# Proposed Application of SHM and/or Proof Testing to Facilitate the Certification of Adhesively Bonded Repairs to Primary Airframe Structure

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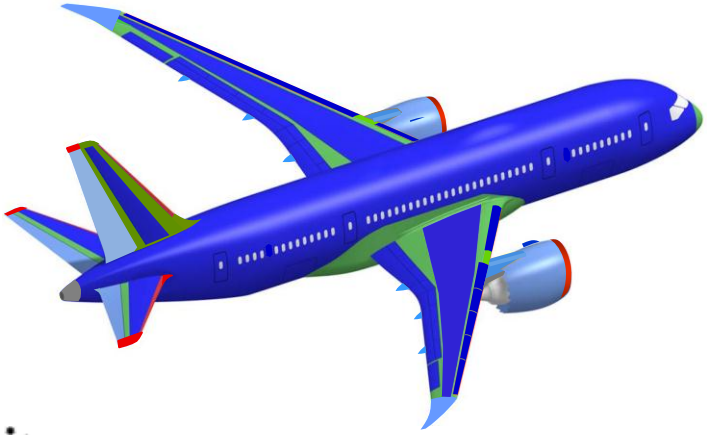

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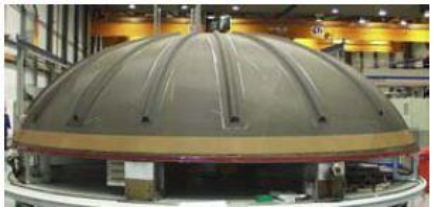
# Aim and Outline of Presentation

To highlight difficulties in validating bond strength and durability for the certification of bonded repairs to aircraft primary structure* and to propose two approaches by which validation may be accomplished

*Large unitised (unreplaceable) primary composite structure is a particular focus as bonded repair maybe the only option*

* *composite or metallic*

# Meeting the Challenge of Composite Fuselage Repair

## Reinforced Plastics, George Marsh, May 2012

While accepting that bolt-on repairs may be familiar and quick, we would prefer bonded repairs if the regulatory hurdles could be overcome. It's certainly a shame to have to make holes [for bolts] in nice continuous fibre lay-ups. But bonded repairs are not yet accepted by the airworthiness authorities except, essentially, as a cosmetic fix.

As one industry insider succinctly put it: "The difficulty at present with a repair bond is knowing exactly what strength you've got. There's no sure way of testing a bond's strength without breaking it, and one has to rely on coupon or sample tests, which might not be fully representative."

# For Certification Need to Demonstrate:

Reinforcement provides required residual strength recovery of F*DLL

- ***Currently F = 1.5 but some relaxation may be possible***

Very high probability of retaining recovered RS for remaining life, *or*

Loss of reinforcing ability can be detected with sufficient warning

# Limitations of Current NDI for Adhesive Bonds

Current Capability e.g. Ultrasonics, Thermography

- ✓ Disbonds
- ✓ Porosity
- ✓ Inclusions
- × Kissing bonds: *no bond but surfaces are in intimate contact*
- × Weak bonds
- × Adhesive under/over-cure

# Current Situation - Certification of Bonded Composite Repairs

**When RS < F × DLL initially or potentially**

Given these limitations of NDI in validating bond strength or durability:

- Repair may be considered acceptable by airworthiness authorities only if implemented under strict factory quality control conditions by experienced technicians
- Very difficult to achieve in most *in situ* repair situations
- Then credit cannot be given to the patch for restoring the required RS or reducing crack growth rate
- ***Thus either the repair is unacceptable or inspection interval would be as for no repair***

# Proposed Options for Certification of Critical Bonded Repairs

- Structural Health Monitoring SHM
  - automatic detection of patch disbonding or damage growth

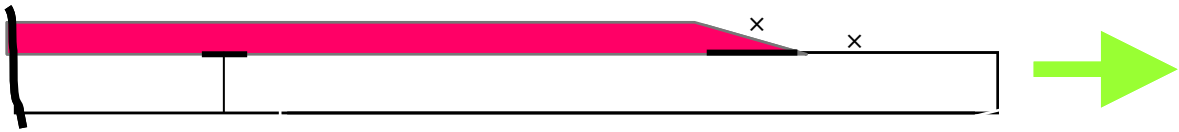
- Proof Testing
  - validation of bond initial and long-term strength

# Some Requirements of a SHM System for Bonded Repairs

- High probability of detection
- Rugged and high reliability

  *Otherwise just exchanging one reliability problem for another one*
- Monitor critical zones:
  - patch*
  - damage

***Patch SHM is the highest priority since this provides the earliest warning of potential loss in patching efficiency***



## Some SHM Options

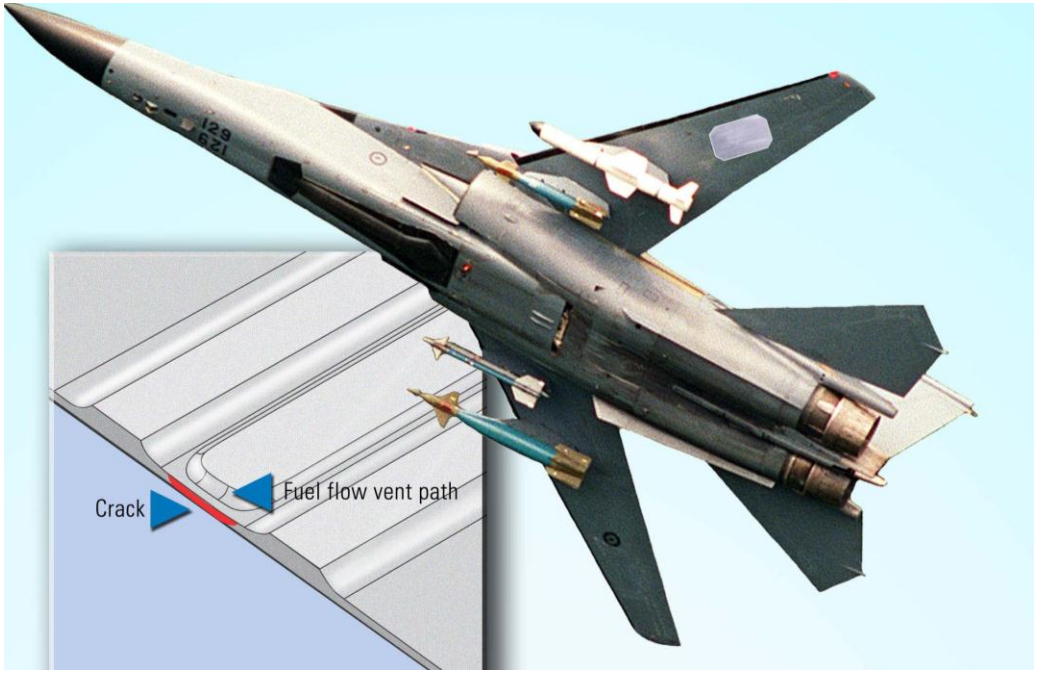
- Patch
  - strain transfer
    - service loads
    - residual strain
  - end displacement
  - disbond growth
  - bond degradation
  - structural response
- Crack*
  - crack growth
  - patch strain

Strain ratio
patch-edge strain/far field strain

Crack growth
ok as backup

**Metals only*

# Australian Case History: SHM of Boron/Epoxy Patch Repair to F111C Wing Fatigue Crack

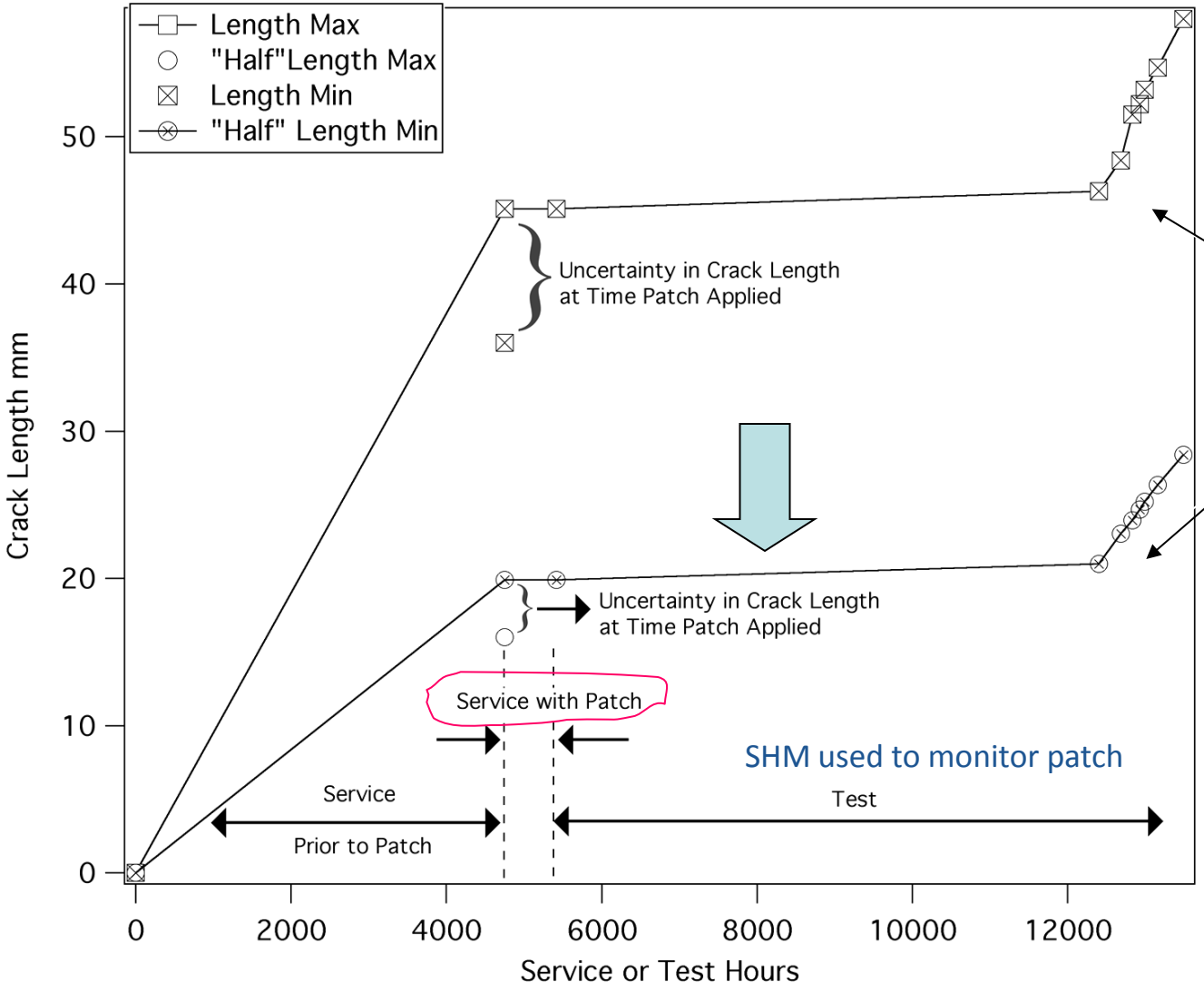


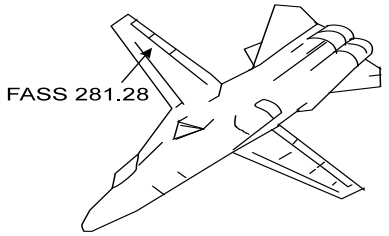

A large fatigue crack was discovered in the lower wing skin of an ADF F-111C

Based on fracture mechanics calculations, the 48 mm crack had reduced RS below DLL

**Bonded boron/epoxy patch most viable option**

# Repair Performance



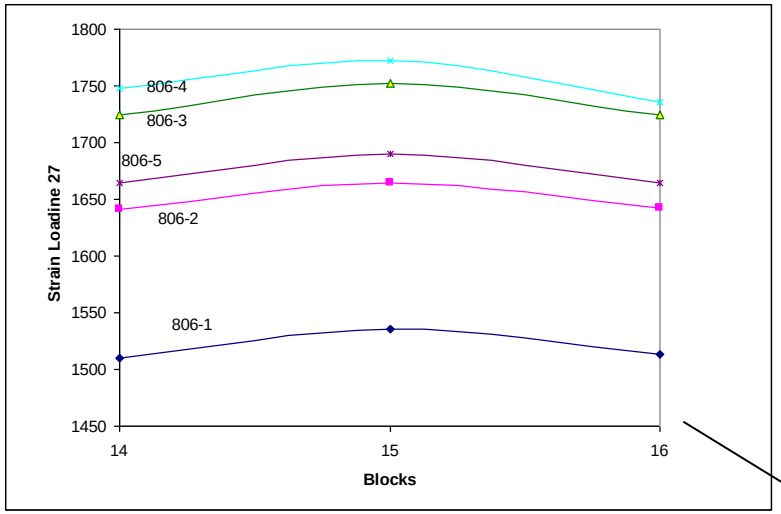



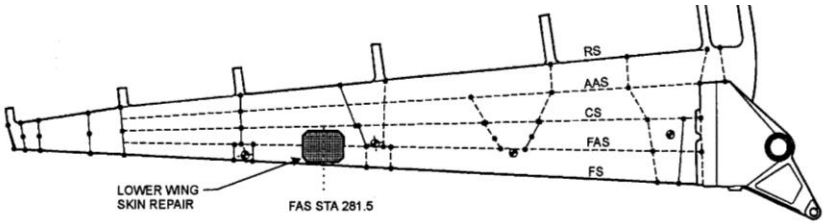

*Failure of aft auxiliary spar*

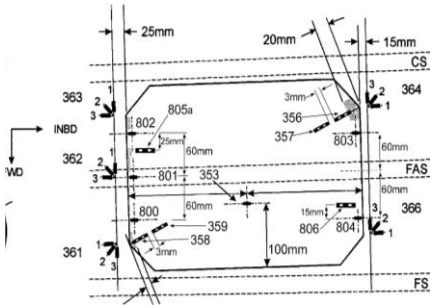
Decision made to fatigue test wing

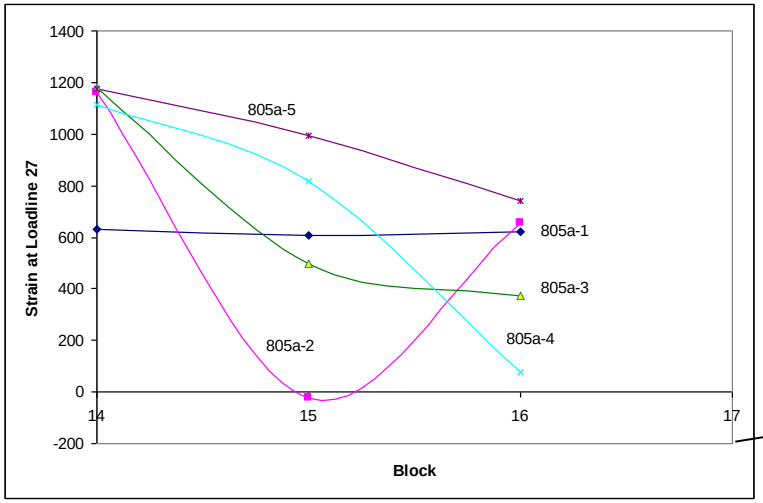
- for DTA analysis fleet
- further substantiate repair
- *Include SHM as NDI indication of local disbonds*

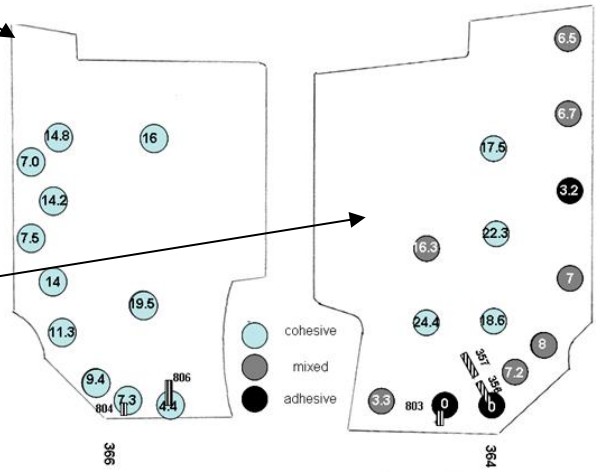
# SHM Comparison with Bond Strength



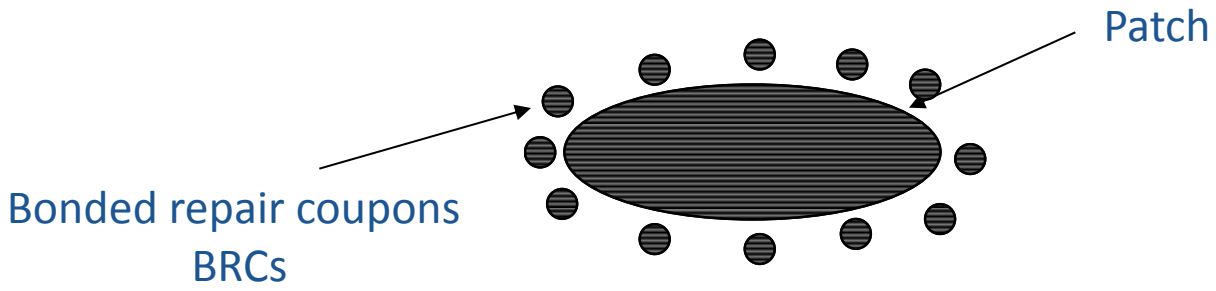

**One block = 500 simulated flying hours*

# Interim Conclusions on Strain-Based SHM

- SHM shown to be viable technically but requires much further development including in-flight demonstration to bring up to the TRL 8 level*
- Reliability a key issue
- Costly but cost would easily be justifiable for large unitised composite or metallic structure

**TRL 8: System completed and proven through successful mission operations*

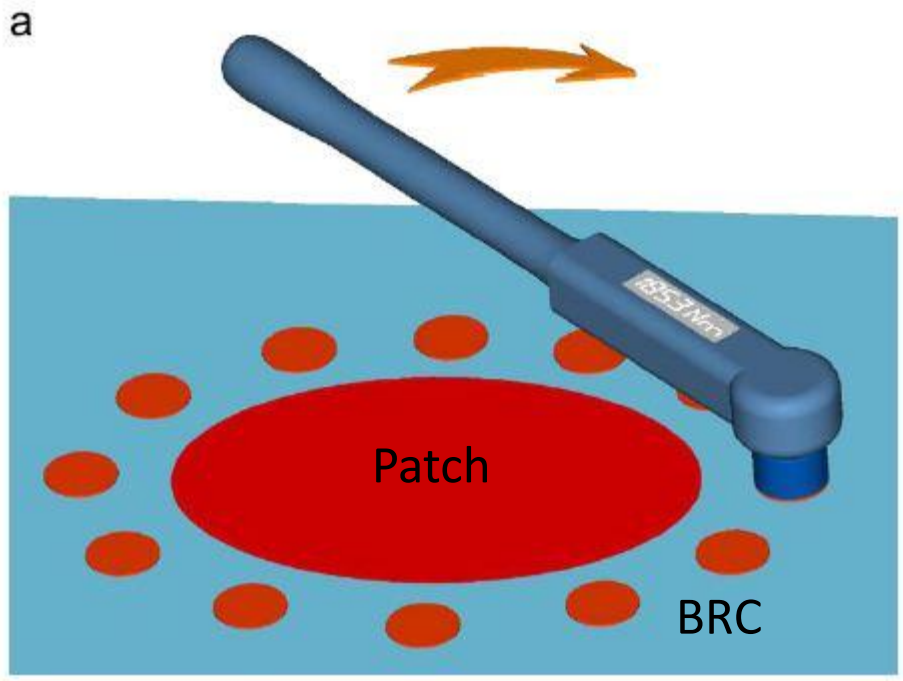
# The Proof Test Concept

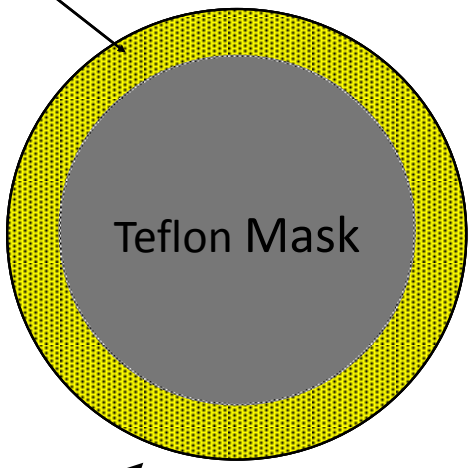


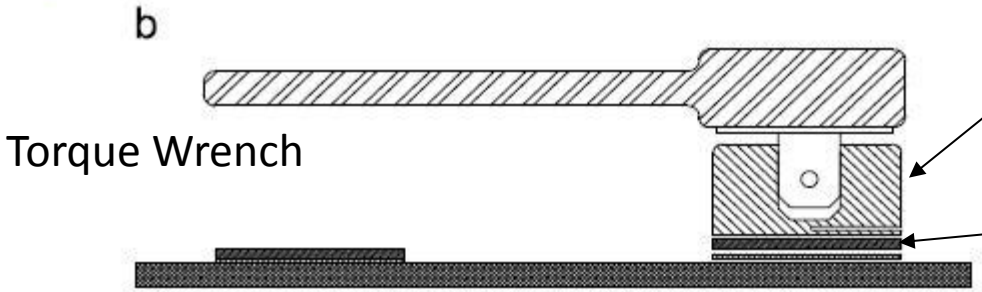

Aim: to validate patch nominal initial and then through-life bond strength

- Based on stress tests on coupons bonded with and therefore under identical conditions to the repair patch
- Stress is applied to a previously determined proof level, determined from tests on standard BRCs
- Proof tests conducted: a) after patch application to assess initial bond strength and then b) periodically through life to assess ongoing bond strength
- Should easily detect: kissing bonds, weak bonds, under-cure, porosity, disbonds, fatigue damage

# The Torque Proof Test Approach

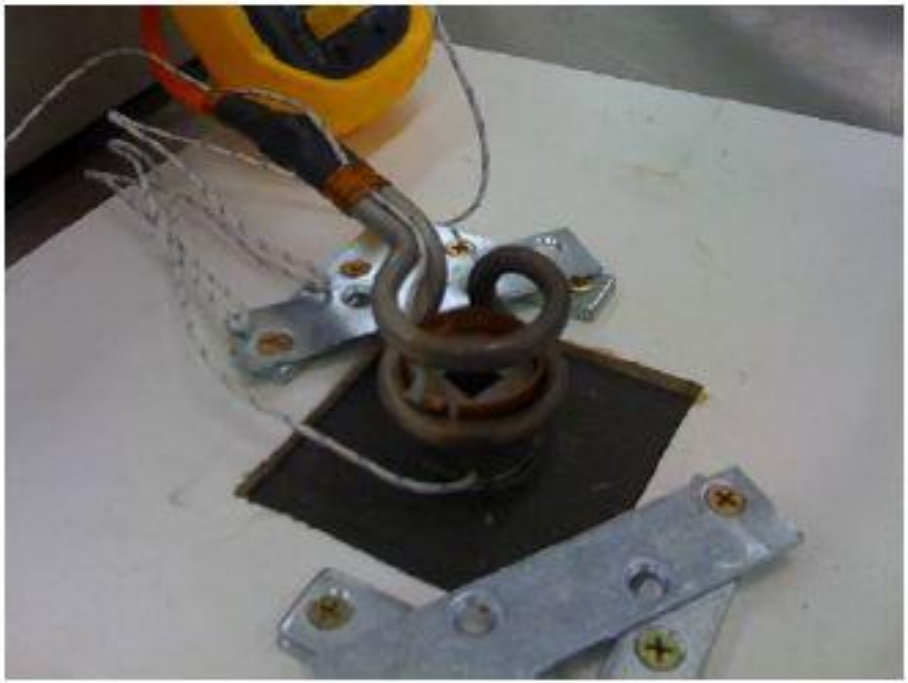


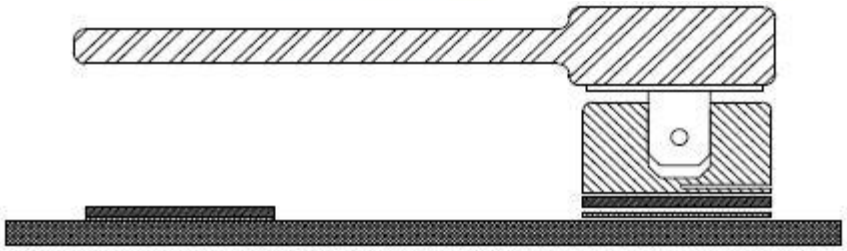




~ 1 mm thick

# Adaptor Removal

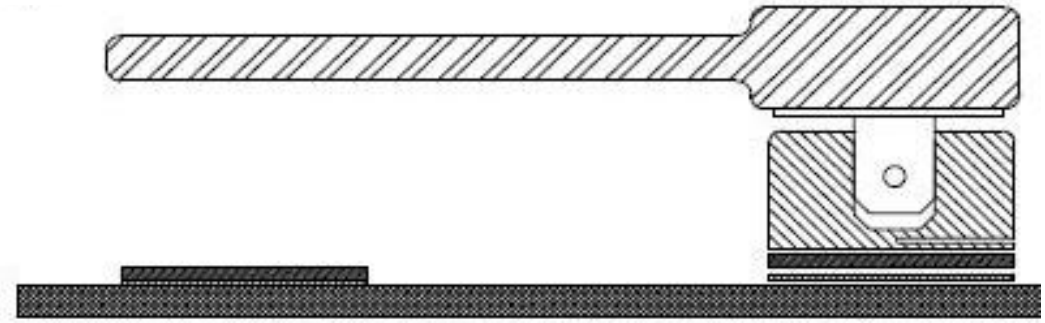


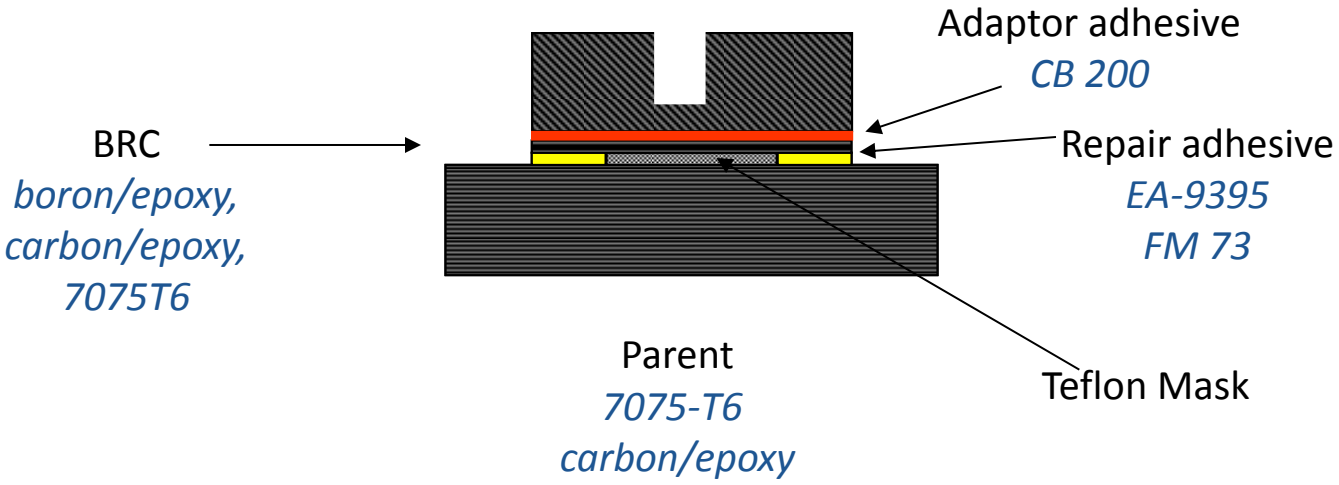
80°C + Torque



Torque Adaptor

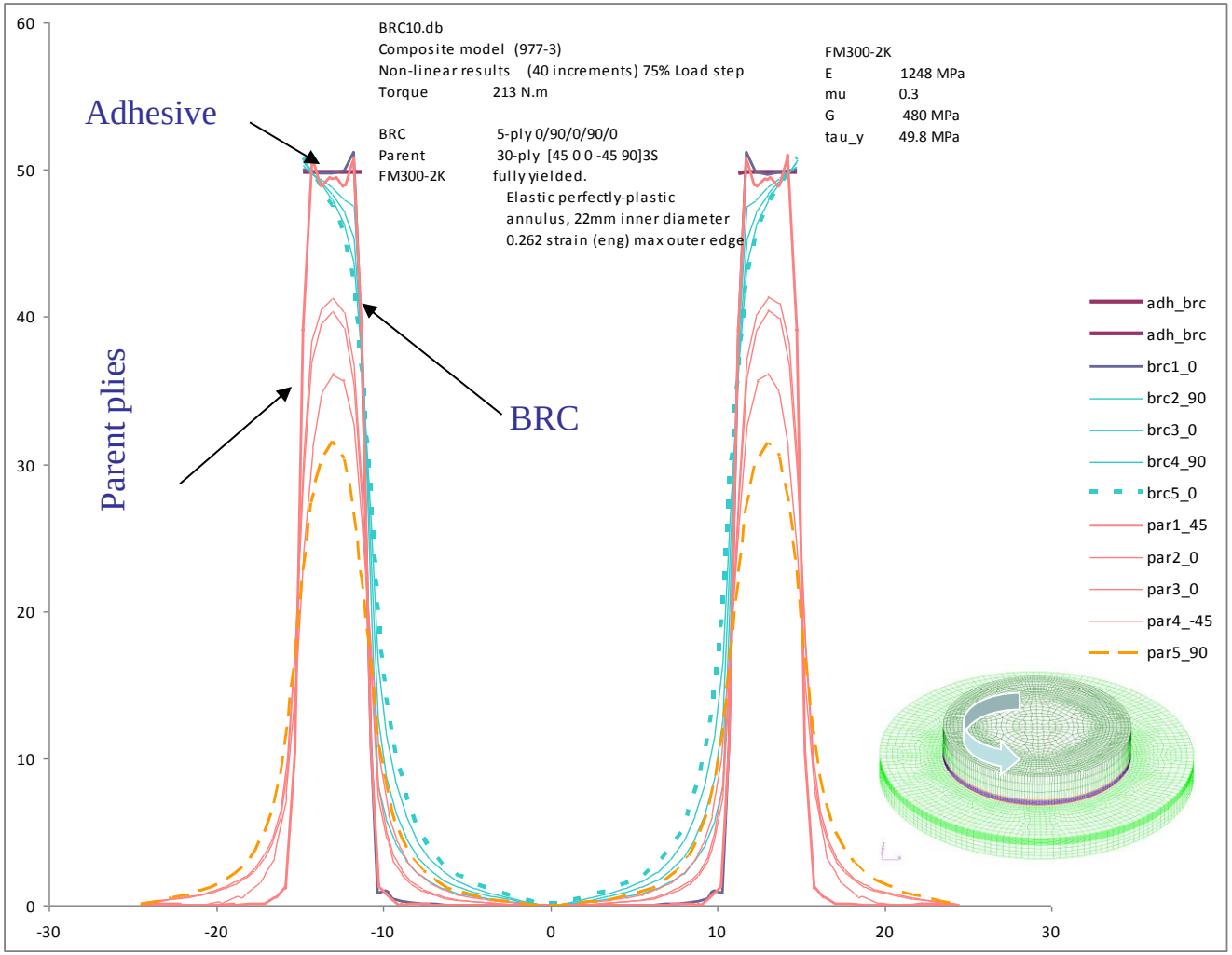
# Test Details






Test variables: *surface treatment, degree of adhesive cure, artificial disbonds*

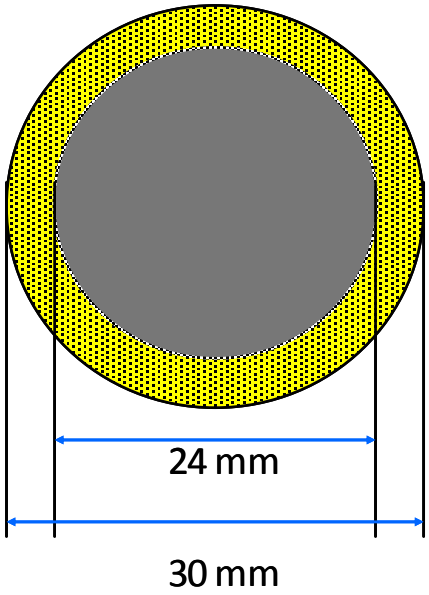
# Non-Linear FE Analysis Shear



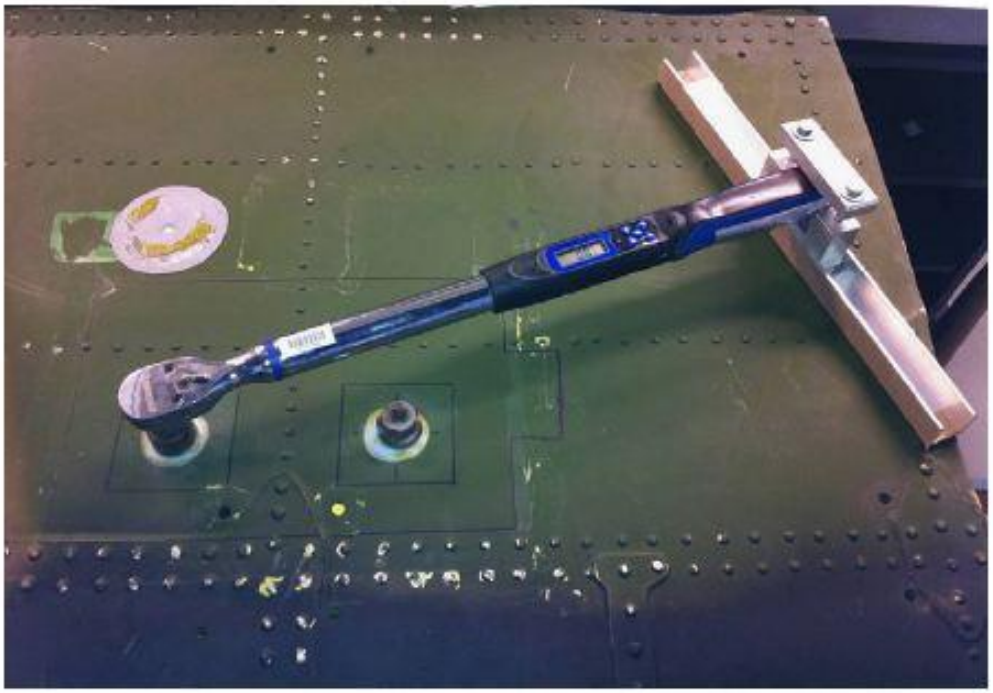

BRC: 0/90/0/90/0

Adhesive: FM 300

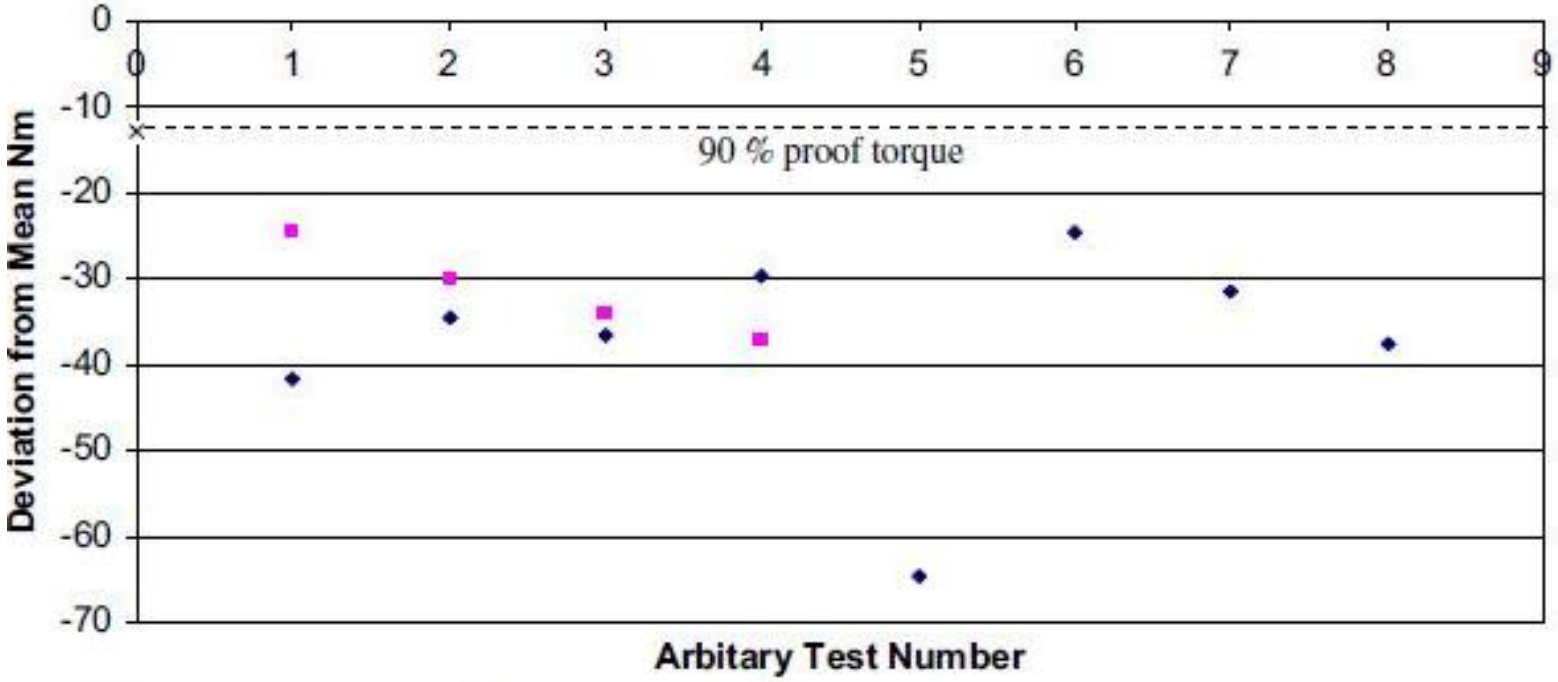
Parent: [45/00/-45/90]3S

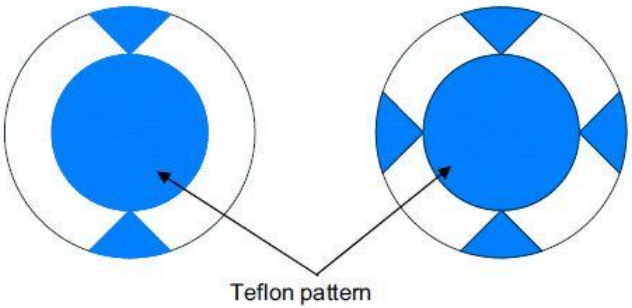


# Proof Test Demonstration on a Component



# Boron-Epoxy/FM73/7075T6 Disbonds




# FM 73 Degree of Cure

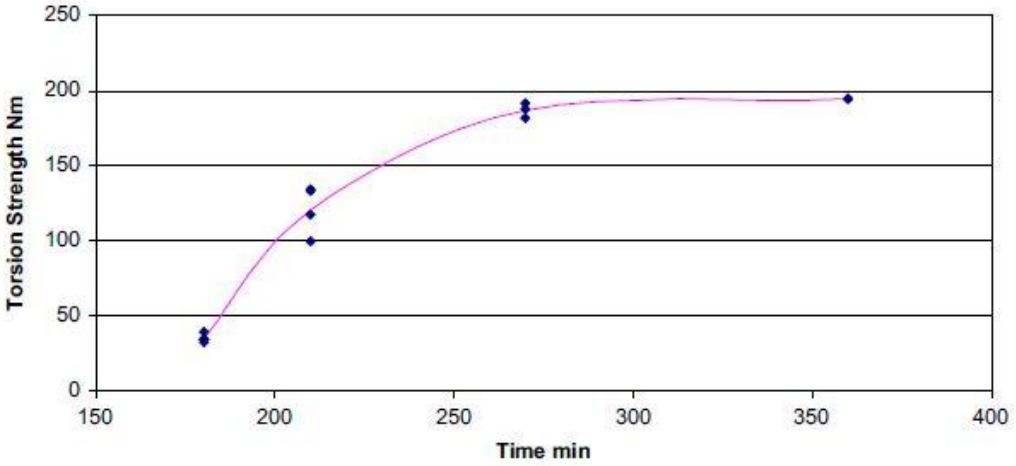
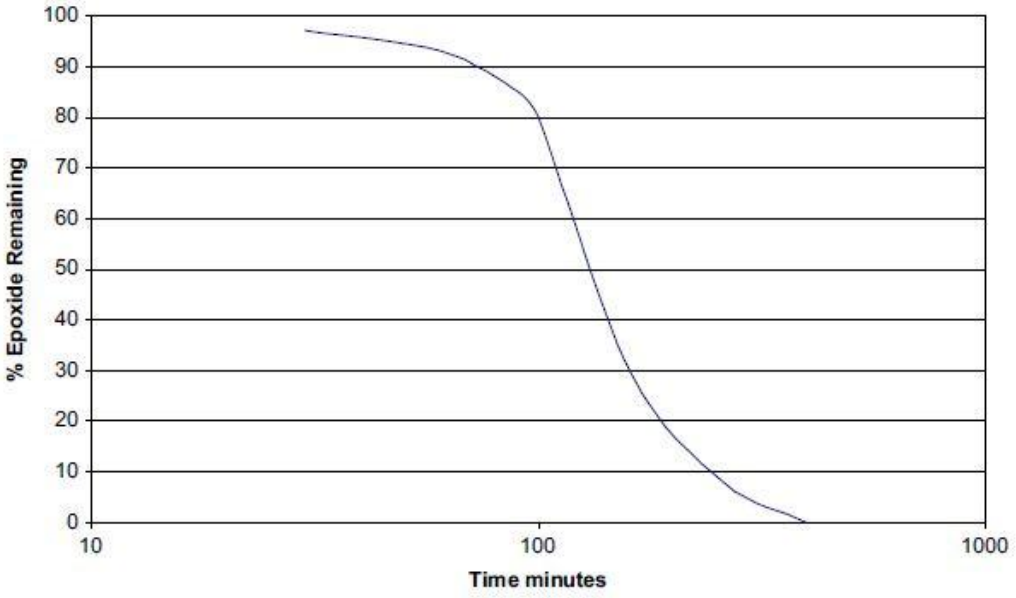



Fig. 9. Plot of torsion strength as a function of cure time for adhesive FM73.



# FAA Advisory Circular (AC) No: 20-107B on Certification of Composite Primary Structure

## Concerning Bonded Joints:

*i) The maximum disbonds of each bonded joint consistent with the capability to withstand the loads [that is, limit loads] must be determined by analysis, tests, or both. Disbonds of each bonded joint greater than this must be prevented by design features; OR*

*(ii) Proof testing must be conducted on each production article that will apply the critical limit design load to each critical bonded joint; OR*

*(iii) Repeatable and reliable non-destructive inspection techniques must be established that ensure the strength of each joint.*

***Key Question: Does Proof Testing satisfy this requirement?***

# Improvements Required to Raise Proof Test to TRL 8*

- Fully developed data base required and in progress
- Check no damage to parent structure with repeated tests
- Check fatigue damage not an issue
- Standardised BRC application to minimise skill requirements
- Automation of testing and recording
- Comprehensive demonstration of practicality under field conditions

**TRL 8 System completed and proven through successful mission operations*

# General Concluding Comments

- Certification will be required, especially to repairs in large unitised composite or metallic primary airframe structure.
- SHM and/or proof testing have considerable potential to alleviate certification concerns
- SHM while effective is expensive, complex and brings its own reliability issues – but may be unavoidable in hidden structure
- The proof test is relatively inexpensive and simple to implement, although it would benefit from automation
- Both approaches require further development to reach Technical Readiness Level 8 where they could be deployed in service

# Acknowledgments

Mr Daniel Bitton, Dr Andrew Gunnion, ACS-A,
Drs John Wang, Paul Chang, DSTO

For very helpful discussions Mr Jean Rouchon: Consultant Aircraft Certification;
Chief Materials and Structures CEAT France, Retired

## References

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*Applied Composite Materials 2010*

"Towards a Practical Health Monitoring System for Patched Cracks in Aircraft Structure"
*Composites A 2009*

"Development of a Proof Test for Through-Life Monitoring of Bond Integrity in Adhesively Bonded Repairs to Aircraft Structure"
*International Journal of Adhesion and Adhesives 2012*

# Thank You

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