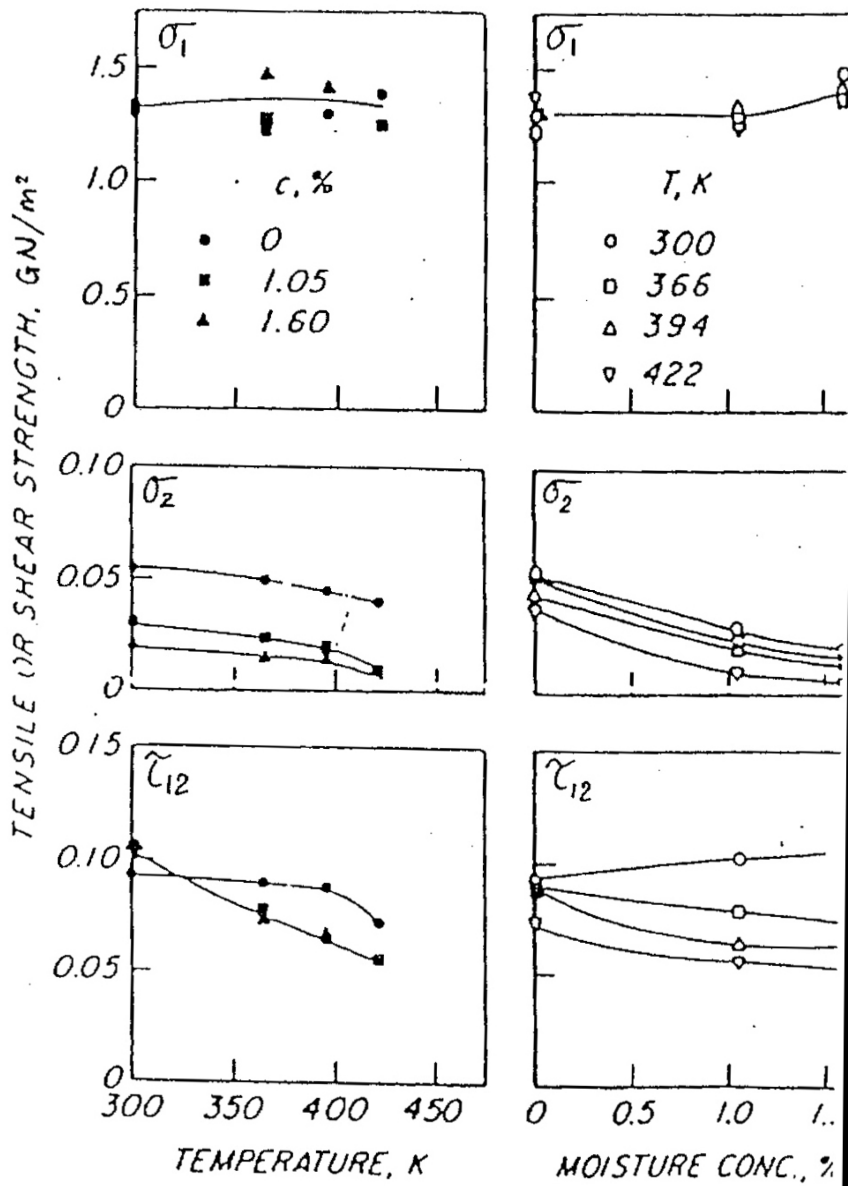
Erosion effects

- Dust, sand, rain

→ surface wear + moisture ingress



Effects of temperature and moisture concentration on tensile and shear moduli of AS/3501



Measurement of Long Term Behaviour

Fatigue

Major structural fatigue test x1

Structure samples x3

Coupons x12+

Cyclic heating!

Low test frequency, long lives!   To prevent excess heating, 5~10Hz cyclic frequency!

Ex) at 5Hz, 10^6 cycles: take 2 days

Moisture

Temperature

Demonstrate no damage growth for:
BVID(“barely visible impact damage”) hot wet compression fatigue

High energy(Flight impacts, Ballistic impacts)

- shock wave

Low energy(Ground operations!)

- Back surface break-out

The problem:

- Elastic to failure
- Low plastic deformation and energy absorption

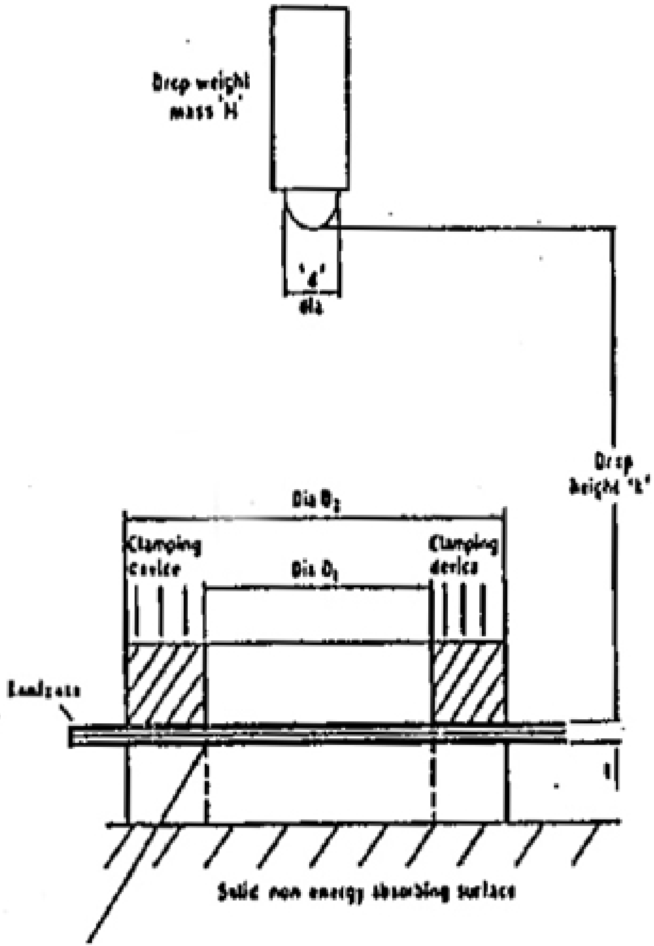
  → poor impact resistance

solutions :

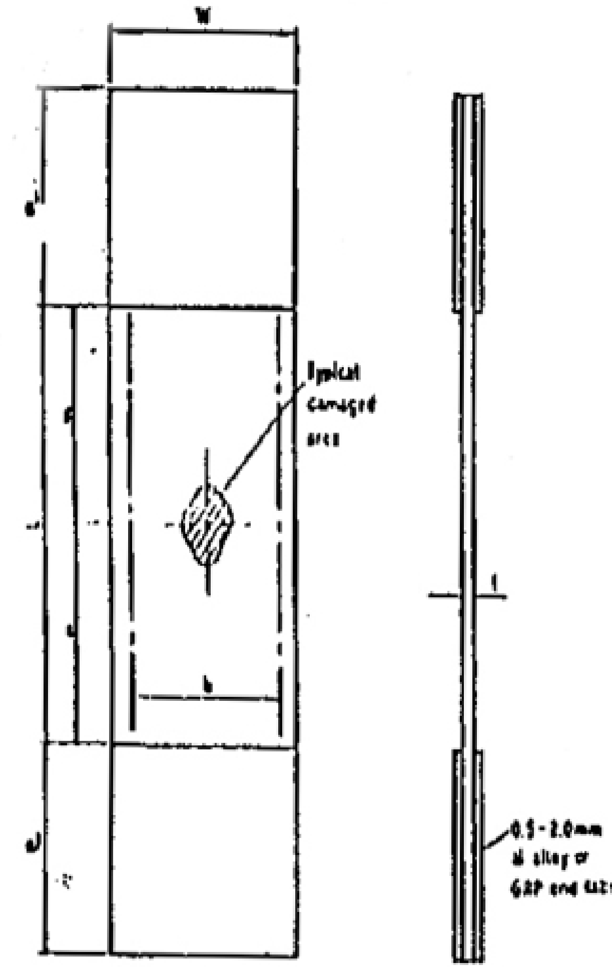
- Kevlar fibre reinforcement
- Hybrid
- Woven forms
- Thermoplastic

Measurement of Impact Resistance (Low energy)

Drop-weight test



Residual compressive strength test



Inspection

In service inspection for :

- surface damage
- Delamination
- Impact damage

Non Destructive Testing, NDT

- Visual
- Ultrasonic scan : Most commonly used method
- X-ray
- Thermal imaging : suitable for glass/epoxy
- Vibration analysis : delamination, skin-core disbonds of sandwich panels

Damage evaluation?
Remaining structural integrity?
Decisions? Repair/replace Structure/cosmetic
Subjective guidelines!

Typical design criteria requires that 1 inch dia damage zone (e.g. delamination or impact damage) has negligible growth in the lifetime of aircraft!

Repair/replace

- Bonded patch kits:
- Directional property match
- Out of place loads : →
- Thermosets :

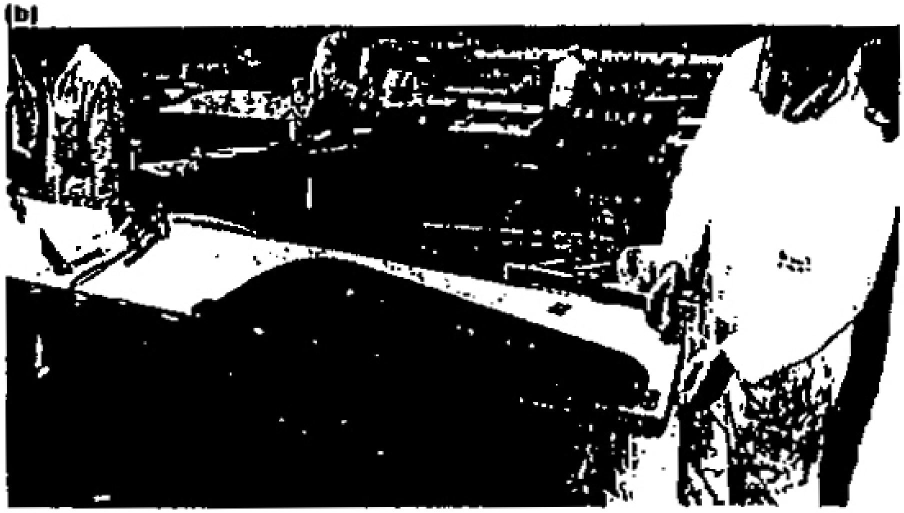
Refrigeration
Layup orientation
Scarf joints
Removal of moisture
Temperature / pressure cure

In Field!
No joint NDT!
Ensure correct procedure

2

(a) Boeing 757 aileron repair after vehicle impact damage

(b)



Joints

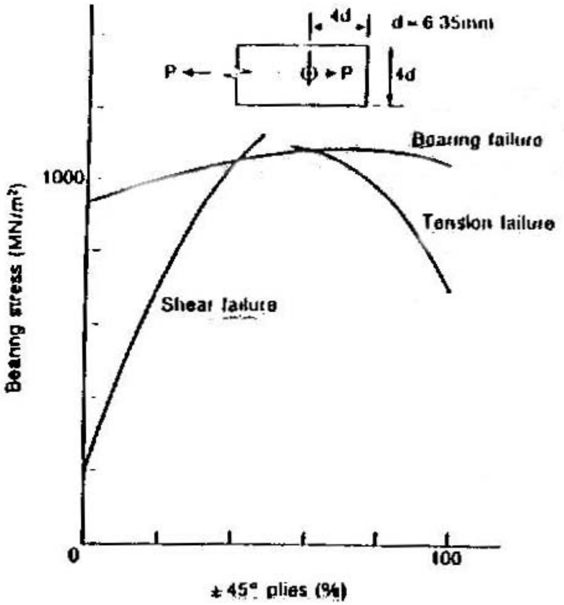
- **Bolted joints**

Design for **bearing**

Cross-section reduction

Stress concentration

Edge peel stresses



Drilling

Higher tolerances than metal

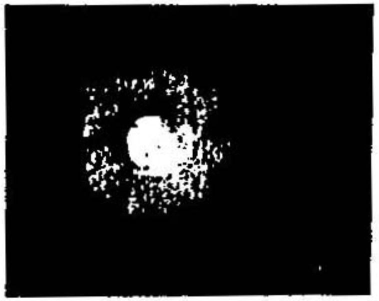
Back face break-out

Misplace holes

Crushing, Galling

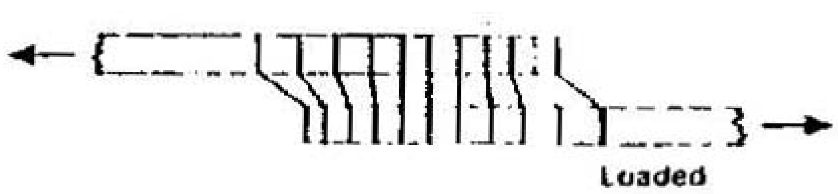
Insert-sandwich panels

Galvanic reaction

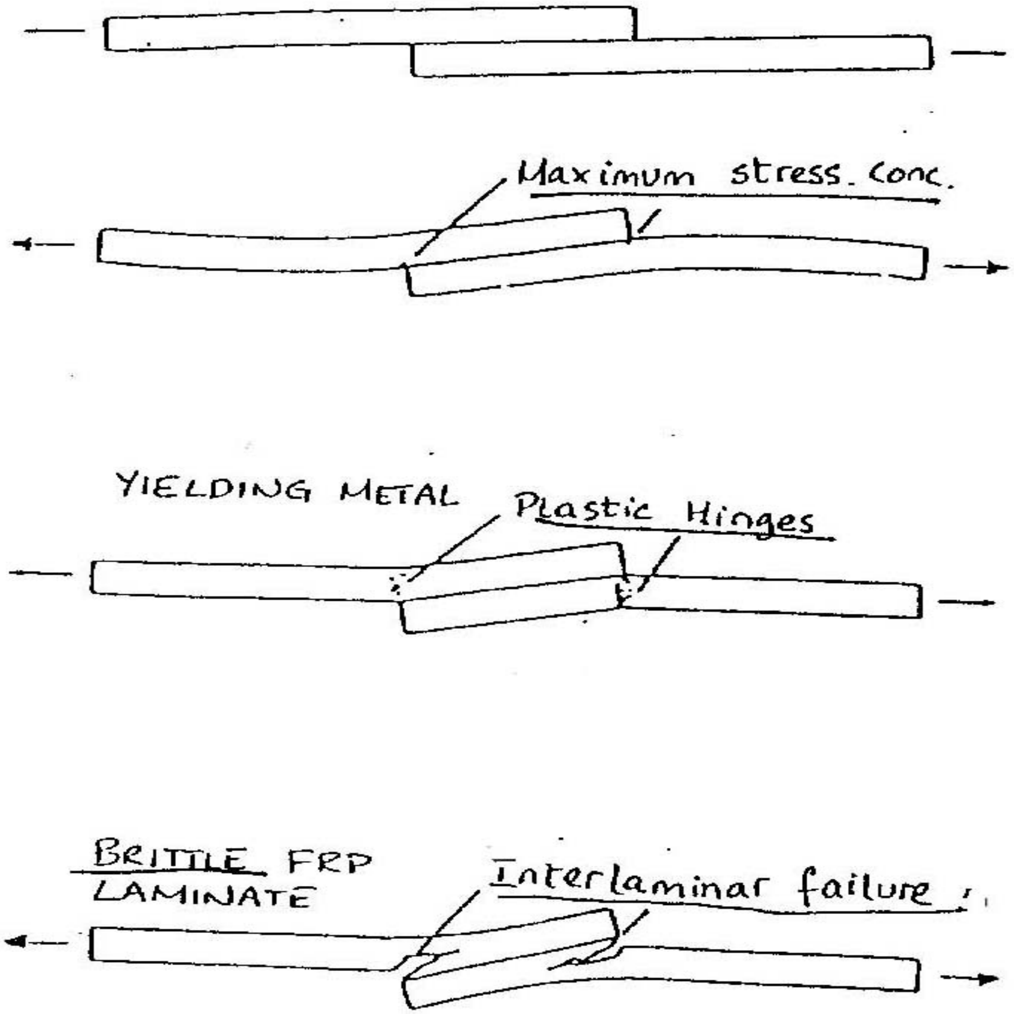


Bonded joints

- Preparation
- Procedure
- No joint NDT
- Mismatch : stiffness, thermal
- Eccentricity



Single lap eccentric loading

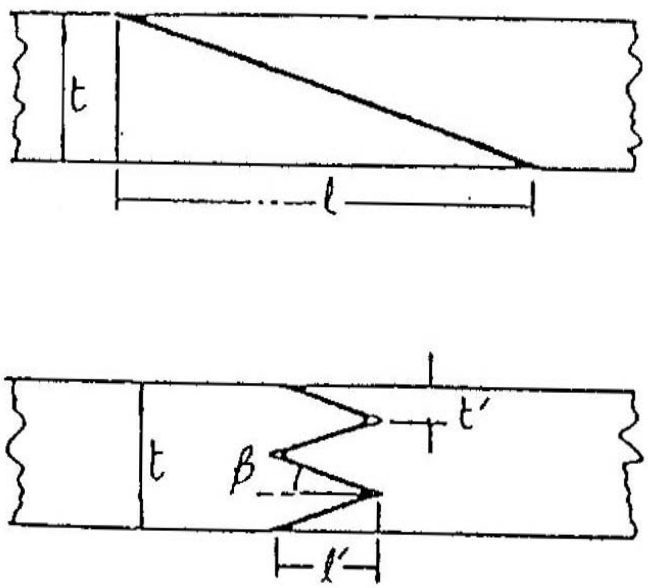


Scarf Joints ★

- Increase bond area

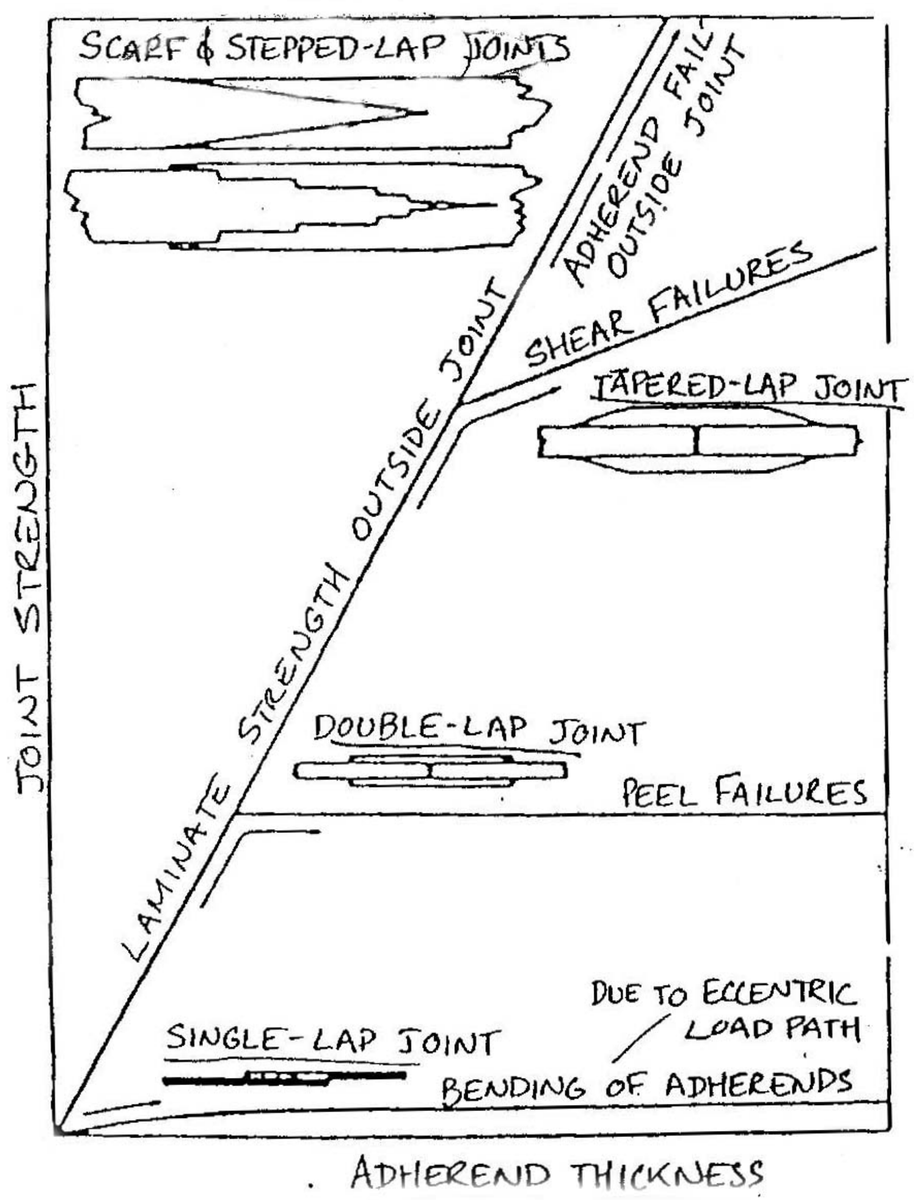
- Decrease effective lengths

- For mismatch of adherends
  (thermal and stiffness)

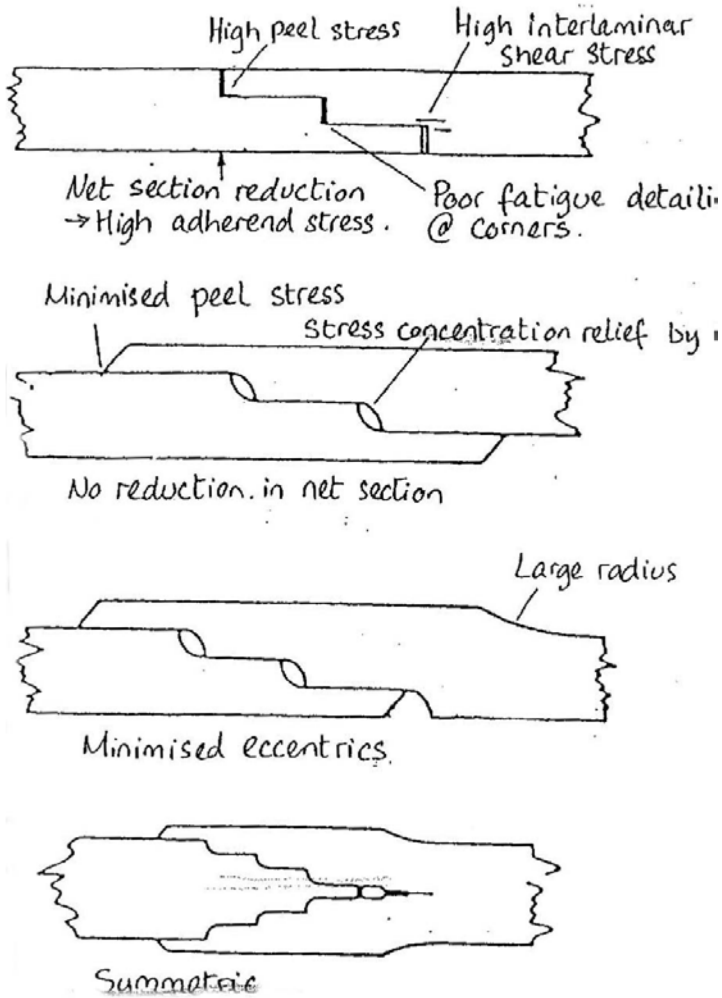


Joint failure for optimum design and geometry

- Influence of joint size and selection of joint configuration



Stepped-lap Joints



ELECTRICAL

Conditionctivity

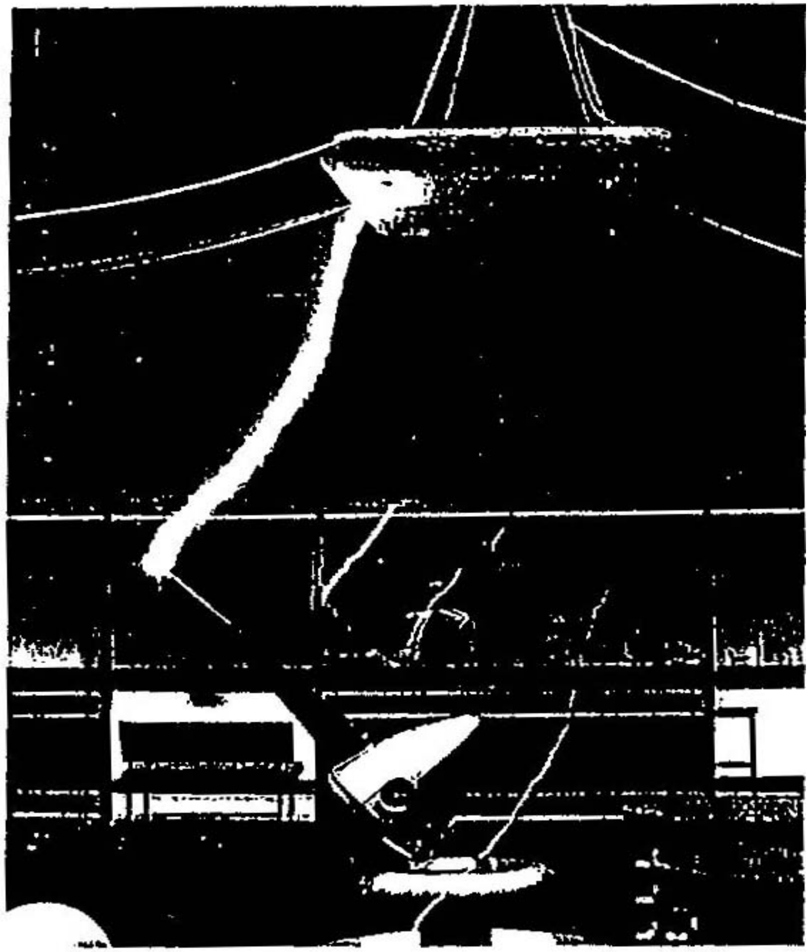
- Low electrical conductivity
- Low thermal conductivity

Lightening strike performance

- Poor energy dissipation
- Require metallic mesh, foil, stainless steel → weight

Electrical use

- Earth returns
- Uni-pole aerials
- Screening



Composite blade undergoing lightning strike test *Dowty Rotol Ltd*

Summary

Advanced Fibre Reinforced Composite Materials :

Special considerations

- ✓ Material made at component stage
- ✓ Different properties in different directions
- ✓ Reinforcing fibres are linear elastic to failure

Incentives

- ✓ Lighter, stiffer , stronger
- ✓ Corrosion resistant, fatigue resistant
- ✓ Optimized directional properties