Composite Materials for Aircraft Structures: A Brief Review of Practical Application

Jared W Nelson,
PhD Candidate

Department of Mechanical and Industrial Engineering
Montana State University

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MONTANA STATE UNIVERSITY
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Mountains & Minds

Introduction

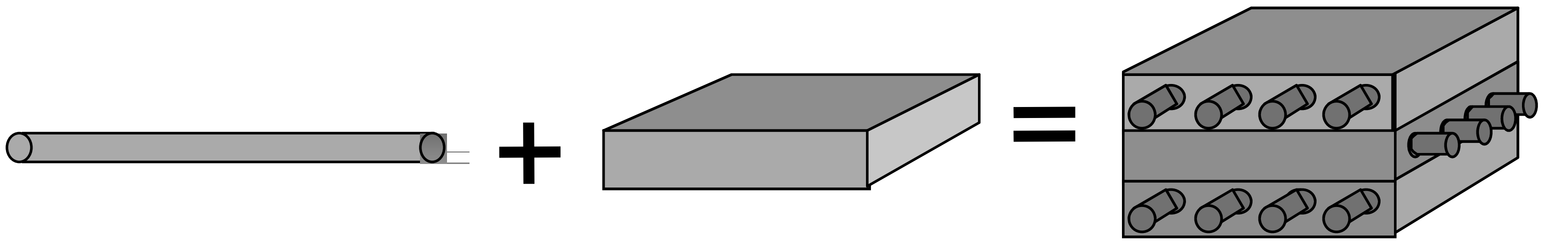
- **Composite materials are used more and more for primary structures in commercial, industrial, aerospace, marine, and recreational structures**



From Last Time

- **Composite parts used for aircraft applications are defined by**
  - Material, process, and manufacturing specifications.
  - Material allowable (engineering definition).
- **All of these have a basis in regulatory requirements.**
- **Most efficient use of advanced composites in aircraft structure is in applications with**
  - Highly loaded parts with thick gages.
  - High fatigue loads (fuselage and wing structure, etc).
  - Areas susceptible to corrosion (fuselage, etc).
  - Critical weight reduction (empennage, wings, fuselage, etc).
- **Use must be justified by weighing benefits against costs.**

Composition of Composites



Fiber/Filament Reinforcement

- High strength
- High stiffness
- Low density
- Carbon, Glass, Aramid, etc

Matrix

- Good shear properties
- Low density
- Thermoset & Thermoplastic
- Epoxy, Polyester, PP, Nylon, Ceramics, etc.

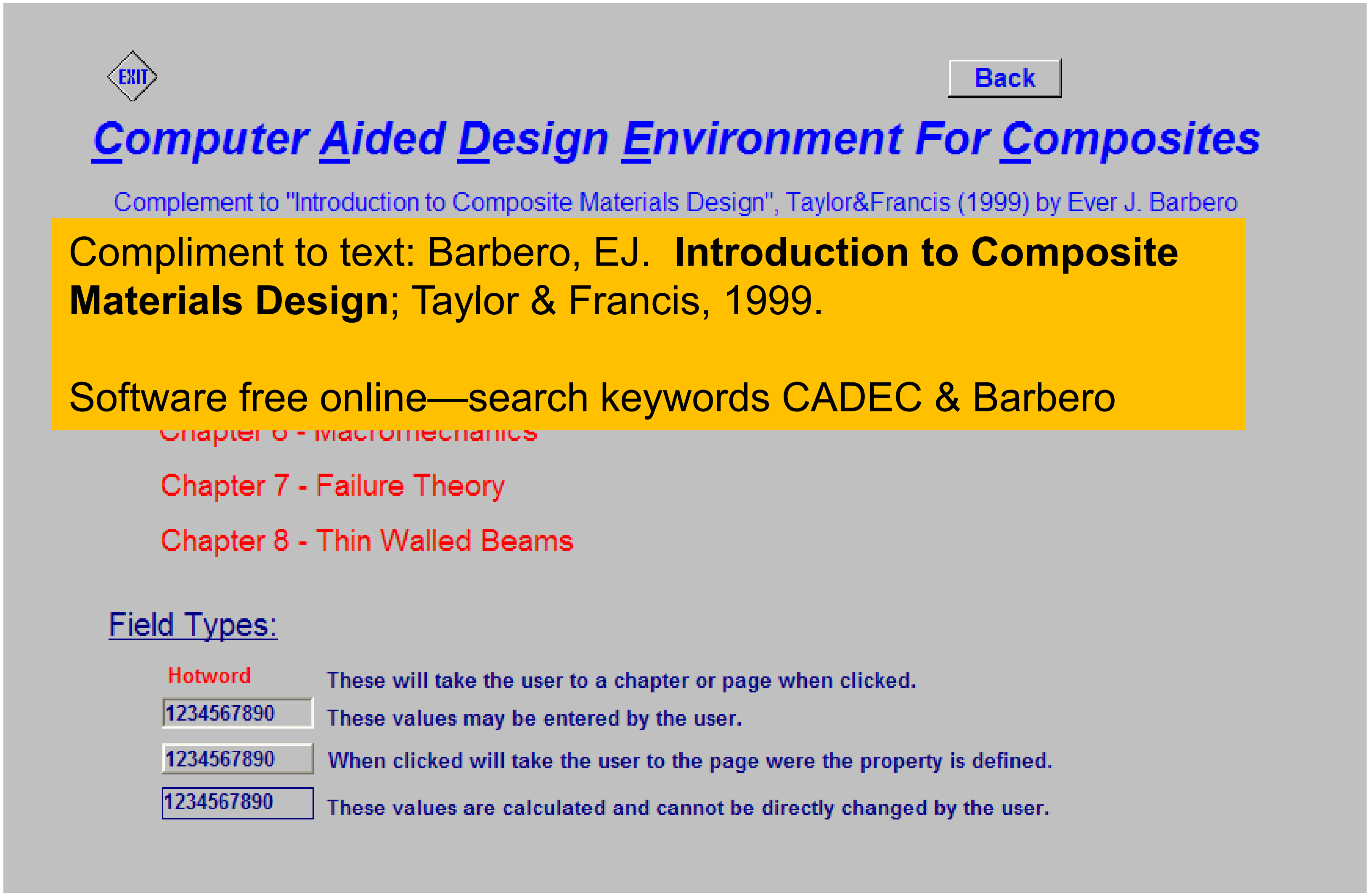
Composite

- High strength
- High stiffness
- Good shear properties
- Low density
- **Anisotropic!**

Overview

- **Micromechanics**
  - Study of mechanical behavior of a composite material in terms of its constituent materials
- **Ply Mechanics**
  - Study of mechanical behavior of individual material plies based on variations from global coordinate system
- **Macromechanics**
  - Study of mechanical behavior utilizing ply mechanics of a homogenized composite material
- **Failure Theories**

CADEC: Introduction



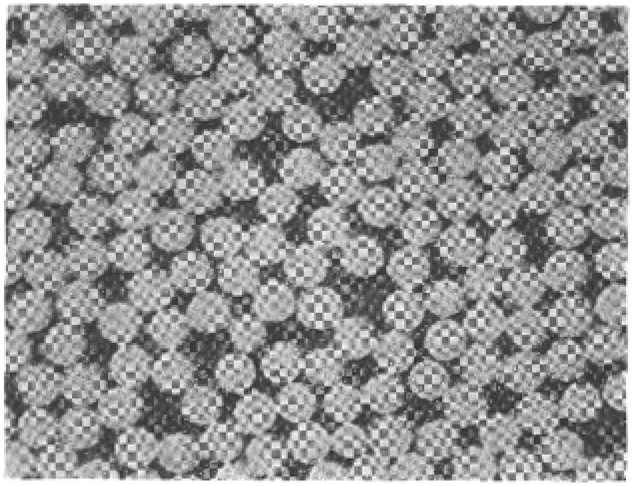
Compliment to text: Barbero, EJ. **Introduction to Composite Materials Design**; Taylor & Francis, 1999.

Software free online—search keywords CADEC & Barbero

Micromechanics: Assumptions

- **Lamina:**
  - Macroscopically homogeneous
  - Linearly elastic
  - Macroscopically Orthotropic
  - Initially stress free

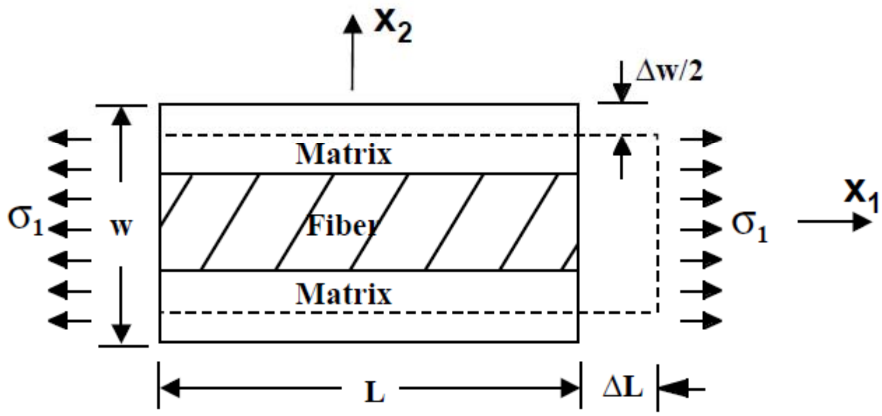
- **Fibers:**
  - Homogeneous
  - Linearly elastic
  - Isotropic/Orthotropic
  - Regularly spaced
  - Perfectly aligned

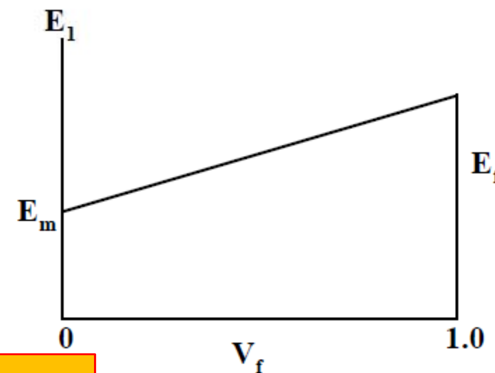


Carbon/epoxy (AS4/3501-6) composite (Vf=.70)

- **Matrix:**
  - Homogeneous
  - Linearly elastic
  - Isotropic
  - Assumptions in Micromechanics of Composites

Micromechanics: Rule of Mixtures





Assumption: Axial (… **fiber and matrix**

$V_{f,max}$ approximately 78%
Common range = 55-67%

∴ **Strain in the composite** $\varepsilon_1 = \frac{\quad}{L} = \varepsilon_f = \varepsilon_m$

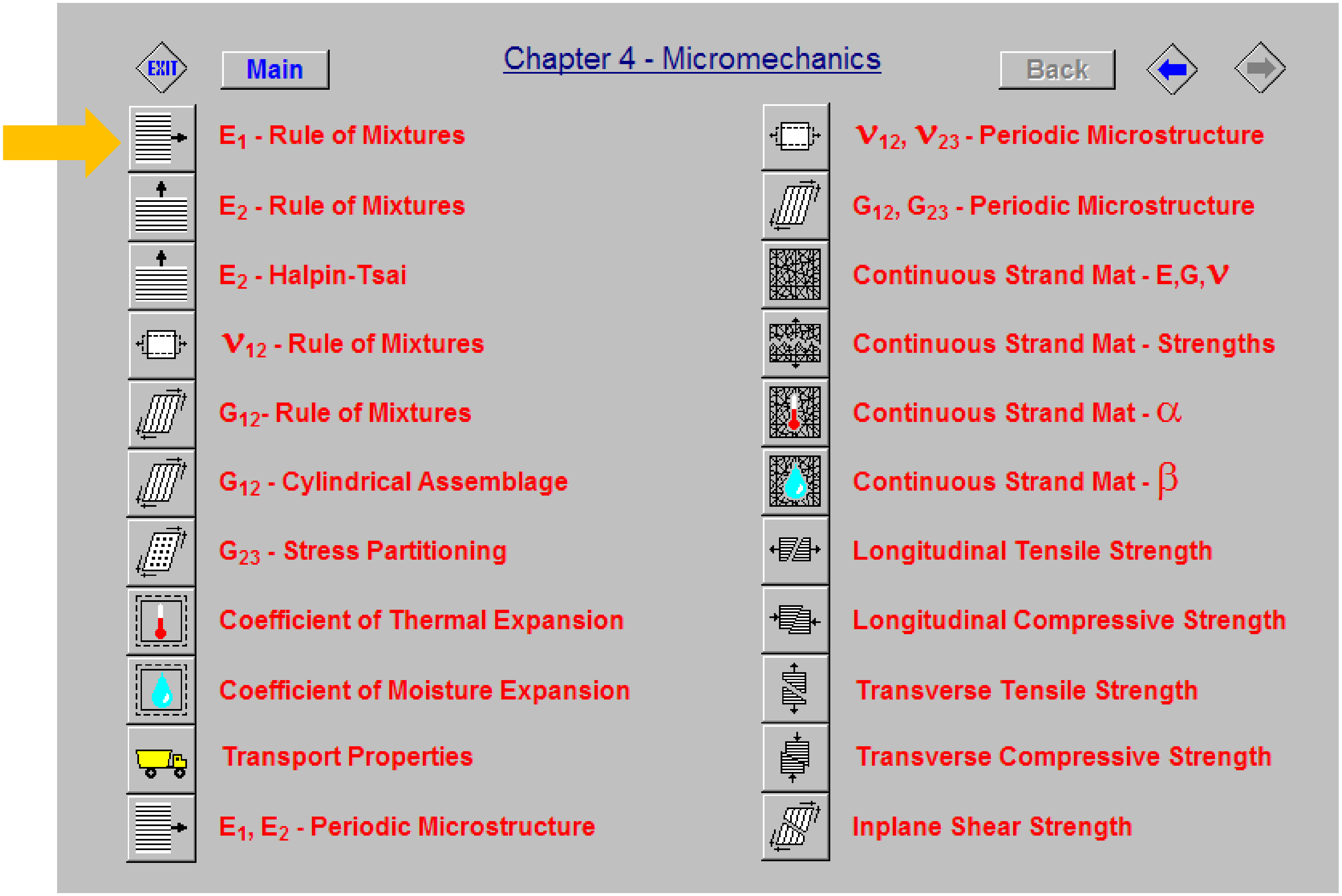
Total force in composite $\sigma_1 A_c = \sigma_f A_f + \sigma_m A_m$

∴ **Stress in the composite** $\sigma_1 = \sigma_f \frac{A_f}{A_c} + \sigma_m \frac{A_m}{A_c} = \sigma_f V_f + \sigma_m V_m$

$$\varepsilon_1 E_1 = \varepsilon_1 E_f V_f + \varepsilon_1 E_m V_m$$

∴ $$E_1 = E_f V_f + E_m V_m$$ ***Rule of mixtures***

Micromechanics: Determining Properties



Micromechanics: Rule of Mixtures (E_1)

EXIT | Main | Chapter | Calculate Value | Update Graph | Back

Longitudinal Modulus - Rule of Mixtures

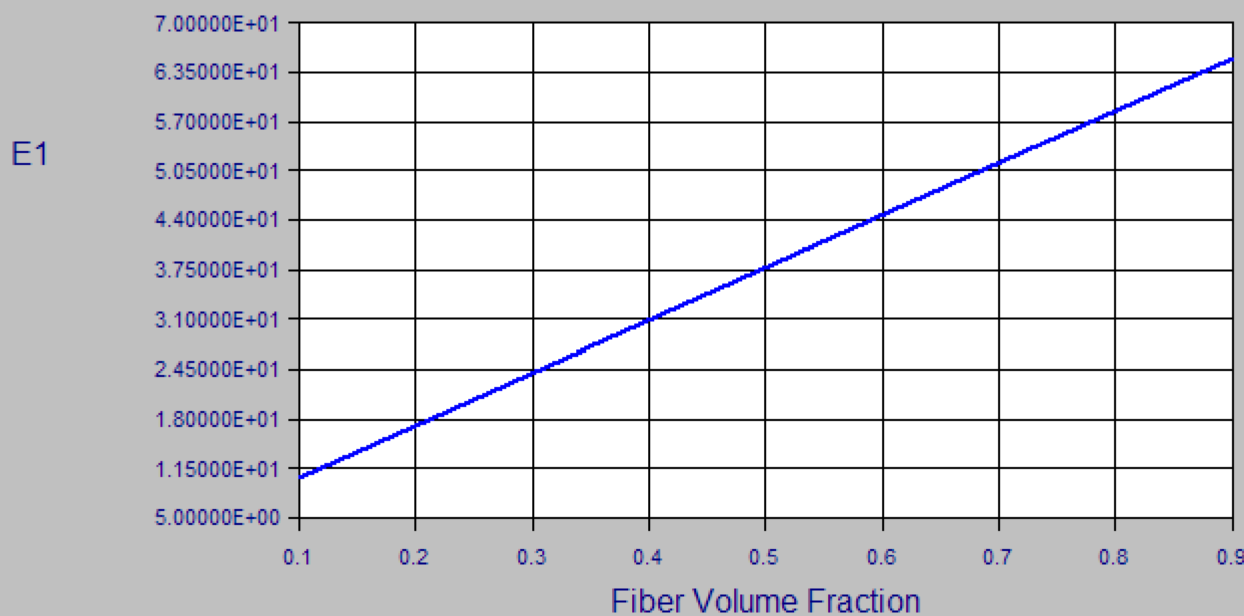
$$E_1 = E_f V_f + E_m V_m$$

E_f = 72.345

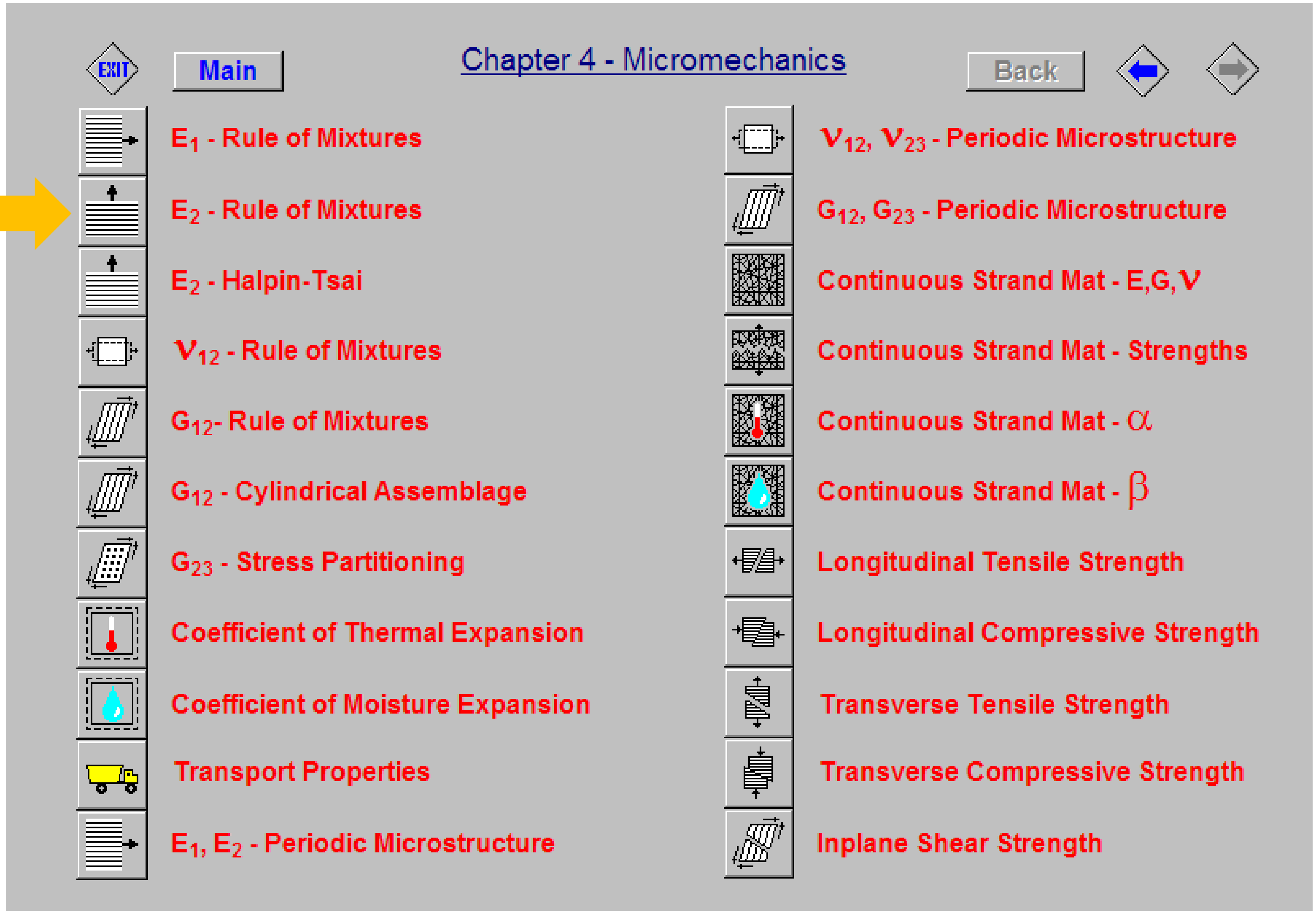
V_f = .5

E_m = 3.4

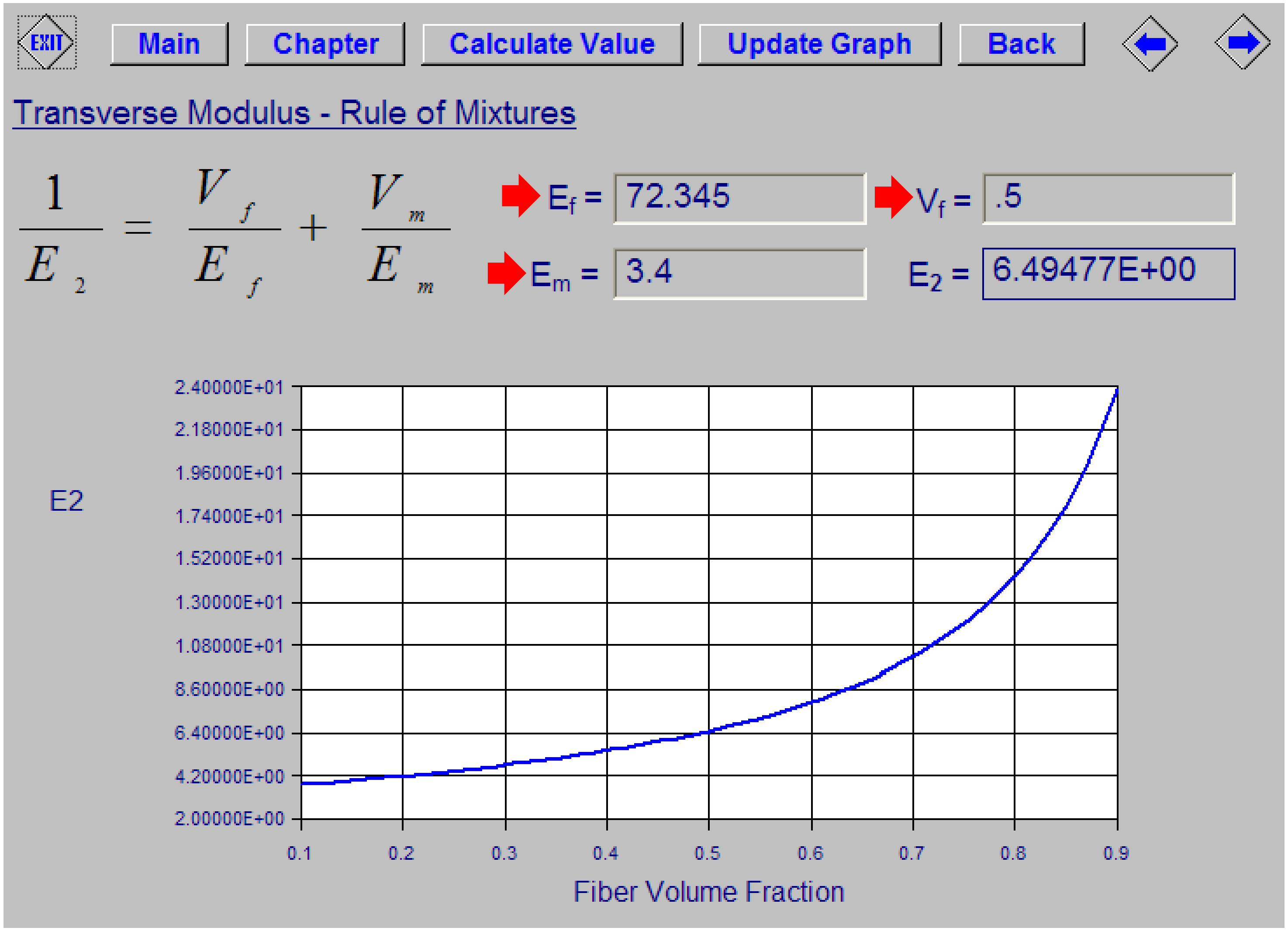
E_1 = 3.78725E+01



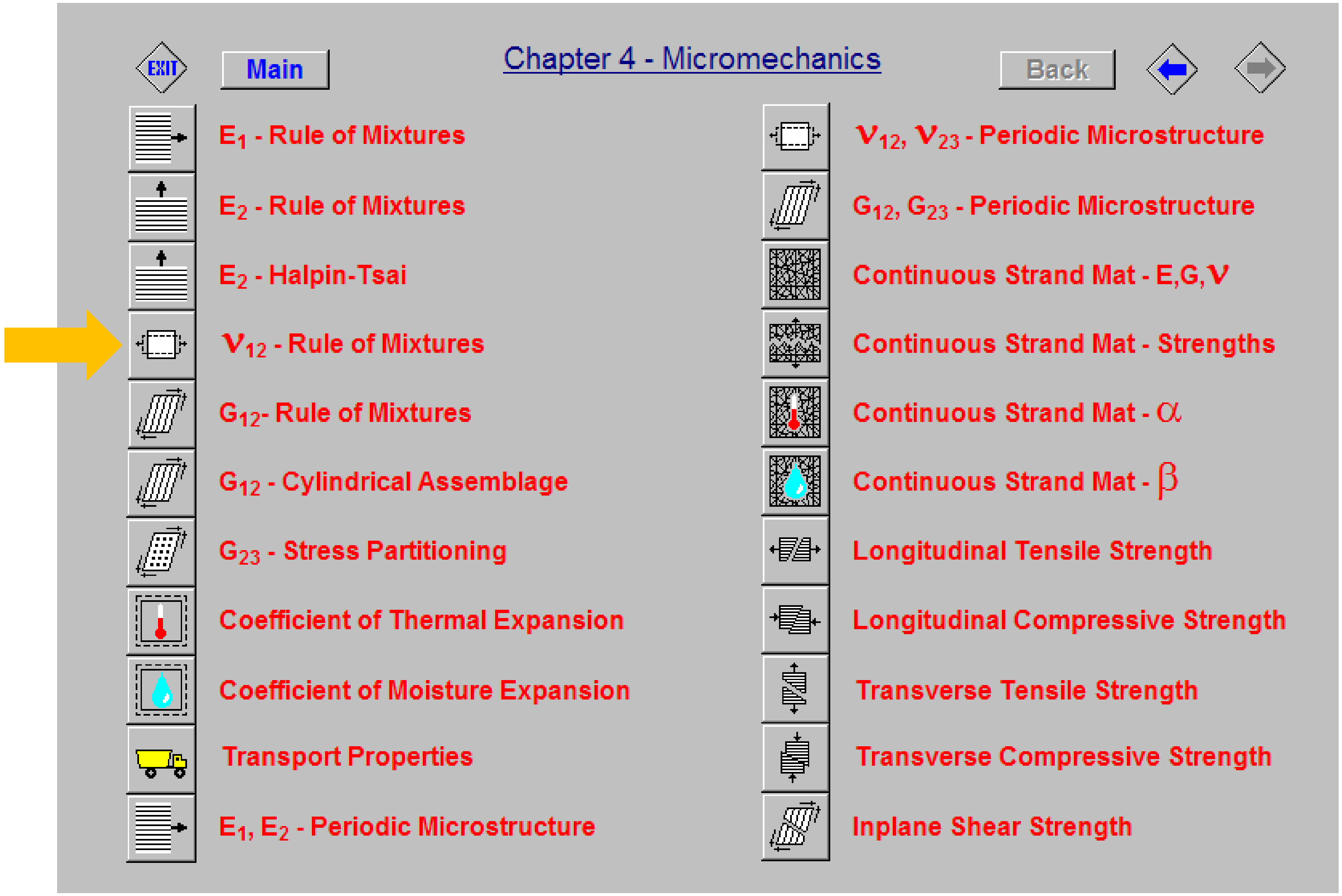
Micromechanics: Determining Properties



Micromechanics: Rule of Mixtures (E_2)



Micromechanics: Determining Properties



Micromechanics: Rule of Mixtures (ν_{12})

EXIT | Main | Chapter | Calculate Value | Update Graph | Back

Inplane Poisson's Ratio - Rule of Mixtures

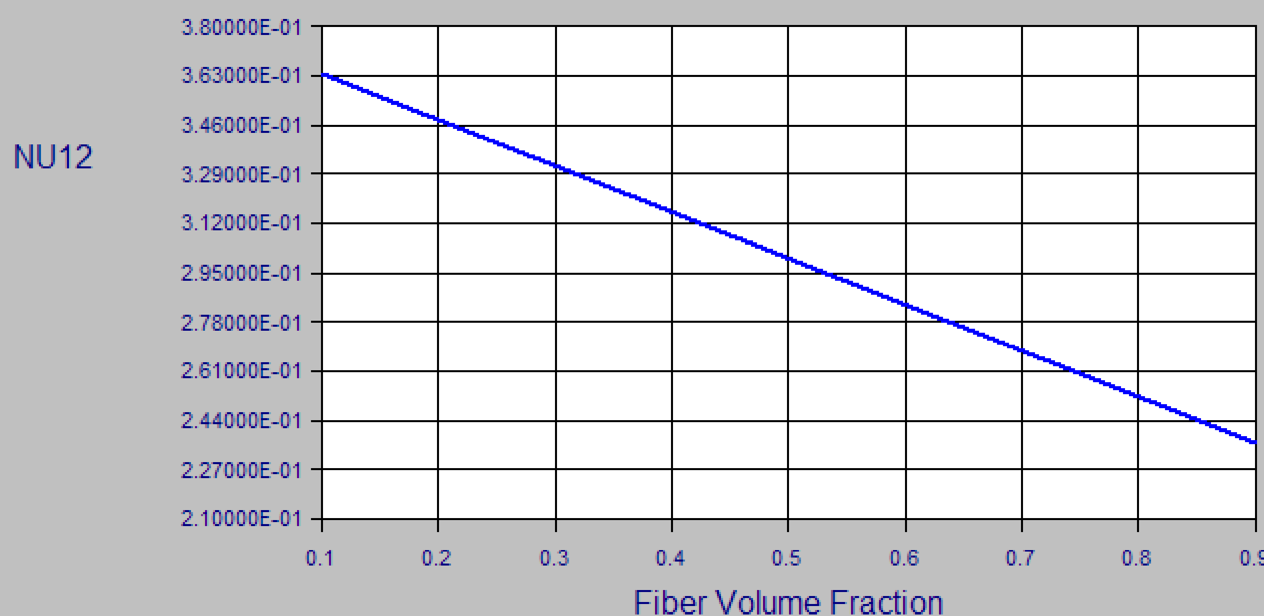
$$\nu_{12} = \nu_f V_f + \nu_m V_m$$

ν_f = 0.22

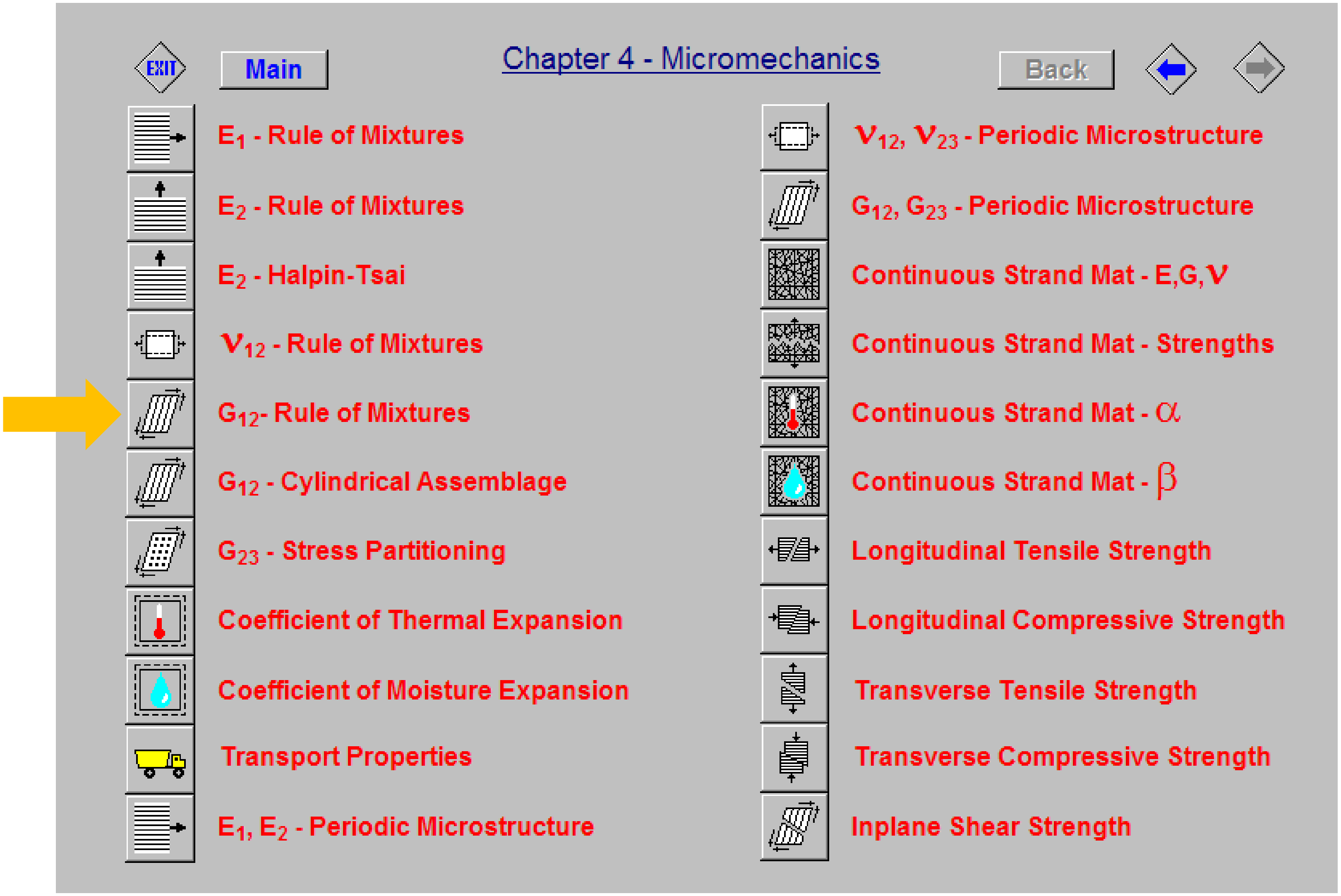
V_f = .5

ν_m = 0.38

ν_{12} = 0.30000



Micromechanics: Determining Properties



Micromechanics: Rule of Mixtures (G_{12})

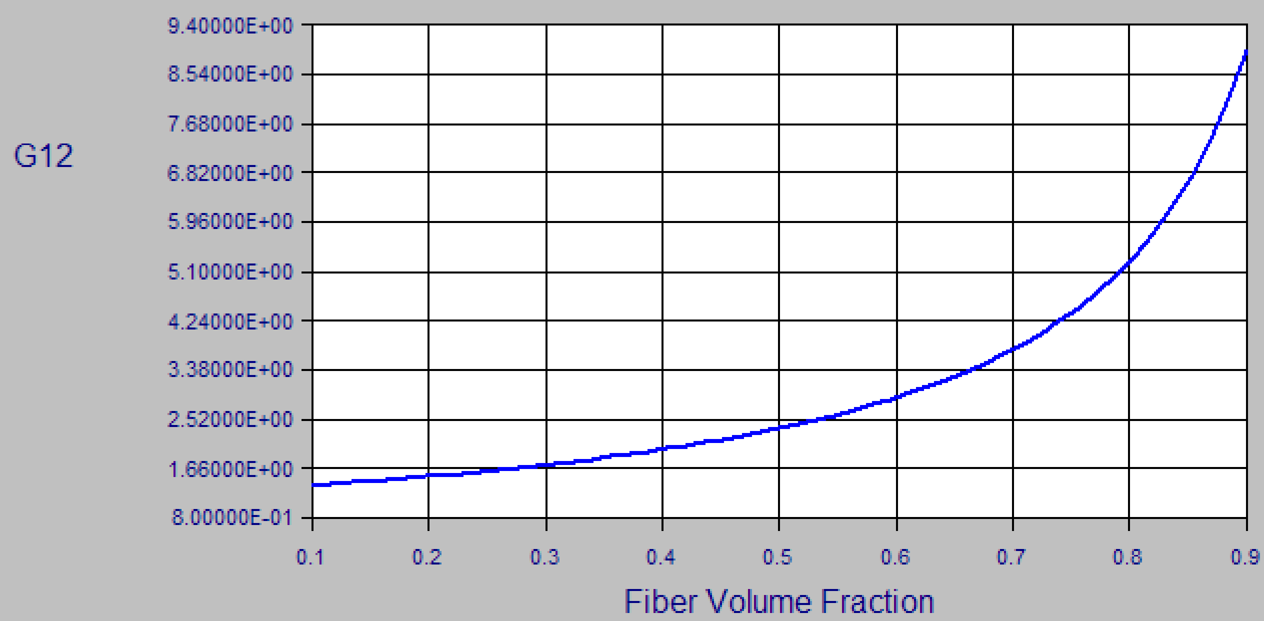
EXIT | Main | Chapter | Calculate Value | Update Graph | Back

Inplane Shear Modulus - Rule of Mixtures

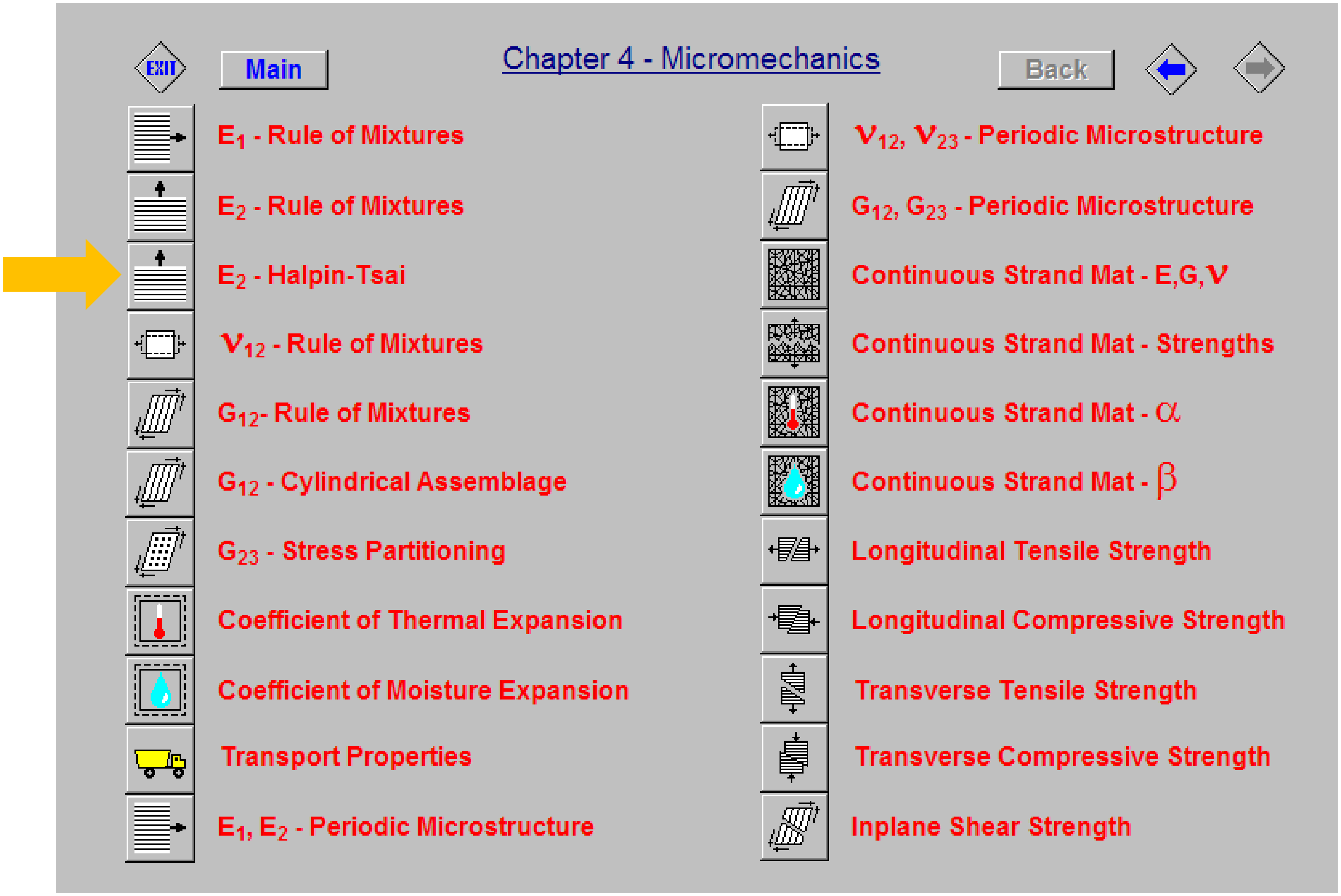
$$\frac{1}{G_{12}} = \frac{V_m}{G_m} + \frac{V_f}{G_f}$$

G_f = 2.96496E+01 V_f = .5

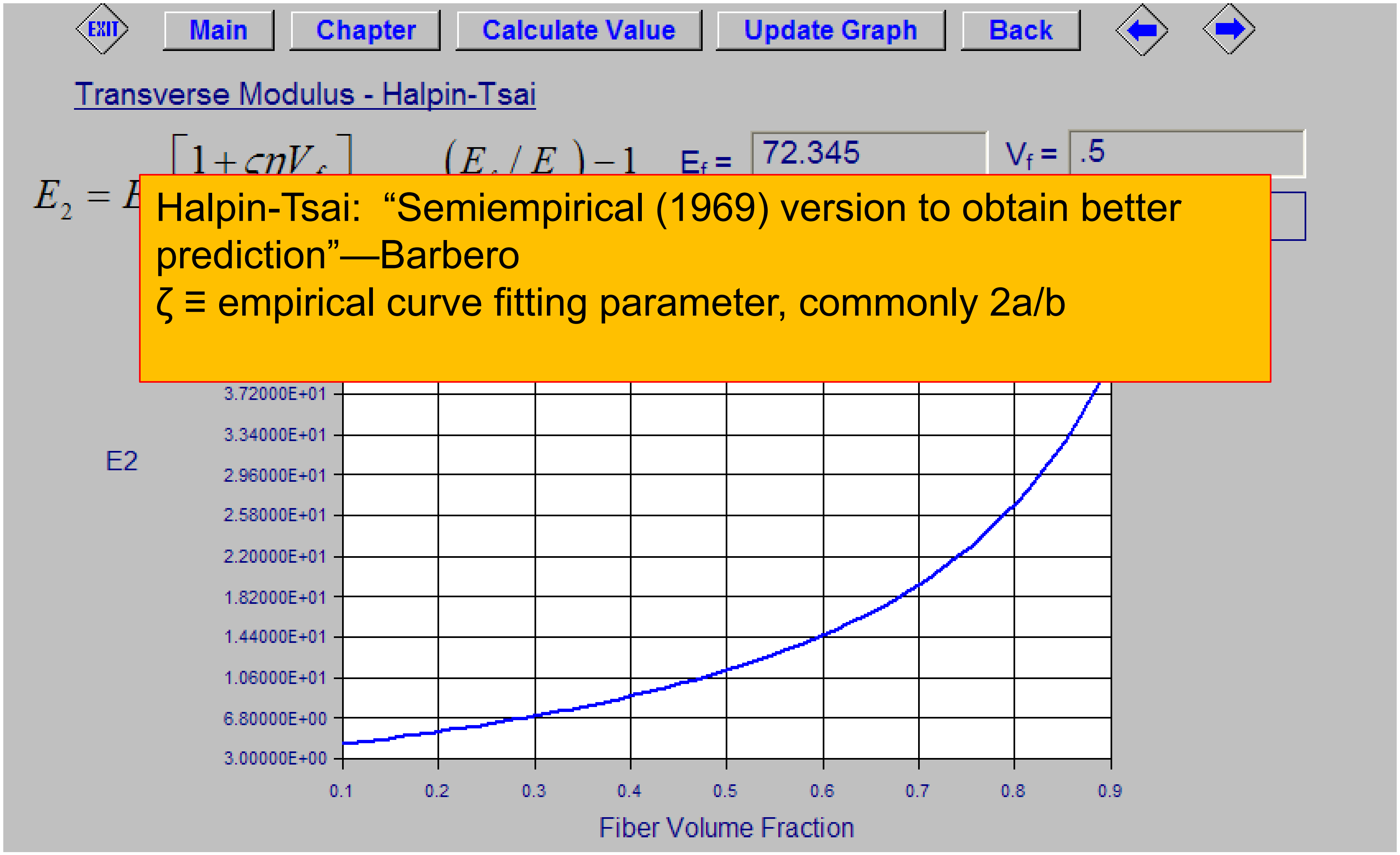
G_m = 1.23188E+00 G_{12} = 2.36548E+00



Micromechanics: Other Methods & Strengths



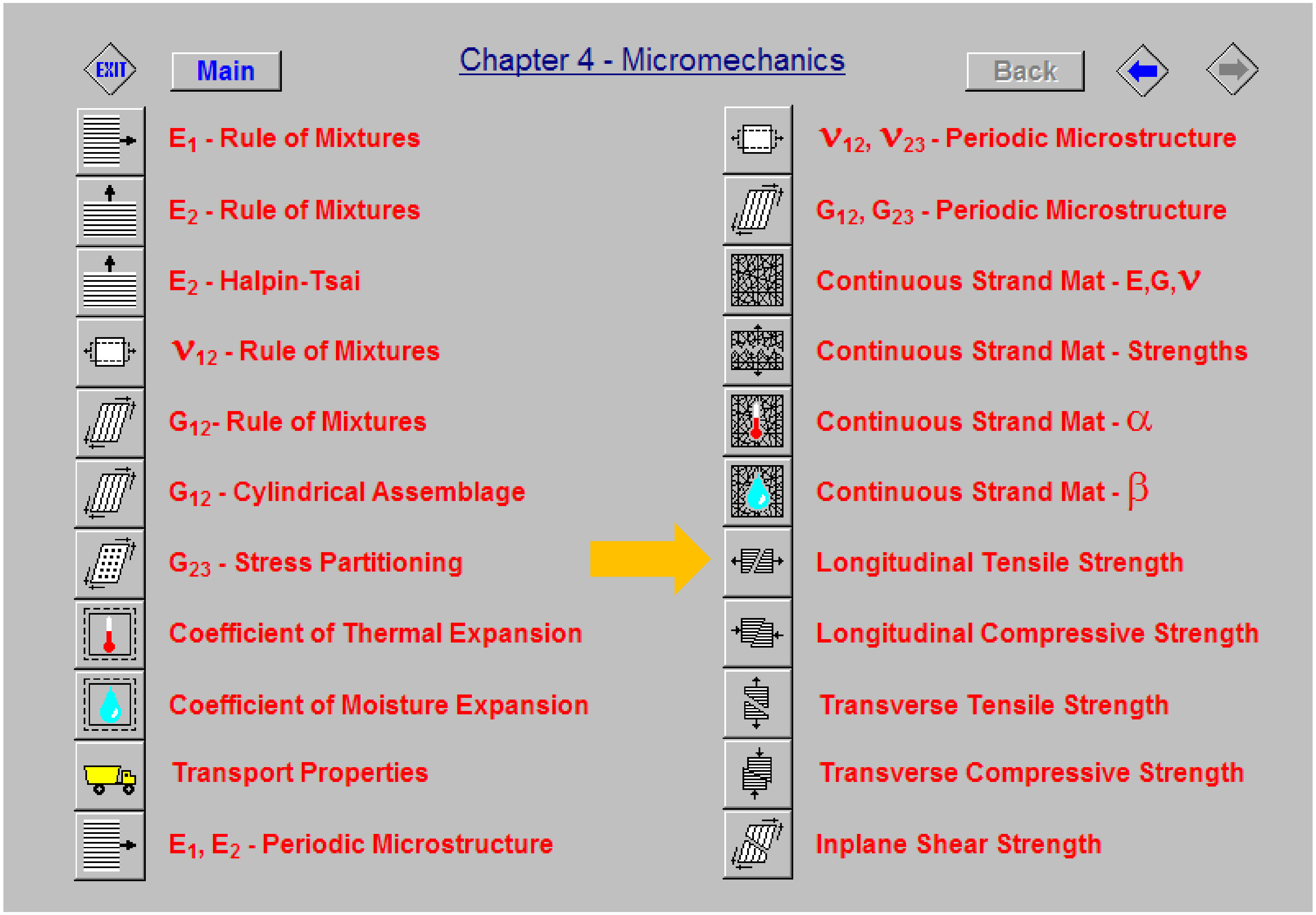
Micromechanics: Halpin-Tsai (E_2)



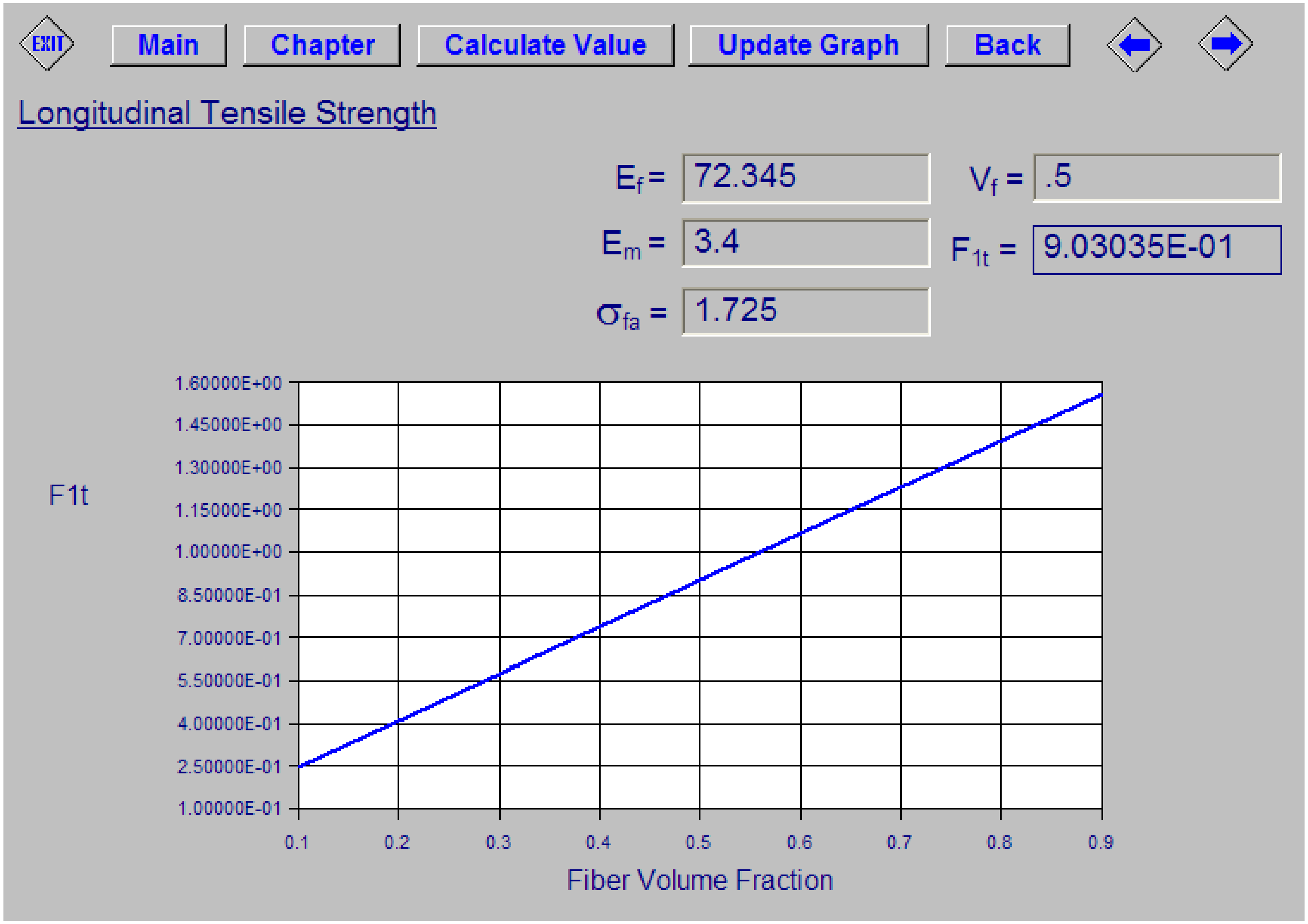
Halpin-Tsai: "Semiempirical (1969) version to obtain better prediction"—Barbero

$\zeta \equiv$ empirical curve fitting parameter, commonly 2a/b

Micromechanics: Determining Properties



Micromechanics: Longitudinal Tensile Strength

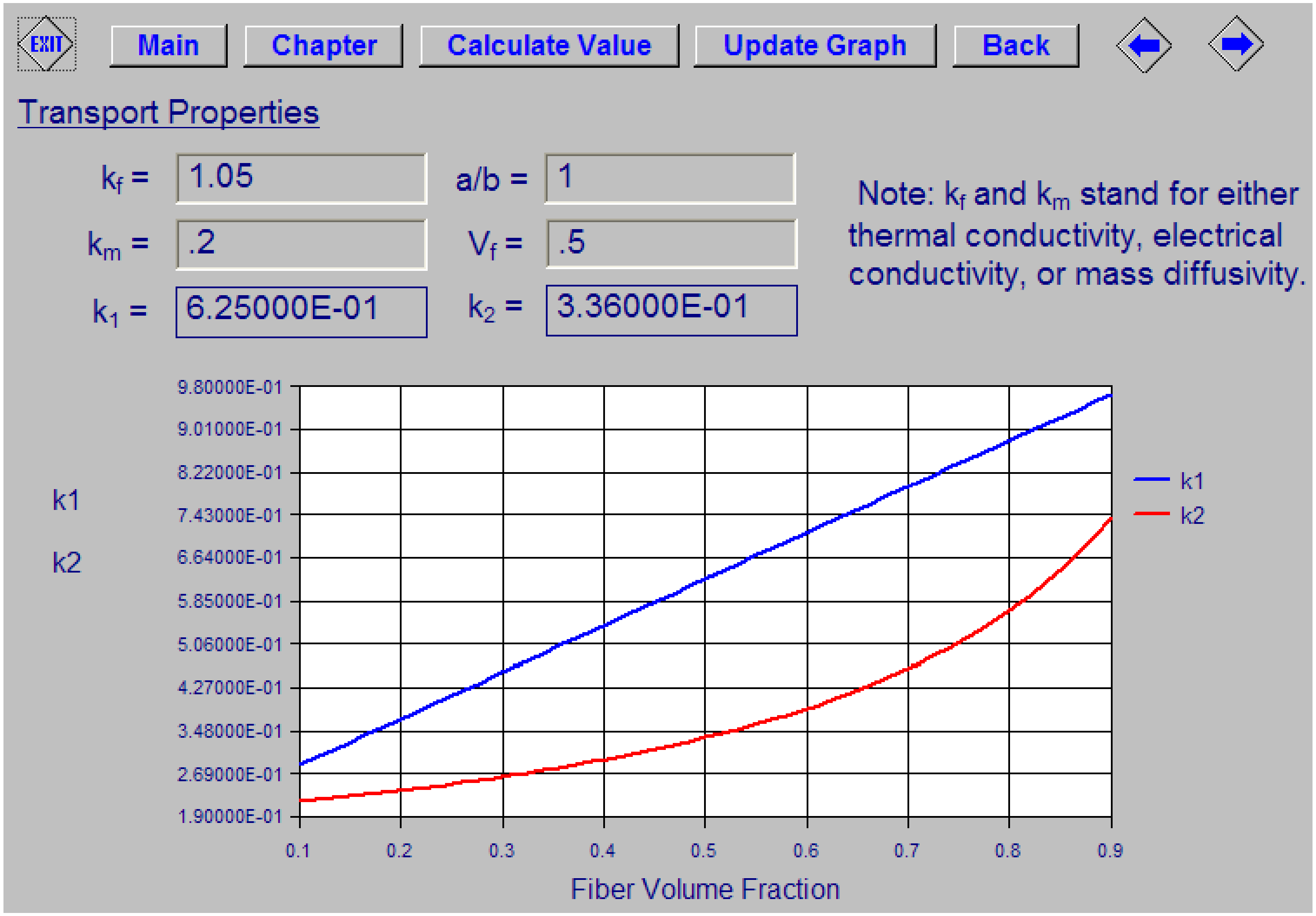


Micromechanics: Determining Properties

EXIT | Main | Chapter 4 - Micromechanics | Back

- E_1 - Rule of Mixtures
- E_2 - Rule of Mixtures
- E_2 - Halpin-Tsai
- ν_{12} - Rule of Mixtures
- G_{12}- Rule of Mixtures
- G_{12} - Cylindrical Assemblage
- G_{23} - Stress Partitioning
- Coefficient of Thermal Expansion
- Coefficient of Moisture Expansion
- Transport Properties
- E_1, E_2 - Periodic Microstructure
- ν_{12}, ν_{23} - Periodic Microstructure
- G_{12}, G_{23} - Periodic Microstructure
- Continuous Strand Mat - E,G,ν
- Continuous Strand Mat - Strengths
- Continuous Strand Mat - α
- Continuous Strand Mat - β
- Longitudinal Tensile Strength
- Longitudinal Compressive Strength
- Transverse Tensile Strength
- Transverse Compressive Strength
- Inplane Shear Strength

Micromechanics: Thermal & Electrical Cond



Ply Mechanics

- **So what happens if we vary the fiber direction angle away from the 1-direction?**
- **CADEC uses Micromechanics results and fiber angle**
  - Plane Stress
  - Transform stress/strain
  - Off-Axis Compliance/Stiffness
  - 3D Constitutive Equ's

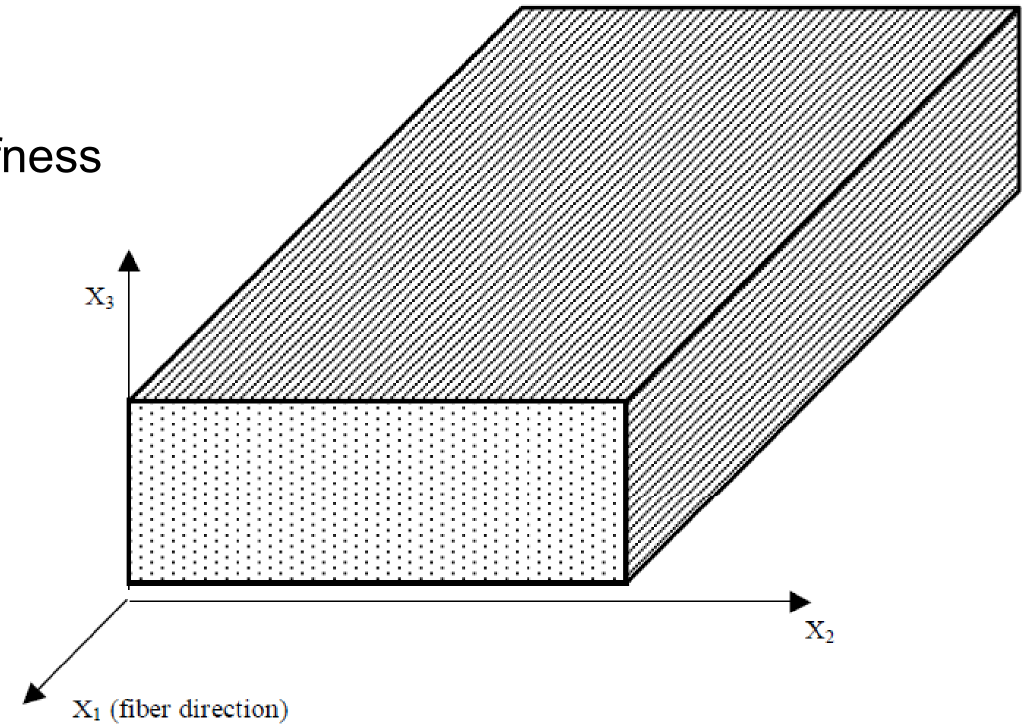
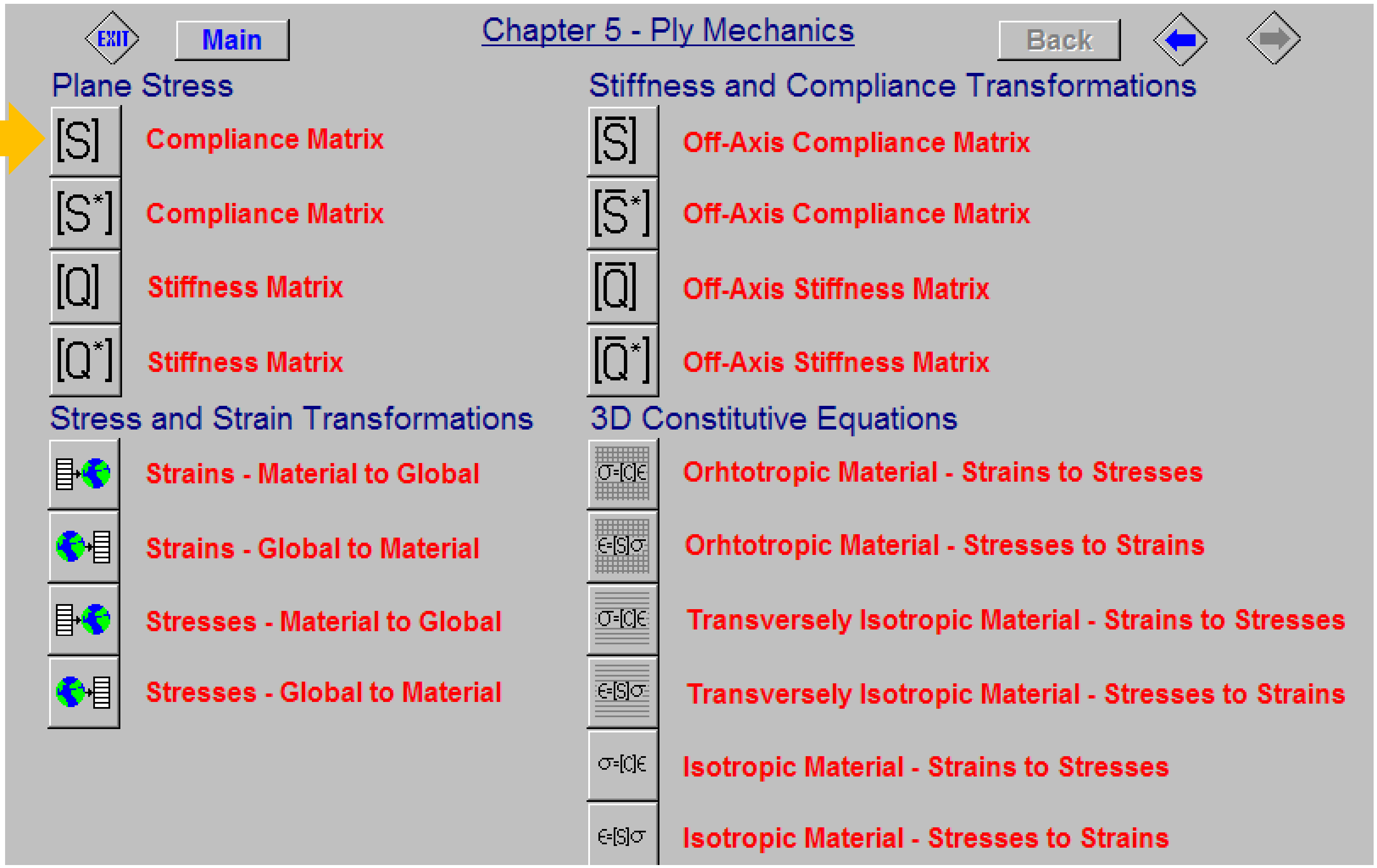


Figure 10.1 Idealized representation of a unidirectional fiber-reinforced material.

Ply Mechanics: CADEC



Ply Mechanics: Compliance Plane Stress

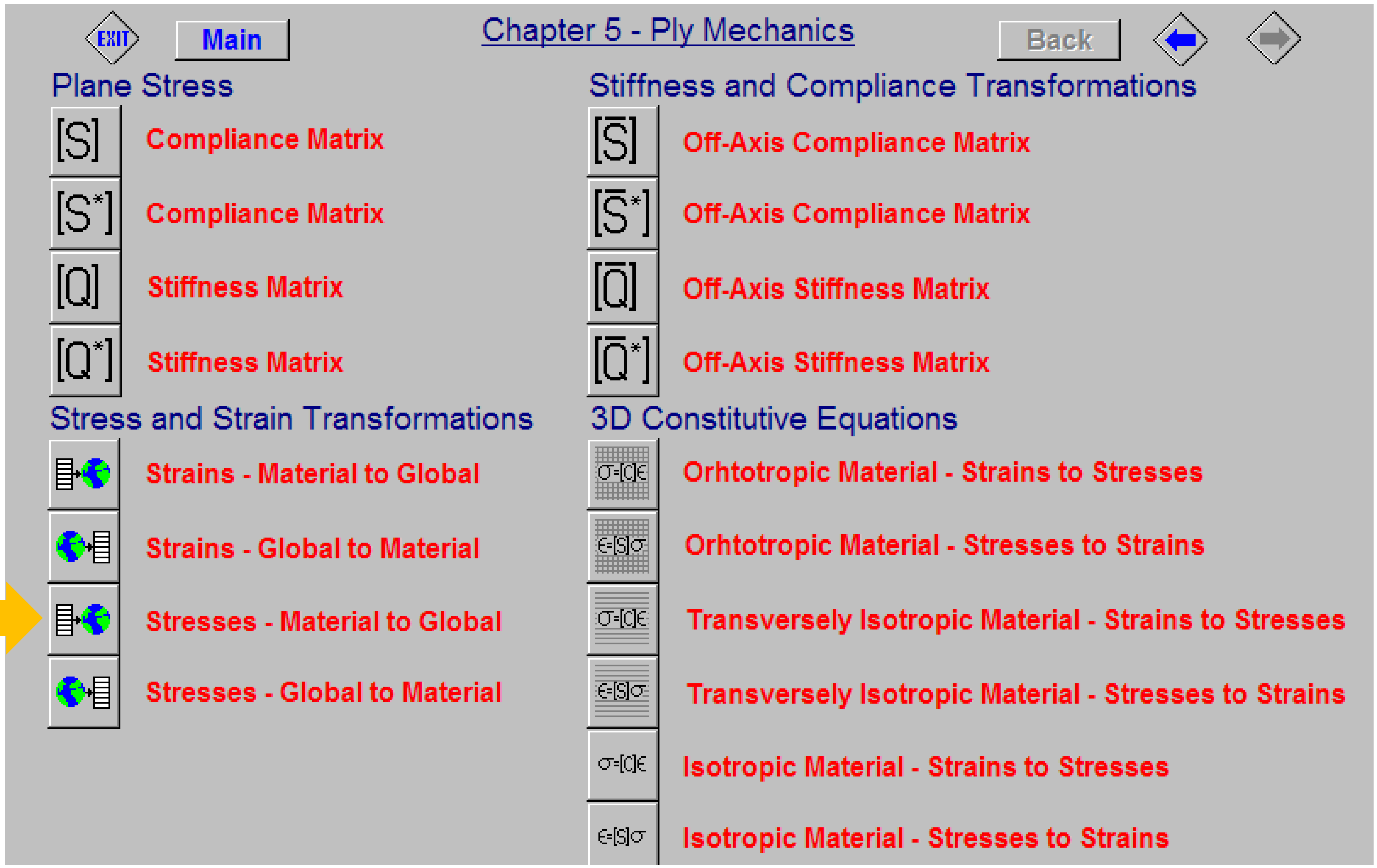
EXIT | Main | Chapter | Calculate Value | Back

Compliance Equations - [S] Matrix

		E_1 =	3.78725E+01				
σ_1 =	1.4e6	E_2 =	1.12711E+01		ε_1 =	2.74606E+04	
σ_2 =	1.2e6	G_{12} =	3.33156E+00		ε_2 =	9.53771E+04	
σ_6 =	.8e6	ν_{12} =	0.30000		γ_6 =	2.40128E+05	

$$\begin{Bmatrix} \varepsilon_1 \\ \varepsilon_2 \\ \gamma_6 \end{Bmatrix} = \begin{bmatrix} 1/E_1 & -\nu_{12}/E_1 & 0 \\ -\nu_{12}/E_1 & 1/E_2 & 0 \\ 0 & 0 & 1/G_{12} \end{bmatrix} \begin{Bmatrix} \sigma_1 \\ \sigma_2 \\ \sigma_6 \end{Bmatrix}$$

Ply Mechanics: CADEC



Ply Mechanics: Transformations

EXIT | Main | Chapter | Calculate Value | Back

Transform Material Stresses to Global Stresses

σ_1 =	100			σ_x =	3.49106E+01
σ_2 =	10	θ is in Degrees (°)		σ_y =	7.50894E+01
σ_6 =	-5	θ =	-55	σ_{xy} =	-4.05761E+01
σ_4 =	6.42909E04			σ_{yz} =	8.95398E+04
σ_5 =	6.42909E04			σ_{xz} =	-1.57883E+04

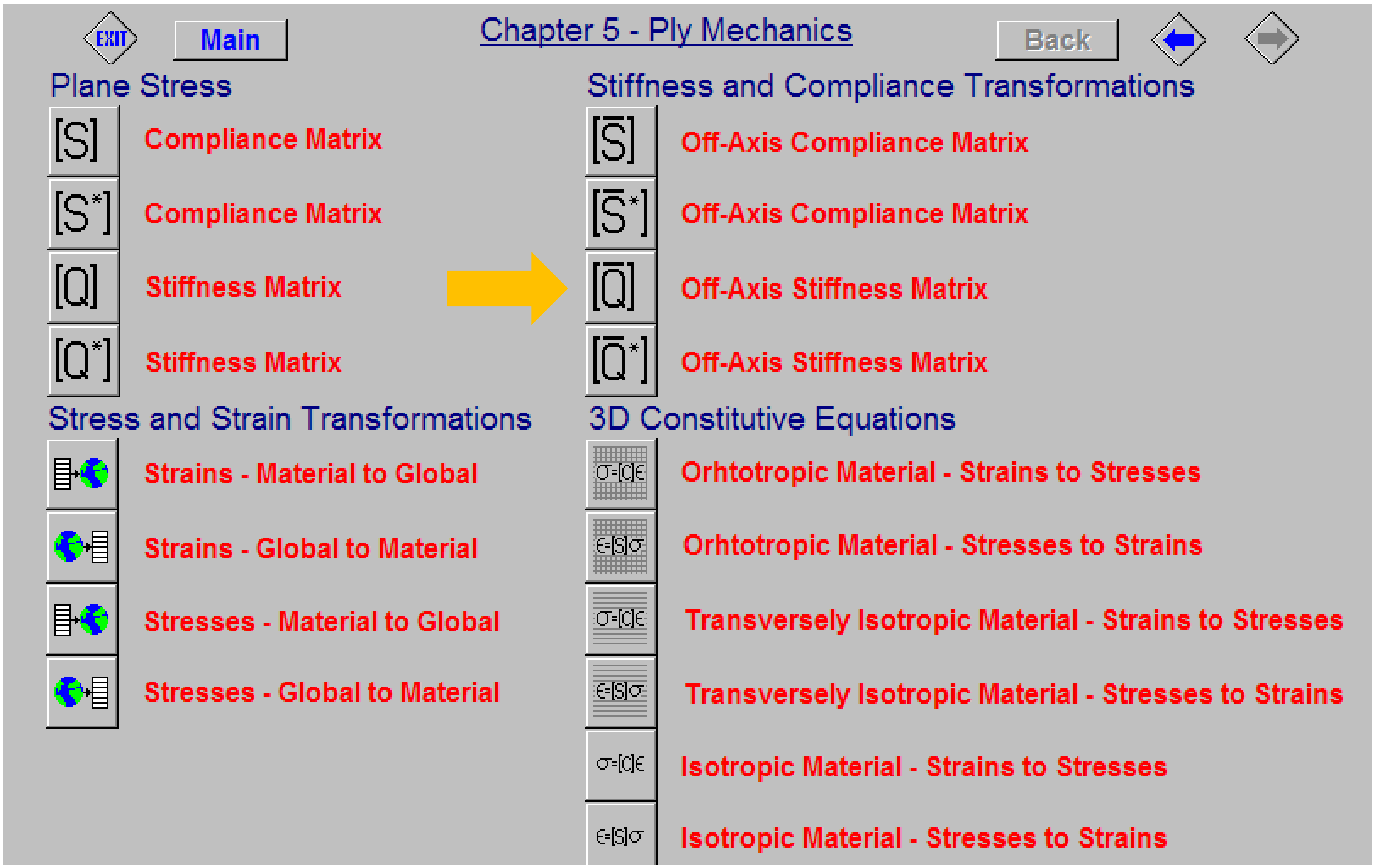
$$\begin{Bmatrix} \sigma_x \\ \sigma_y \\ \sigma_{xy} \end{Bmatrix} = \begin{bmatrix} m^2 & n^2 & -2mn \\ n^2 & m^2 & 2mn \\ mn & -mn & m^2 - n^2 \end{bmatrix} \begin{Bmatrix} \sigma_1 \\ \sigma_2 \\ \sigma_6 \end{Bmatrix}$$

$$\begin{Bmatrix} \sigma_{yz} \\ \sigma_{xz} \end{Bmatrix} = \begin{bmatrix} m & -n \\ n & m \end{bmatrix} \begin{Bmatrix} \sigma_4 \\ \sigma_5 \end{Bmatrix}$$

$m = \cos(\theta)$

$n = \sin(\theta)$

Ply Mechanics: CADEC



Ply Mechanics: Off-Axis Stiffness Matrices

EXIT | Main | Chapter | Calculate Value | Back

Off-Axis Stiffness Matrix - $[\bar{Q}]$

E_1 = 3.78725E+01

ε_x = 1.50706E+00 | E_2 = 1.12711E+01 | σ_x = 4.95793E+01

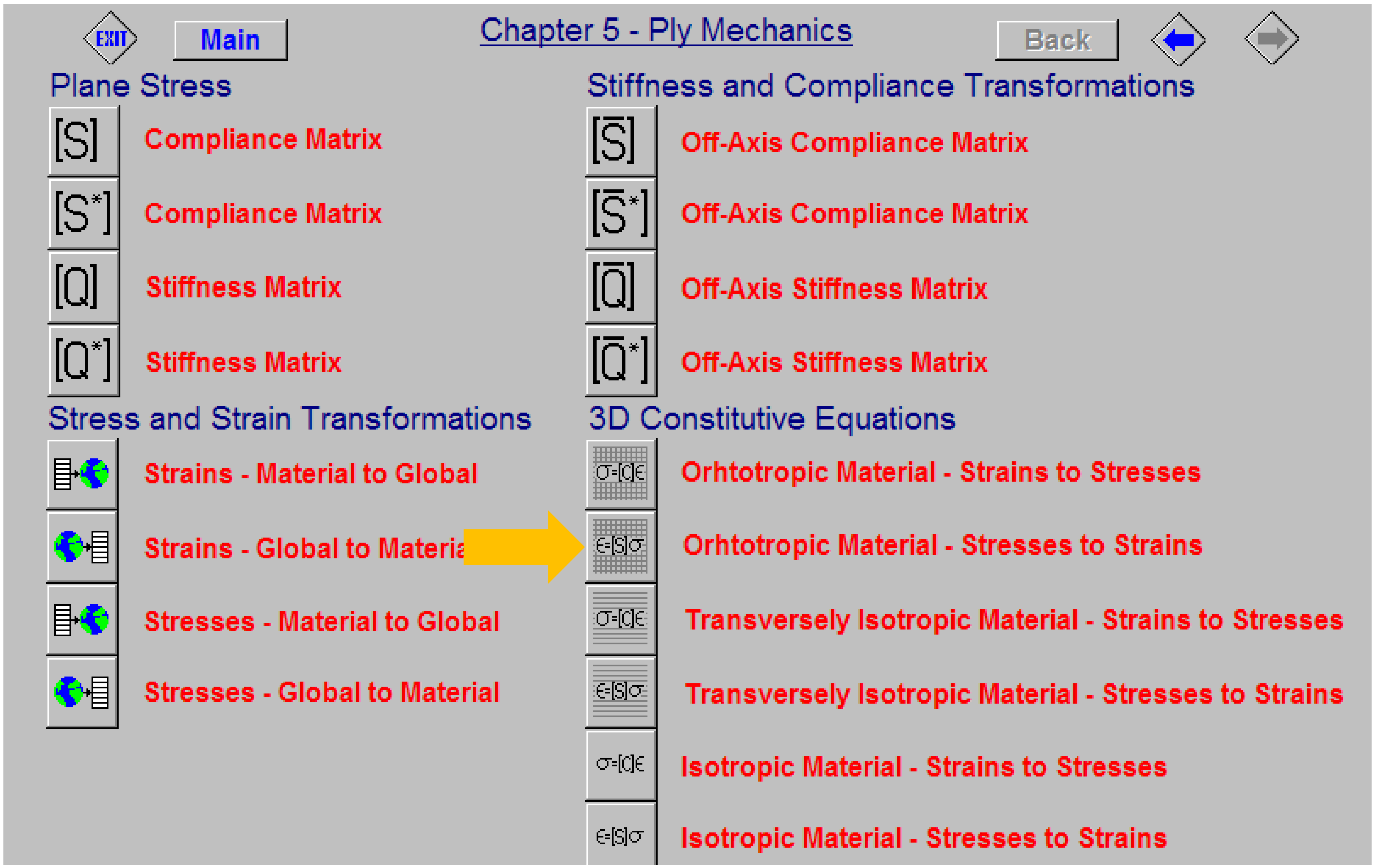
ε_y = 2.05733E+00 | G_{12} = 3.33156E+00 | σ_y = 5.43611E+01

γ_{xy} = 6.42909E-01 | ν_{12} = 0.30000 | σ_{xy} = 3.13561E+01

$\theta°$ = 45

$$\begin{Bmatrix} \sigma_x \\ \sigma_y \\ \sigma_{xy} \end{Bmatrix} = [\bar{Q}] \begin{Bmatrix} \varepsilon_x \\ \varepsilon_y \\ \gamma_{xy} \end{Bmatrix} = \begin{bmatrix} 1.76928E+01 & 9.00294E+00 & 6.83338E+00 \\ 9.00294E+00 & 1.76928E+01 & 6.83338E+00 \\ 6.83338E+00 & 6.83338E+00 & 1.08868E+01 \end{bmatrix} \begin{Bmatrix} \varepsilon_x \\ \varepsilon_y \\ \gamma_{xy} \end{Bmatrix}$$

Ply Mechanics: CADEC



Ply Mechanics: Stress-Strain Relationships

- **Stress-Strain Relationship:** $\sigma_{ij} = C_{ij}\varepsilon_{ij}$

- **With 3 planes → C_{ij} has 81 terms, but since:** $\sigma_{ij} = \sigma_{ji}$
  and: $\varepsilon_{ij} = \varepsilon_{ji}$ **only 36 terms**

- **Orthotropic material (2 planes of symmetry) reduces to 9 terms:**

$$\begin{Bmatrix} \sigma_{xx} \\ \sigma_{yy} \\ \sigma_{zz} \\ \sigma_{yz} \\ \sigma_{zx} \\ \sigma_{xy} \end{Bmatrix} = \begin{bmatrix} C_{11} & C_{12} & C_{13} & 0 & 0 & 0 \\ & C_{22} & C_{23} & 0 & 0 & 0 \\ & & C_{33} & 0 & 0 & 0 \\ & & & C_{44} & 0 & 0 \\ & \text{sym} & & & C_{55} & 0 \\ & & & & & C_{66} \end{bmatrix} \begin{Bmatrix} \varepsilon_{xx} \\ \varepsilon_{yy} \\ \varepsilon_{zz} \\ \varepsilon_{yz} \\ \varepsilon_{zx} \\ \varepsilon_{xy} \end{Bmatrix}$$

Ply Mechanics: Orthotropic Material

EXIT | Main | Chapter | Calculate Value | Back

Orthotropic Material - Stresses to Strains ☑ Activate Micromechanics

σ_1 =	0.1	E_1 =	3.78725E+01	ε_1 =	1.84831E-03
σ_2 =	0.1	$E_2 = E_3$ =	1.12711E+01	ε_2 =	8.08012E-03
σ_3 =	0	$G_{12} = G_{13}$ =	3.33156E+00	ε_3 =	-5.87904E-03
σ_4 =	0	$\nu_{12} = \nu_{13}$ =	0.30000	γ_4 =	0.00000E+00
σ_5 =	0	G_{23} =	3.03426E+00	γ_5 =	0.00000E+00
σ_6 =	0	ν_{23} =	0.57335	γ_6 =	0.00000E+00

$$\begin{Bmatrix} \varepsilon_1 \\ \varepsilon_2 \\ \varepsilon_3 \\ \gamma_4 \\ \gamma_5 \\ \gamma_6 \end{Bmatrix} = \begin{bmatrix} S_{11} & S_{12} & S_{13} & 0 & 0 & 0 \\ S_{12} & S_{22} & S_{23} & 0 & 0 & 0 \\ S_{13} & S_{23} & S_{33} & 0 & 0 & 0 \\ 0 & 0 & 0 & S_{44} & 0 & 0 \\ 0 & 0 & 0 & 0 & S_{55} & 0 \\ 0 & 0 & 0 & 0 & 0 & S_{66} \end{bmatrix} \begin{Bmatrix} \sigma_1 \\ \sigma_2 \\ \sigma_3 \\ \sigma_4 \\ \sigma_5 \\ \sigma_6 \end{Bmatrix}$$

Macromechanics

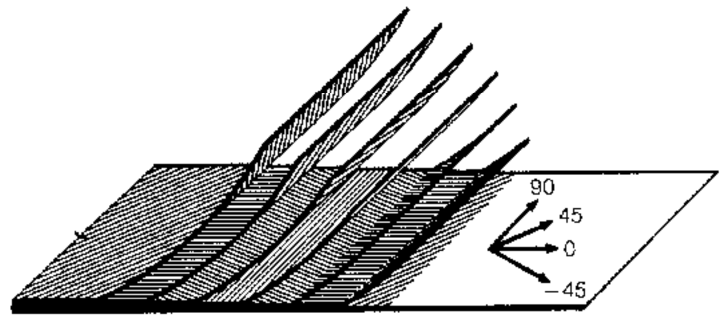
- **What if there are multiple lamina at differing angles?**

- **CADEC uses Micromechanics and Ply mechanics to determine:**
  - Stiffness and Compliance Equations
  - Laminate Moduli
  - Global and Material Stresses and Strains
  - Strains and Curvatures
  - Thermal and Hygroscopic loads
  - For both Intact and Degraded materials

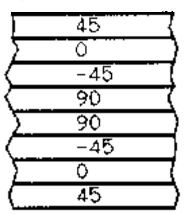
- **Assumes:**
  - Plane sections remain plane
  - Symmetry about a neutral surface
  - No shear coupling
  - Perfect bonding

Shorthand Laminate Orientation Code

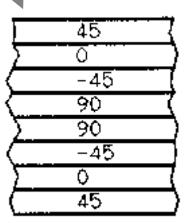
Tapes or Undirectional Tapes



[45/0/-45/90_2/-45/0/45



[45/0/-45/90]$_s$



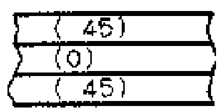
Tapes or undirectional tapes

- Each lamina is labeled by its ply orientation.
- Laminae are listed in sequence with the first number representing the lamina to which the arrow is pointing.
- Individual adjacent laminae are separated by a slash if their angles differ.
- Adjacent laminae of the same angle are depicted by a numerical subscript indicating the total number of laminae which are laid up in sequence at that angle.
- Each complete laminate is enclosed by brackets.
- When the laminate is symmetrical and has an even number on each side of the plane of symmetry (known as the midplane) the code may be shortened by listing only the angles from the arrow side to the midplane. A subscript “S” is used to indicate that the code for only one half of the laminate is shown.

Shorthand Laminate Orientation Code

Fabrics and Tapes and Fabrics

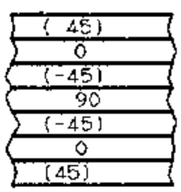
[(45)/(0)/(45)]



Midplane

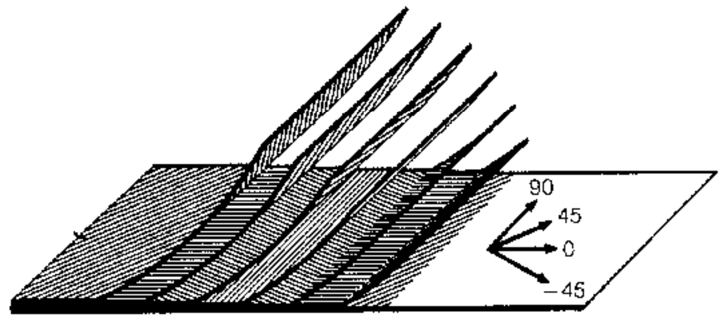
Fabrics

[(45)/0(-45)/90]



Midplane

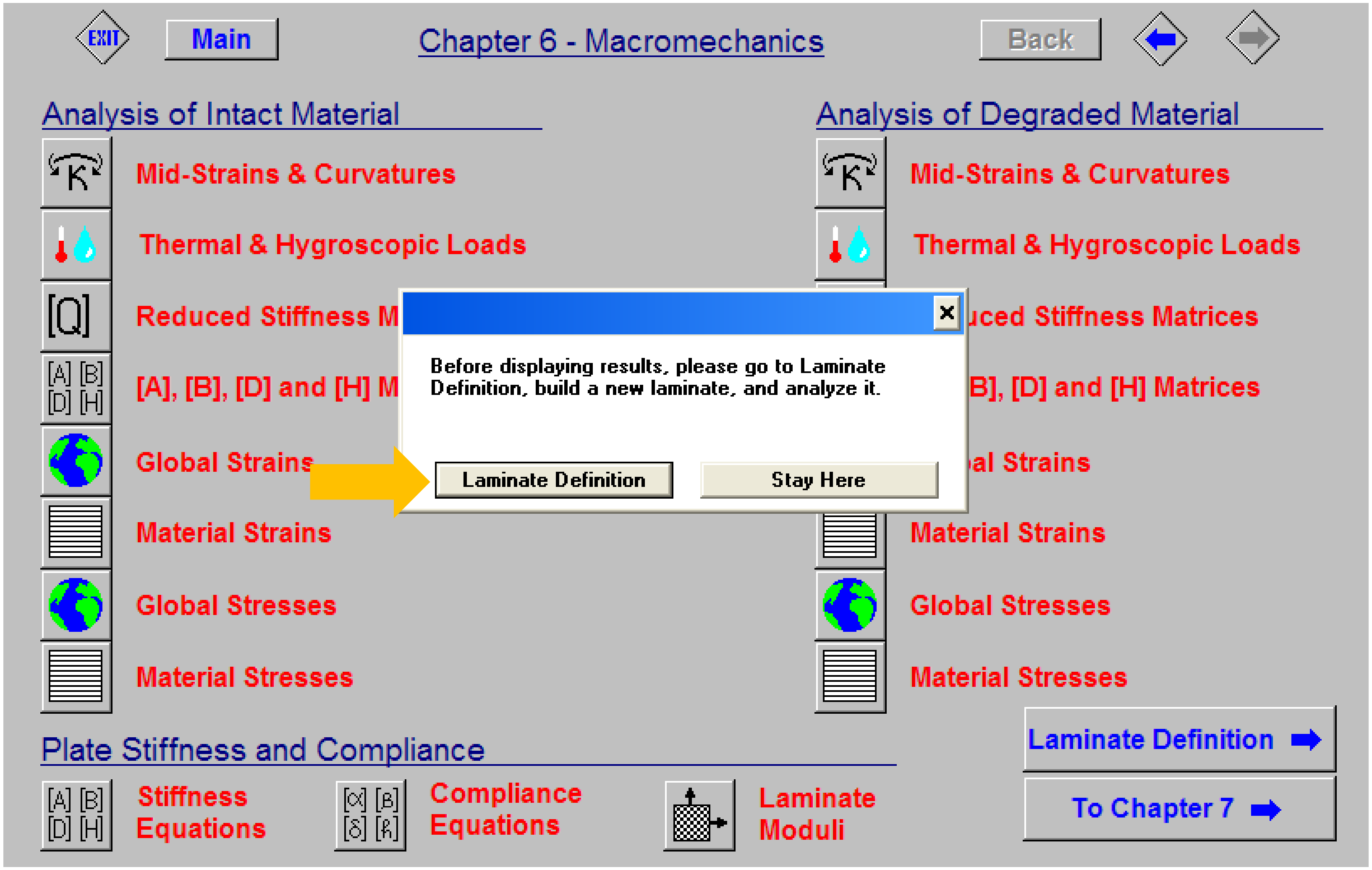
Tapes & Fabrics



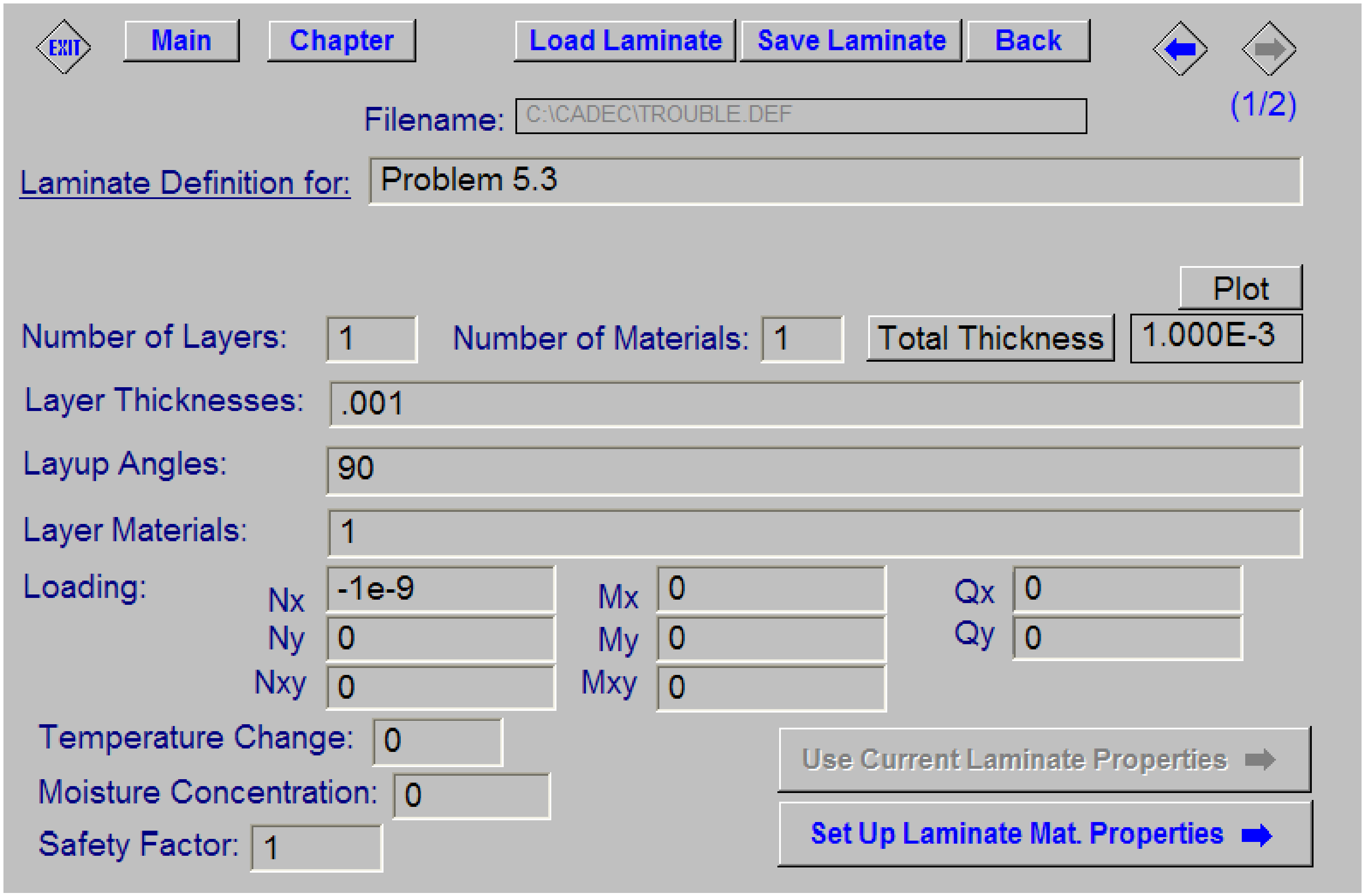
- When plies of fabric are used in a laminate. The angle of the fabric warp is used as the ply direction angle. The fabric angle is enclosed in parentheses to identify the ply as a fabric ply.
- When the laminate is composed of both fabric and tape plies (a hybrid laminate). The parentheses around the fabric plies will distinguish the fabric plies from the tape plies.
- When the laminate is symmetrical and has an odd number of plies, the center ply is overlined to indicate that it is the midplane.

Macromechanics: CADEC

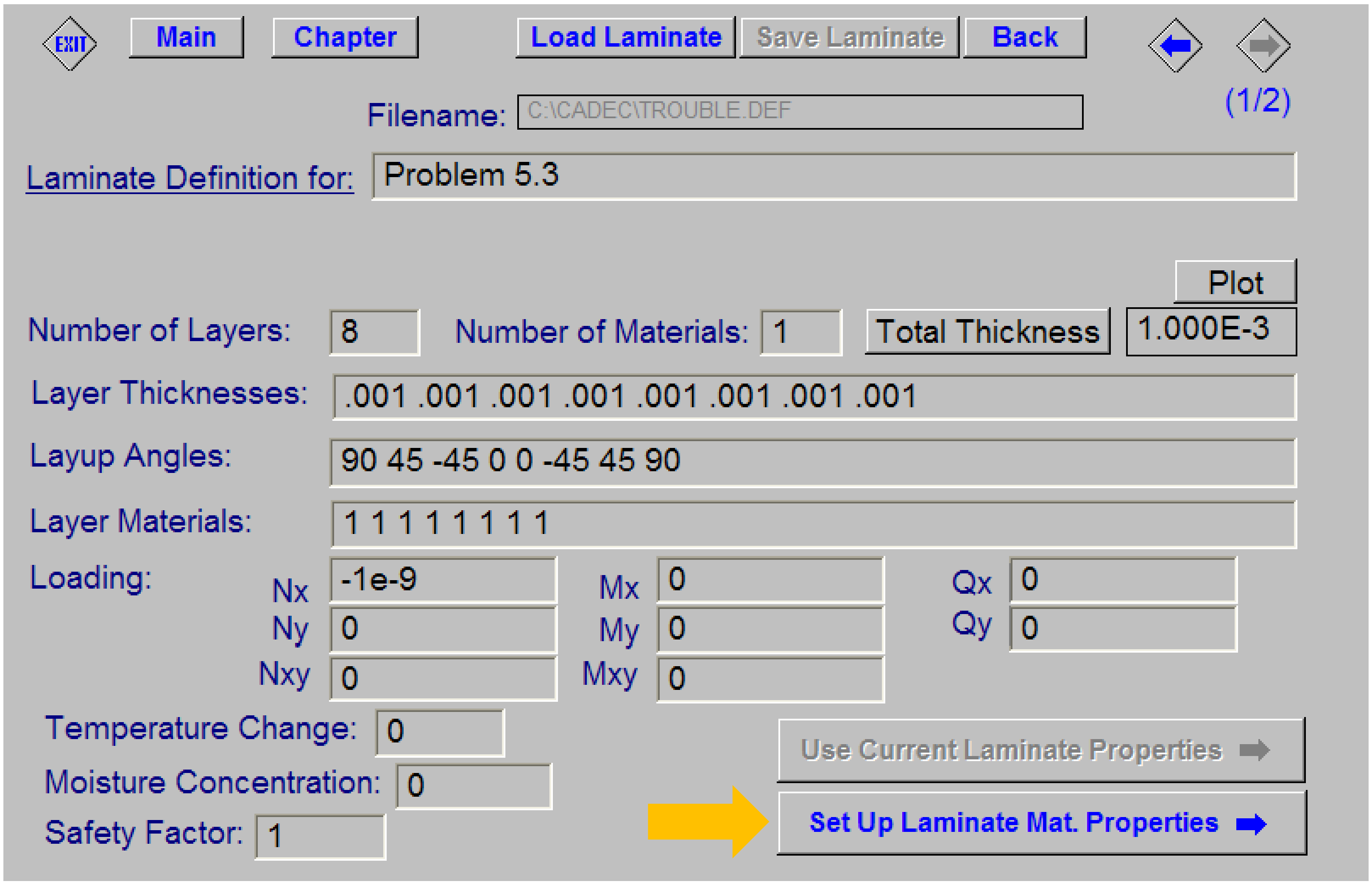
Macromechanics: CADEC



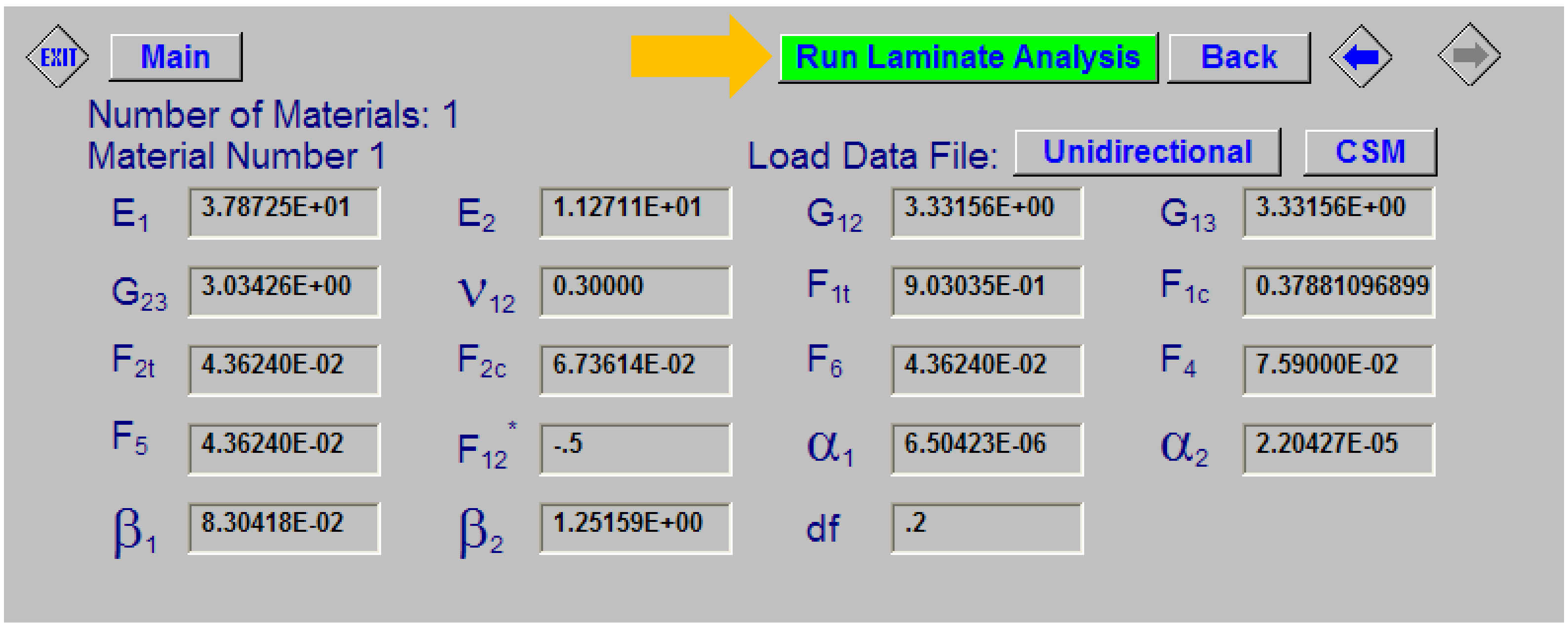
Macromechanics: Defining Laminate



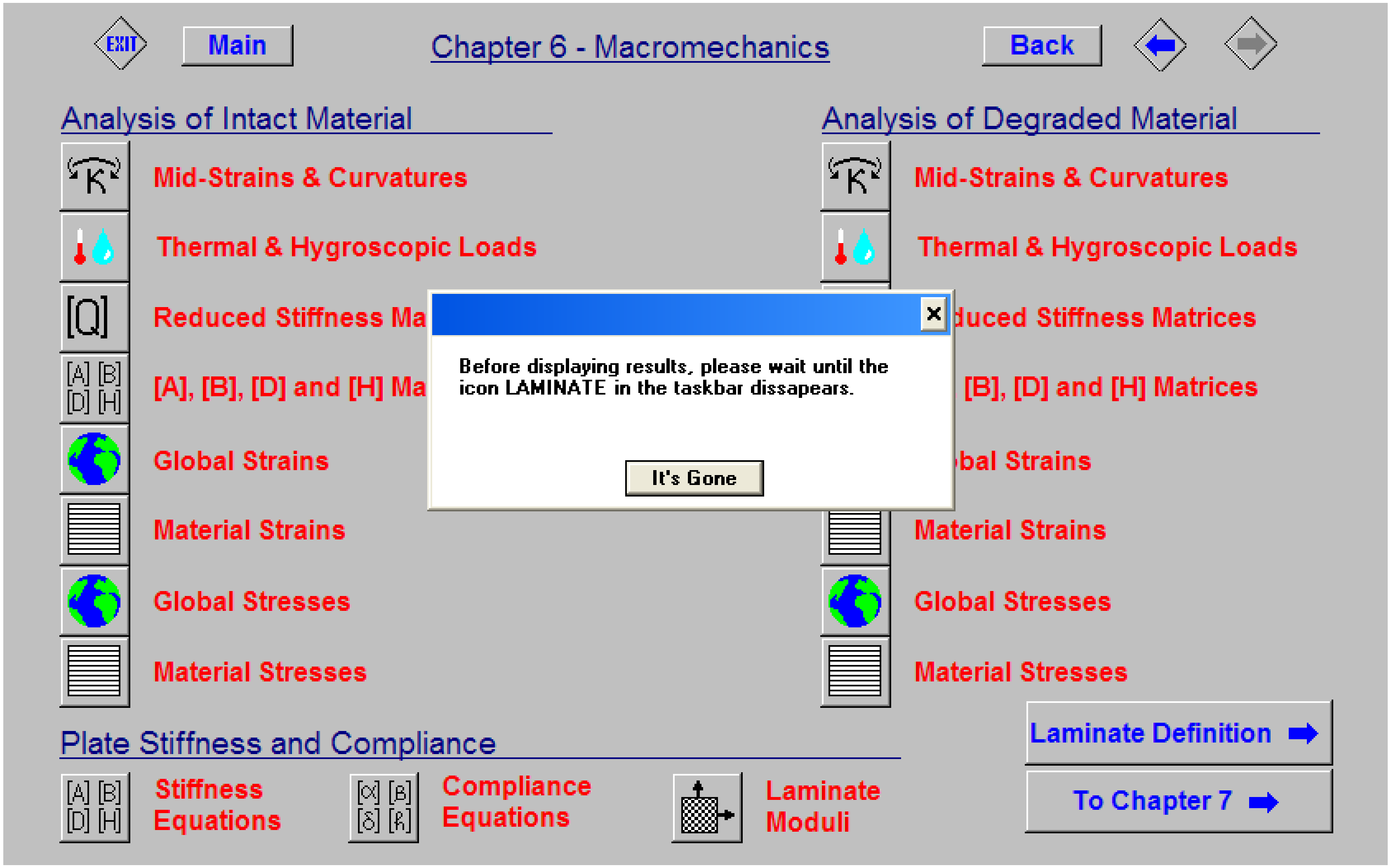
Macromechanics: Defining Laminate



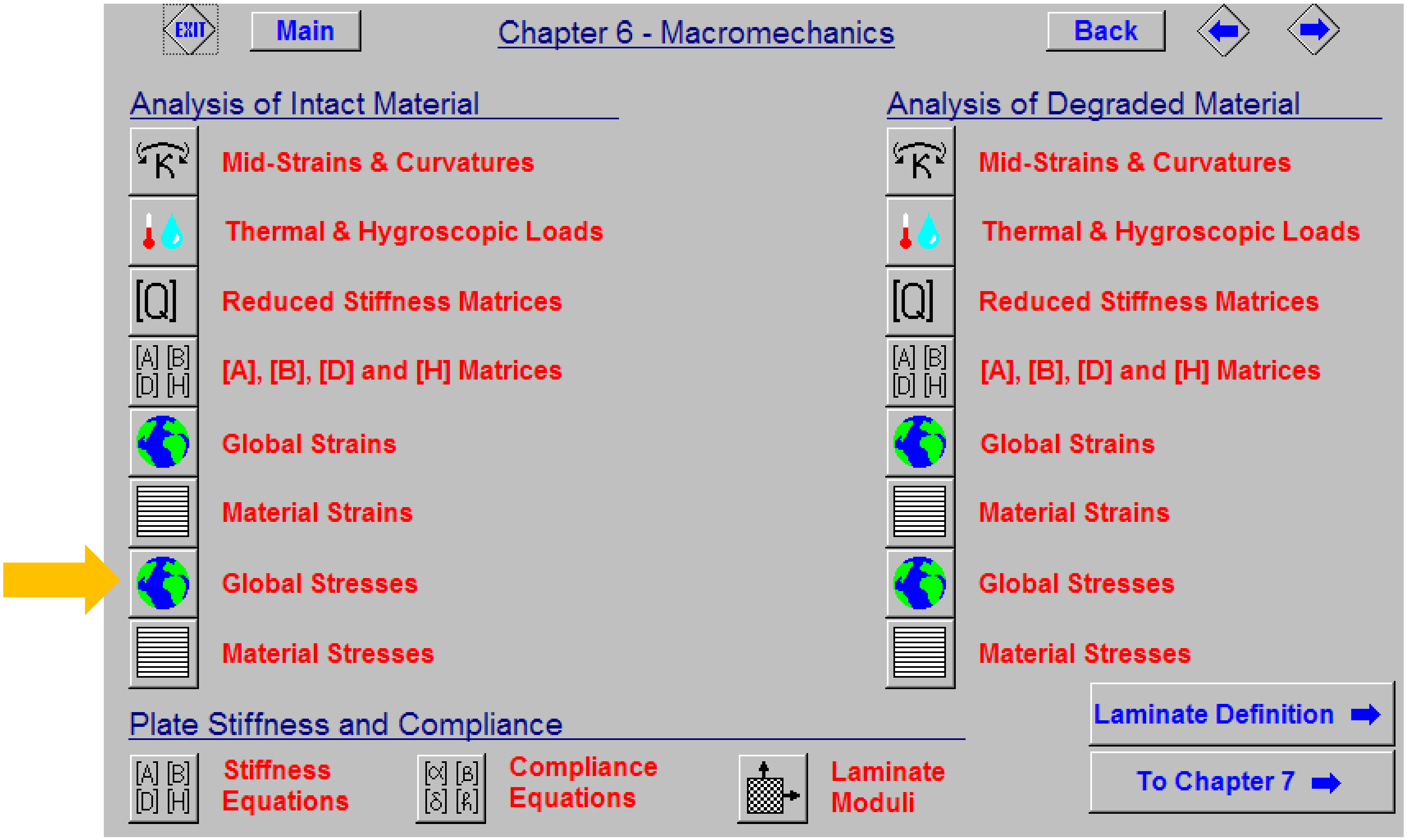
Macromechanics: Material Properties



Macromechanics: CADEC Quirkiness



Macromechanics: Review Outputs



Macromechanics: Global Stresses

EXIT | Main | Chapter | Back

Global Stresses - Intact Material

Hygrothermal Stress (Print)

Ply	Face	σ_x	σ_y	σ_{xy}	σ_{yz}	σ_{xz}
8	TOP	.000	.000	.000	.000	.000
8	BOT	.000	.000	.000	.000	.000
7	TOP	.000	.000	.000	.000	.000
7	BOT	.000	.000	.000	.000	.000
6	TOP	.000	.000	.000	.000	.000

Mechanical Stress (Print)

Ply	Face	σ_x	σ_y	σ_{xy}	σ_{yz}	σ_{xz}
8	TOP	-.684E-07	.635E-07	.383E-23	.000	.000
8	BOT	-.684E-07	.635E-07	.387E-23	.000	.000
7	TOP	-.918E-07	-.332E-07	-.297E-07	.000	.000
7	BOT	-.918E-07	-.332E-07	-.297E-07	.000	.000
6	TOP	-.918E-07	-.332E-07	.297E-07	.000	.000

Total Stress (Print)

Ply	Face	σ_x	σ_y	σ_{xy}	σ_{yz}	σ_{xz}
8	TOP	-.684E-07	.635E-07	.383E-23	.000	.000
8	BOT	-.684E-07	.635E-07	.387E-23	.000	.000
7	TOP	-.918E-07	-.332E-07	-.297E-07	.000	.000
7	BOT	-.918E-07	-.332E-07	-.297E-07	.000	.000
6	TOP	-.918E-07	-.332E-07	.297E-07	.000	.000

Macromechanics: ABD Matrices

- **Stiffness of composite where:**
  - [*A*] = in-plane stiffness.
  - [*D*] = bending stiffness.
  - [*B*] relates in-plane strains to bending moments and curvatures to in-plane forces—bending-extension coupling.
  - [*H*] relates transverse shear strains to transverse forces.

$$\begin{Bmatrix} N_x \\ N_y \\ N_{xy} \\ M_x \\ M_y \\ M_{xy} \end{Bmatrix} = \begin{bmatrix} A_{11} & A_{12} & A_{16} & B_{11} & B_{12} & B_{16} \\ A_{12} & A_{22} & A_{26} & B_{12} & B_{22} & B_{26} \\ A_{16} & A_{26} & A_{66} & B_{16} & B_{26} & B_{66} \\ B_{11} & B_{12} & B_{16} & D_{11} & D_{12} & D_{16} \\ B_{12} & B_{22} & B_{26} & D_{12} & D_{22} & D_{26} \\ B_{16} & B_{26} & B_{66} & D_{16} & D_{26} & D_{66} \end{bmatrix} \begin{Bmatrix} \epsilon_x^0 \\ \epsilon_y^0 \\ \gamma_{xy}^0 \\ \kappa_x \\ \kappa_y \\ \kappa_{xy} \end{Bmatrix} \quad (6.15)$$

$$\begin{Bmatrix} V_y \\ V_x \end{Bmatrix} = \begin{bmatrix} H_{44} & H_{45} \\ H_{45} & H_{55} \end{bmatrix} \begin{Bmatrix} \gamma_{yz} \\ \gamma_{xz} \end{Bmatrix}$$

with

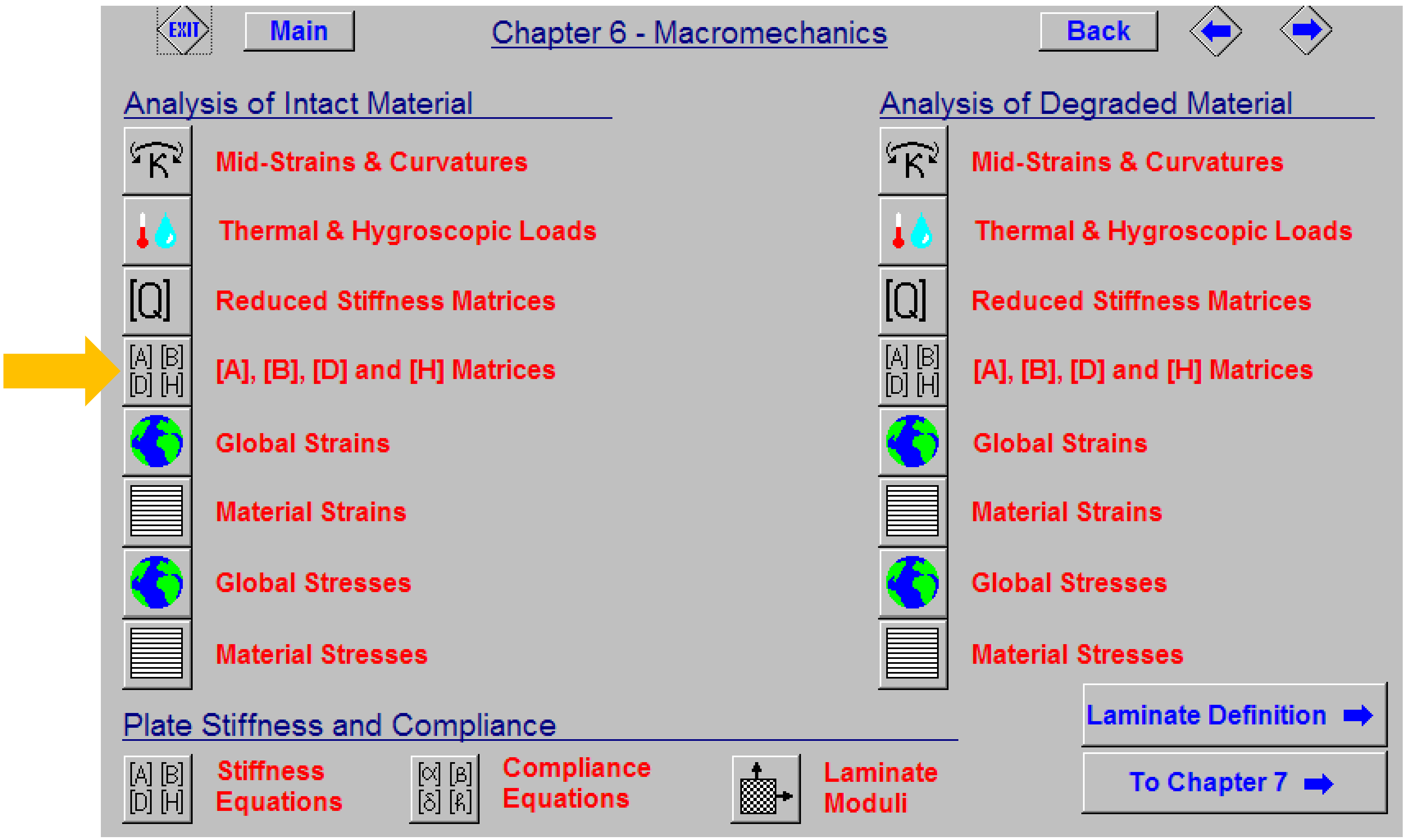
$$A_{ij} = \sum_{k=1}^{N} (\overline{Q}_{ij})_k (z_k - z_{k-1}) = \sum_{k=1}^{N} (\overline{Q}_{ij})_k t_k \qquad i, j = 1, 2, 6$$

$$B_{ij} = \frac{1}{2} \sum_{k=1}^{N} (\overline{Q}_{ij})_k \left(z_k^2 - z_{k-1}^2\right) = \sum_{k=1}^{N} (\overline{Q}_{ij})_k t_k \bar{z}_k \qquad i, j = 1, 2, 6$$

$$D_{ij} = \frac{1}{3} \sum_{k=1}^{N} (\overline{Q}_{ij})_k \left(z_k^3 - z_{k-1}^3\right) = \sum_{k=1}^{N} (\overline{Q}_{ij})_k \left(t_k \bar{z}_k^2 + \frac{t_k^3}{12}\right) \qquad i, j = 1, 2, 6$$

$$H_{ij} = \frac{5}{4} \sum_{k=1}^{N} (\overline{Q}_{ij})_k \left[t_k - \frac{4}{t^2}\left(t_k \bar{z}_k^2 + \frac{t_k^3}{12}\right)\right] \qquad i, j = 4, 5 \quad (6.16)$$

Macromechanics: ABD Matrices



Macromechanics: ABD Matrices

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Analysis for Intact Material

[A] =

.171763	.580162E-01	-.884037E-19
.580162E-01	.171763	.349677E-17
-.884037E-19	.349677E-17	.568736E-01

[B] =

.677626E-20	.169407E-20	-.309413E-21
-.169407E-20	.000000	-.609111E-21
-.309413E-21	-.609111E-21	.169407E-20

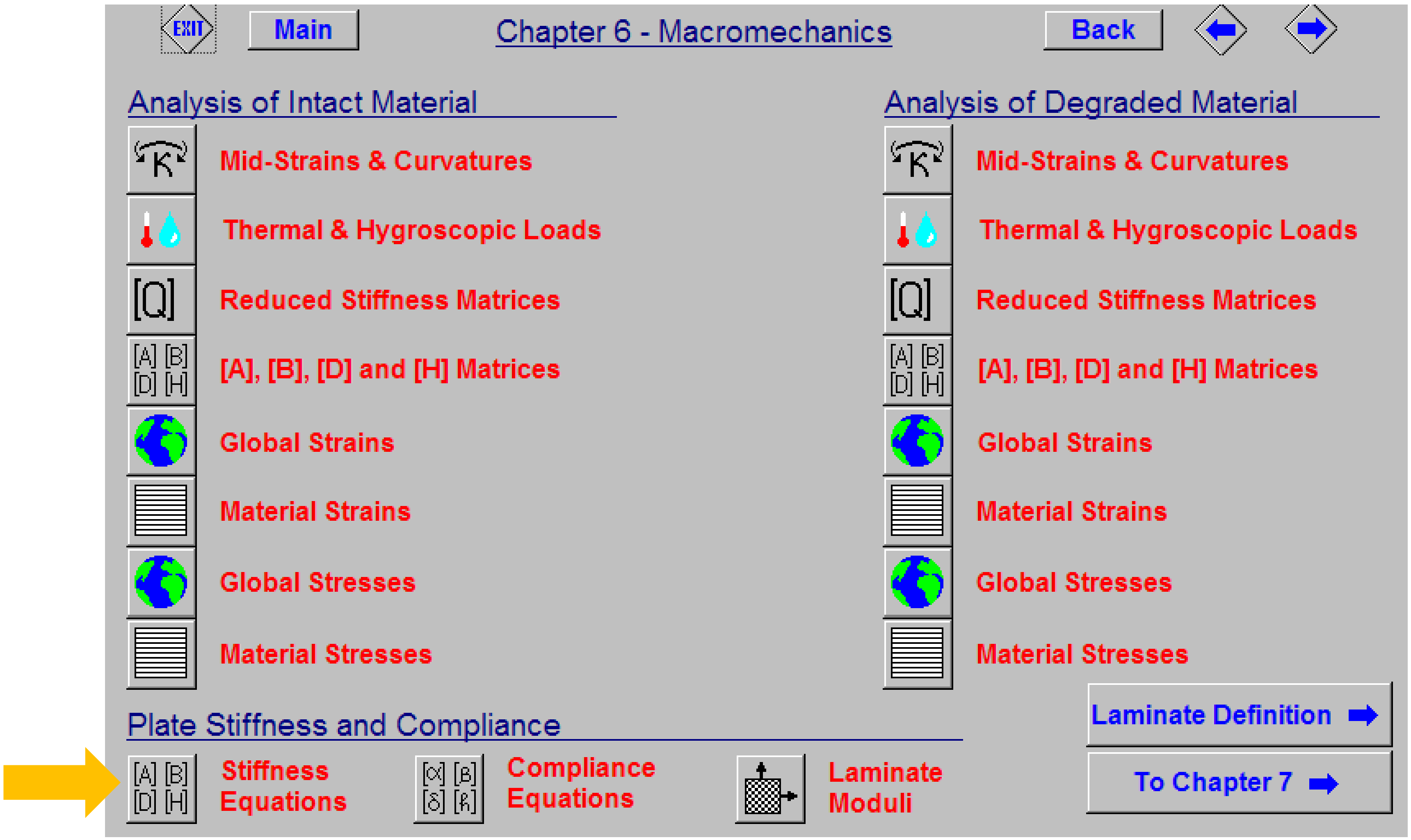
[D] =

.618290E-06	.279199E-06	.546670E-07
.279199E-06	.127429E-05	.546670E-07
.546670E-07	.546670E-07	.273105E-06

[H] =

.209407E-01	-.929063E-04
-.929063E-04	.214981E-01

Macromechanics: Stiffness Equations



Macromechanics: Stiffness Equations

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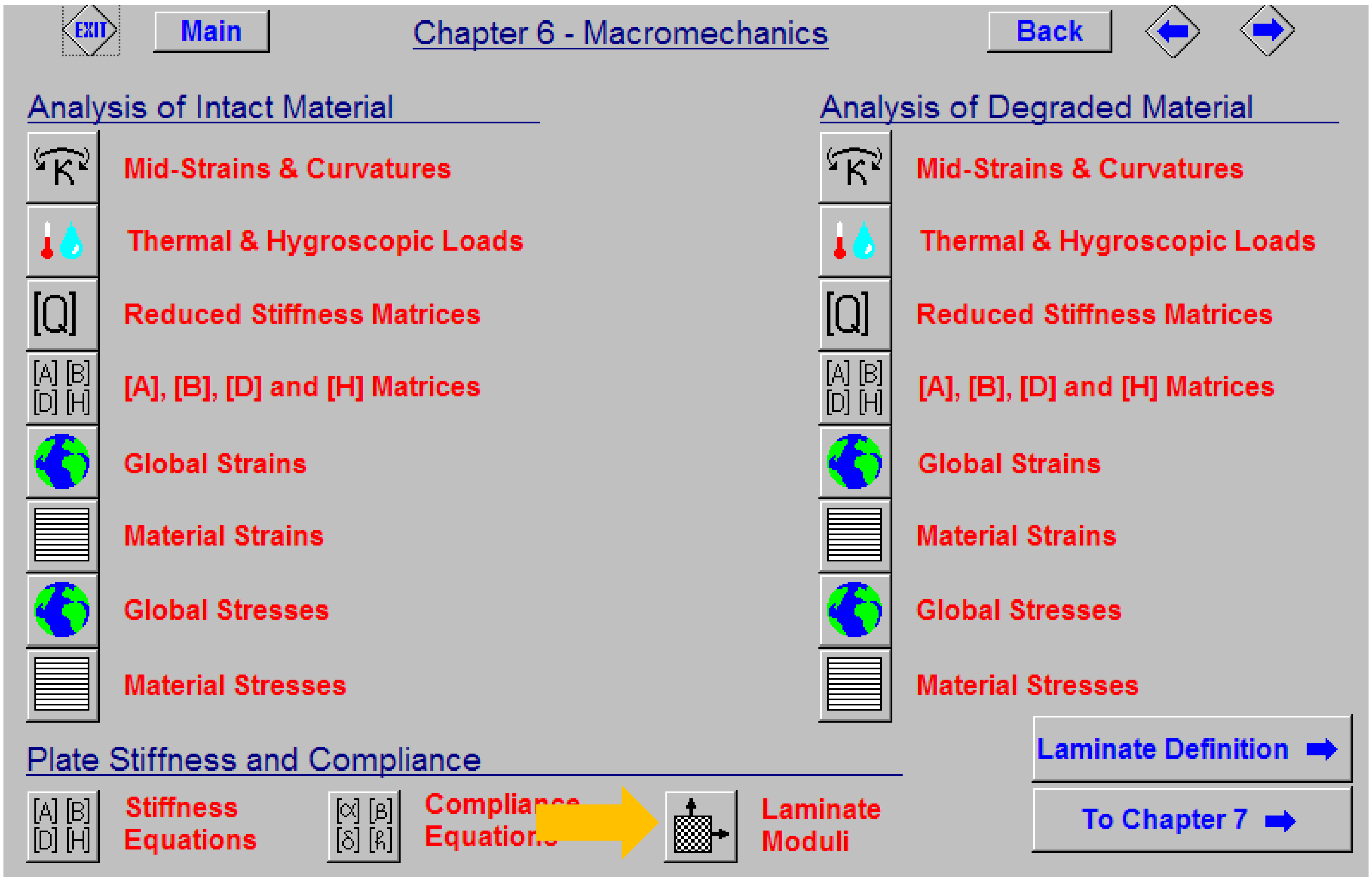
Plate Stiffness Equations

		=								
N_x =	-9.99687E-10		1.72E-01	5.80E-02	-8.84E-20	6.78E-21	1.69E-21	-3.09E-22	ε^o_x =	-.657E-08
N_y =	1.47426E-13		5.80E-02	1.72E-01	3.50E-18	-1.69E-21	0.00E+00	-6.09E-22	ε^o_y =	.222E-08
N_{xy} =	-1.67775E-29		-8.84E-20	3.50E-18	5.69E-02	-3.09E-22	-6.09E-22	1.69E-21	γ^o_{xy} =	-.147E-24
M_x =	3.06089E-32		6.78E-21	1.69E-21	-3.09E-22	6.18E-07	2.79E-07	5.47E-08	κ_x =	.786E-22
M_y =	-2.22681E-29		-1.69E-21	0.00E+00	-6.09E-22	2.79E-07	1.27E-06	5.47E-08	κ_y =	-.254E-22
M_{xy} =	1.12259E-32		-3.09E-22	-6.09E-22	1.69E-21	5.47E-08	5.47E-08	2.73E-07	κ_{xy} =	-.131E-22

		=				
V_y =	0.00000E+00		2.09E-02	-9.29E-05	γ_{yz} =	0.00000E+00
V_x =	0.00000E+00		-9.29E-05	2.15E-02	γ_{xz} =	0.00000E+00

Matrices [ABD] and [H] are based on the current intact material, or the last matrix file loaded.

Macromechanics: Laminate Moduli



Macromechanics: Laminate Moduli

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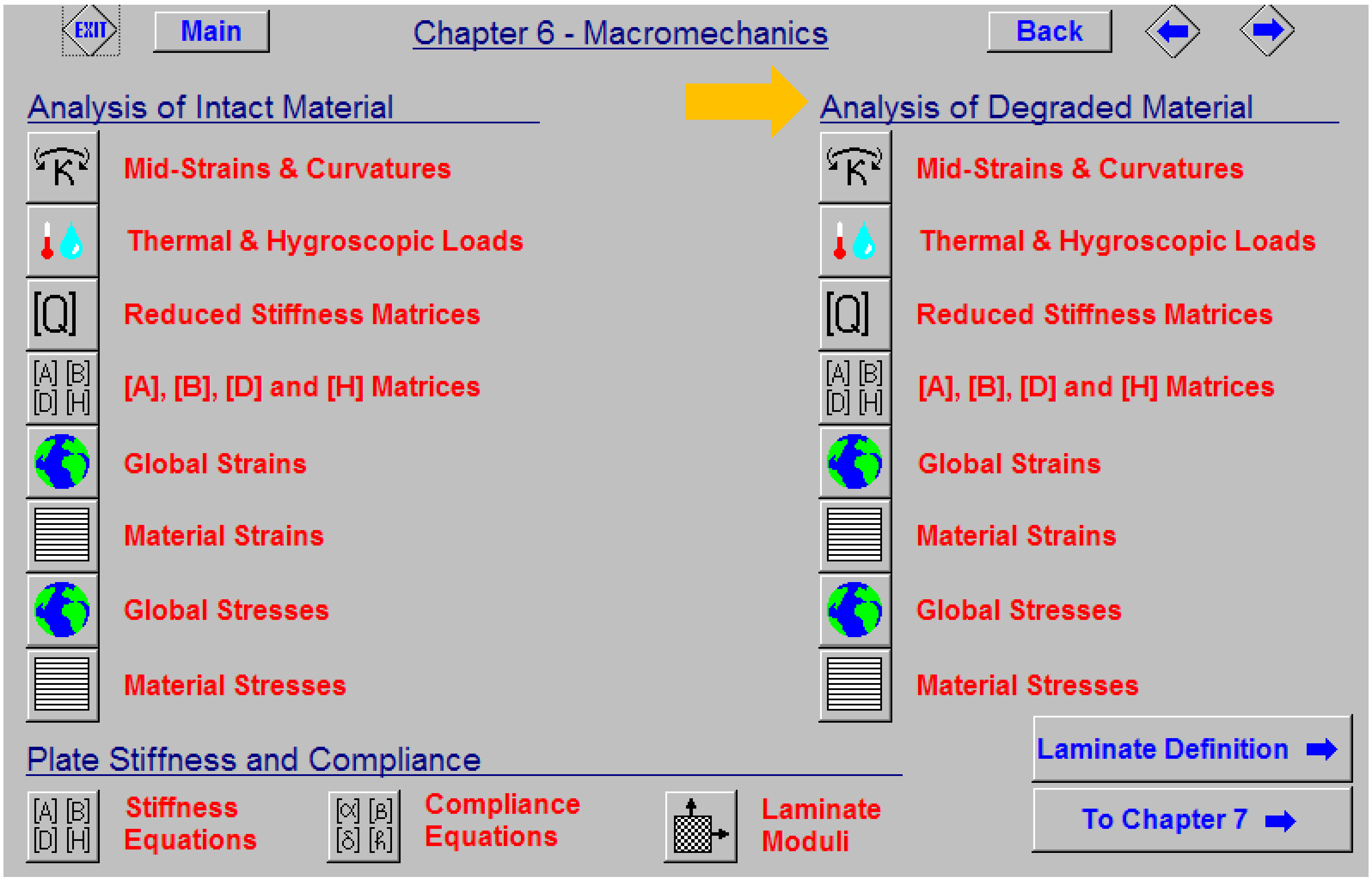
LaminateModuli:

Inplane		Bending	
E_x =	2.278E+1	$E^b{}_x$ =	3.243E+1
E_y =	1.779E+1	$E^b{}_y$ =	1.567E+1
G_{xy} =	6.329E+0	$G^b{}_{xy}$ =	3.774E+0
ν_{xy} =	0.335	$\nu^b{}_{xy}$ =	0.244

Macromechanics: Degraded Material



Macromechanics: Degraded Material

- **What is a degraded material?**

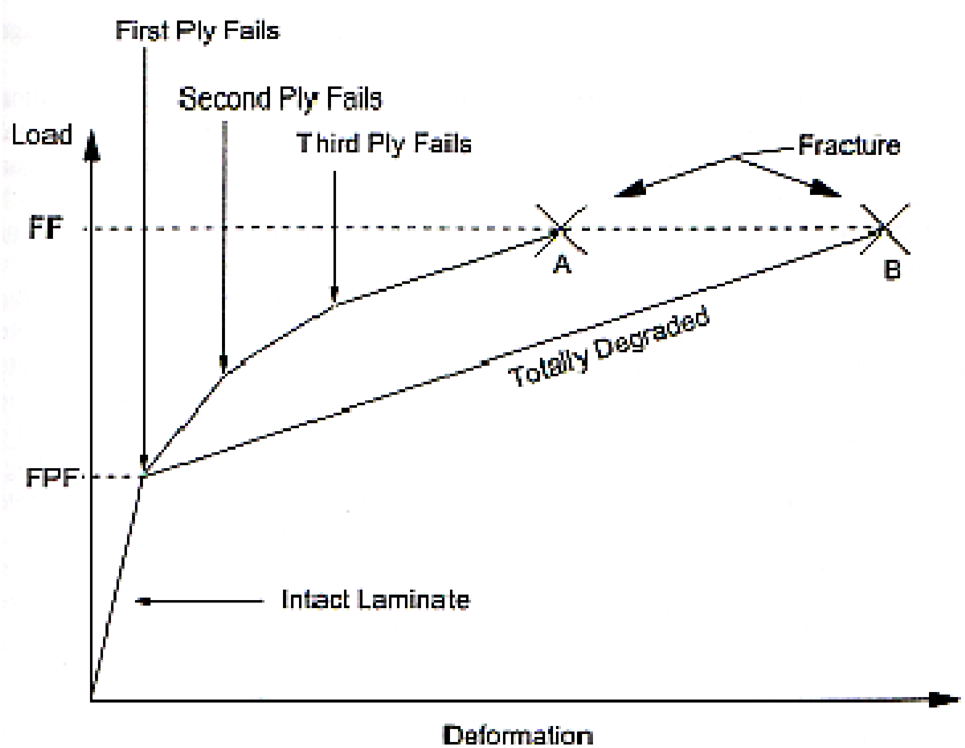
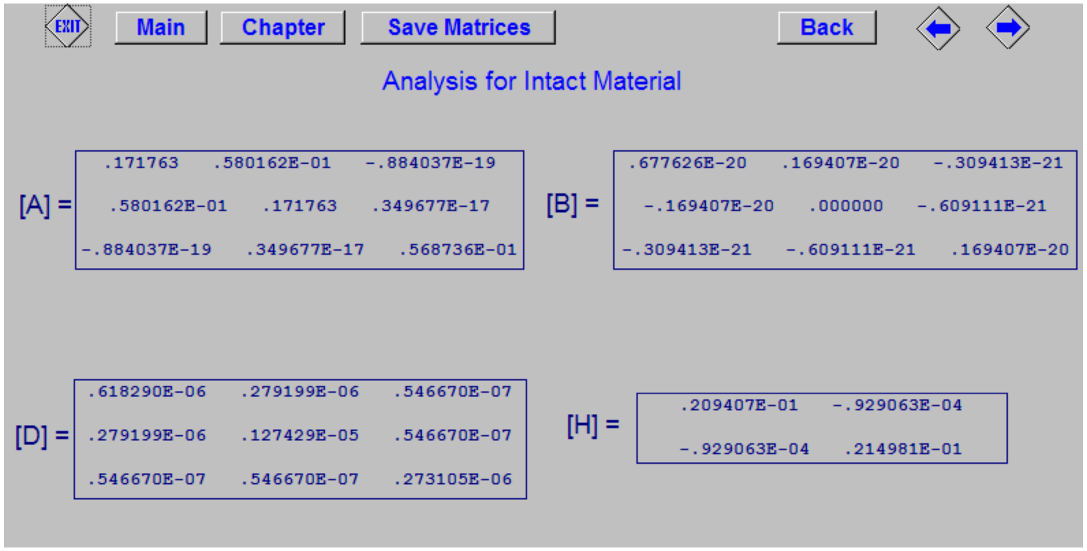
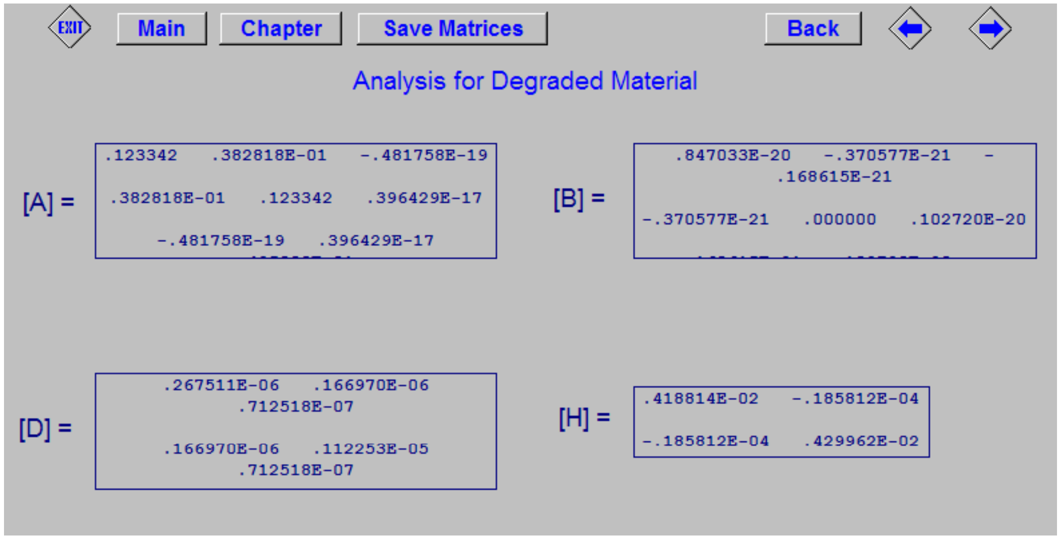


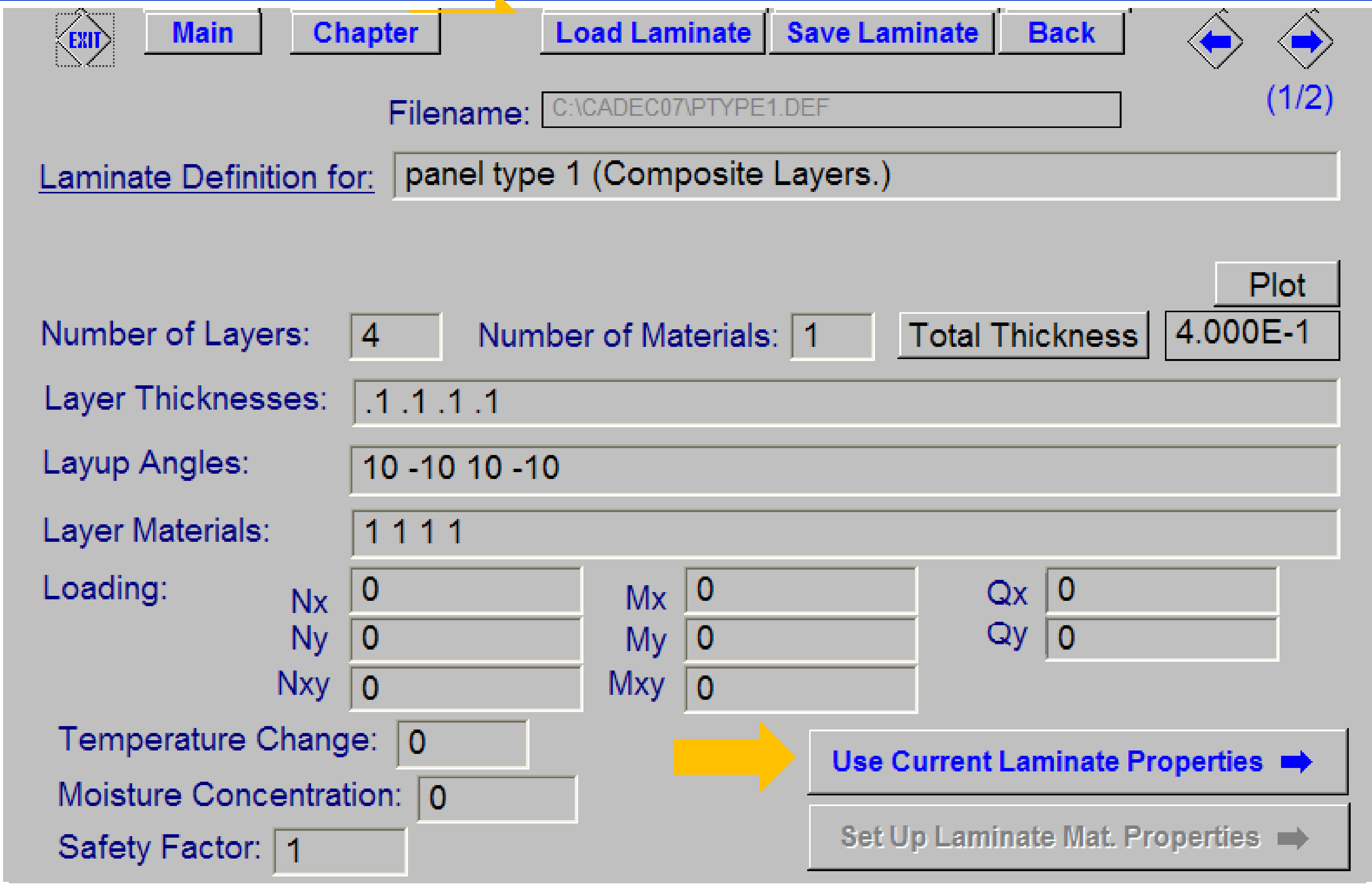
Figure 7.7 Determination of FF load by an incremental and two-step approach.

Macromechanics: ABD Comparison





Macromechanics: CADEC Alt Methods



Failure Theories

- **Many failure criteria, most popular:**
  - Maximum stress criterion
  - Maximum strain criterion
  - Tsai-Hill failure criterion
  - Tsai-Wu failure criterion

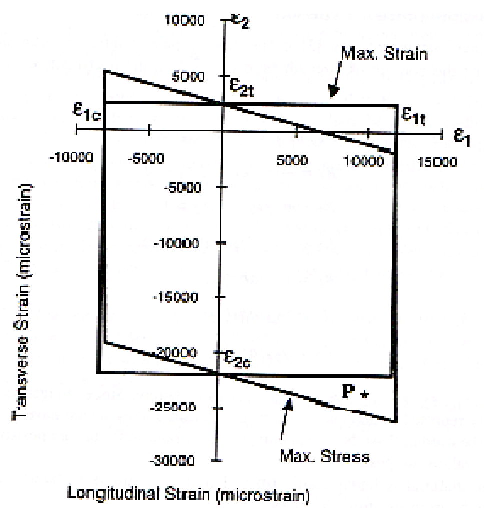


Figure 7.3 Failure envelopes using the maximum strain and maximum stress criteria in strain space $\varepsilon_1 - \varepsilon_2$ for carbon–epoxy.

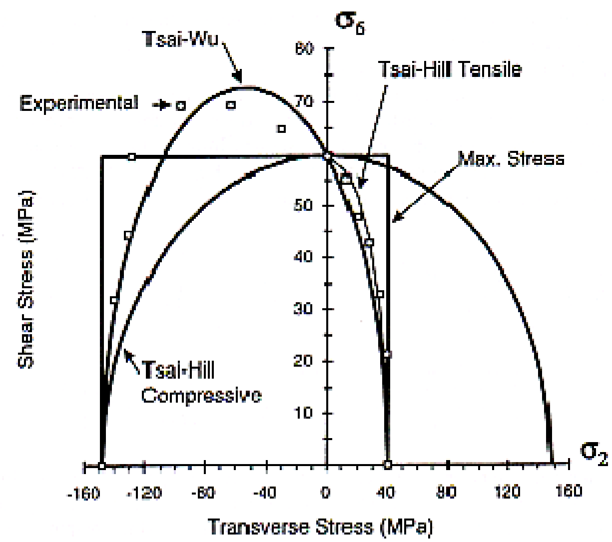
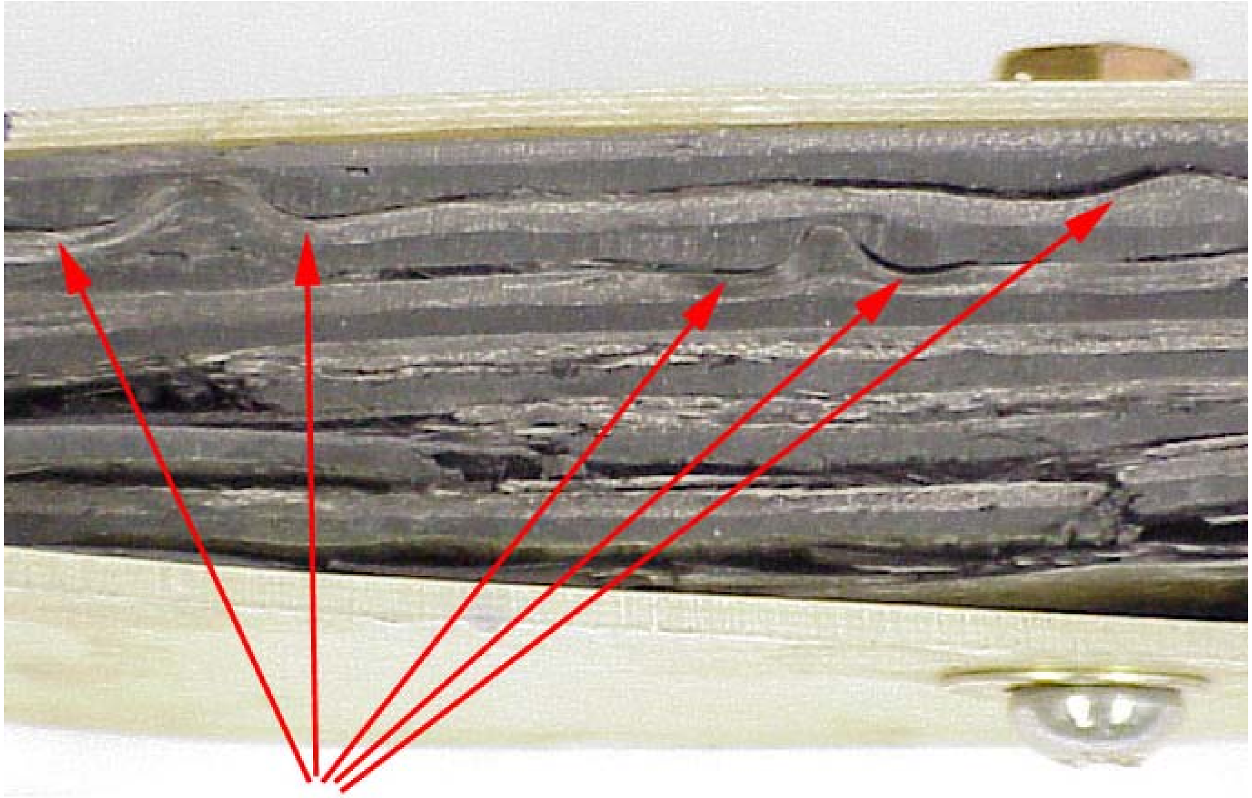


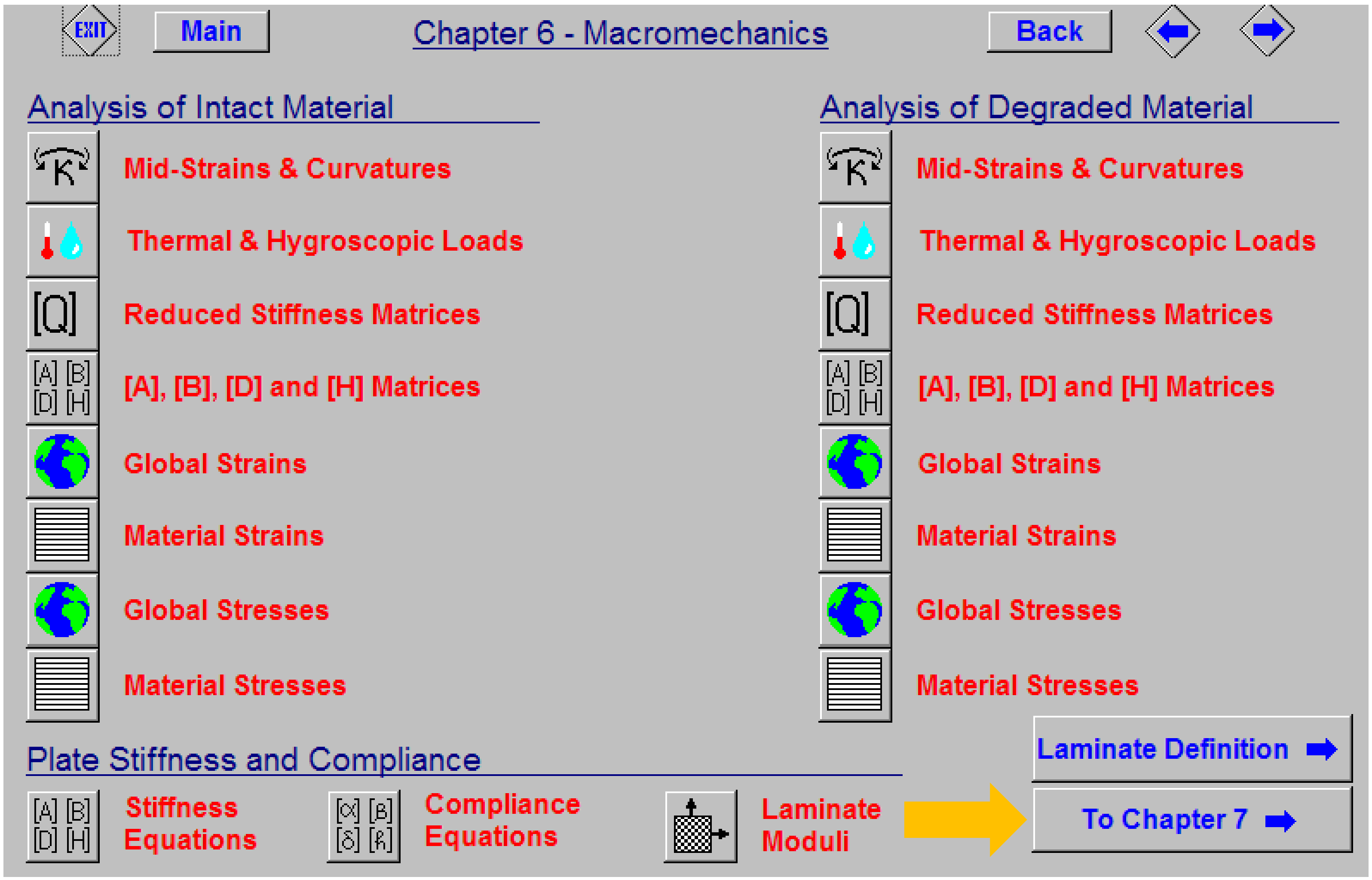
Figure 7.5 Various failure envelopes in $\sigma_6 - \sigma_2$ space compared with experimental data.

Not Just An Academic Exercise

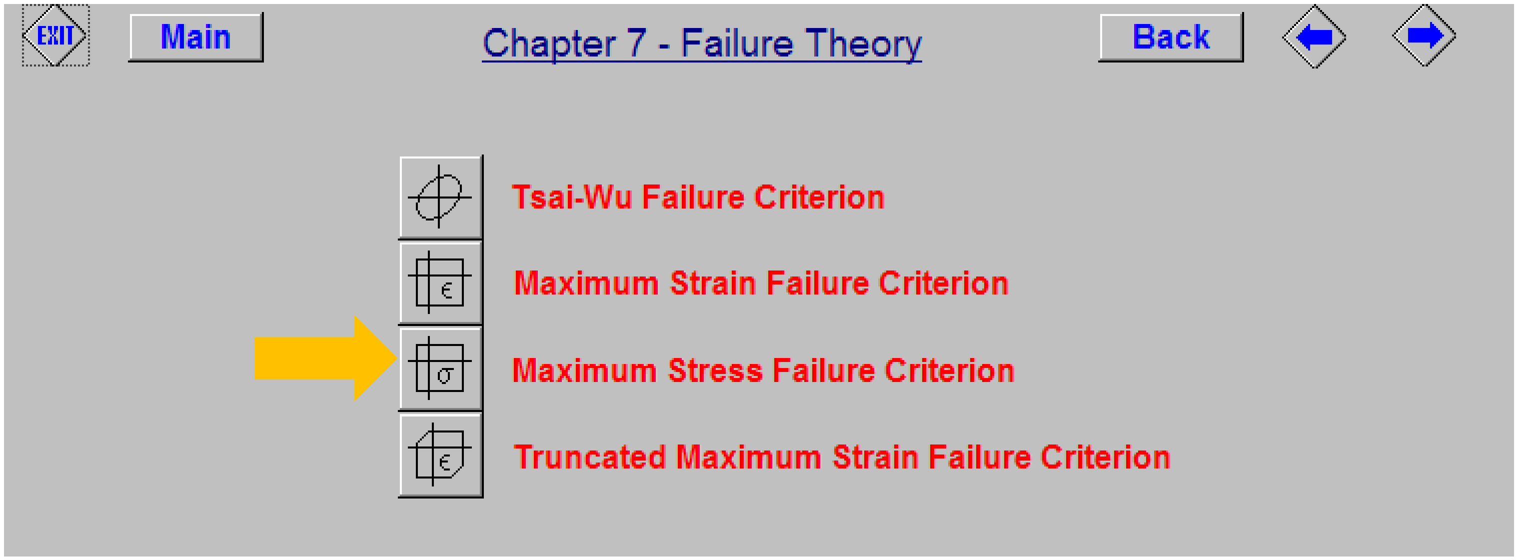


Consequence of Misalignment in Large, Composite Structure

Failure Theories: CADEC



Failure Theories: CADEC



Failure Theories: Max Stress Criterion

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Maximum Stress Failure Criterion

		Intact Laminate		Degraded Laminate	
Ply	Angle	Rint-Top	Rint-Bot	Rdeg-Top	Rdeg-Bot
4	-10.	1000. (1)	1000. (1)	1000. (1)	1000. (1)
3	10.	1000. (1)	1000. (1)	1000. (1)	1000. (1)
2	-10.	1000. (1)	1000. (1)	1000. (1)	1000. (1)
1	10.	1000. (1)	1000. (1)	1000. (1)	1000. (1)

Print

FPF 1000. FF 1000.

Failure Theories: Tsai-Wu Criterion

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Tsai-Wu Failure Criterion

		Intact Laminate		Degraded Laminate	
Ply	Angle	Rint-Top	Rint-Bot	Rdeg-Top	Rdeg-Bot
4	-10.	.000	.000	.000	.000
3	10.	.000	.000	.000	.000
2	-10.	.000	.000	.000	.000
1	10.	.000	.000	.000	.000

Print FPF .000 FF .000

CADEC Demo

Concluding Remarks

- **Composite design fairly simple**
  - Assumptions lead to simplified analysis
  - Idealized
  - Real-world?
- **CADEC**
  - Begin with component properties
  - Micromechanic, Ply and Macromechanic analysis
  - Apply loads and match against failure criteria
  - Simple structures (Not covered)
  - Software options: COMPRO, MSExcel, Matlab, MathCAD, etc.
- **Composites still require significant analysis and physical testing**
- **Parts/Structures are only as good as the manufacturing**
  - “You can never make good parts with bad materials, but you can easily make bad parts with good materials!”