Proof of Concept for New Composite Structures of Aircraft

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ABSTRACT

The integration technology by composite materials yielded big part-count reductions in addition to weight savings .The strength of wing box was evaluated by ultimate static test and residual strength tests with full scale component after exposed to BVID, VID and DSD. The no-growth of BVID was proved by PIF test with a 3-stringer panel. A-SCAN and visual inspection are useful as FITs for foam core sandwich outer panes of cockpit. The integrated composite sandwich panel can be expected to reduce remarkably the assembly cost. FAA DER gave us many good comments and effective recommendations for compliance with FAR 25.

So far, it is said that there are thresholds for PIF/BVID at about 60% of initial strength. However, recent test data show no threshold and under-threshold criteria is not applicable to PIF/BVID. On the other hand, no-growth design can be done as for flawed holes and matrix crack initiation. Fastener joints and materials itself have finite fatigue lives. For these items, the speedy TTSP/ATM approach is effective to estimate. As concerns wing and cockpit structures, no-growth design was not completed, but 2-life design target is attained because of the low working strains. More mass save is expected if working strains are increased under the high threshold given by resign toughness

1. INTRODUCTION

Many composite materials are applied to new aircraft to be developed before long, e.g., EASA (Environmental Adapted Small-size Airplane) or B7E7. The percentage of composite materials is expected as over 40% of structural weight by application not only to tail and control surfaces but also to primary structures such as wing. Some new design approaches are needed to use new material instead of aluminum alloy reigning the aircraft structures for long in order to ensure flight safety.

For the purpose of proof of concept (POC) for new composite structures, a composite wing box made by Resin Transfer Molding (RTM) and pressurized composite/foam core sandwich panels for cockpit structure applicable to civil transports were developed under a five-year (1999-2003) R&D program entrusted by NEDO (New Energy and Industrial Technology Development Organization). The targets of wing box was 50% part-count reduction and 20% weight saving, and those of cockpit structure was 80% part-count reduction and 20% weight saving.

Authors have proposed the necessity [1] and the methodology [2] for new design approaches about durability and damage tolerance, especially predictable damage growth design, of composite primary structures. The first of the proposed approaches is time- temperature superposition principle (TTSP) approach based on accelerated test method (ATM) [3] which gives speedy estimation of long-term fatigue life. The second is post impact fatigue (PIF) approach for barely visible impact damage (BVID) [4] and the third is under-threshold criteria approach for hole flaws or matrix crack initiation [4].

2. COMPOSITE STRUCTURES DEVELOPED

2.1 Wing Box

(1) Materials Parts of the wing box were made with new epoxy resin and carbon fiber 3-D textiles T800S/TR-A33, e.g. front and rear spars by RTM, stringers by Vacuum assisted RTM (VaRTM) and with prepreg T800S/3900-2B for upper & lower skin panels. These parts were finally co-bonded to get entire wing box [5]. The feature are showed in Fig.1.

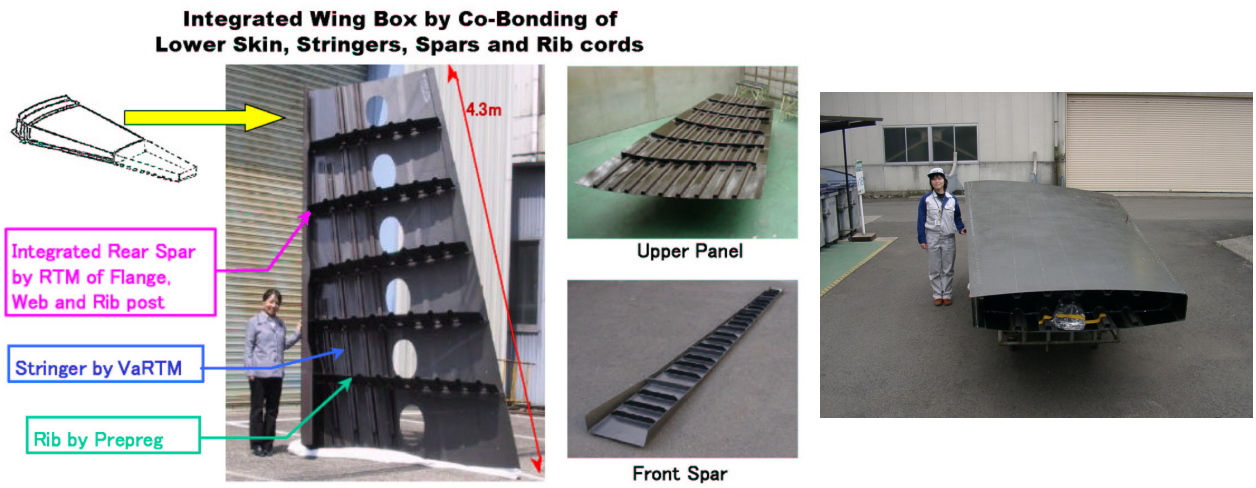


Fig.1 Co-bonded Wing Box

(2) Wing Box Fabrication Process The process from parts fabrication and NDI on the spot to co-bonded assembly and final assembly is shown in Fig.2.

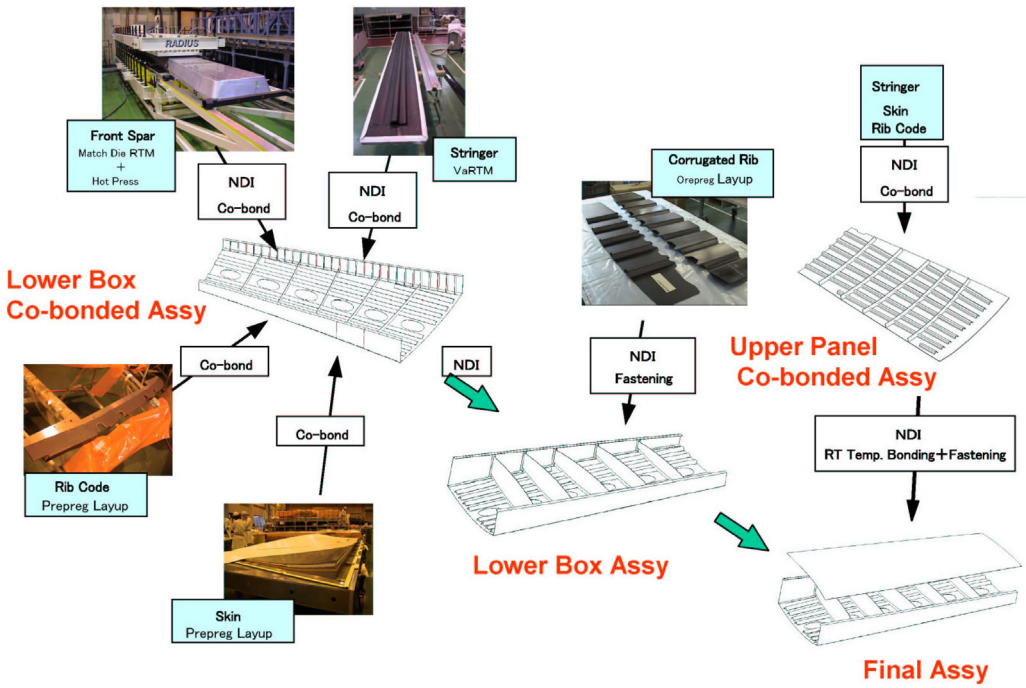


Fig.2 Wing Box Fabrication Process

2.2 Cockpit Outer Panels

(1) Materials The cockpit outer panels were sandwich of foam core and prepreg UT500/#135 skin [6]. The sandwich construction under high internal pressure was selected to reduce part-count. The foam cores having various shapes and curvatures were formed with an automated multi-support mold by heating and pressing. The formed cores and prepreg skins were finally co-cured to get sandwich panels. The features are showed in Fig.3.

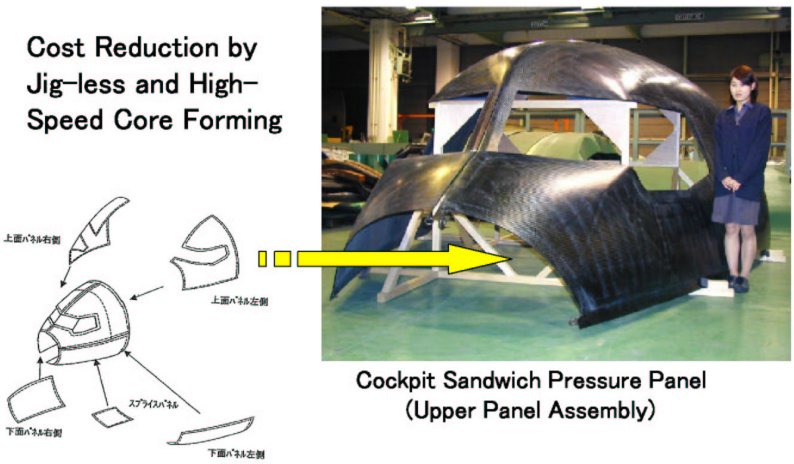


Fig.3 Cockpit Upper Sandwich Panel

(2) Cockpit Outer Panel Fabrication Process

A multi-support device controlled by computer was tried for foam core forming. This device shown in Fig.4 yielded the considerable reduction of jig number in comparison with the conventional mold-type jigs. The process from prepreg layup and bagging to autoclave curing is a conventional method as shown in Fig.5.

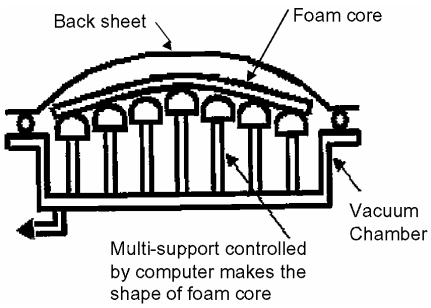


Fig.4 Multi-support device

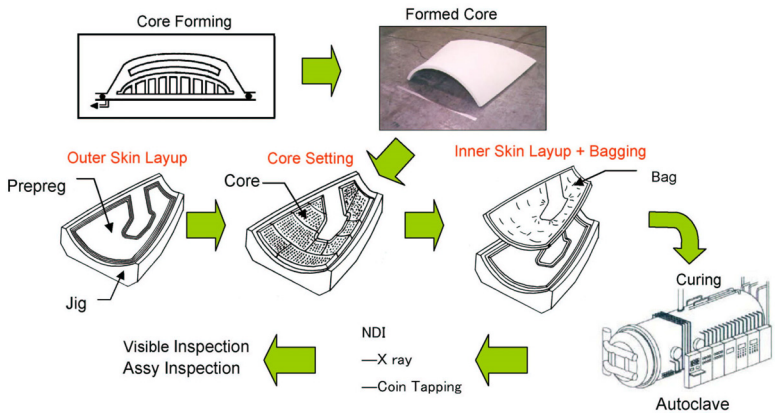


Fig.5 Cockpit Outer Panel Fabrication Process

2.3 Development Derivatives

(1) Part–count Reduction and Weight Saving Table 1 shows the comparison of targets and results. All results satisfied the targets.

Table 1 Comparison of Targets and Results

	Wing Box		Cockpit Outer Panel	
	Target	Result	Target	Result
Part-count Reduction	50%	54%	80%	98%
Weight Saving	20%	27%	20%	23%

(2) Wing Box Full Scale Static and Residual Strength Tests Full scale wing box test were conducted in the following sequence. Strain survey up to 60%LMT load / Stiffness & Residual deformation test up to 100%LMT / Stiffness & Residual deformation test after impacted to BVID at 100%LMT / Ultimate static test with BVID at 150%LMT / Residual strength test after impacted to VID (Visible Impact Damage) at 100%LMT / Residual strength test after exposed to DSD (Discrete Source Damage) up to get home load 70%LMT.

(3) Cockpit Panel NDI Test Some kinds of NDI technique were examined to detect impact damades with full scale cockpit panels. The result is shown in Fig.6. Tapping and MIA (Mechanical Impedance Analysis) gave poor performance for the foam core sandwich panel. A-SCAN, C-SCAN and visual inspection gave good performance. A-SCAN and visible inspection are useful for FITs (Field Inspection Techniques) [7] by its handiness.

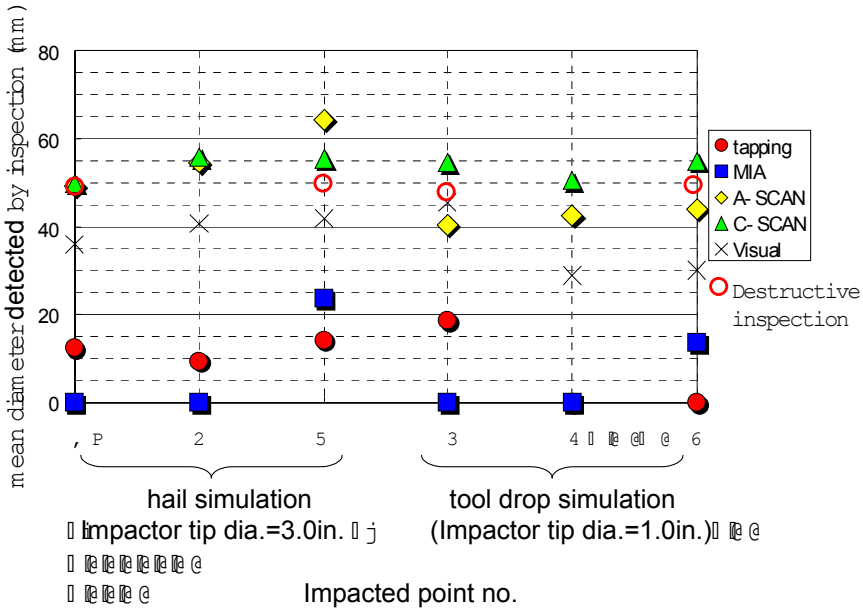


Fig.6 Comparison of NDI Detection Performance

(4)Cockpit Structure Final Assembly The final assembly is shown in Fig.7. Outer panels were assembled by only approximately 700 fasteners. In comparison with about 4,200 rivets

necessary to assemble skins and frames in the conventional aluminum alloy build-up structure, integrated composite sandwich panel can reduce remarkably the assembly cost.

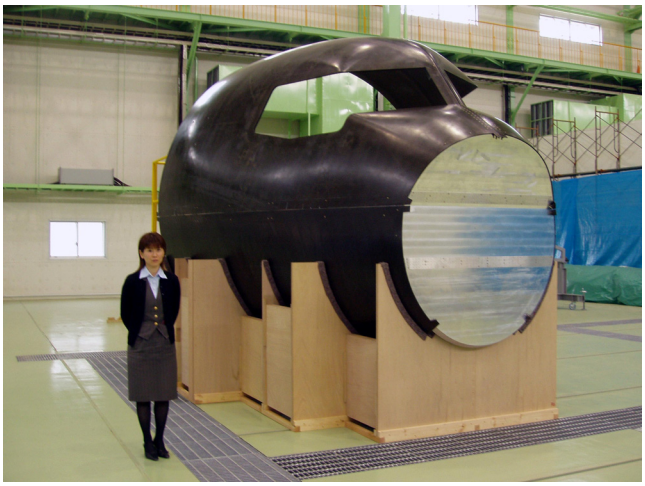


Fig.7 Cockpit Structure Final Assembly

3. DURABILITY AND DAMAGE TOLERANCE EVALUATION

3.1 Design Condition

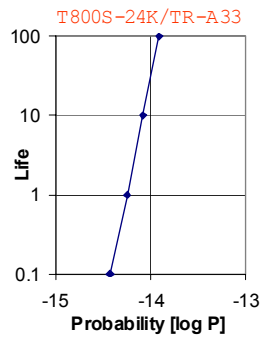
The design conditions are 1 hour / flight, 75,000flights / 20years. A standard load sequence for transport aircraft wing structures TWIST [8] was used for the wing evaluation. For the cockpit, a constant amplitude pressure is repeated each flight. The temperature is repeated each flight from +45 degree C (on ground) to –54 degree C (in flight).

3.2 Fatigue Life Estimation

(1) Fastener Joint Fatigue lives were estimated from S-N data, The B-basis fatigue life of wing root upper joint is 3.4 life. The B-basis fatigue life of cockpit sandwich panel miter joint is 250 life and the endurance up to 4.8 life (not failed) was verified by fatigue test of joint specimen.

(2) Creep Creep endurances of the three materials were estimated by ATM/TTSP. Looking at one result in Fig.8, probability of creep rupture is extremely remote (under 10^{-14}).

(3) PIF/BVID BVID grows under repeated loading and the residual strength reduces. However, the PIF lives are very long because of the low working strains. The B-basis PIF life of wing upper panel stringers is estimated as 2,400 life (Fig.9) and no-growth was confirmed up to 2 lives by the fatigue test of 3-stringer panel under enhanced loading by factor 1.15. PIF life of cockpit sandwich panel is extremely long as the main repeated load is tensile under pressurization.



Fog.8 Creep Endurance

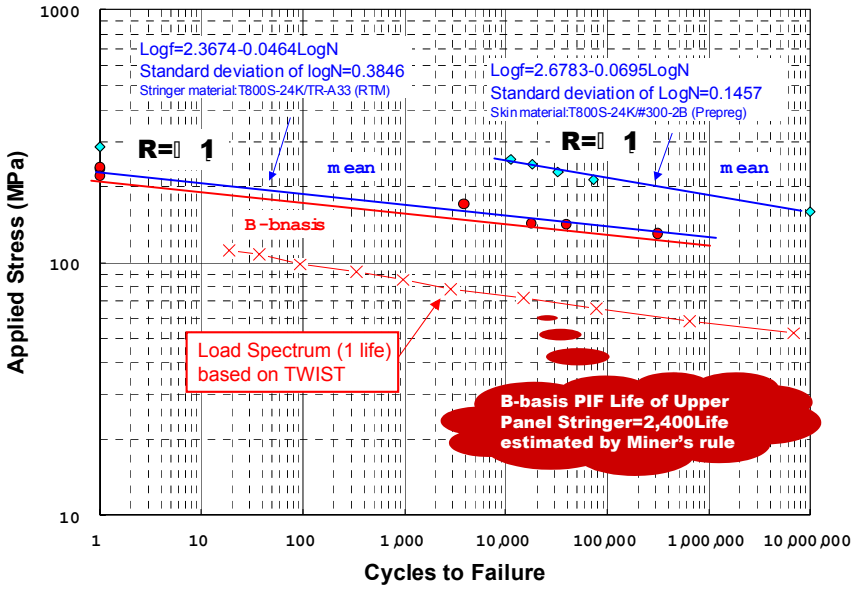


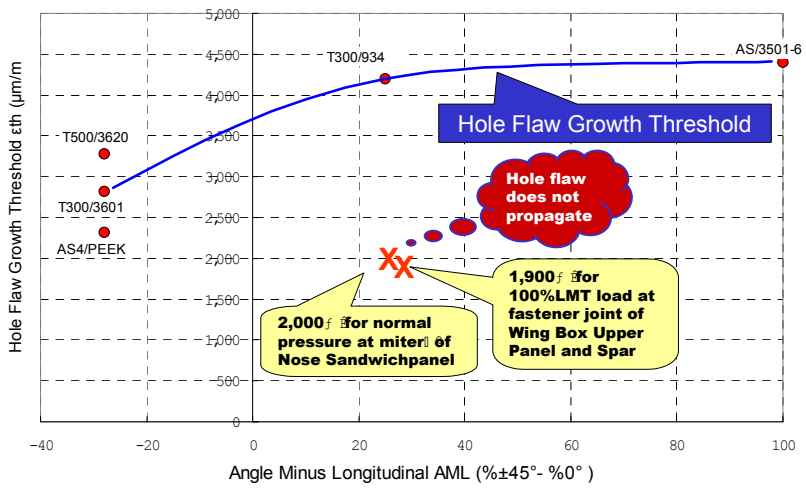
Fig.9 PIF/BVID of Wing Upper Panel

3.2 Damage Tolerance Evaluation

Thresholds under which damage does not grow are very high as to composite materials in comparison with aluminum alloy. Therefore, under-threshold design, i.e. no-growth design, can be done by setting the working strain under the threshold.

(1) Flawed Hole by Drilling The working strains of wing upper panel and spar, and nose sandwich panel are below the threshold of flawed holes specified by the hole accept or reject criteria. The threshold against working strains is showed in Fig.10.

(2) Matrix Crack Initiation As showed in Fig.11, the working strains wing lower panel and cockpit sandwich panel are below the threshold.



.Fig.10 Flawed Hole Growth Threshold and Working Strain

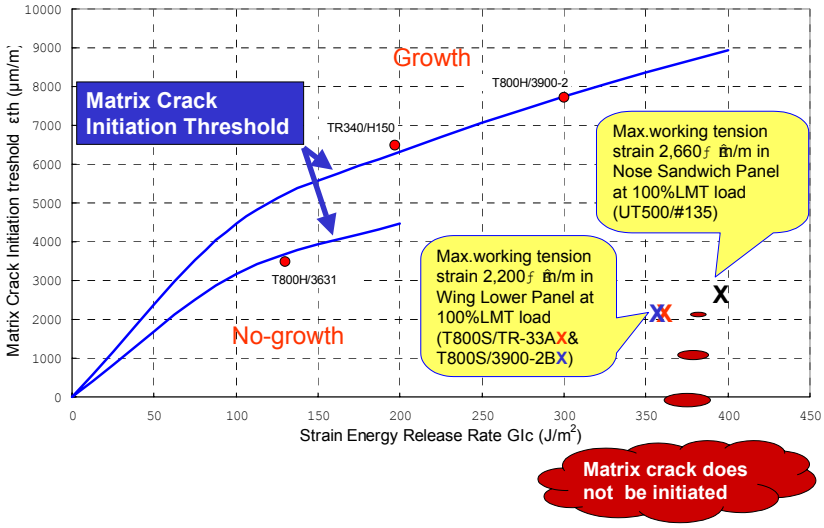


Fig 11 Matrix Crack Initiation Threshold and Working Strain

4. RESULTS & DISCUSSION

(1) Composite Structures Developed

(a) The integration technology by composite materials yielded big part-count reductions (54% - 98%) in addition to weight savings (27% - 23%).

(b) The strength of wing box was evaluated by ultimate static test and residual strength tests with full scale component after exposed to BVID, VID and DSD. The no-growth of BVID was proved by PIF test with a 3-stringer panel.

(c) A-SCAN and visual inspection are useful as FITs for foam core sandwich outer panes of cockpit. The integrated composite sandwich panel can be expected to reduce remarkably the assembly cost

(d) FAA DER (Designated Engineering Representative) reviewed our activities and gave us many good comments and effective recommendations for compliance with FAR 25.

(2) Durability and Damage Tolerance Evaluation

(a) So far, it is said that there are thresholds for PIF/BVID at about 60% of initial strength. However, recent test data in Fig.12 show no threshold and under-threshold criteria, i.e. no-growth design is not applicable to PIF/BVID.

(b) On the other hand, no-growth design can be done as for flawed holes and matrix crack initiation (Fig.10, Fig.11).

(c) Fastener joints, materials itself (repeated loads, creep, moisture absorb/exhaust, cyclic temperature, etc.) have finite fatigue lives. For these items, the speedy time-temperature shift factor approach is effective to estimate.

(d) As concerns wing and cockpit structures, no-growth design was not completed, but 2-life design target is attained because of the low working strains. More mass save is expected if

working strains are increased under the high threshold given by resign toughness.

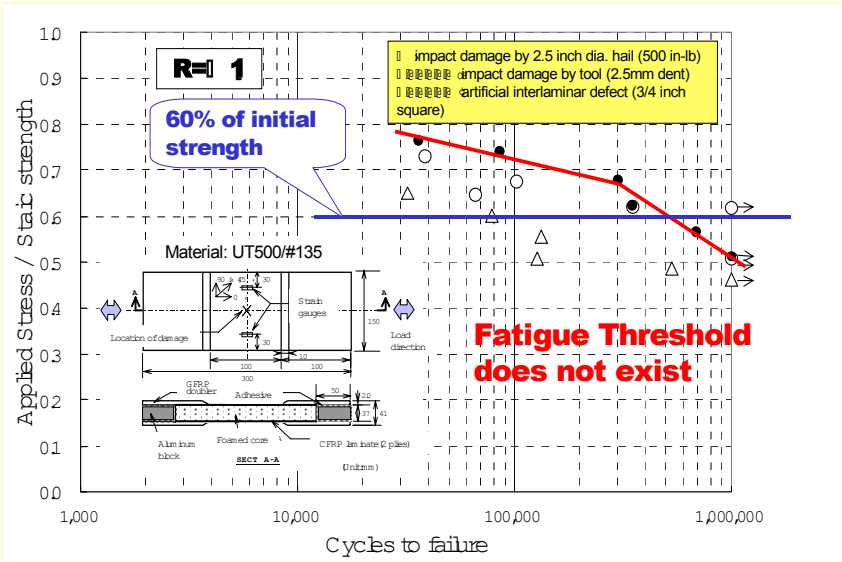


Fig.12 Recent Test Data of PIF

References

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