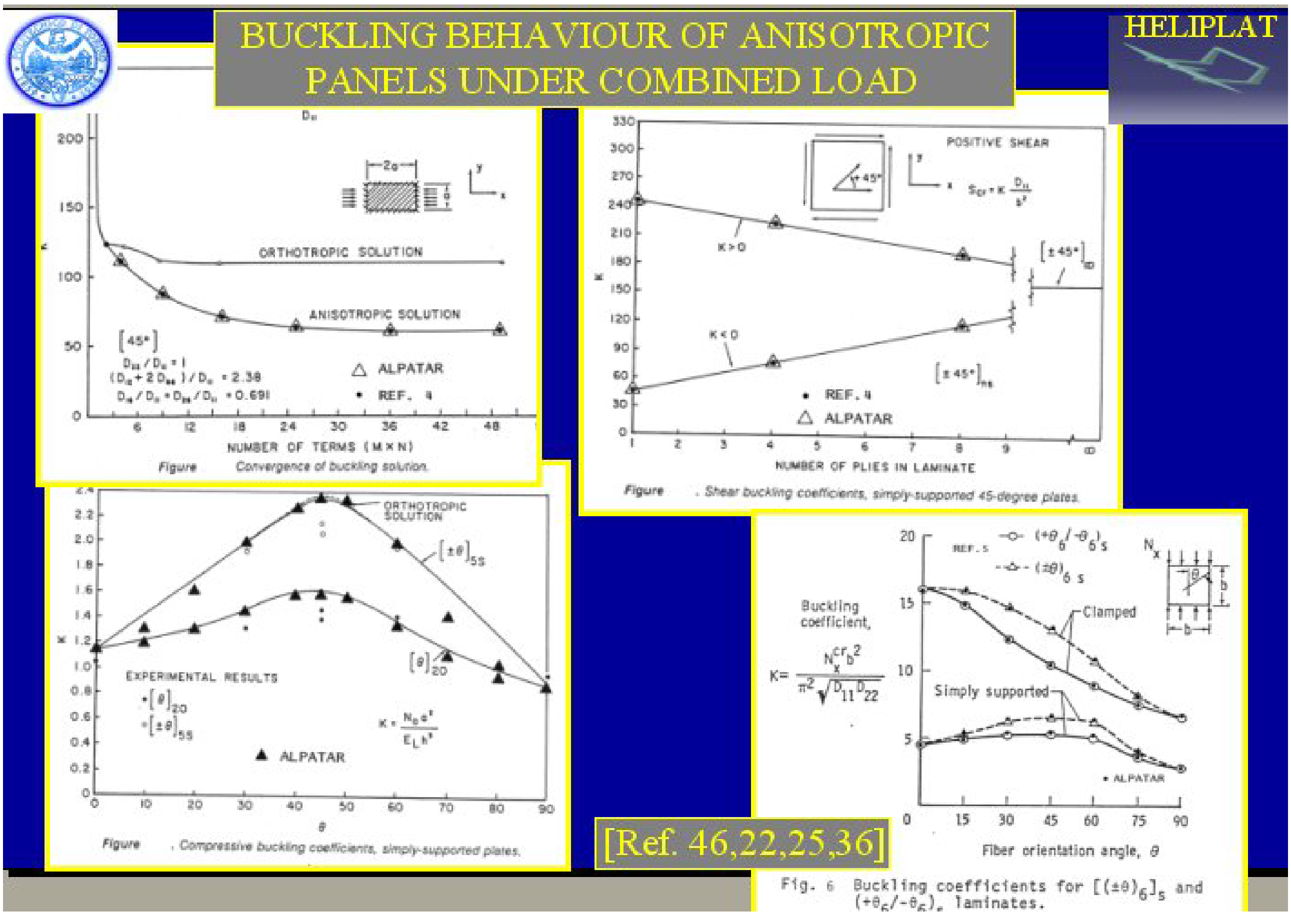
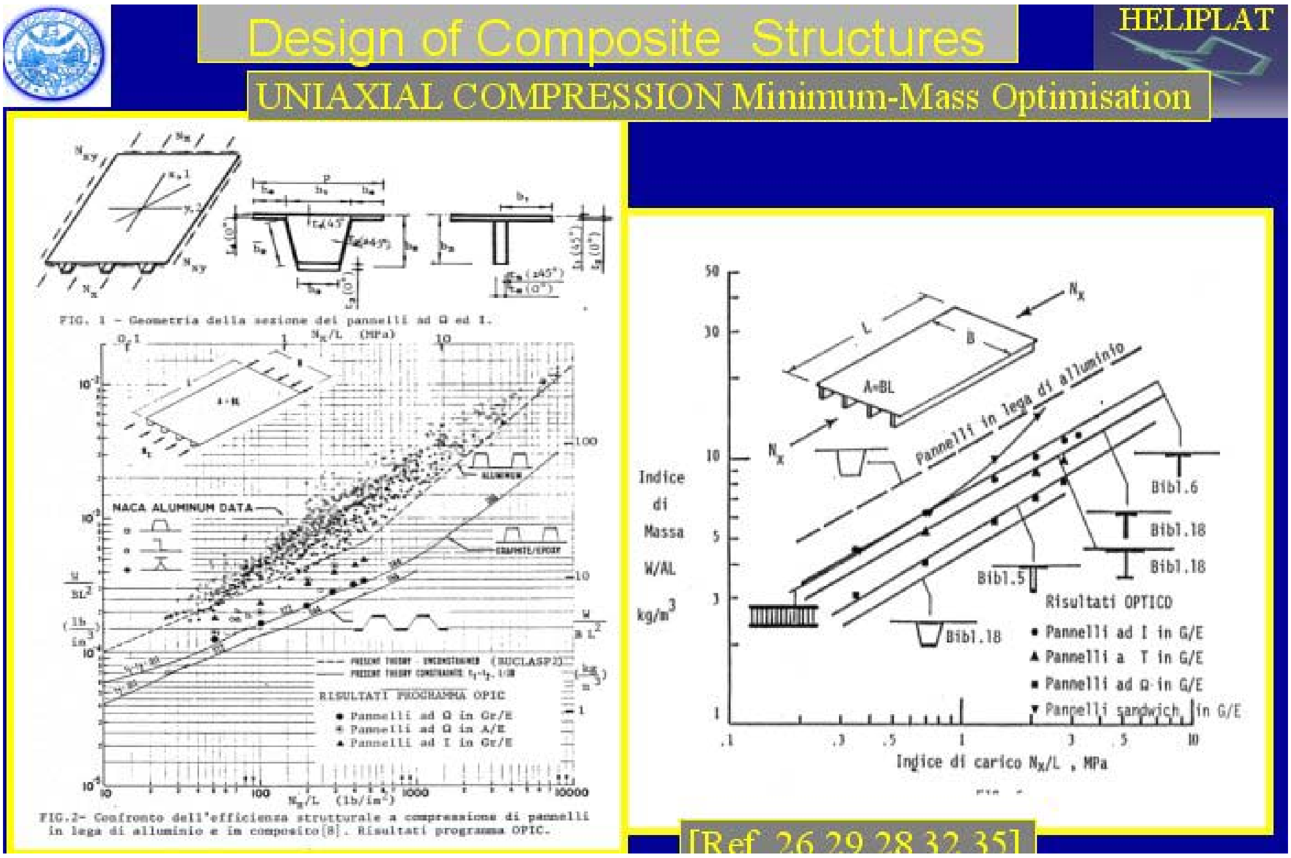
TITLE OF THE RESEARCH: Numerical/Experimental Analysis and Design of composite structural components for aerospace applications

RESEARCH LEADER : Prof. Ing. Giulio ROMEO

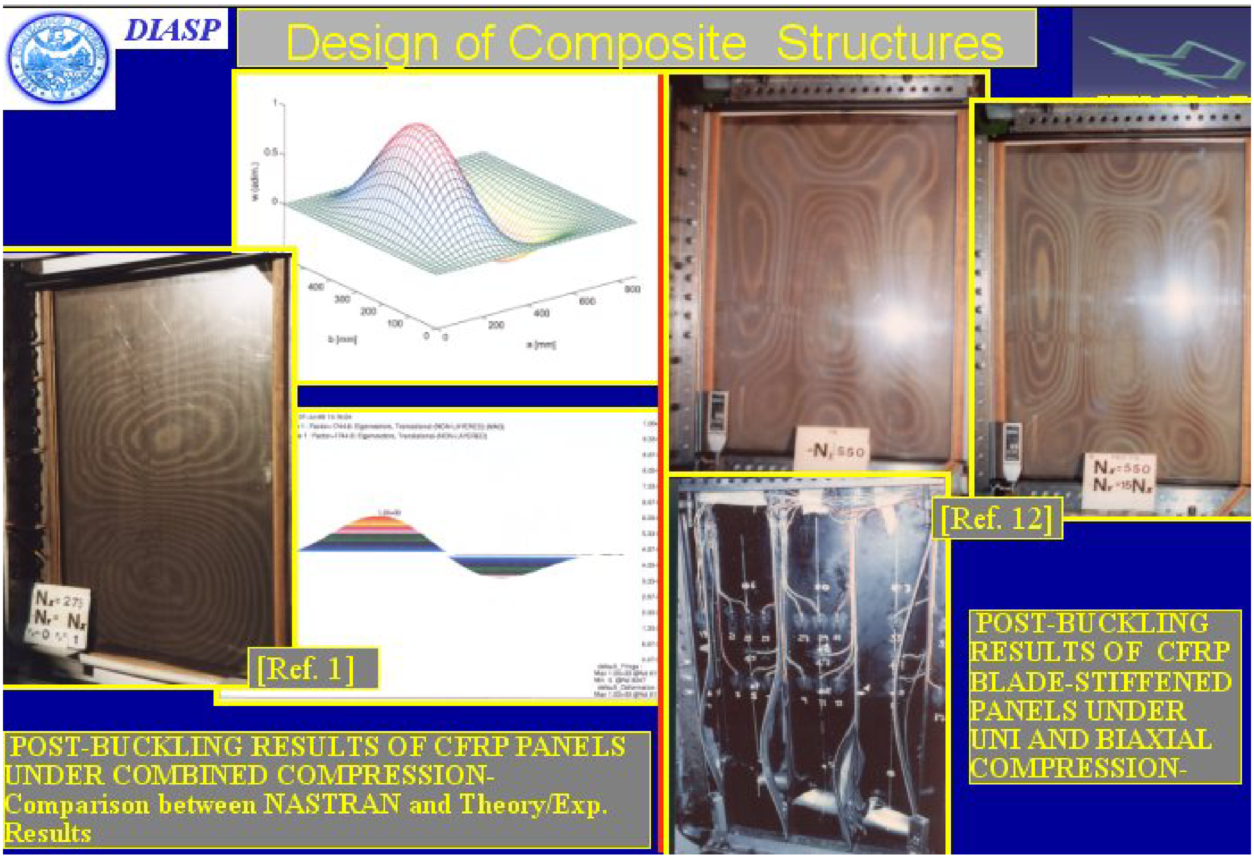
DESCRIPTION:

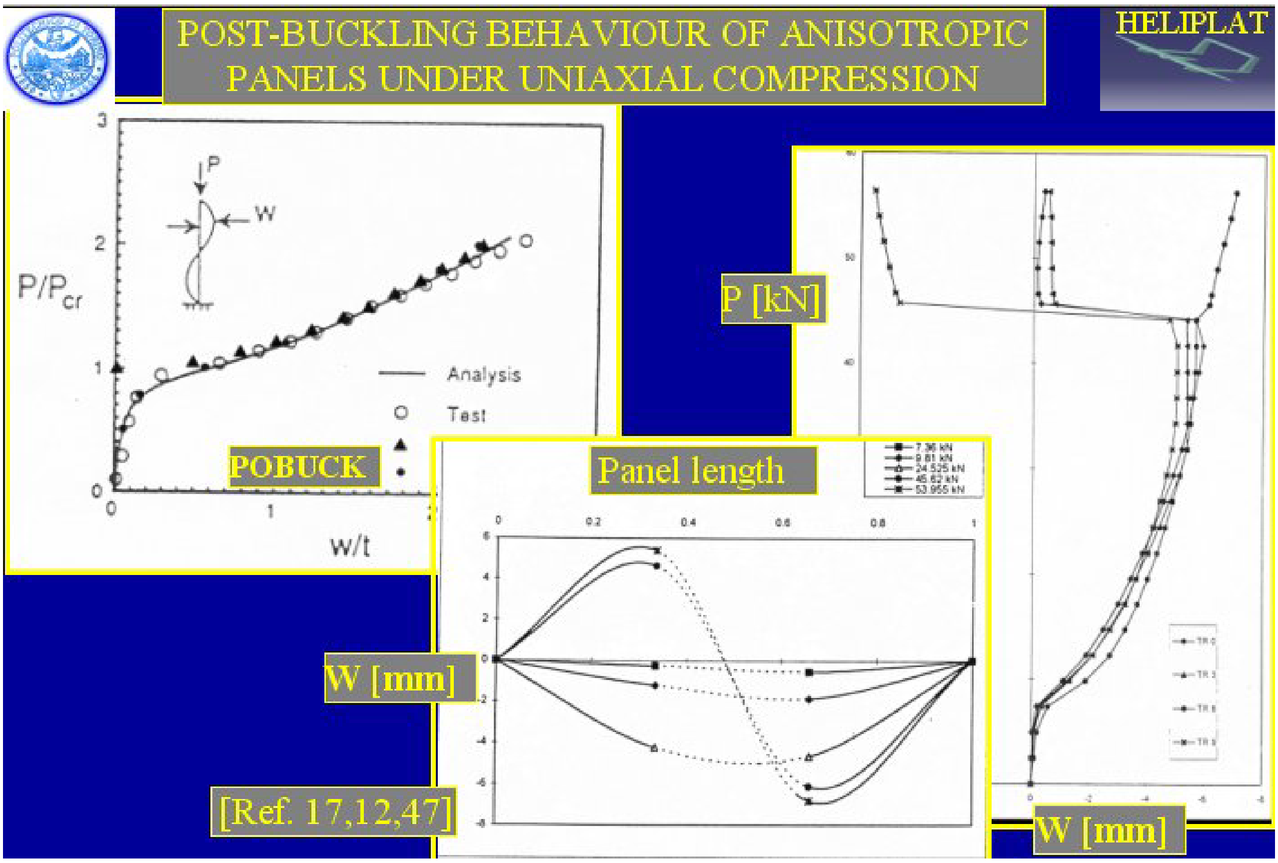
Composite material exhibits a very attractive characteristics from the structural design point of view. Low specific mass connected to high mechanical properties make them as one of the main interesting materials for aerospace applications. The increase of the structural efficiency on the basis of the mission and airworthiness requirements has to be determined at the end of the structural design process. Weight reduction is reached by means of specific shape design of the structural component addressing it to a thin-shape structural concept taking into consideration their critical behavior. While metallic constructions are usually designed to work in the post-critical field up to limit load conditions, the composite counterpart are required to be absent by critical phenomena up to 90% the limit load condition also if experimental results show that composite structures have a wide post-critical behavior before failure.
Buckling analysis of flat/curved panels have been developed by means of Donnell approximate theory and Galerkin method. The approximate solution is valid for width to thickness ratio higher than 40 and curvature radius to thickness ratio higher than 500. In the other cases the transverse shear effect have to be included in the analysis in order to determine its effect on the buckling behavior. A first approximate solution takes into consideration the linear contribution of the transverse shear deformation. Higher order solution are considered such as indicated by Reddy where such effect is included into the displacement field up to cubic order. The shear effect is important in the investigation of buckling and post-buckling of sandwich panels.



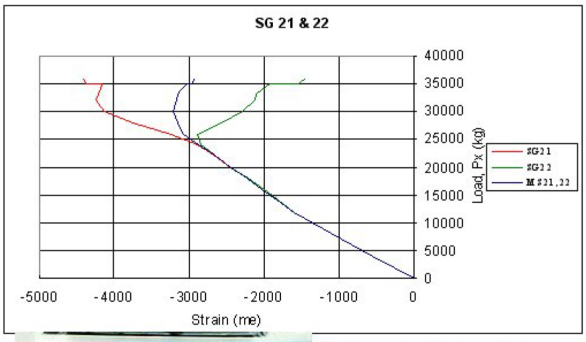


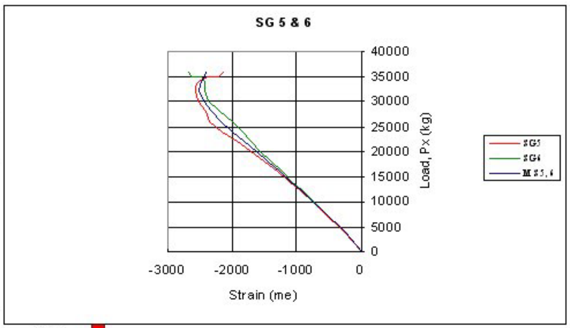
An analytical solution has been developed in order to investigate post-buckling behavior of simply supported and/or fully clamped anisotropic plates under combined biaxial compression and shear load. Since panels are often not perfectly manufactured, initial imperfections are also included in the analysis. Such imperfections in fact considerably influence the out-of-plane displacement behavior of the panel and cannot be ignored. The classical laminate theory has to be modified for post-buckling analysis. The non-linear part of the strain-displacement relation is taken into consideration on the basis of the Marguerre approximate nonlinear theory including the effect of the initial imperfection.

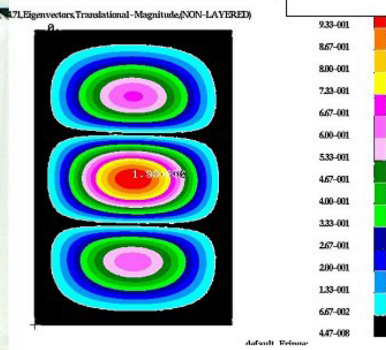




-) Biaxial load experimental results (-1,-0.3,0)

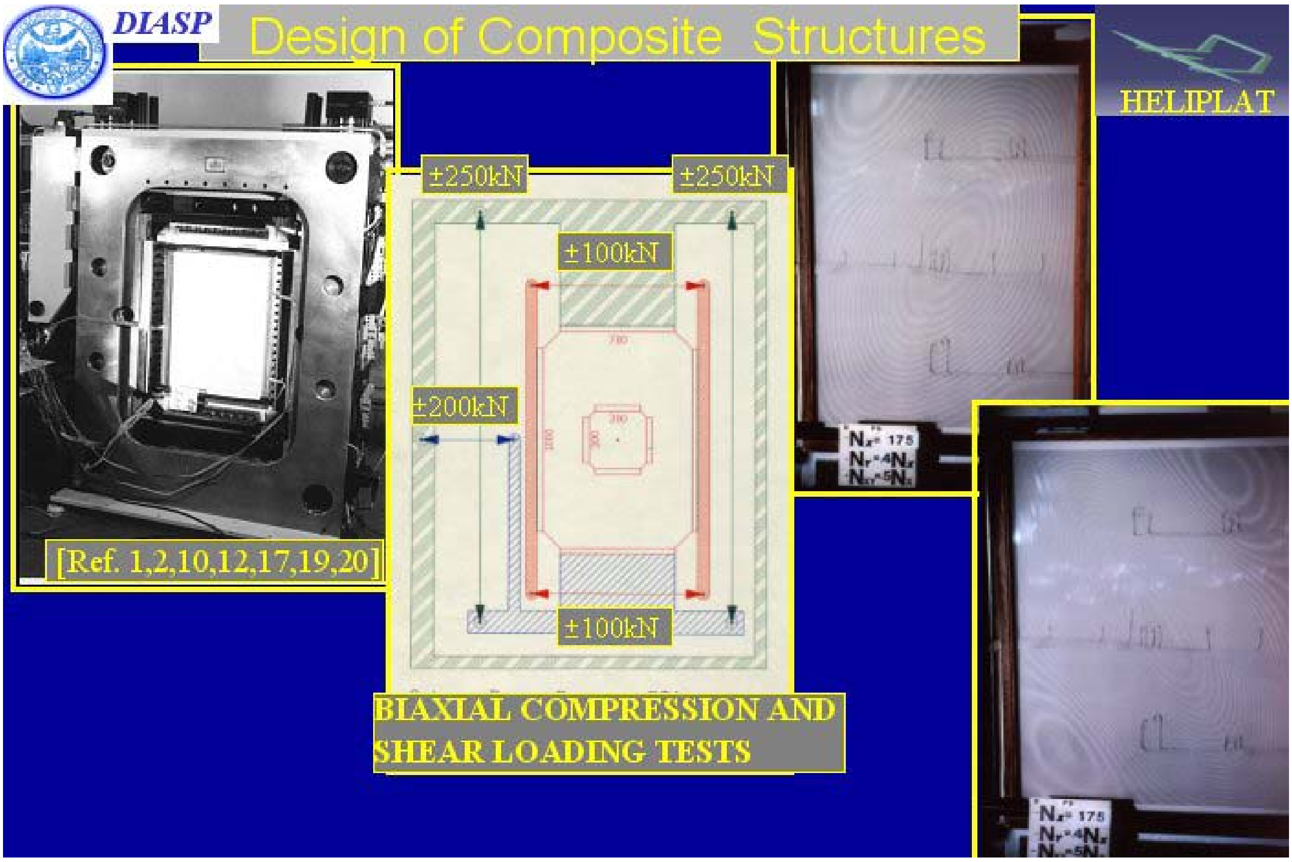






Strain results (mid longitudinal position).

experimental buckling load : 294kN



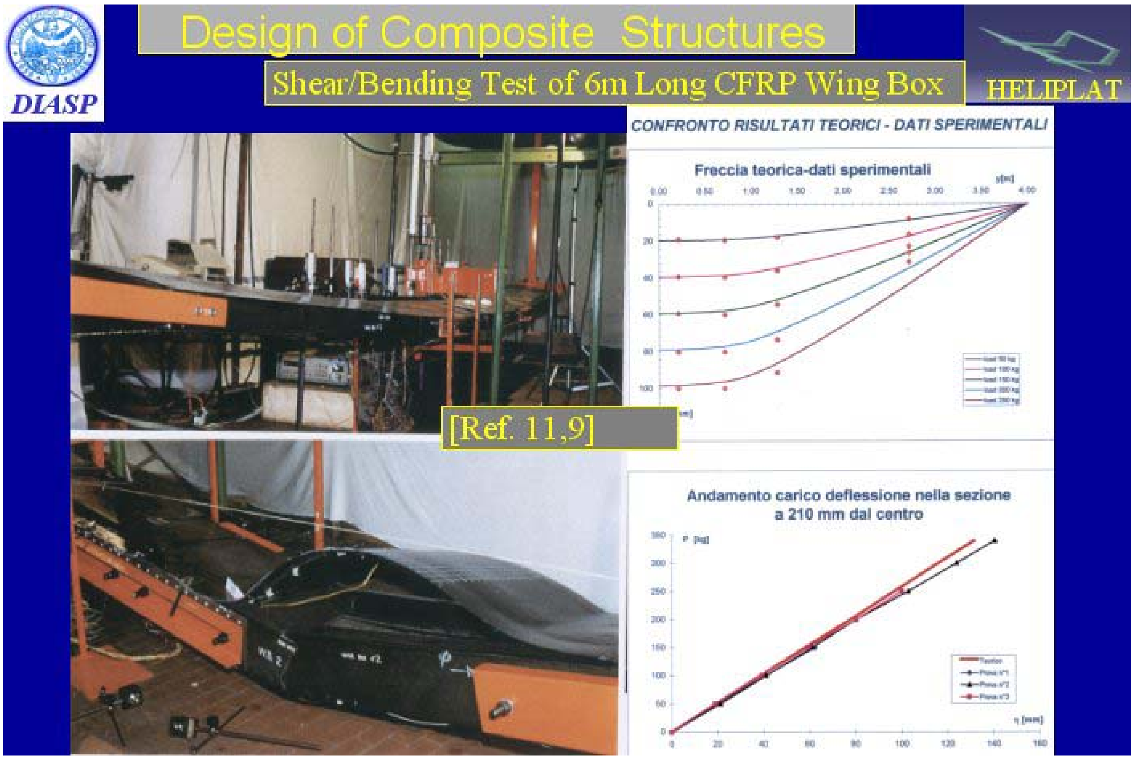
The stationary value of the total potential energy produces the three governing equilibrium equations and boundary conditions. By means of Airy function the in-plane equilibrium equations are identically satisfied while the third has to be solved with the aid of compatibility equation. A two equation in two unknown is finally worked out. Four kind of boundary conditions are considered: ends and sides simply supported, sides simply supported ends clamped, ends simply supported sides clamped, ends and sides clamped. The approximate assumed functions are chosen accordingly and the approximate solution on the basis of Galerkin procedure is worked out. The algebraic non linear system obtained by Galerkin method is solved out using a library subroutine. The obtained results correlate very well with the ones found in the open literature including the effect of the initial imperfection and post-buckling shape change. The problem of the correct loading distribution is also considered assuming that two kind of loading situation arise: load control case and displacement control case. In the case of loading control case, the compressive load is assumed at different level and the post-buckling path is determined. In the case of displacement control case, the end shortening of the panel is assumed and the post-buckling path is determined. In this second case the condition for the ends and sides to be straight causes the load distribution to change after buckling condition has been passed. This fact is quite important in the determination of transverse deflection that can be quite lower than the load control case in the post-buckling regime. In specific situation a different load configuration has to be considered: a linearly varying load along the panel edges. It is a typical loading condition for various structural elements such as box beam webs, fuselage panels and launch vehicle. Buckling behavior of such kind of loaded anisotropic panel was determined. The Rayleigh-Ritz method was used to determine the buckling load under different boundary condition and loads. Several interaction curves were obtained for various plate aspect ratios and load combinations. The characteristic clamped-clamped beam functions have been used to satisfy the boundary conditions in a better way.

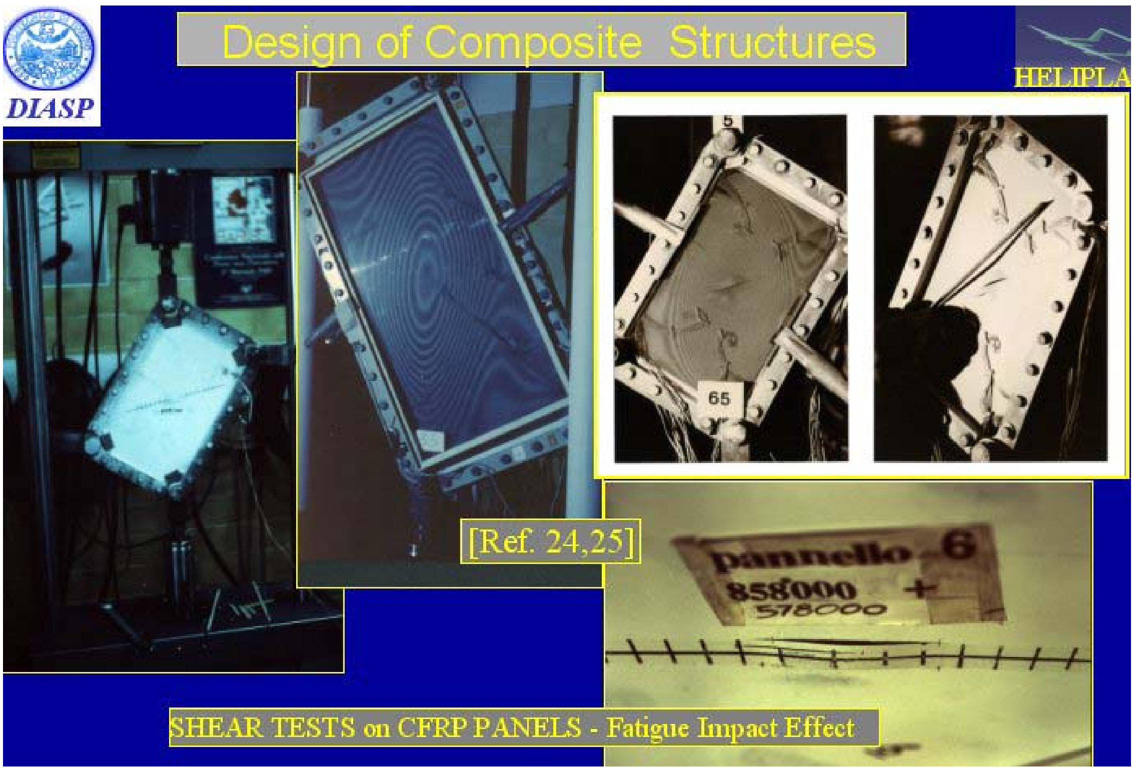
Stiffened composite panels were designed in the past to not allow skin local buckling; however a lower mass could be achieved if it is permitted to the skin panel to work in the post-buckling range where a wide post-buckling behavior is determined before failure. The presence of the stiffeners allows the panel to be subjected to a control displacement situation causing a variation in loading distribution after panel buckling. The longitudinal compression is applied along the centroidal axes of the stiffened section while the transverse load is applied to the skin resulting in an eccentricity between the loading case: a couple is introduced in the system. Several types of buckling have to be taken into consideration in studying stiffened panels: overall buckling, buckling of the skin between two stiffeners, torsional or twisting buckling. The overall buckling can be determined by Galerkin method and smeared stiffness approximate formulation, that is not of course strictly sound but a rapid solution is possible for optimization procedure design. The torsional buckling has to include the bending-torsion effect (Wagner theory) and the skin reaction over the classical theory contribution in order to give an exact estimate of the critical condition.

Standard approach for buckling and post-buckling of aircraft shells assumes them as in-plane loaded and separated from the main structures. Some important effect arise when they are considered included into a global structure such as a box beam or a fuselage and so on. In the case of wing boxes under pure bending there is the presence of a typical crushing pressure function of applied bending moment that influences the transversal members design and the out of plane skin behavior and has to be included in a non-linear formulation of the skin problem in order to define in a better way the boxes behavior. A similar crushing pressure is originated by torsion moment applied to wing boxes under pure torsion with different shear load regimes according to pre or post-critical conditions occur.

A new test facility was built at the Polytechnic of Turin in order to apply simultaneously both biaxial compression and shear loads. The longitudinal load is applied separately from the transversal and shear loads and are separately controlled via electronic modules that closed the loop by means of load cells and transducers. A maximum of 600kN longitudinal (tension/compression) ,200 kN transversal and shear loads can be applied by the machine to 1000x700mm solid/stiffened panels. The possibility of a cyclic load condition is possible at low frequency.

A wide testing activity has been performed over structural component in full scale such as wing boxes (24m span) and primary structure aircraft. Typical shear bending test by means of the treeloading configuration have been developed including testing set-up and manufacturing. Fatigue loading condition are also investigated over single details such as junctions and panels. The advanced junction configuration was designed and tested under cyclic load in order to verify the well behaved static behavior also in presence of fatigue load. The average load applied to the pin-hole junction correspond to the failure load of the composite joint without the reinforcement. Maximum load higher than the previous is maintained during fatigue. A satisfactory result is determined up to 10^6 cycles.



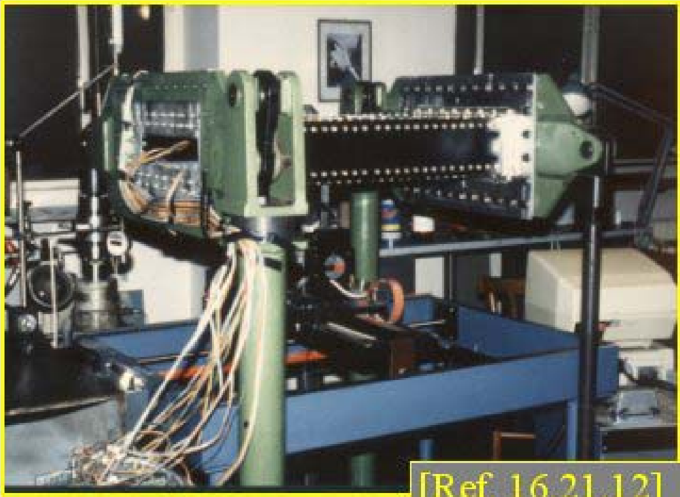


DIASP

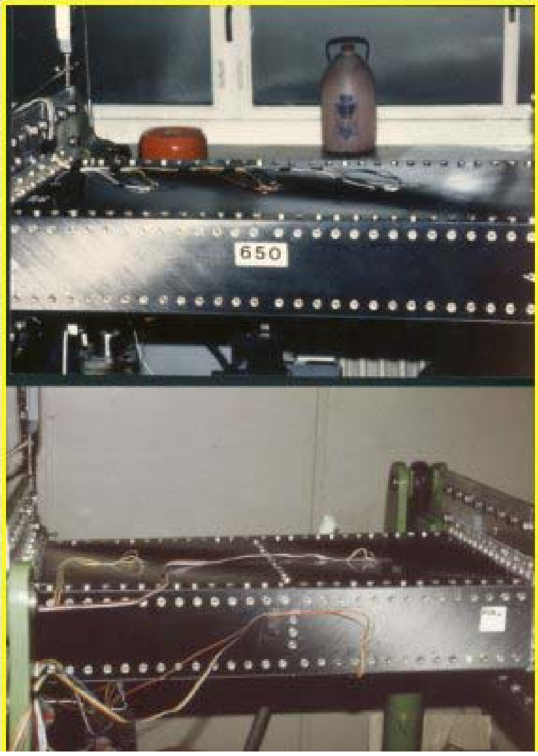
Design of Composite Structures

CFRP Wing Box in TORSION

HELIPLA



[Ref. 16,21,12]

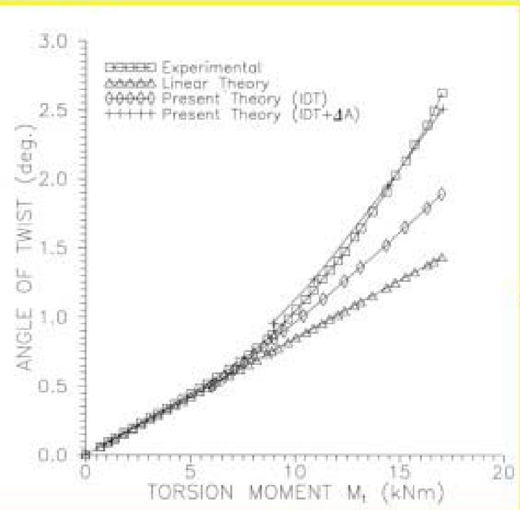


DIASP

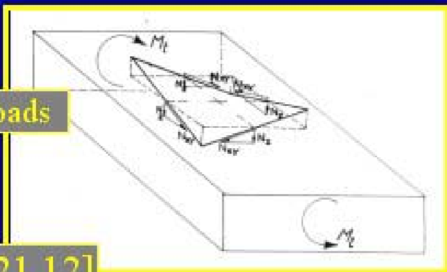
Design of Composite Structures

CFRP Wing Box in TORSION

HELIPLAT



Crushing Loads



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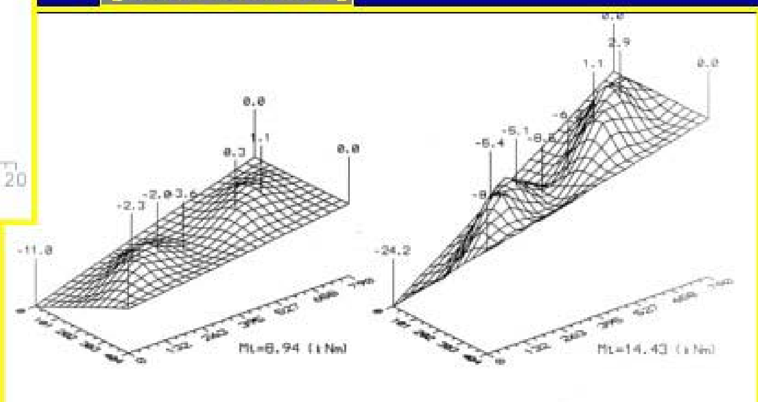
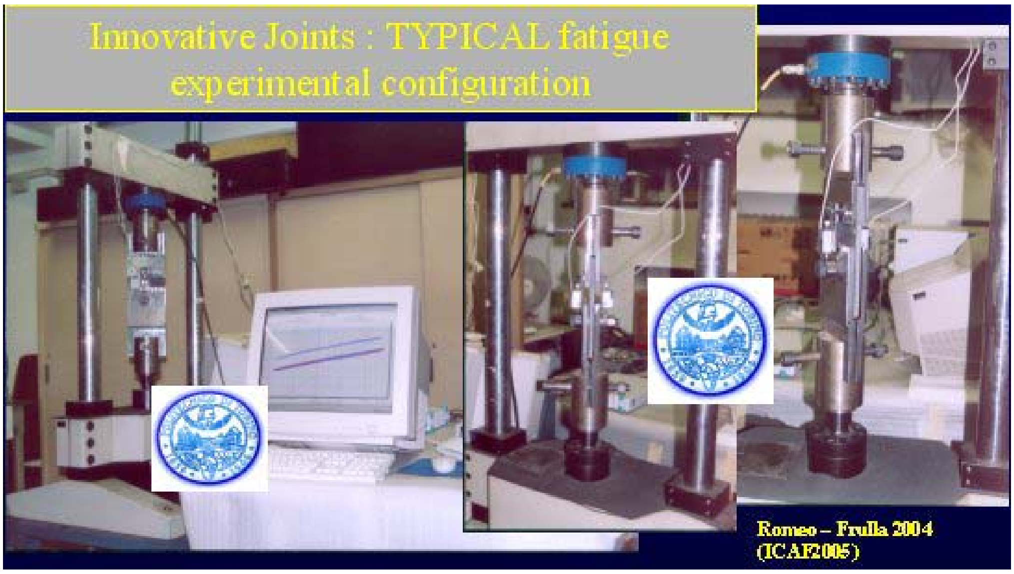
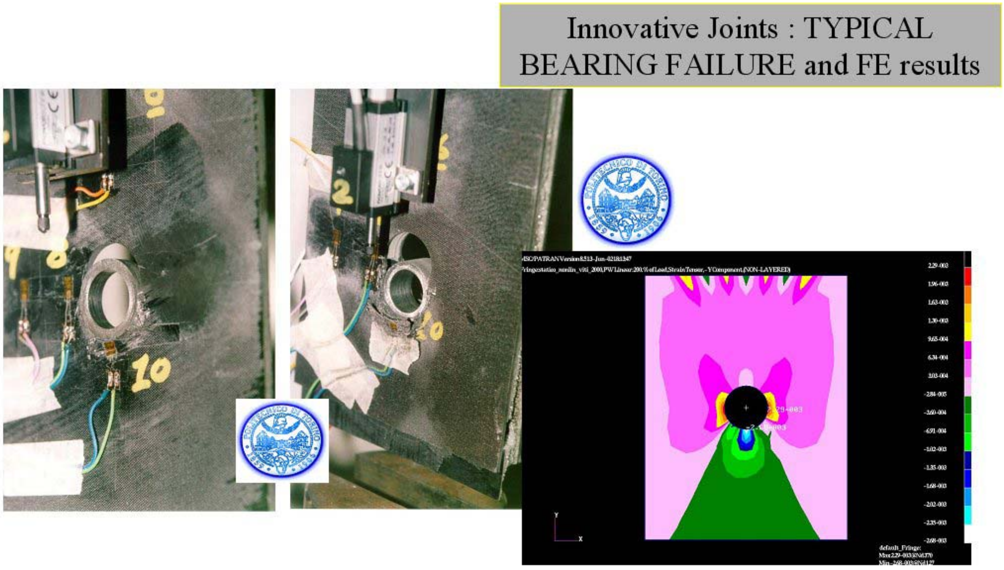
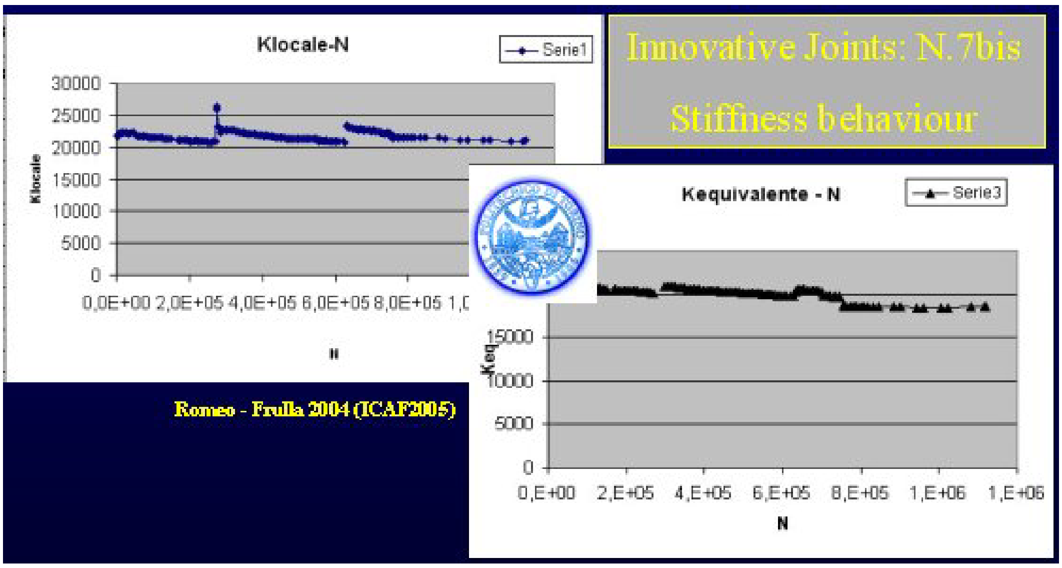



Innovative Joints : TYPICAL fatigue experimental configuration
Romeo – Frulla 2004 (ICAF2005)


Innovative Joints : TYPICAL BEARING FAILURE and FE results


Klocale-N
Serie1
Klocale
30000
25000
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0,0E+00 2,0E+05 4,0E+05 6,0E+05 8,0E+05
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Innovative Joints: N.7bis
Stiffness behaviour
Romeo - Frulla 2004 (ICAF2005)
Kequivalente - N
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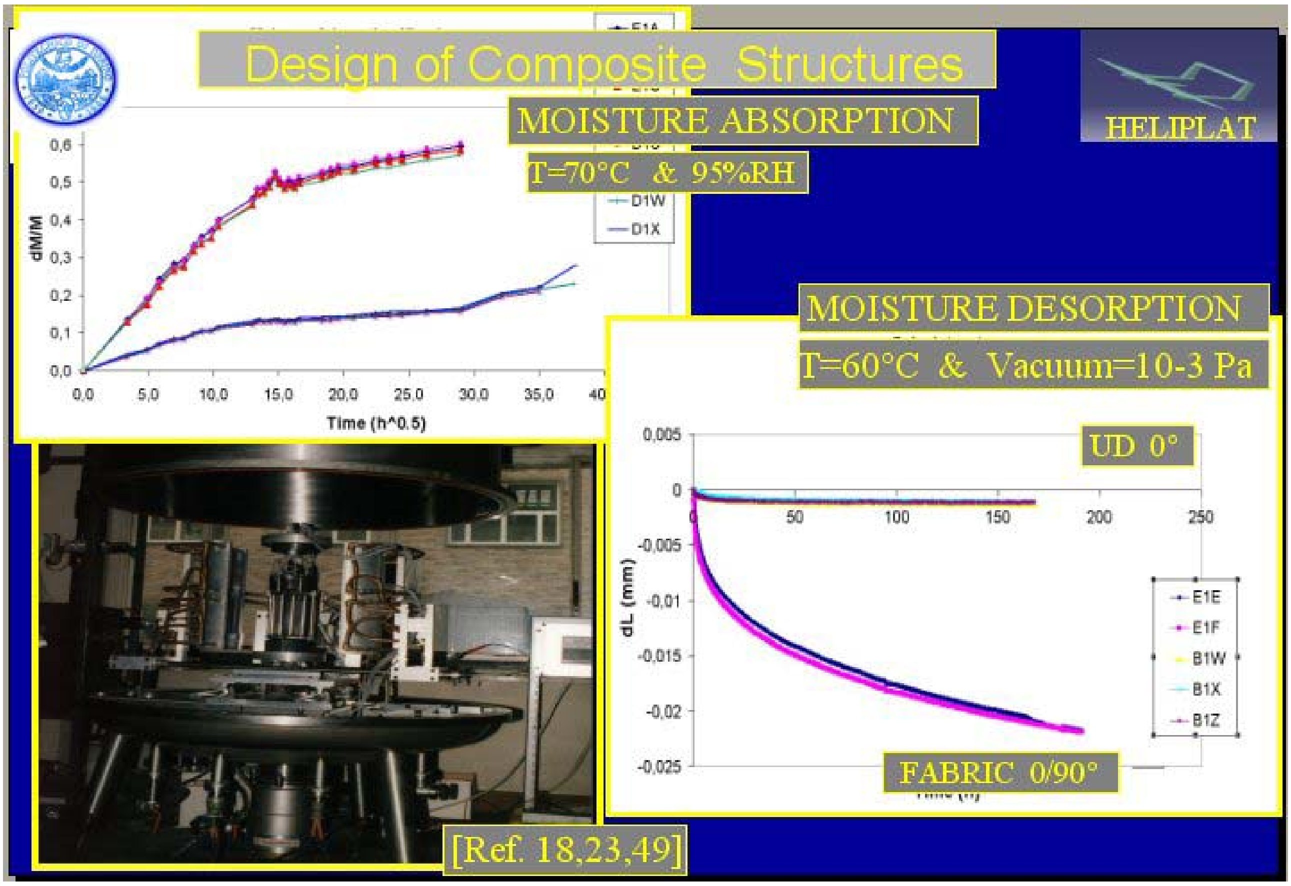
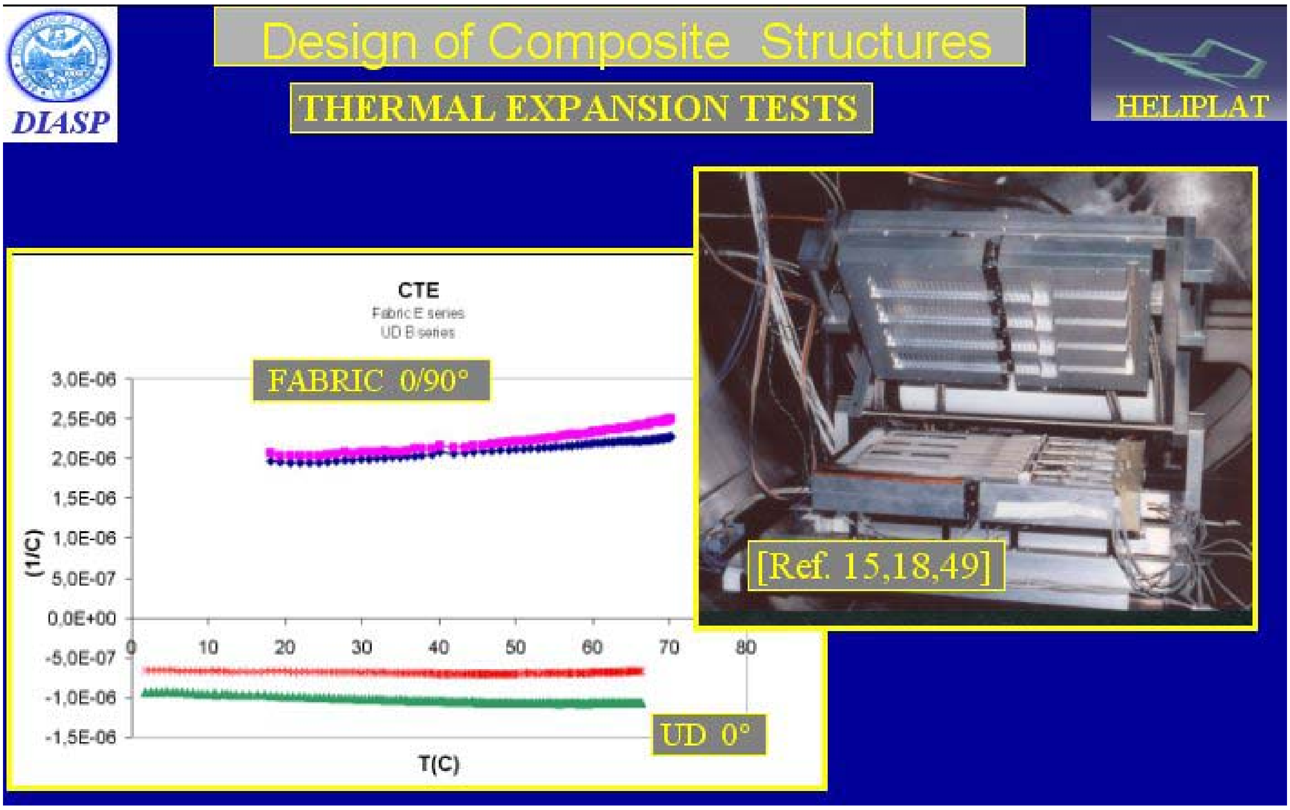


Design of Composite Structures
MOISTURE ABSORPTION
T=70°C & 95%RH
HELIPLAT
D1W
D1X
dM/M
Time (h^0.5)
MOISTURE DESORPTION
T=60°C & Vacuum=10-3 Pa
UD 0°
dL (mm)
E1E
E1F
B1W
B1X
B1Z
FABRIC 0/90°
[Ref. 18,23,49]



Design of Composite Structures
DIASP
THERMAL EXPANSION TESTS
HELIPLAT
CTE
Fabric E series
UD B series
FABRIC 0/90°
(1/C)
T(C)
UD 0°
[Ref. 15,18,49]

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