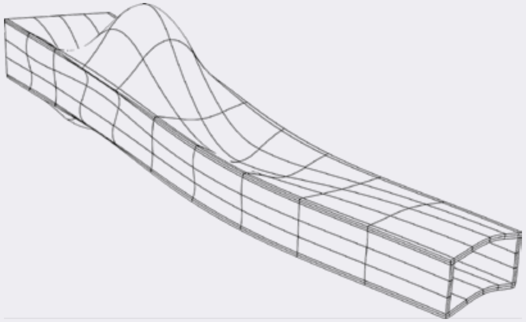
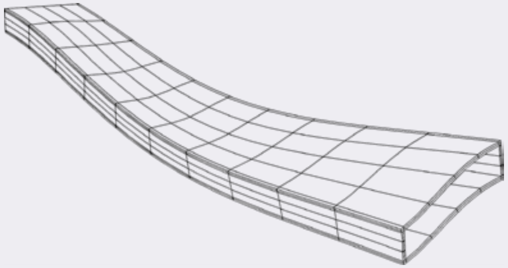
POLITECNICO
DI TORINO
Department of MEchanical
and AeroSpace Engineering

Low Fidelity and High Fidelity Structural Models
for
Metallic and Composites Aircraft Structures





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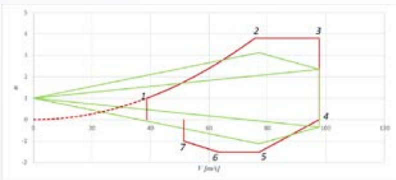
Torino 11-12 October 2017

Summary

TIVANO: a classical FEM analysis.

Loads evaluation

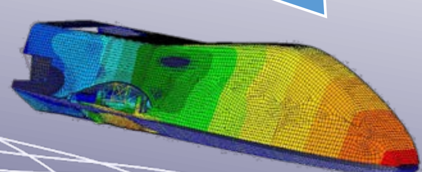


Fuselage CAD analyses.

Modelling using Commercial Softwares

FE structural analyses.

Results & Reporting



Advanced FEM Models

1-d High Fidelity model

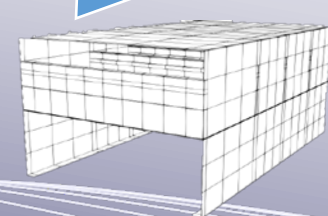
Tapered shape

Composite material

Multi-component structures

Damage & Tailoring analyses

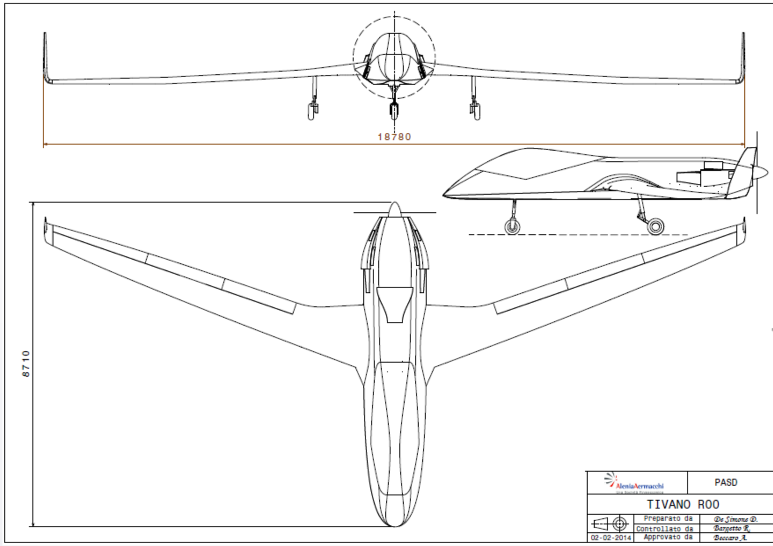
Tow Angle Placement

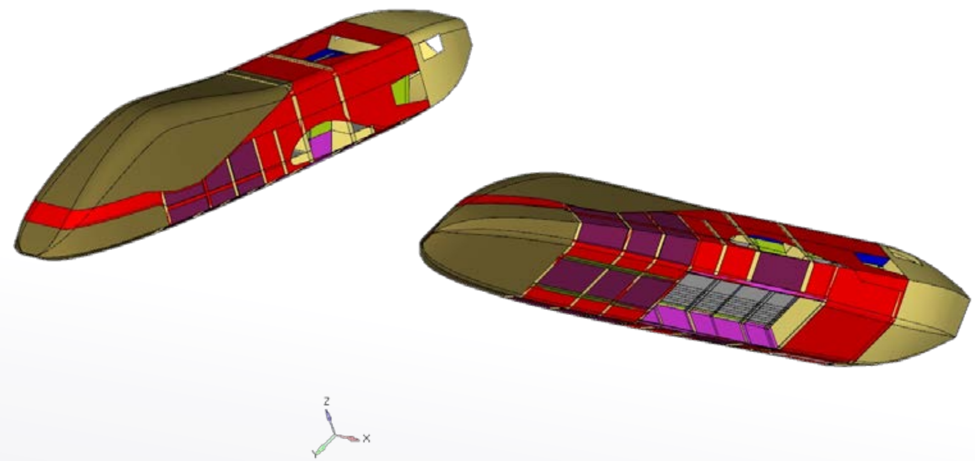


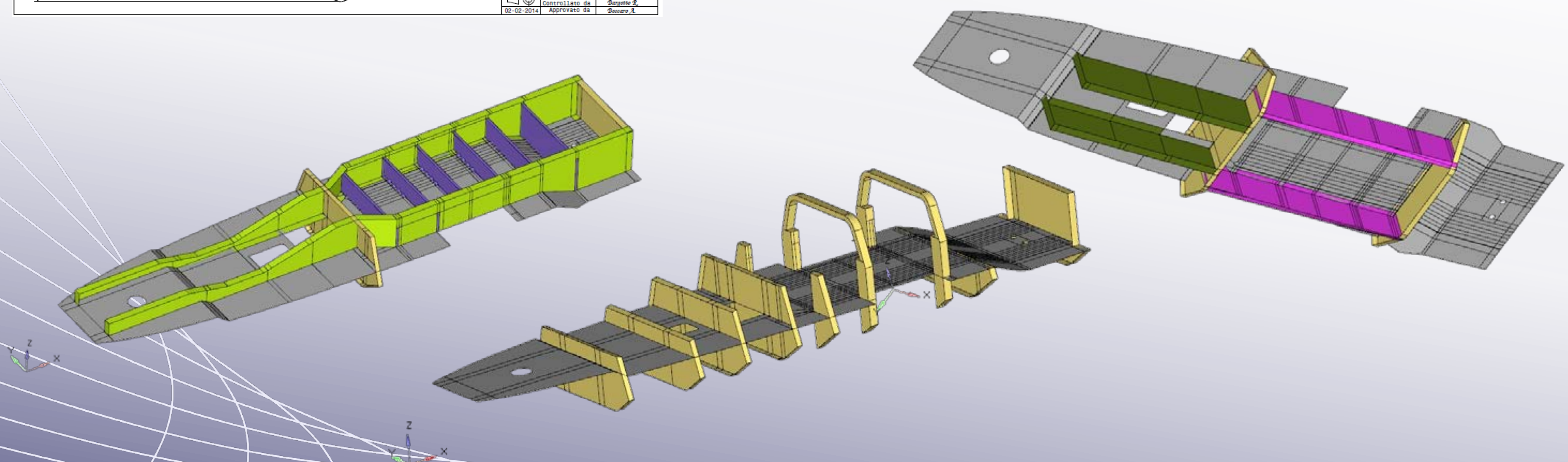
TIVANO fuselage

Complex
Aeronautical
Structures

TIVANO


18780
8710
AleniaAermacchi
PASD
TIVANO R00
Preparato da
Controllato da
Approvato da
02-02-2014
De Simone D.
Bargetto R.
Beccaro A.

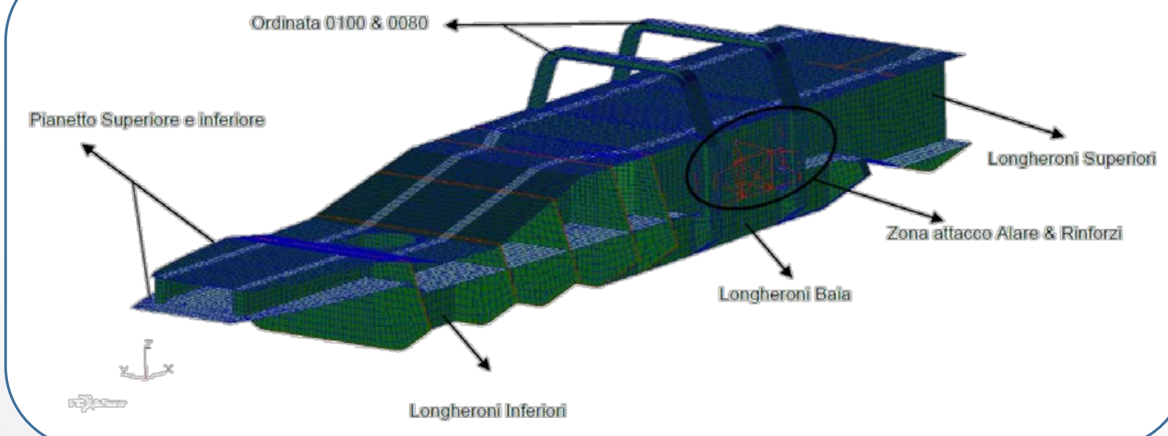




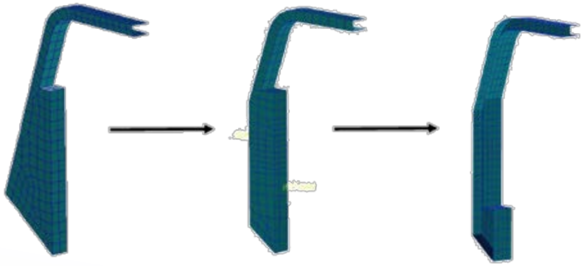
Work Flow

1. Preliminary load evaluation.
2. CAD analysis:
   Geometry cleaning process and optimization for the FEM modelling.
3. Import to pre-processing software and preliminary mesh of the structure components.
4. Introduction of the loads, BCs, materials and layup. Definitive mesh.
5. Design and modification of the structure
6. Model Checks:
   free edges, free body check, rigid modes…
7. Static analysis
   Inertial analysis
8. Optimization of the structure considering the obtained results (thickness, reinforcments…).
9. Reporting

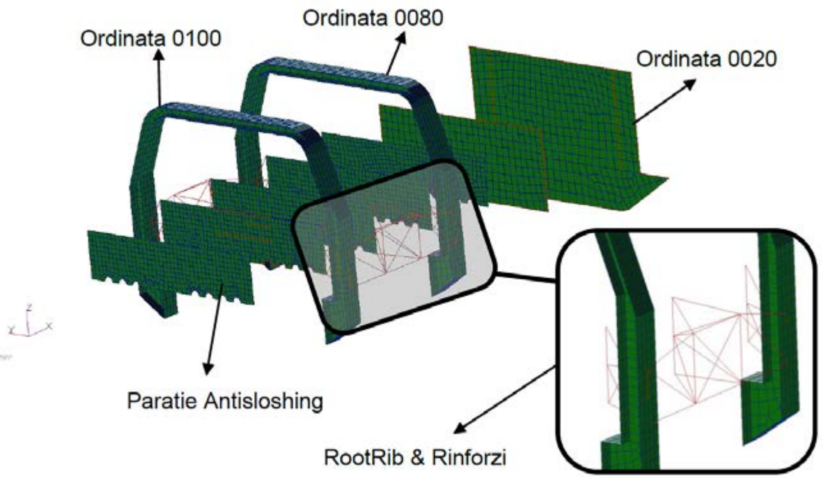
TIVANO Model



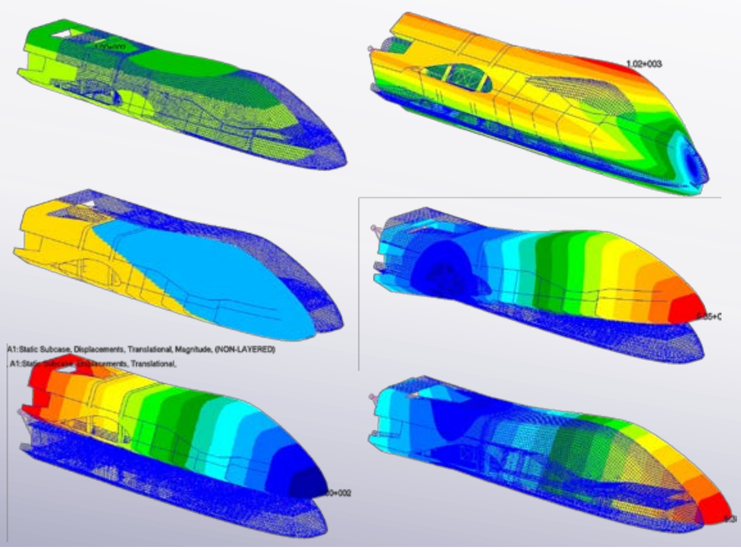
Modification of the main frames.



Metallic part



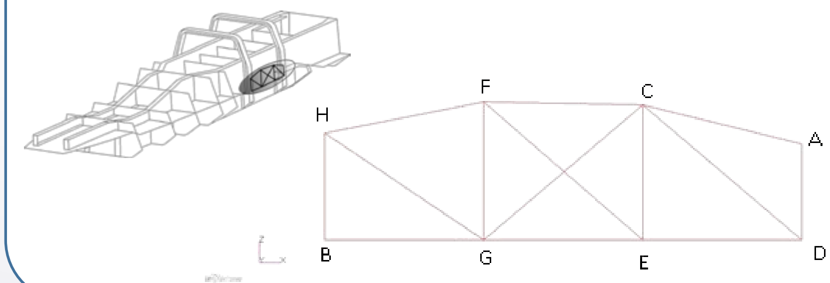
Rigid body check.



TIVANO: load cases and boundary conditions.

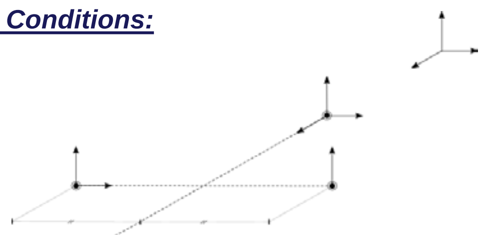
Inertial Load: $n_x = 0$ $n_y = 0$ $n_z = 3.57$

Wing Load:

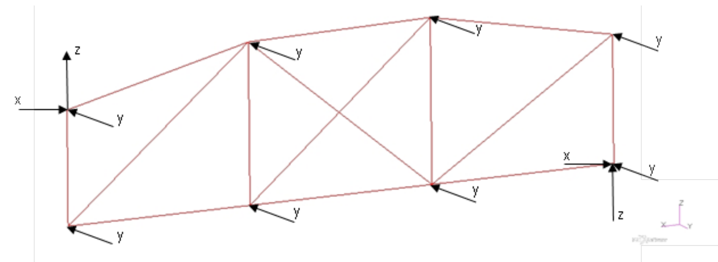


	Boundary Reactions[N]						
HOLE	F_X	F_Y	F_Z	HOLE	F_X	F_Y	F_Z
A	79084.5	-276675	-2704.5	E	/	+168097.5	/
B	-70993.5	+324768	-52629	F	/	-197487	/
C	/	-224803.5	/	G	/	+153168	/
D	/	+282144	/	H	/	-224604	/

Boundary Conditions:



Wing Load



Inertial Load

Allowables:

Metallic alloy $\sigma_R = 524$ Mpa

Composite tape $\mu\varepsilon_C = 10750$ $\mu\varepsilon_T = 7000$

$RF_{UL} = \sigma_{allowable} / \sigma_{applied} \geq 1.00$

$RF_{UL} = \mu\varepsilon_{allowable} / \mu\varepsilon_{applied} \geq 1.00$

At Room Temperature and Ultimate load.

TIVANO: Inertial Load Results

Figure: Maximum stress.

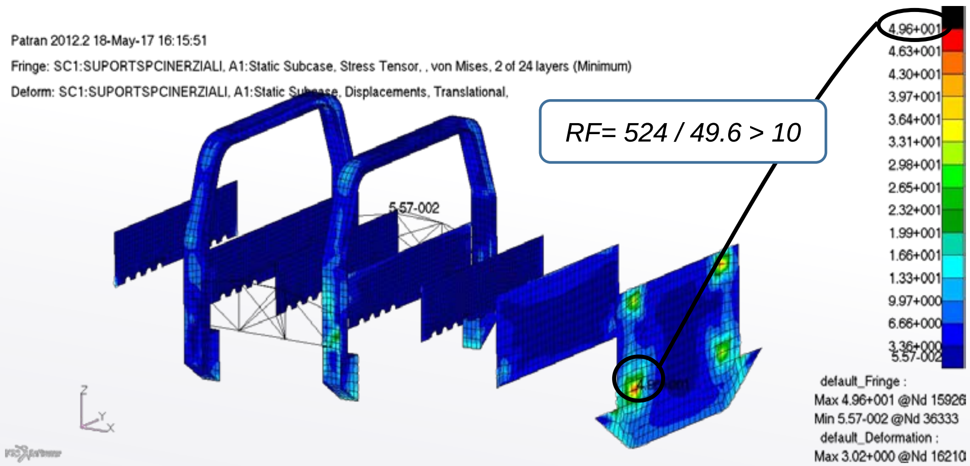


Figure: Displacements of the fuselage.

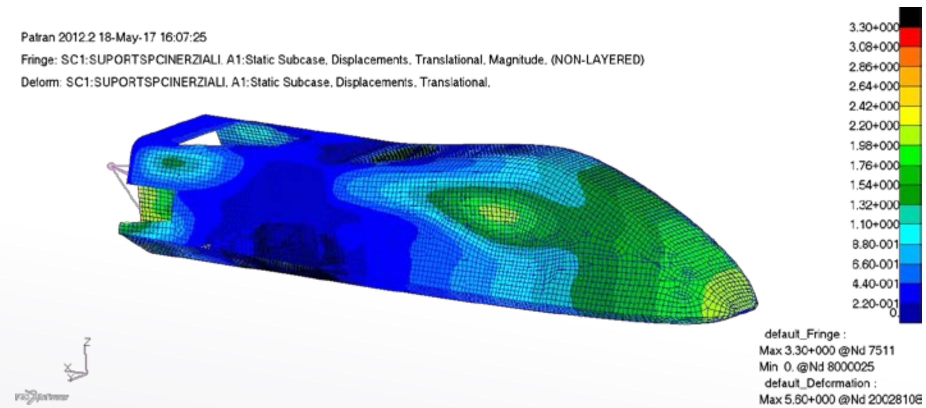
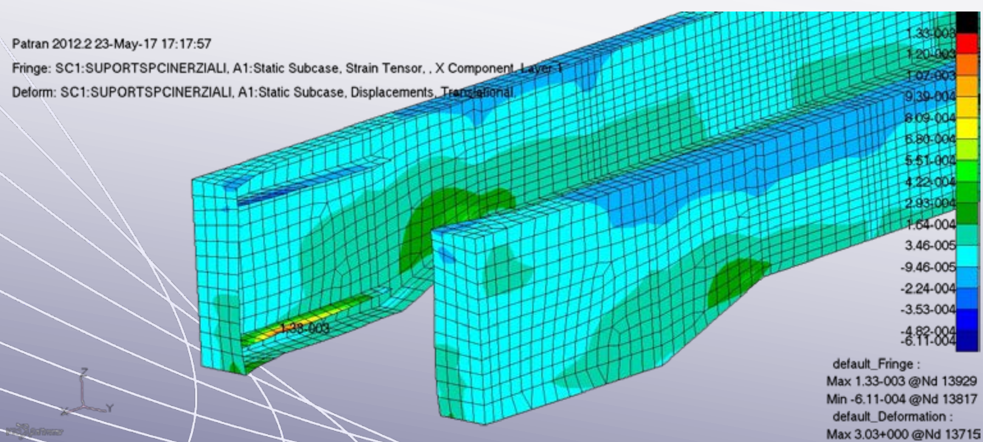
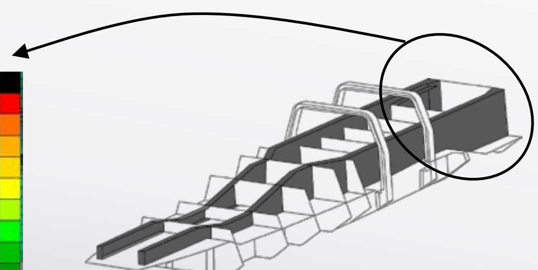


Figure: Maximum traction strain.





$RF_c = 7000/1070 > 6$

$RF_T = 10750/1330 > 8$

TIVANO: Wing Load Results

Figure: Maximum stress.

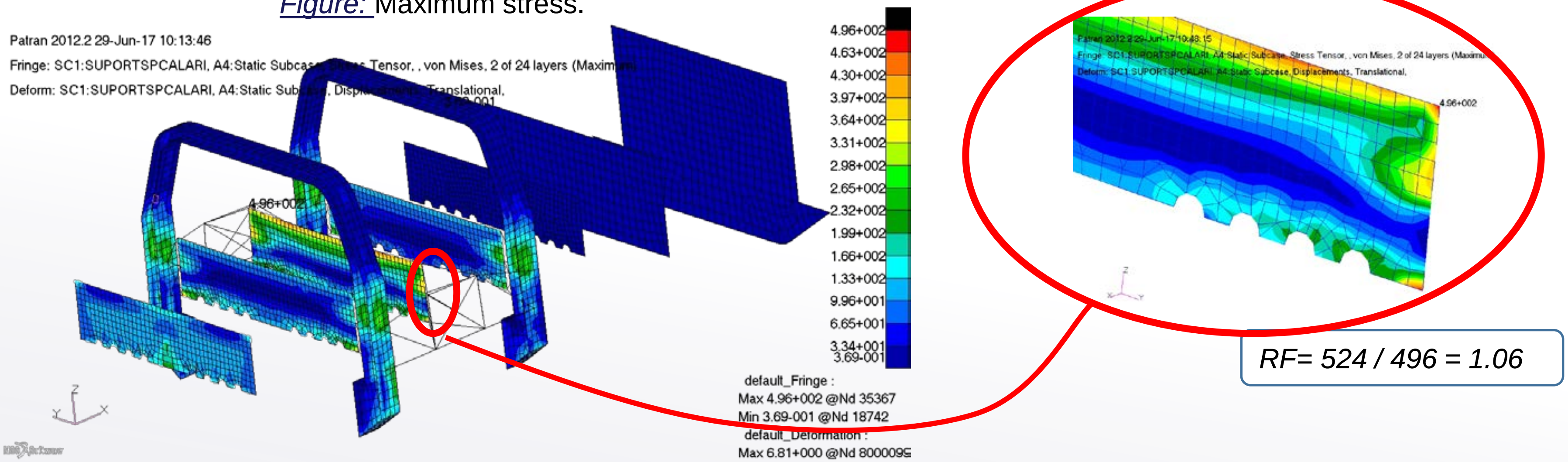
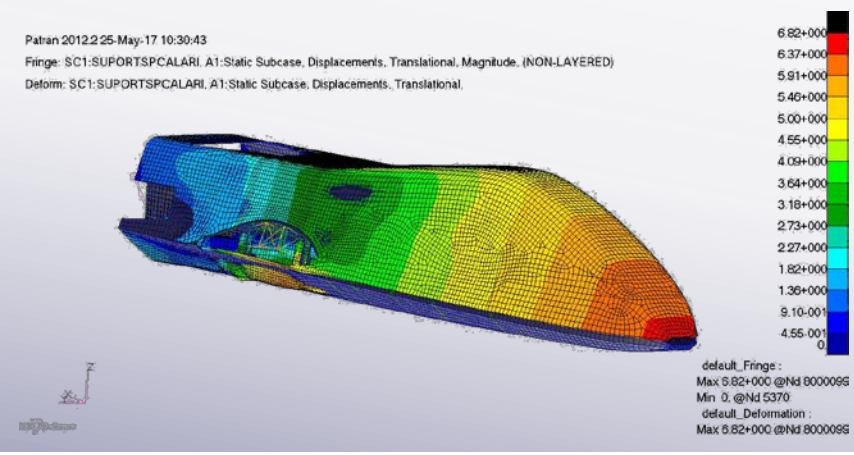


Table: Reserve factors of the components.

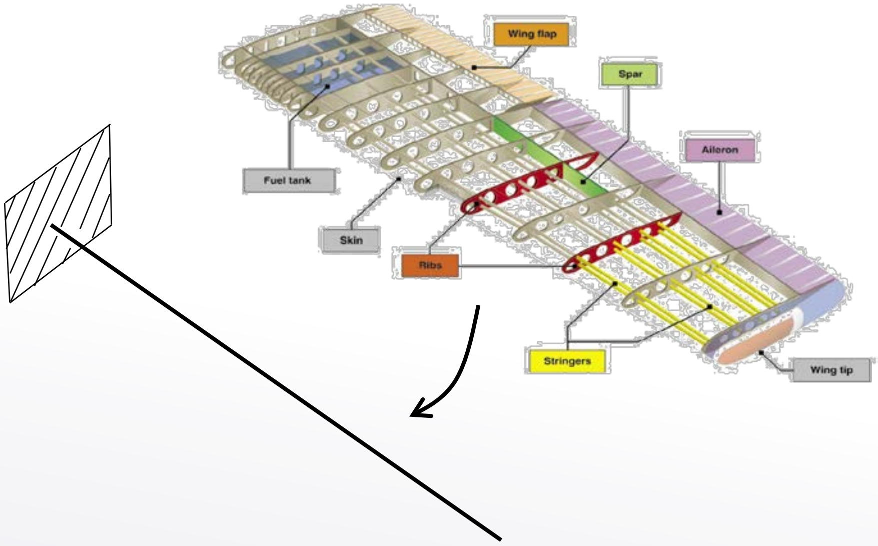
Componente:	RF	Componente:	RF
Skin		Ordinate	
A trazione	HIGH	A trazione	HIGH
A compressione	HIGH	A compressione	3.2
Longheroni Superiori		Ordinata 125	
A trazione	3.1	A trazione	3.38
A compressione	HIGH	A compressione	3.91
Longheroni Inferiori		Pianetto Superiore	
A trazione	HIGH	A trazione	HIGH
A compressione	HIGH	A compressione	2.6
Longheroni Baia		Pianetto Inferiore	
A trazione	3.02	A trazione	2.78
A compressione	3.55	A compressione	3.55

Figure: Displacements of the fuselage.



Introduction to the High Fidelity Model

In the aeronautical field the structural analyses require huge computational costs due to the complexity of the structure and the large use of composite materials.
Usally particular shapes are used in the structure, as the tapered one or twist angle, to satisfy aerodynamics requirements. These features introduce coupling effects that are difficult to be detected by the classical models.



It’s important to develop efficiently tools in order to provide accurate results reducing the computational cost. These tools have to be able to describe complex structures considering their not-prismatic shapes and their multicomponent scenarios.
In addition, a correct design of the composite laminate can be strongly affects the mechanical behavior of the structure. An advanced model allows a Layer-wise approach to be implemented in order to study in the details the behaviors of a laminate and allows global coupling effects to be detected.

Preliminaries and material definition

$$\mathbf{u}(x,y,z)=\mathbf{F}_\tau(x,z)\mathbf{u}(y)$$

$$\mathbf{u}(x,y,z)=N_i(y)F_\tau(x,z)\mathbf{q}_{\tau i}$$

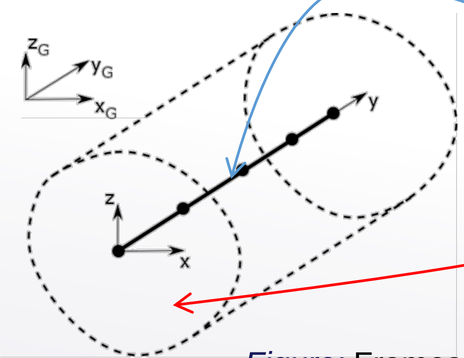
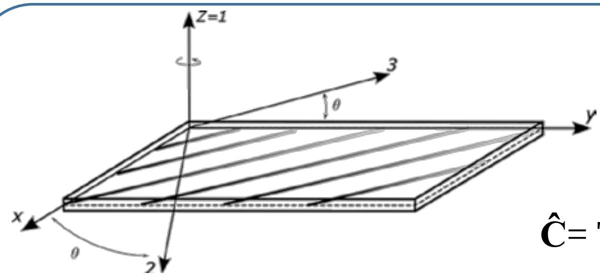


Figure: Frames.



$$\hat{C}= TCT^T$$

Figure: Material coordinate frames.

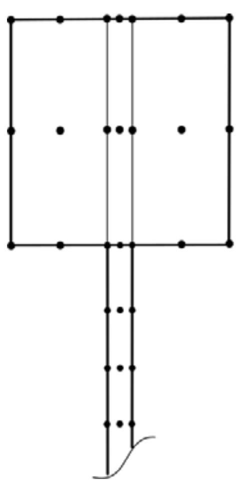


Figure: Cross-section described by L9 elements.

$$\delta L_{int} = \int_V \delta\boldsymbol{\varepsilon}^T \boldsymbol{\sigma}\, dV \longrightarrow \mathbf{K}$$

$$\delta L_{ine} = \int_V \delta\mathbf{u}^T \rho\ddot{\mathbf{u}}\, dV \longrightarrow \mathbf{M}$$

Static & Free Vibration Problems

Derivation of the $k^{\tau sij}$:

$$\boldsymbol{\varepsilon}(x,y,z) = \mathbf{b}\mathbf{u}(x,y,z)$$

$$\delta\boldsymbol{\varepsilon}(x,y,z) = \mathbf{b}N_j(y)F_s(x,z)\delta\mathbf{q}_{sj}$$

$$\boldsymbol{\sigma}(x,y,z) = \mathbf{C}\boldsymbol{\varepsilon}(x,y,z)$$

$$\boldsymbol{\sigma}(x,y,z) = \mathbf{C}\mathbf{b}N_i(y)F_\tau(x,z)\mathbf{q}_{\tau i}$$

$$\delta L_{int} = \delta\mathbf{q}_{sj} \int_V F_s(x,z)N_j(y)\underbrace{\left[\ \mathbf{b}^T\ \right]}_{[3x6]}\underbrace{\left[\ \mathbf{C}\ \right]}_{[6x6]}\underbrace{\left[\ \mathbf{b}\ \right]}_{[6x3]} N_i(y)F_\tau(x,z)dV\, \mathbf{q}_{\tau i}$$

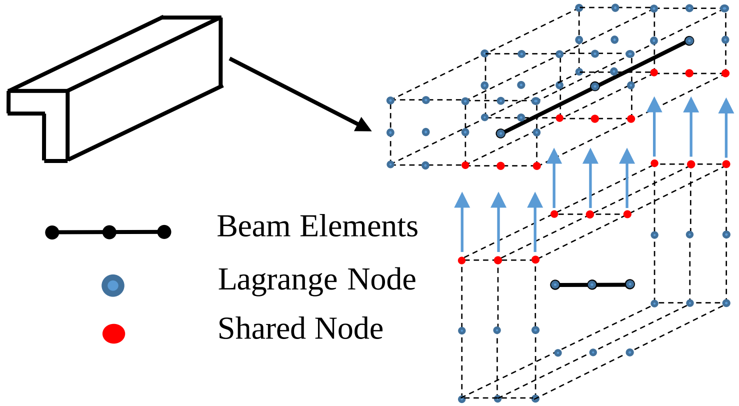
[3x3]

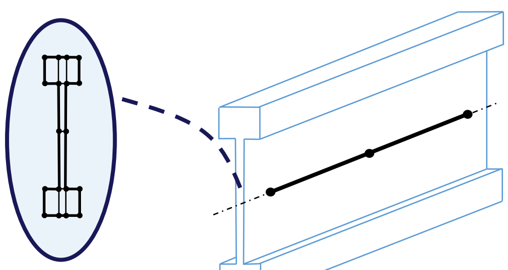
Fundamental Nucleus $[\mathbf{k}^{\tau sij}]$

Assembling of different beams & tapered shapes.

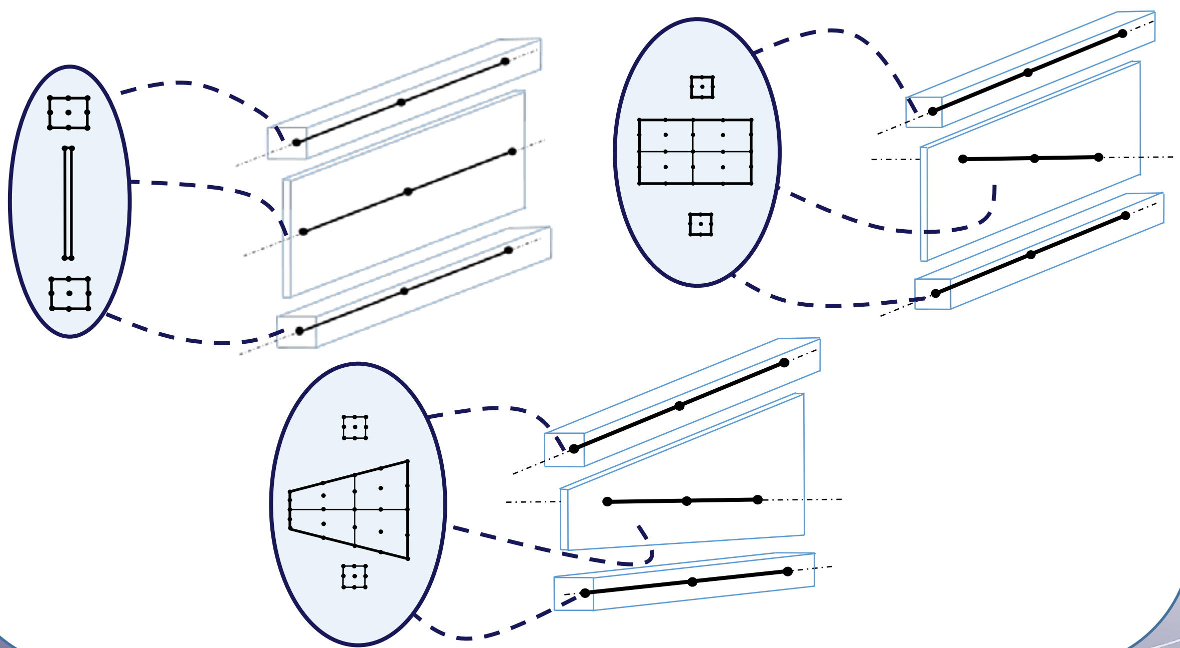
Different structures with LE expansion can be assembled very easily because in this formulation there are only pure displacements. In this way the structures can be joined without incurring in issues.

The assembling can be done by imposing the congruence of the displacements.

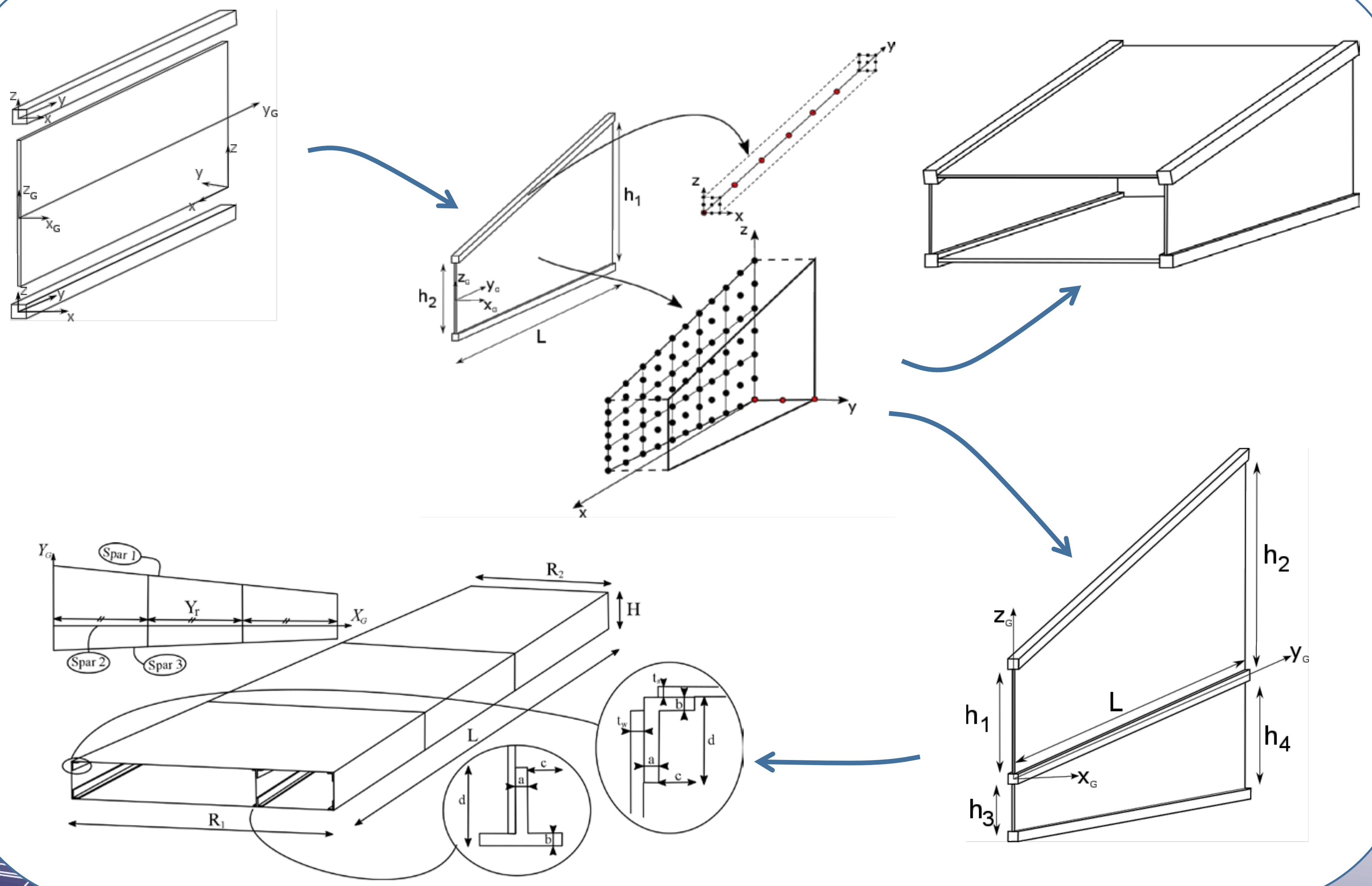




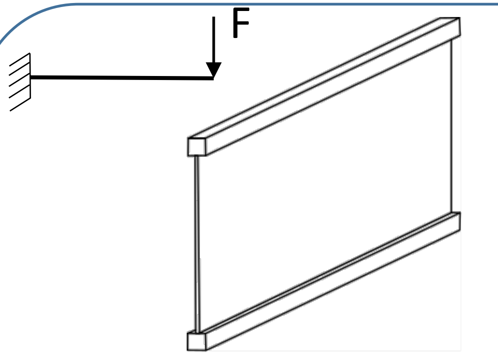
Component wise approach



To the complex structure using the Component Wise approach...



Static analyses



	DOF	Point A	Point B
20L9	4521	-4.179	-4.36
Nastran *Solid + Shell*	~ 15000	-4.219	-4.487
Nastran *Beam + Shell*	~ 7000	-4.367	-4.671

Table: Displacements.

Figure: σ_{zy} over the height of the panel.

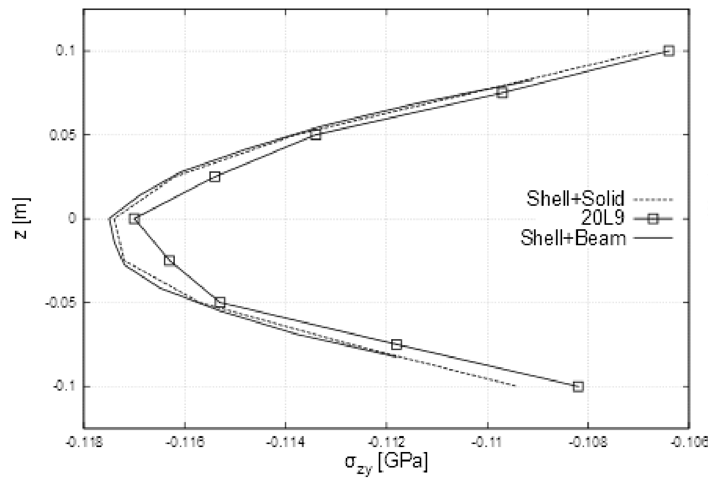
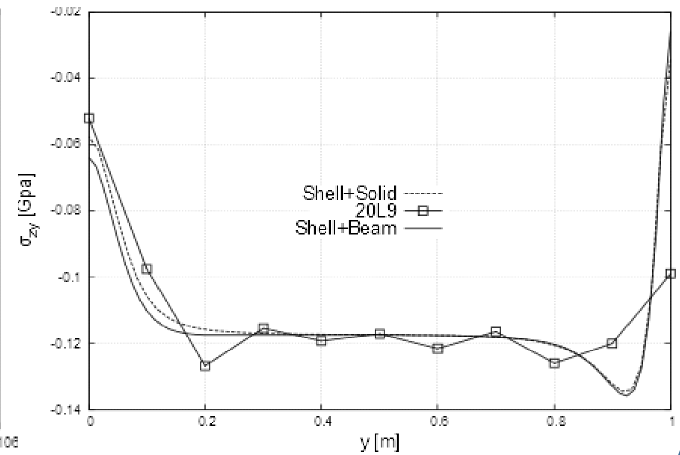
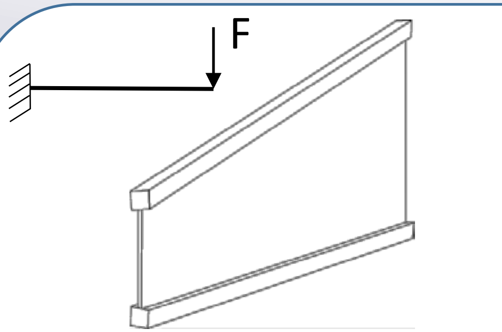


Figure: σ_{zy} over the lenght of the panel.





	DOF	Point A	Point B
20L9	4521	-4.535	-4.413
Nastran *Solid + Shell*	~ 22000	-4.586	-4.409
Nastran *Beam + Shell*	~ 15700	-4.823	-4.599

Table: Displacements.

Figure: σ_{zy} over the height of the panel.

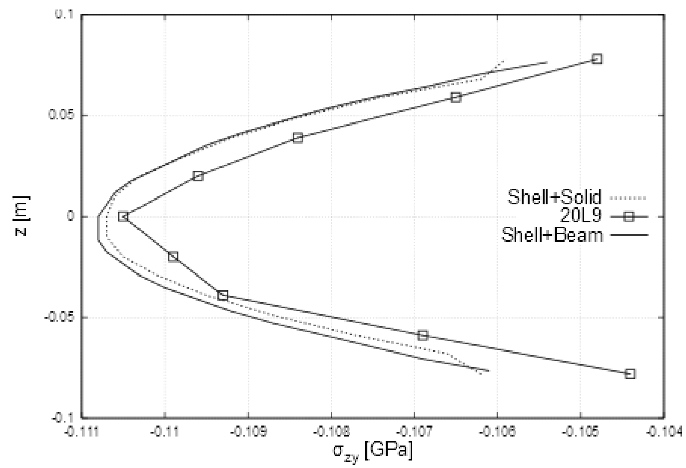
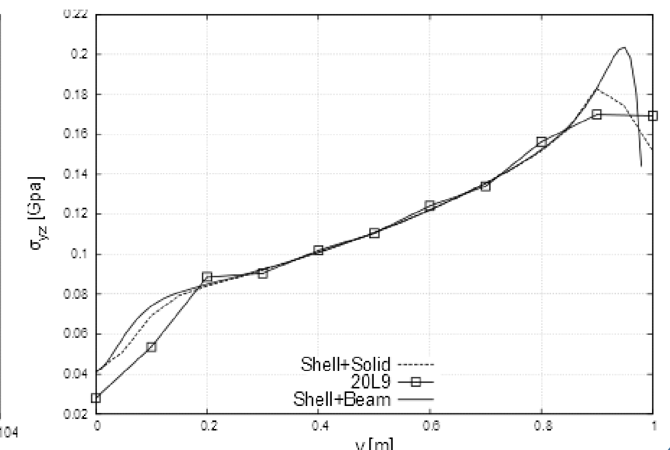


Figure: σ_{zy} over the lenght of the panel.



Three spar panel made of composite laminate

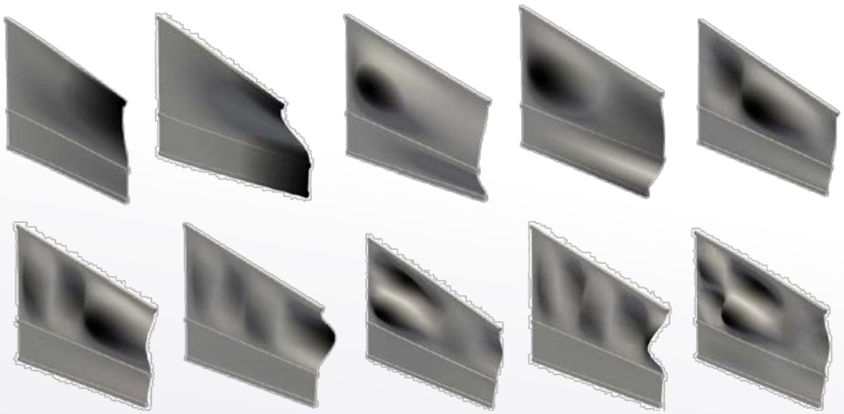


Figure: First 10 Modal Shapes.

Table: Frequencies.

	C-W Model	$NAS_{Beam+Shell}$	NAS_{Solid}
DOF's	13167	37296	179700
f_1	7,14	6,97	7,14
f_2	7,89	7,72	7,96
f_3	12,31	10,51	13,58
f_4	13,02	11,09	14,53
f_5	18,02	15,25	20,49
f_6	23,75	19,79	26,62
f_7	28,44	23,34	30,26
f_8	32,54	24,78	32,99
f_9	36,07	30,17	38,43
f_{10}	41,33	31,88	42,35
f_{11}	46,97	-	46,73
f_{12}	47,57	-	47,80
f_{13}	49,62	40,75	50,10
f_{14}	50,85	-	50,85
f_{15}	52,98	-	54,18

Layer-wise description

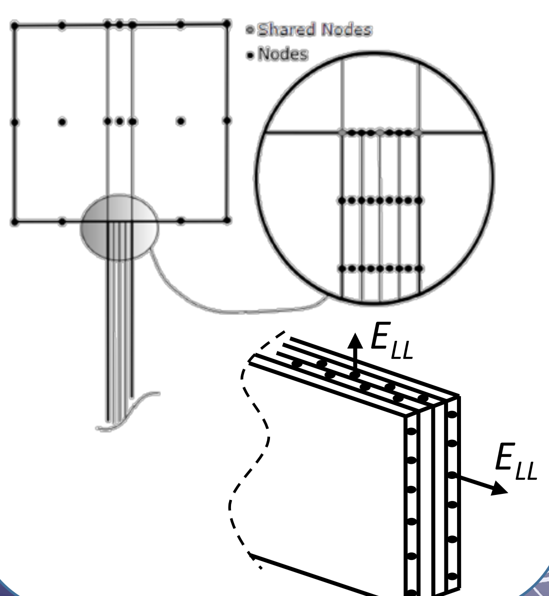
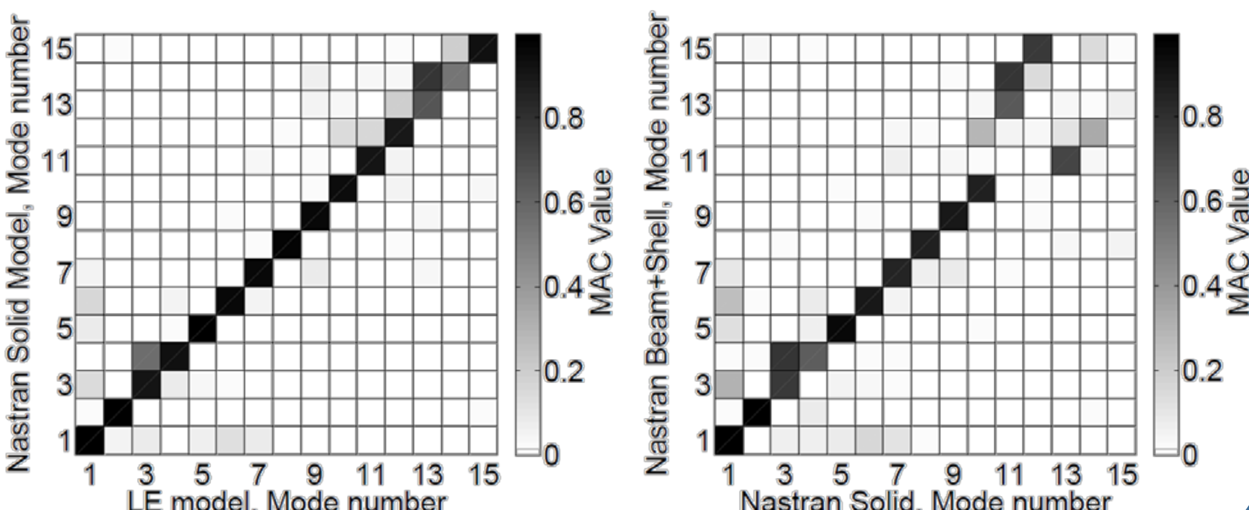


Figure: Modal shape comparison between the models.



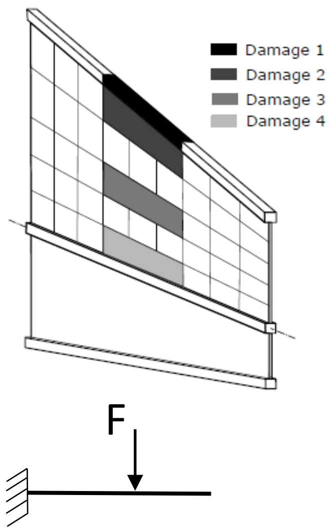
Introduction of the damage.

The damage is implemented by the degradation of the mechanical material proprieties.

The damage level is defined through the parameter d, according to the formula.

$$E_d = d \times E \qquad 0 < d < 1$$

Once a component fails, the load is redistributed on the other structural elements. The understanding of the evolution of the stress distribution inside the structure is of primary importance in the design process and for the maintenance program



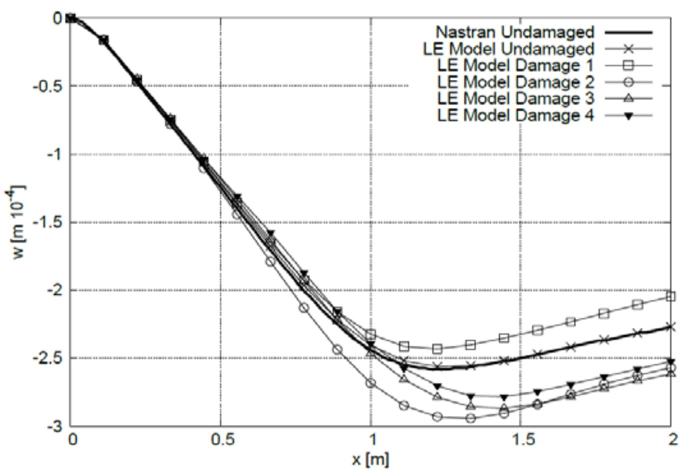


Figure: Displacements of the central stringer.

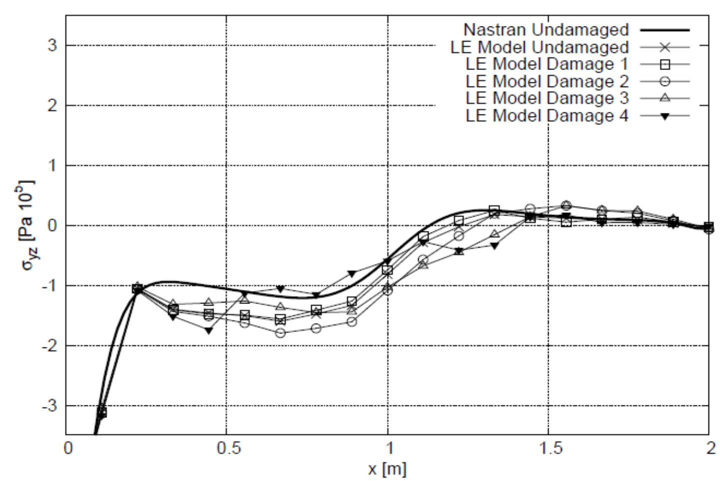
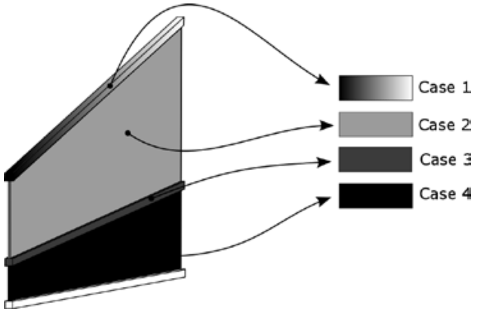


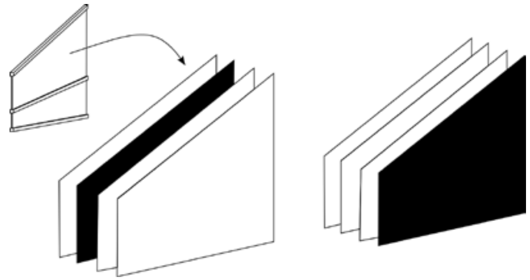
Figure: σ_{zy} of the central stringer.

In the aeronautical field, due to the complexity of the structure and the particular shapes to ensure the aerodynamic characteristics, the evaluation of the stress evolution is very difficult. For these reasons the timely damage detection is a crucial point in the maintenance process. Through the comparison of the damaged natural frequencies with a wide damage database which collects the information about the damage and the natural frequencies, an estimation of the damage location can be performed. In this way the efficiency of the damage detection tests can be increased.

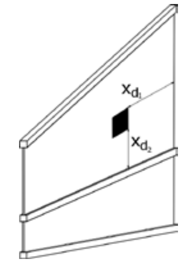
CUF for the damage detection.



Whole damaged components



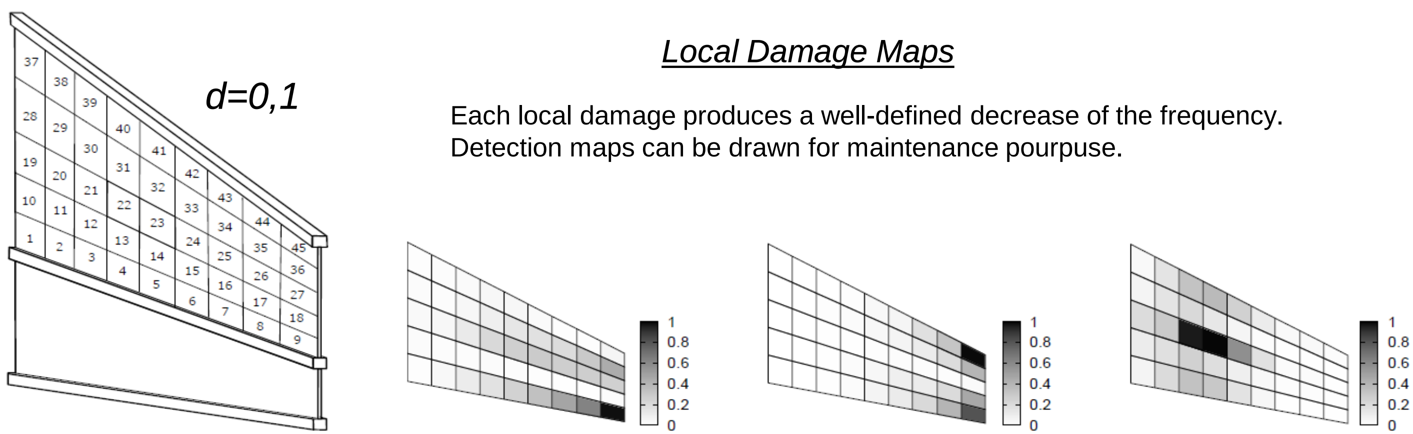
Damaged layer



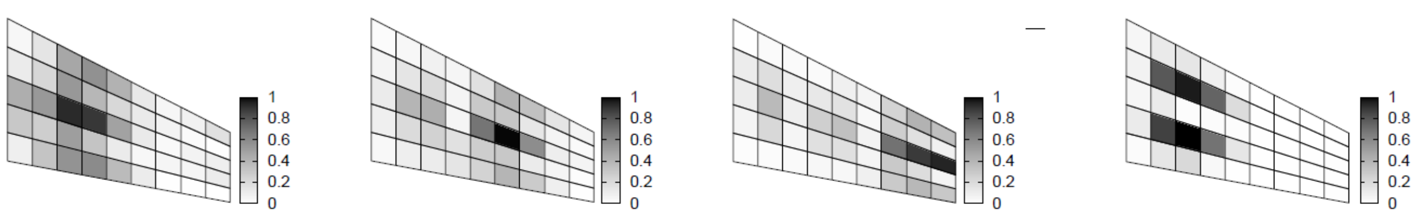
Local Damage

<u>*Local Damage Maps*</u>

Each local damage produces a well-defined decrease of the frequency.
Detection maps can be drawn for maintenance pourpuse.

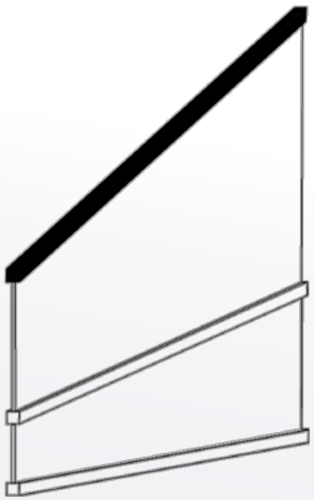


(a) Mode 1 (b) Mode 1 (c) Mode 2 (d) Mode 3

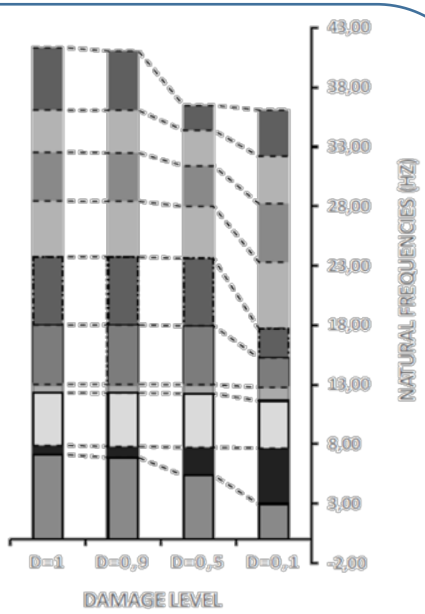


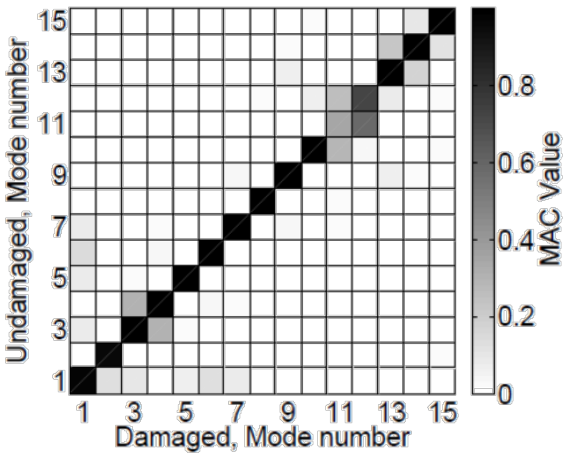
(e) Mode 4 (f) Mode 5 (g) Mode 6 (h) Mode 8

CUF for the damage detection: the damaged stringer case.

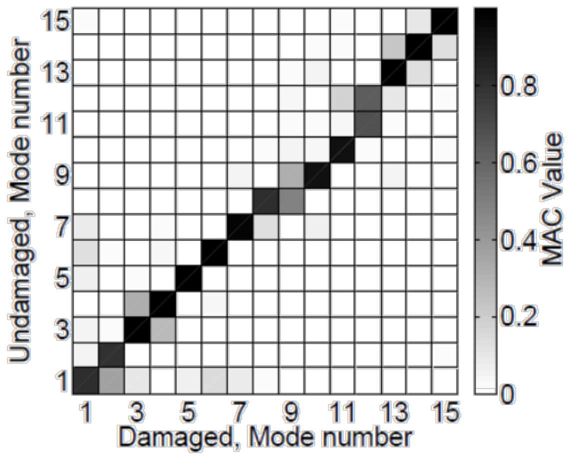


	Case 1			
	d=0	d=0.9	d=0.5	d=0.1
f_1	7.14	6.88	5.37	2.96
f_2	7.89	7.81	7.71	7.66
f_3	12.31	12.3	12.24	11.65
f_4	13.02	13.02	13	12.76
f_5	18.02	18.01	17.95	15.25
f_6	23.75	23.74	23.62	17.73
f_7	28.44	28.41	27.97	23.3
f_8	32.54	32.49	31.39	28.22
f_9	36.07	36.05	34.39	32.2
f_{10}	41.33	41.05	36.47	36.09
f_{11}	46.97	45.39	42.16	40.44
f_{12}	47.57	47.27	47.22	42.55
f_{13}	49.62	49.57	49.52	47.14
f_{14}	50.85	50.03	50.01	49.49
f_{15}	52.98	52.92	52.71	50

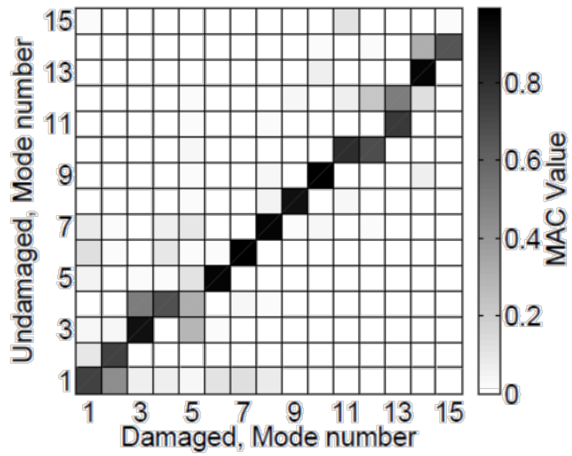




d=0,9



d=0,5



d=0,1

CUF for the damage detection: the complex wing-box.

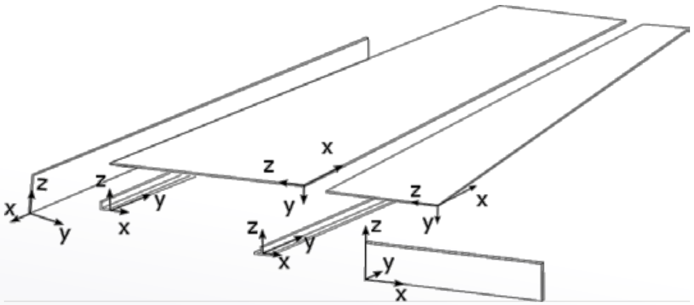


Figure: Beam description of the components.

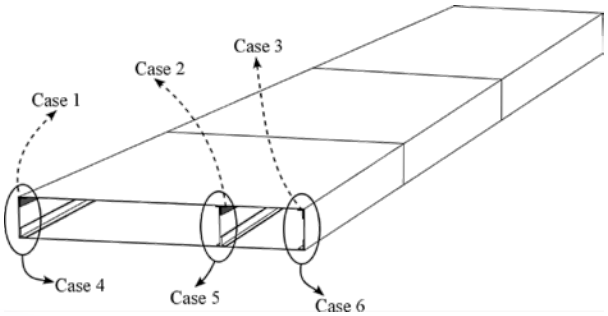
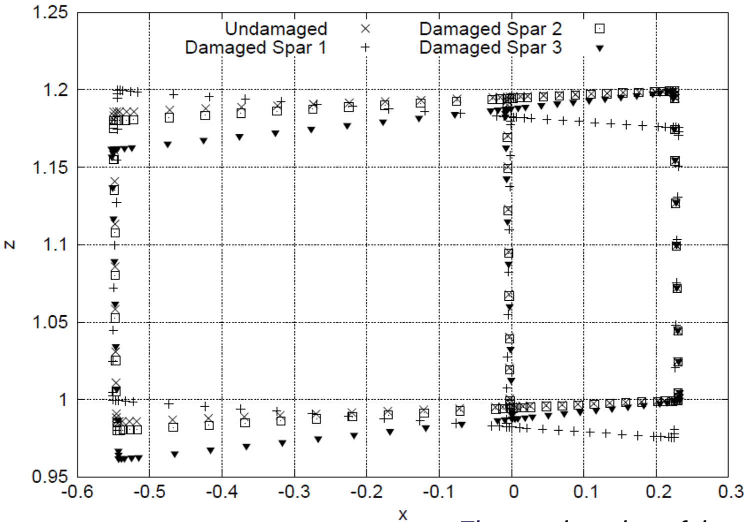


Figure: Damaged Cases.

	NAS$_{SOLID}$	LE$_{MODEL}$	Damage case Case 1 Spar Caps 1	Case 2 Spar Caps 2	Case 3 Spar Caps 3	Case 4 Spar 1	Case 5 Spar 2	Case 6 Spar 3
1^{ST} Bending	9,57	9,58	9,35	9,3	9,35	8,96	8,87	9,01
1^{ST} Torsional	38,5	39,4	38,91	39,15	39,18	35,7	37,73	37,67

Table: Frequencies.



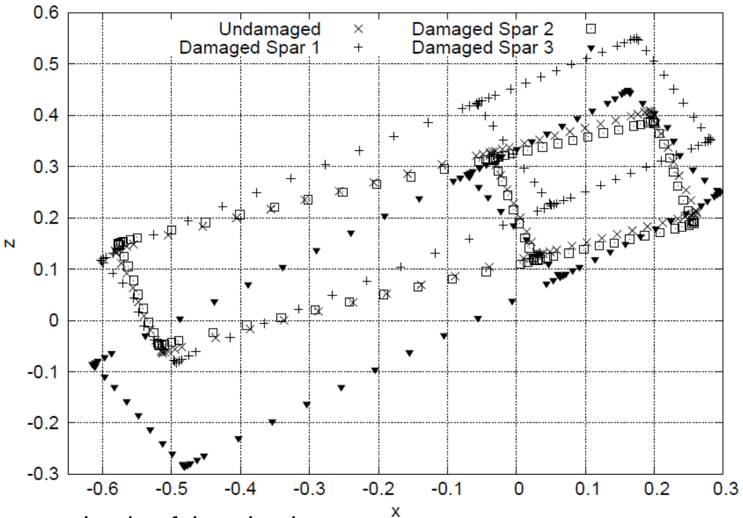


Figure: alteration of the modal shapes at the tip of the wing box.

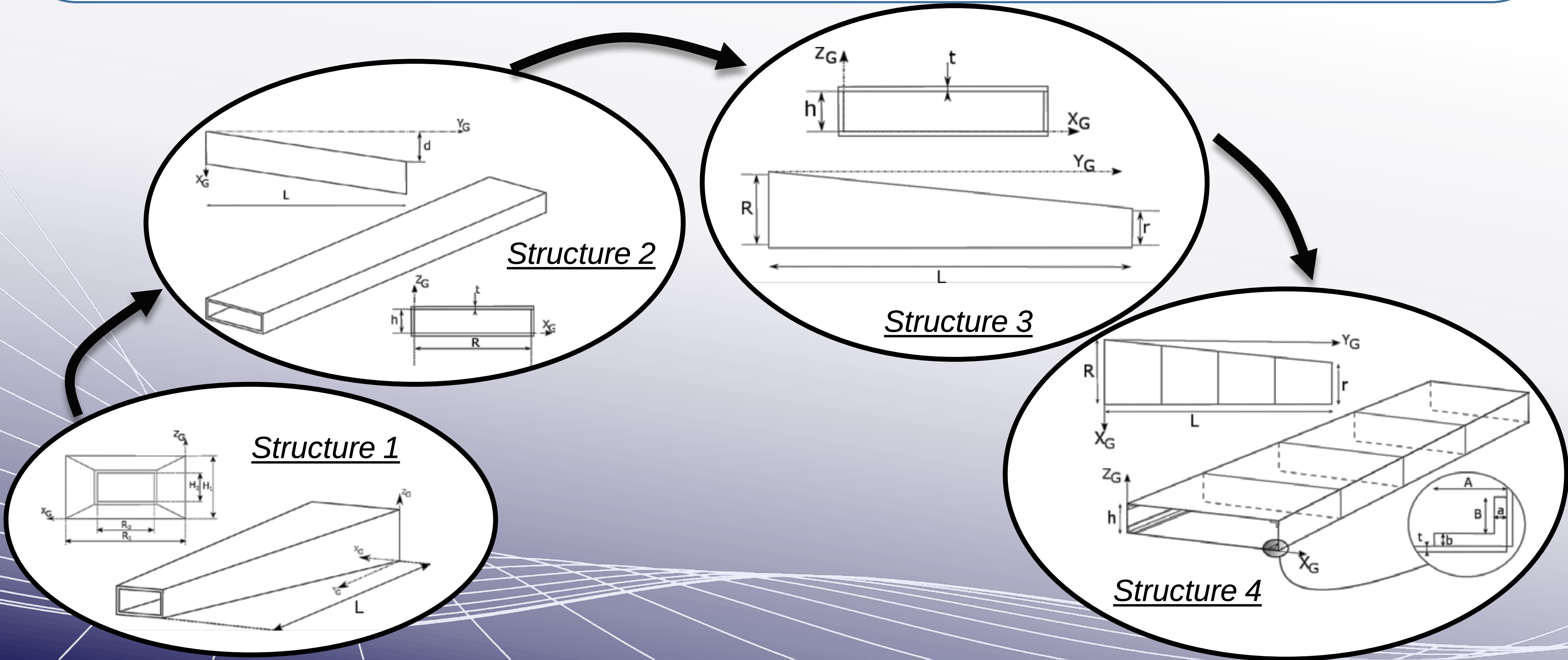
CUF for Tailoring Analysis

Laminate composite may have a high anisotropic behavior, that is, the mechanical proprieties can be different in each direction.

Layers with different fiber direction an be overlapped in a process called ***tailoring*.**

In this way a laminate with desired proprieties can be obtained.

Tailoring is largely used in the aeronautical field to address, for example, aeroelastic stability problems. In fact, using the tailoring, the designer can introduce coupling effects between torsion and bending or, can influence the effects due to the geometric coupling.

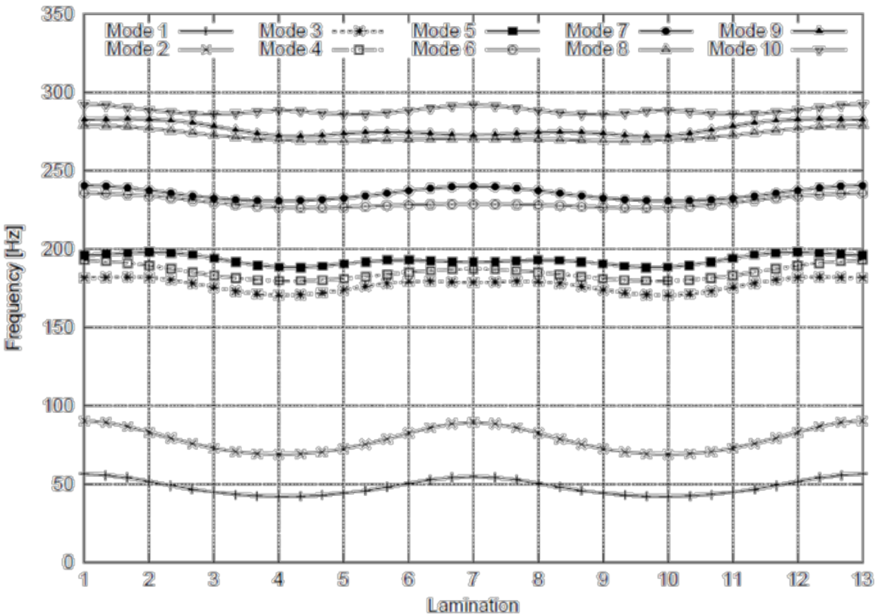


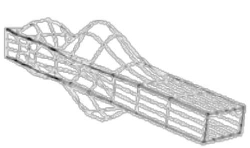
Tailoring analysis: structure 1 and structure 2.

Table: Lamination

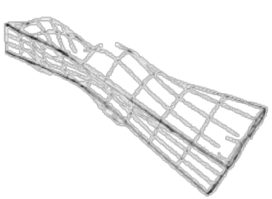
	1	2	3	4	5	6	7	8	9	10	11	12	13
θ_1	90°	75°	60°	45°	30°	15°	0°	−15°	−30°	−45°	−60°	−75°	−90°
θ_2	0°	−15°	−30°	−45°	−60°	−75°	−90°	−105°	−120°	−135°	−150°	−165°	−180°

Structure 1

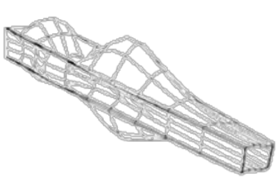




90°/0°

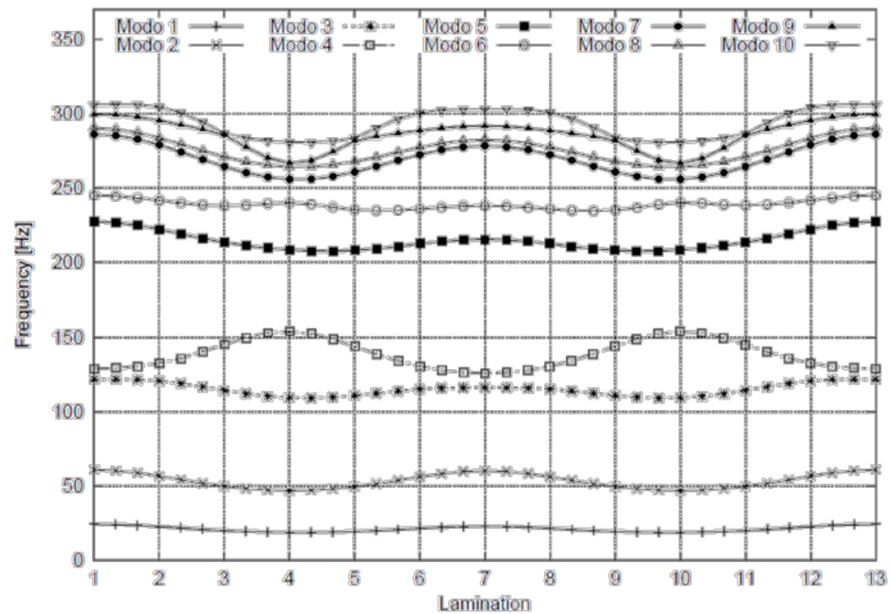


60°/-30°



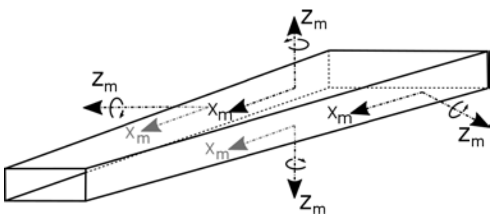
45°/-45°

Structure 2

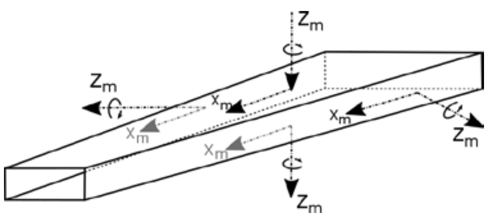


	Lamination	Present Model	Solid Nastran
DOF		11844	130000
1^{st} Mode	4	18,97	17,39
	7	23,04	21,51
2^{nd} Mode	1	60,97	57,76
	10	47,04	43,66
3^{rd} Mode	4	109,3	101,11
	7	116,09	107,04
4^{th} Mode	1	128,86	123,79
	4	153,71	148,29

Tailoring analysis: structure 3.

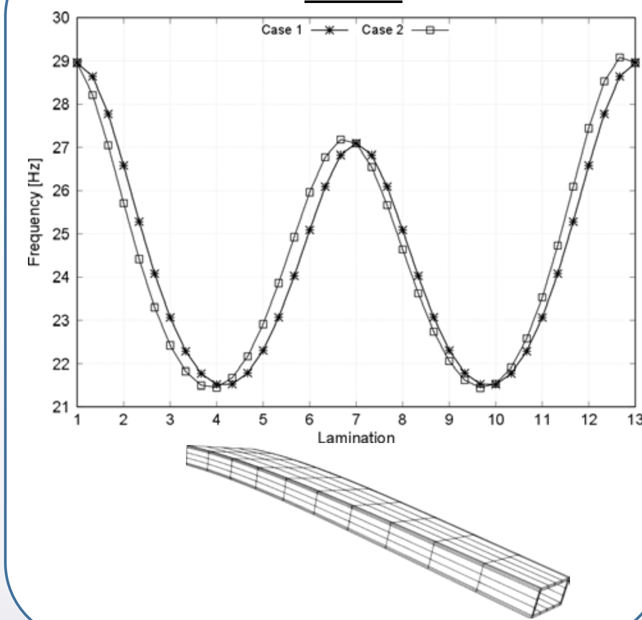


Case 1

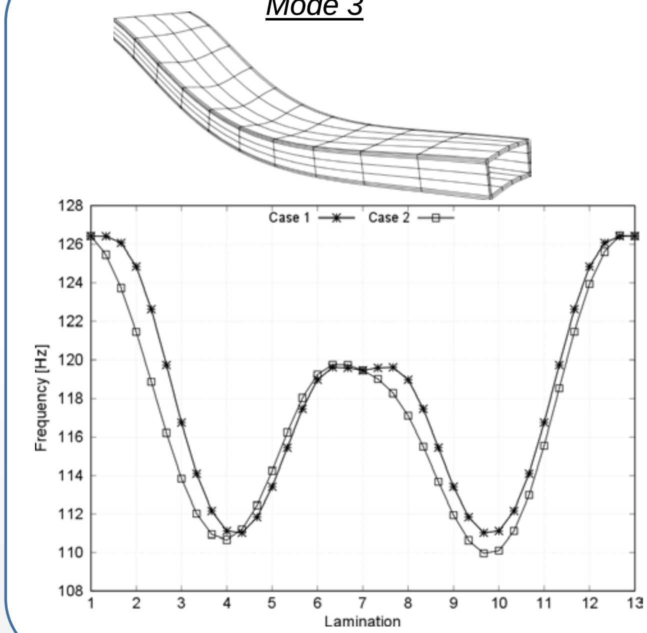


Case 2

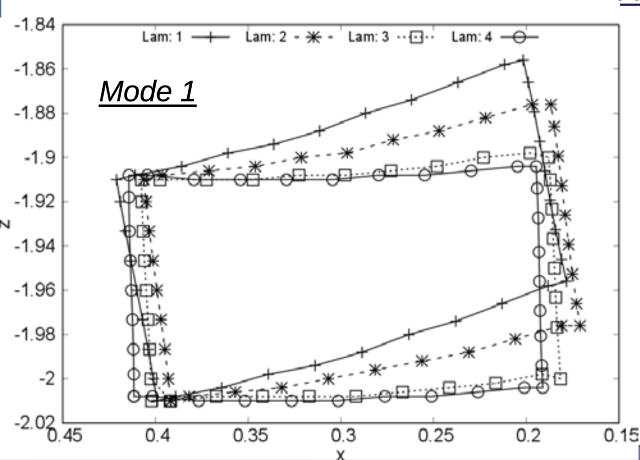
Mode 1

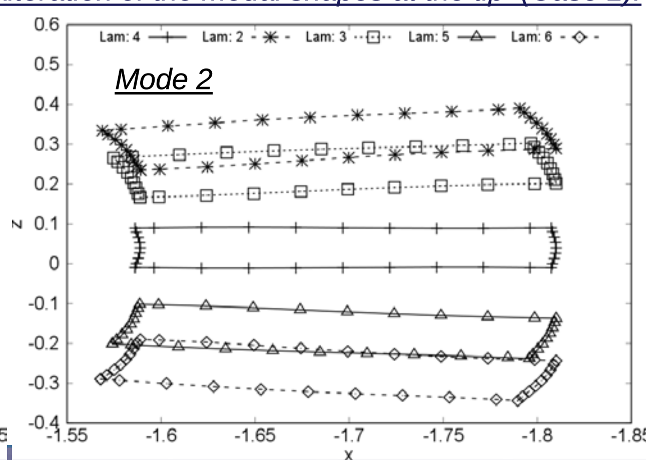


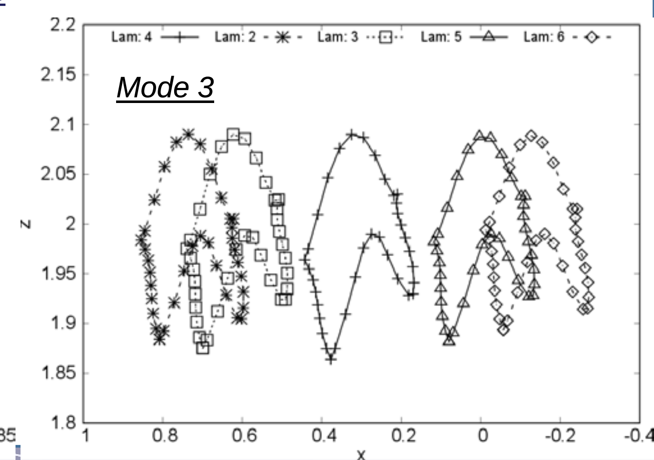
Mode 3



Alteration of the modal shapes at the tip (Case 1).







Tailoring analysis: structure 3.

Mode 1: the behavior is governed by the stiffness of the panel.

Mode 2: the behavior is influenced by both the webs and the panels. The highest frequencies are obtained with a cross play lamination in the components.

Mode 3: As the first mode. The tailoring of the web have yet weak effects.

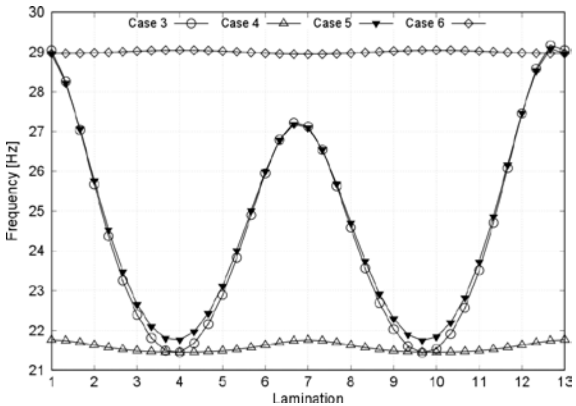
Mode 4: The torsional modes is maximized by an angle play configuration.

	1	2	3	4	5	6	7	8	9	10	11	12	13
θ_1	90°	75°	60°	45°	30°	15°	0°	−15°	−30°	−45°	−60°	−75°	−90°
θ_2	0°	−15°	−30°	−45°	−60°	−75°	−90°	−105°	−120°	−135°	−150°	−165°	−180°

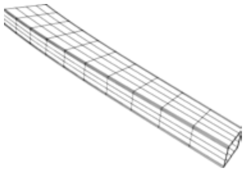
	Panels		Webs	
	Inner Layer	Outer Layer	Inner Layer	Outer Layer
Case 3	θ_1	θ_2	45°	−45°
Case 4	45°	−45°	θ_1	θ_2
Case 5	θ_1	θ_2	90°	0°
Case 6	90°	0°	θ_1	θ_2

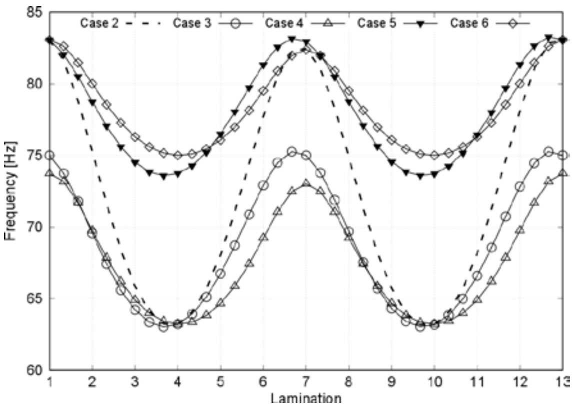
Table: Particular Cases

Mode 1

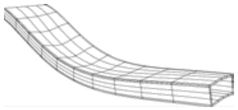


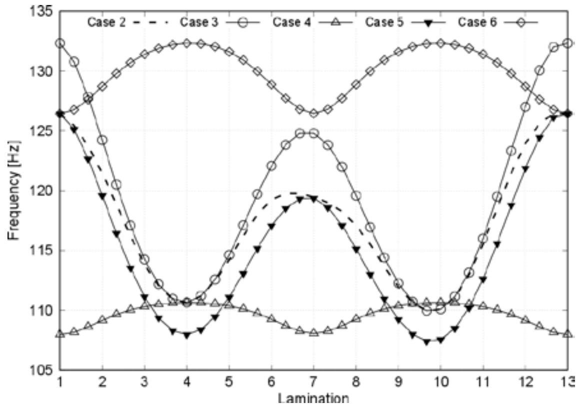
Mode 2



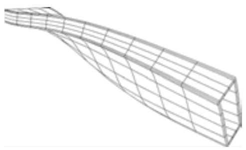


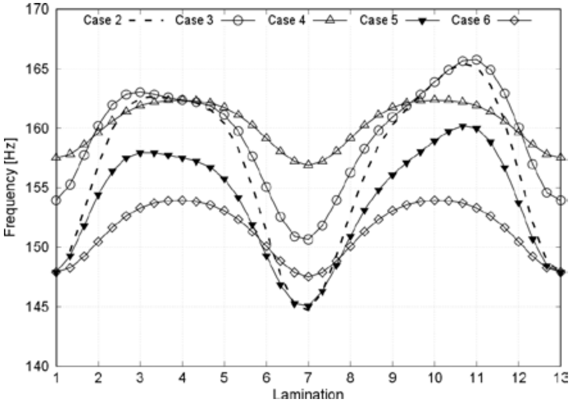
Mode 3





Mode 4





Tow Angle Placement

VSCL Variable stiffness composite laminate: a laminate can be developed providing different stiffness according to the considered area through a variable lamination over the structure.

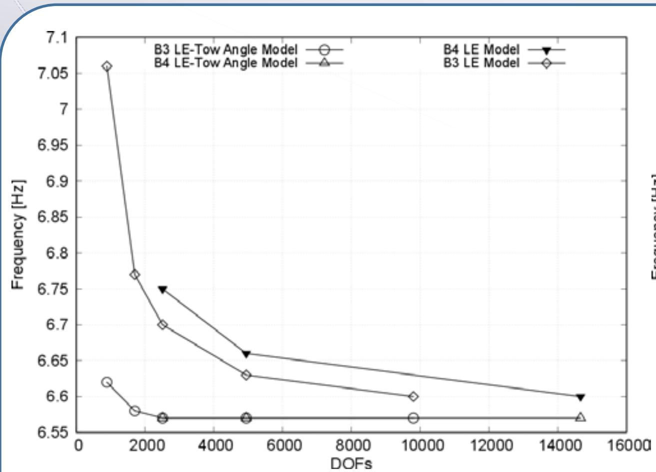
$$\mathbf{k}^{\tau sij} = \int_V F_s(x,z) N_j(y) \left[\ \mathbf{b}^T\ \right]\left[\ \mathbf{C}\ \right]\left[\ \mathbf{b}\ \right] N_i(y) F_\tau(x,z)\, dV$$

C is function of (x,y,z) and have to be integrated.

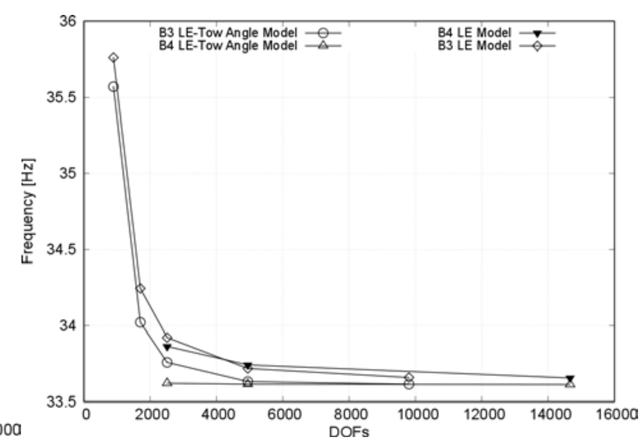
θ(y) = 0 + (y/L)*90

Tow angle LE model: the material proprieties are integrated.

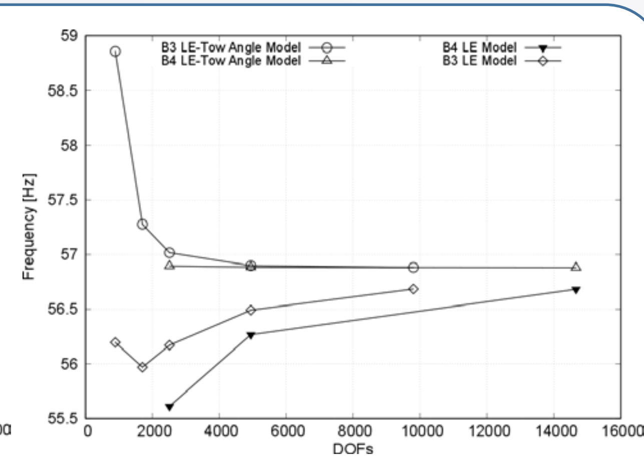
LE model: lamination law discretized by step. Each beam element introduces a step of the lamination.



Mode 1



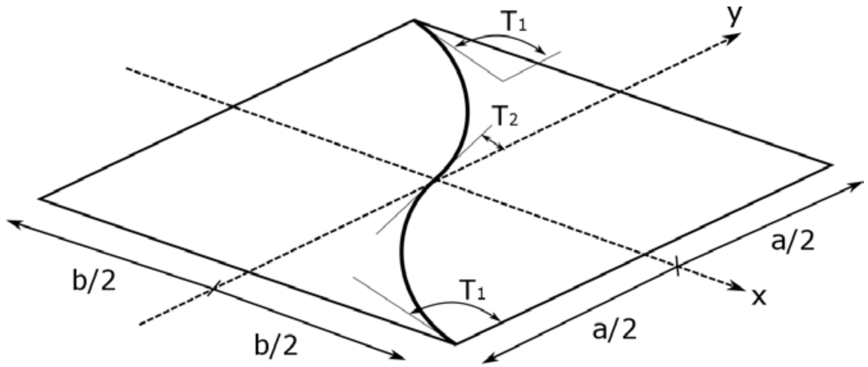
Mode 2



Mode 3

Tow Angle Placement: 3-layer assessment

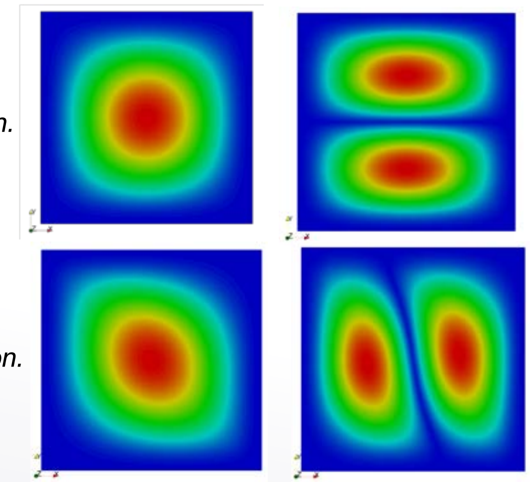
$$\theta = 2(T_2 - T_1)|x|/a + T_1$$



Mode 1 Mode 2

45/90/-45 Lamination.

Tow angle Lamination.



h/a		Mode 1	Mode 2	Mode 3	Mode 4	Mode 5	Mode 6	Mode 7	Mode 8	Mode 9
		[<0,45>,<-45,-60>,<0,45>]								
0.01	Reference	92,26	130,82	195,19	237,86	274,99	282,67	340,09	389,10	431,02
	Present Model	94,98	136,82	209,12	247,71	288,46	312,12	363,04	437,19	479,90
0.1	Reference	614,11	909,55	1233,02	1338,63	1485,64	1798,60	1932,28	1965,59	2152,26
	Present Model	613,22	909,97	1218,36	1334,12	1472,13	1775,87	1933,61	1937,41	2116,02
		[<90,45>,<60,30>,<90,45>]								
0.01	Reference	113,18	145,25	212,66	269,06	292,47	316,49	362,78	392,79	465,04
	Present Model	113,93	148,23	221,81	278,49	303,42	331,16	380,58	425,19	507,75
0.1	Reference	682,20	917,49	1304,68	1313,59	1466,64	1714,97	1920,80	1991,02	2001,10
	Present Model	671,60	906,88	1269,61	1299,70	1443,23	1692,44	1829,70	1904,92	1943,44

Reference: Akhavan H., Ribeiro P.. «Natural modes of vibration of variable stiffness composite laminates with curvilinear fibers» *COMPOSITE STRUCTURE,* Vol.93, pp.3040-3047, 2011.

Final Remarks

- These activities have led me to compare different methods to model an aeronautical structure exploring the characteristics of each method.

- Through the *training on the job*, the company work flow has been learned overcoming the different difficulties that may arise. The structure of a complex fuselage has been analyzed using commercial software. It has been optimized in order to be able to deal with the wing loads.

- A CUF 1-D model based on the Lagrange expansion is extended for the analyses of complex structure characterized by tapered shapes and multicomponent scenarios using a Component-Wise approach. The results confirm that the present model can deal with this topic providing accurate 3D-like results comparable with those obtained from an expensive analysis performed with a commercial code.

- A deeply damage analyses are performed on different aeronautical structure for damage detection purpose. Simple tapered structure and complex wing box are investigated. The model is able to detect the effects on the frequencies and on the modal shapes due to local and global damages. The following notes can be listed:
  - Low damage can introduce effects only on the high frequencies. For this reason, a model for the damage detection has to be able to detect with accuracy a wide range of frequencies.
  - The natural frequency analyses must be performed considering both the frequencies and the modal shapes. In fact, damage can affect some frequencies and not the related mode. The opposite is also possible.
  - In a wing structure, a damage can influence the bending-torsional coupling effects. It's important to evaluate if the new condition is suitable in term of aerodynamic and aeroelasticity. Thanks to its efficiency and accuracy, the model can be used to create a database of multiple damage scenarios which can be useful for maintenance reasons and damage detection.

- Tapered shapes and sweep angle have been implemented in several complex structures for tailoring analyses. The torsional-bending couplings and the possible alterations induced by the tailoring process is also investigated. Using particular laminations, the frequencies can be shifted to the suitable values without modifying the geometry and the coupling effects can be emphasized or reduced. These feature can be of great interest in the aeroelastic design of wing structures.

- The model is extended to deal with TOW placement problem. Results confirm the capability to describe a structure with the fiber direction governed by a control law. In this way the structure can be designed with a variable stiffness. Problems of VSCL (Variable Stiffness Composite Laminate) are investigated providing very good results. This technology is of great interest for the control of the coupling effects and the aeroelastic behavior without affect the stiffness in other areas of a wing box (for example the root of the wing).