BONDED REPAIR OF CORROSION

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(Topic area : Bonded Repair)

Introduction

The US Air Force is operating a large fleet of aging aircraft. Some examples of problems that aging aircraft can encounter, among others, are fatigue and corrosion. This research focuses on bonded repair of exfoliation corrosion in the upper wing skin of a military transport aircraft. In the past, bonded repairs have shown to be a very effective repair method, in situations where riveted repairs did not work, and/or replacement of the part was cost prohibitive or impossible due to the lack of replacement parts.

Upper wing skins on this transport, made out of 7000 series aluminum, have shown extensive exfoliation damage, mainly around the fastener holes, see Figure 1. Exfoliation corrosion is a form of intergranular corrosion associated with high strength aluminum alloys. Heavily worked alloys with a microstructure of elongated grains are especially susceptible to exfoliation. Corrosion products will build up along the grain boundaries and eventually result in layers of the material peeling or flaking off. Exfoliation often initiates at end grains encountered in locations such as machined holes, and can progress further into the material.

The current approach to deal with this damage is to remove the exfoliated material by grinding it out, and not replacing the removed material. Depending on the location on the wing, local grind-outs are allowed up to 25% of the skin thickness. However, if the exfoliation extends beyond this limit, this approach is no longer allowed since too much material is lost. Replacement of the wing plank might be necessary, making this damage very costly.

The purpose of the current research is to develop a bonded repair method that restores structural integrity to corrosion grind-outs deeper than 25% of the skin thickness, so that severe corrosion cases can be handled without the possible need for replacement. Specimens were tested in fatigue in order to come to the most efficient and durable repair concept.

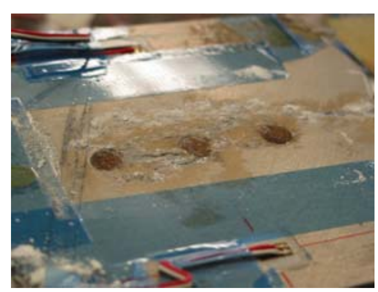


Figure 1: Example of exfoliation corrosion on upper wing skin

Specimen Design

Basic Specimen

The specimen is designed to simulate the loading conditions in the upper wing skin. Since the exfoliation occurs mainly around fastener holes, a mechanically fastened joint had to be designed with less than 10% load transfer and less than 10% secondary bending.

Aluminum 7075-T6 bare with a thickness of 4.826 mm (0.19") was used as specimen material, combined with HL19PB-6-6 Hi-Loks.

The specimen was designed for the following loading conditions:

- Maximum compressive stress of -193 MPa (-28 ksi)
- Maximum tensile stress of 115.8 MPa (16.8 ksi)

The specimen had to be easy to manufacture and easy to inspect. It was therefore decided to use an unstiffened panel. Figure 2 shows the dimensions of the specimen. The dimensions are based on the size of the defect, in this case a grind-out with repair (see next paragraph). The specimen has to be long and wide enough to allow for proper load introduction and load bypass/attraction, resulting in a relatively long specimen.

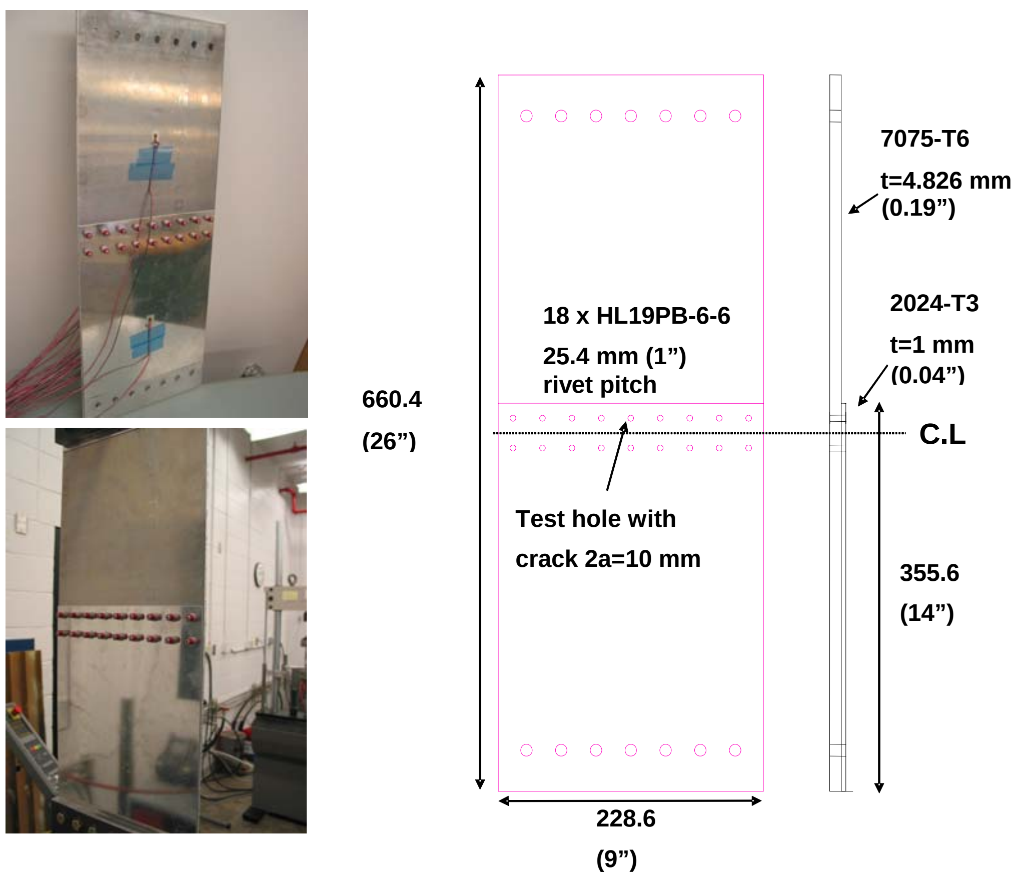


Figure 2: Specimen Dimensions

In order to create the necessary load transfer, a sheet of 2024-T3 bare aluminum with a thickness of 1 mm (0.04”) was attached to the specimen over half its length, see Figure 2. It was fastened using two rows of nine hi-loks, centered around the horizontal centerline of the specimen. The rivet pitch was 25.4 mm (1”).

Since the specimen would not be able to withstand the high compressive loads, anti-buckling plates combined with four I-beams were used to support the specimen, see Figure 3.

The two steel anti-buckling plates, each with a thickness of 12.7 mm (0.5”), were attached on either side of the specimen. The sheets were covered with Teflon to prevent friction build-up. The plates were attached using two rows of bolts that are hand-tightened. It is important not to clamp these plates onto the specimen, to avoid friction between the anti-buckling plates and specimen. It is therefore not possible to eliminate all bending since there is a limited amount of play between the plates and specimen. The anti-buckling plates were shorter than the specimen gage length, to prevent loading the plates in compression.

The small gap that was allowed for this, was cause for some bending near the grips. For that reason four I-beams were bolted onto the grips, fixed to the grips on one side, and sliding with slots on the other side of the specimen. Strain gage measurements were taken along the length of the specimen to verify that the loading requirements were met.

As can be seen in Figure 2, the top center rivet hole had a through-crack on both sides of the hole. There were several reasons for making a starter crack in this location:

- To be able to compare the effectiveness of the different types of repairs, it was decided to look at crack growth rates.
- If no crack would be present, one would be looking at a crack nucleation event, which can have large amounts of scatter.
- Having a starter crack represents a worst case scenario: a crack has initiated in the substrate. Can the repair control this crack over the remaining lifetime?

The total length of the crack, including the rivet hole, is 10 mm (0.4”). All the starter cracks were pre-cracked at 80 MPa (11.6 ksi), R=0.05, to avoid large (what does “large” mean in this context?) plastic zones. The grind-outs were made after pre-cracking.

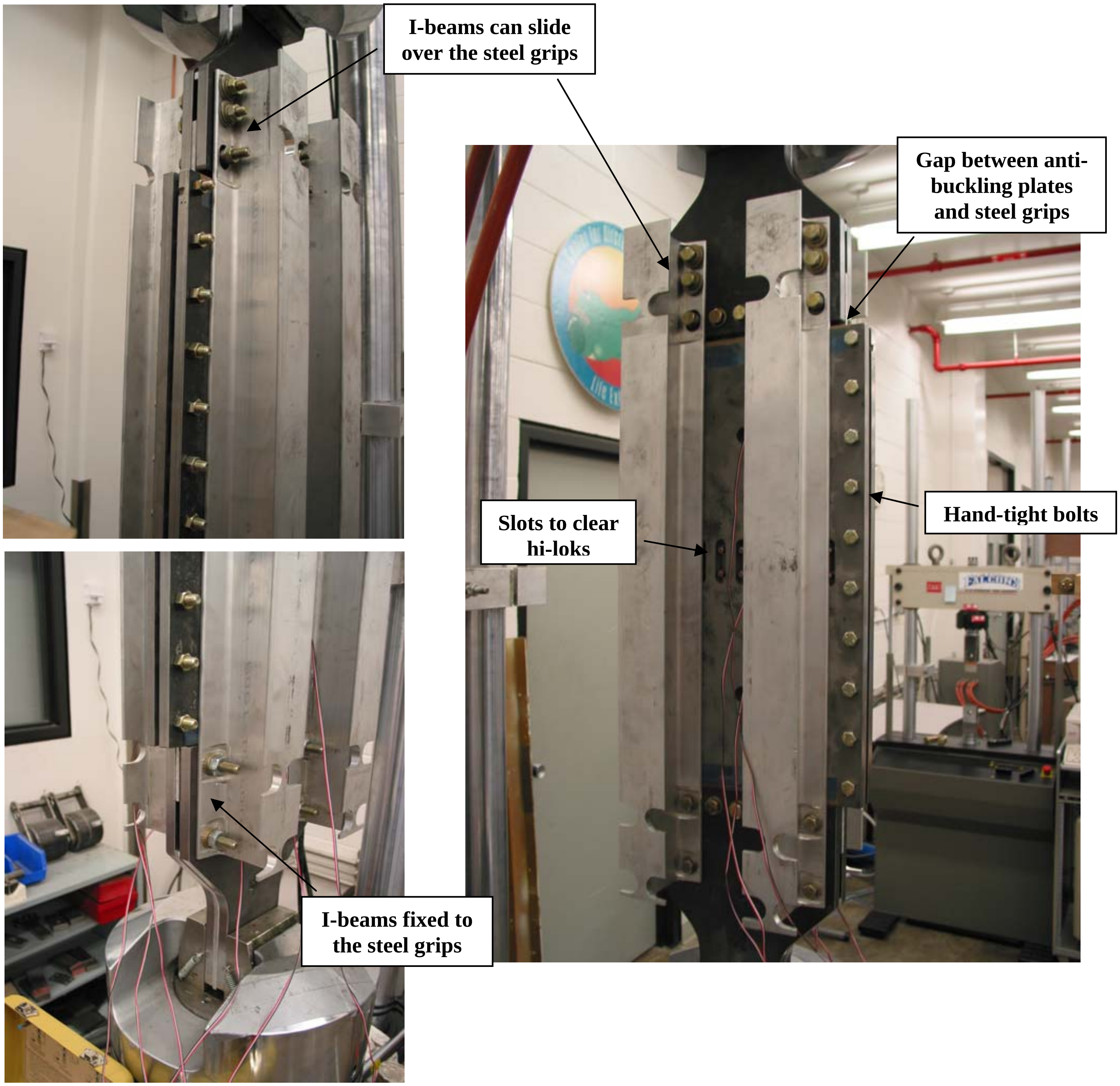


Figure 3: Anti-buckling test set-up

Corrosion Grind-Outs

Since the dimensions of the specimen are driven by the size of the damage, in this case the grind-out, it was necessary to look at field data in order to determine the dimensions of a representative grind-out. Analysis of the grind-out dimensions of in-service aircraft showed that, in the area under consideration, a corrosion grind-out with a diameter of 50.8 mm (2") or less will cover approximately 92% of the cases seen in the field.

In order to significantly expand the current T.O. limits for corrosion grind-outs, it was decided to increase the grind-out depth from 25% to 66% of the skin thickness. The specimen material for this project was 7075-T6 bare, with a 4.826 mm (0.19") thickness, therefore resulting in a 3.175 mm (0.125") deep grind-out.

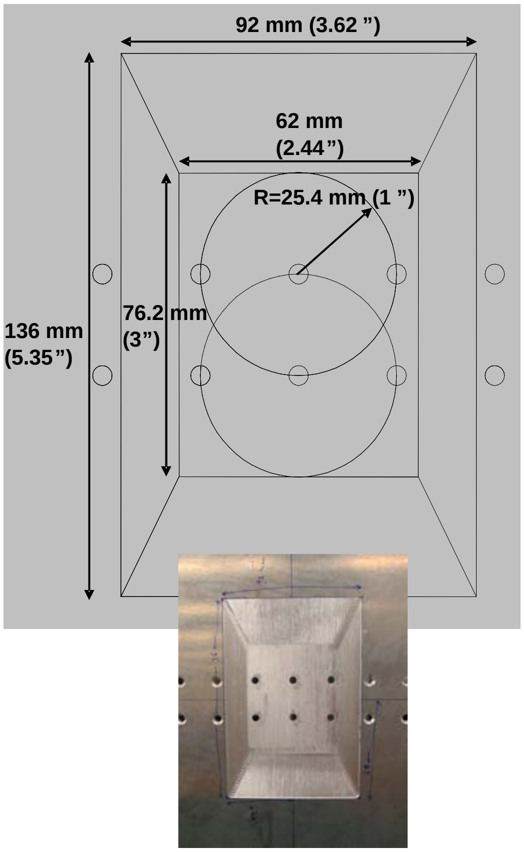
Current T.O. guidelines prescribe the following requirements for corrosion grind-outs:

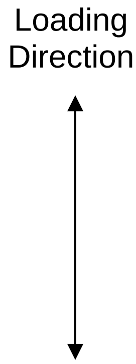
- Remove all corrosion
- Add blend-out length of 10 x grind-out depth in loading direction (spanwise?)
- Add blend-out length of 5 x grind-out depth perpendicular (chordwise")
- Grind-out is elliptical in shape
- Grind-out needs to be terminated between rivets
- Re-countersink fastener if countersink <= 0.75% of thickness remains

To shape the grind-outs for this project, it was assumed that two adjacent fastener holes have corrosion damage with a radius of 25.4 mm (1"), see the overlapping circles in Figure 4 and Figure 5. The area covered by these 2 circles was then approximated by a rectangular area in which the corrosion depth was assumed constant with a depth of 3.175 mm (0.125"). The blend out lengths were then added.

Following the T.O. requirements of tapering in two directions resulted in a grind-out that is shaped as a pyramid-without-top (Figure 4). By omitting the taper perpendicular to loading direction, the second type of grind-out was created (Figure 5).

Both grind-outs had a taper of 10:1 in loading direction. Leaving out the taper in width direction (5:1 taper ratio) makes it easier to fit the composite and hybrid patches (see next paragraph). Static strain gage measurements in and around these two grind-outs showed no significant overall differences since the specimens are only loaded in one direction. The remaining thickness in the bottom of the grind-out is sufficient to allow re-countersinking of the hi-loks without creating a knife-edge.





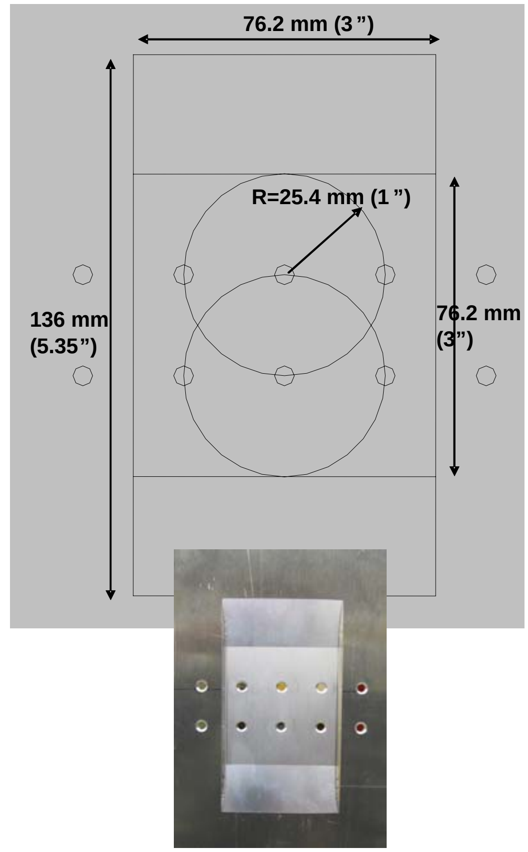


Figure 4: Grind-out for aluminum repair

Figure 5: Grind-out for boron and hybrid repair

Design of Repairs

In order to design the repairs for the grind-outs that are shown above, several requirements had to be met:

- Repairs must be flush with the contour of the wing. No permanent external patches are allowed, which limits the possibilities for repairs.
- Repair must fit within the grind-out:
  - It was decided to follow the current T.O. guidelines for taper angles when making the grind-outs.
  - Only the “corroded” material is removed so that the grind-out is kept as small as possible. The grind-out is not made any larger to facilitate the bonded repair, which is not ideal. Therefore the repairs shown in this paper are designed to sustain the loading conditions mentioned previously, repairs could need re-sizing depending on specific location/loading/operating conditions on the aircraft.
- Repair is sized to carry loads that are normally carried by the removed material, assuming that the remaining material is undamaged.
- In order to size the repairs, a combination of the Composite Repair of Metallic Structure (CRMS) manual [1] and the RAAF Engineering Standard C5033 [2] was used.

Three different flush repairs were designed:

1. Aluminum repair
2. Boron-epoxy repair
3. Hybrid repair: aluminum combined with boron-epoxy

1. Aluminum Repair

Using an aluminum repair has several advantages and disadvantages.

Advantages:

- No, or limited thermal residual stresses after bonding the patch
- Rivets can be countersunk through the patch

Disadvantages:

- The extensional stiffness of the repair is limited by the flushness requirement. Only the extensional stiffness of the removed material can be replaced
- Crack nucleation in the patch is a possible failure mode
- Patch has to be the perfect positive of the grind-out
- Patch needs surface preparation similar to skin before bonding

For repairing corrosion grind-outs where the remaining substrate is undamaged, the C5033 procedure is to restore the original extensional stiffness of the structure. However, the CRMS manual allows the Stiffness Ratio of the patch/substrate combination to be up to 1.5, resulting in the following guideline:

$$StiffnessRatio: 1.0 \leq \frac{E_{repair}\, x t_{repair} + E_{substrate}\, x t_{substrate}}{E_{skin}\, x t_{skin}} \leq 1.5$$

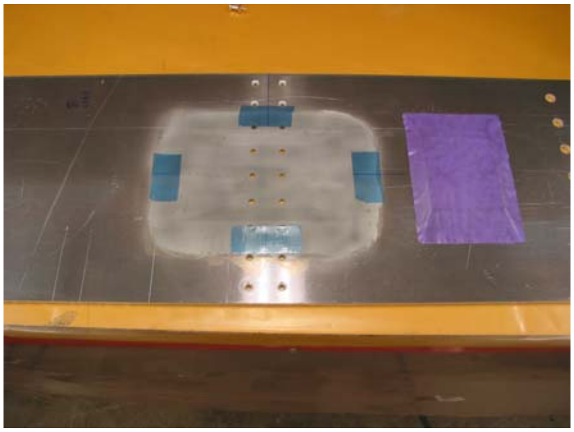
where Ext is known as the extensional stiffness

If the “x” in the equation above is “multiplication” then just remove it. Applying a repair that is too stiff will result in unwanted load attraction into the repaired area, possibly leading to problems at the patch tips. Nucleating a crack at the run-out of the repair is a very dangerous situation that should be avoided. Applying a repair that is not stiff enough will allow for stresses in the grind-out region that will be higher than desired, and could lead to crack nucleation in the substrate under the repair.

The aluminum repairs that were used were made of 2024-T3 bare aluminum, with a thickness of 3.175 mm (0.125”). The aluminum repair is used in combination with the pyramid shaped grind-out (taper in two directions). Both the patch and grind-out were made using CNC milling equipment and therefore the patch was the exact positive of the grind-out.

Since the removed material is replaced with material with similar thickness and Young’s modulus, the Stiffness Ratio of the aluminum repair is 1.0.

Figure 6 shows the specimen before and after bonding in the aluminum patch. As can be seen, the rivets have been re-countersunk through the repair.



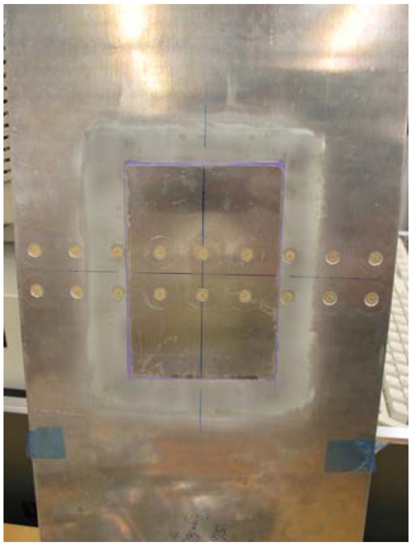


Figure 6: Aluminum repair before and after bonding

2. Boron-Epoxy Repair
Using a boron-epoxy repair has several advantages and disadvantages.

Advantages:
- Higher Young's modulus allows for a higher stiffness ratio within in the grind-out
- Co-curing the patch makes it easier to get a good fit in the grind-out
- No patch cracking issues

Disadvantages:
- Thermal (residual) stresses after bonding and due to thermal cycling in service
- Potential issues with fasteners through the patch, fibers will be cut

The boron-epoxy repair used in this project consisted of the following:
- 13 plies of 5521/F4 boron-epoxy patch (Specialty Materials), ply thickness 0.136 mm (0.005")
- The stiffness ratio of the patch-substrate combination was 1.34.
- This repair was used in combination with the grind-out with only taper in loading direction. The patch follows the taper of the grind-out.
- The lay-up was uni-directional, inverted wedding cake, where the shortest ply is 76.2 mm (3") long and the longest ply is 136 mm (5.35")
- The hi-loks were countersunk at the bottom of the grind-out, after which the boron was bonded over the fasteners.

Figure 7 shows the repair after bonding.

Figure 7: Boron-epoxy repair after bonding

3. Hybrid Repair

Using a boron-epoxy/aluminum hybrid repair has several advantages and disadvantages.

Advantages:

- Higher Young's modulus of the two combined materials allows for a higher stiffness ratio within in the grind-out
- No patch cracking issues (Fiber Metal Laminates principle)
- Possibility to locally tailor the thermal stresses to obtain the most beneficial stress situation

Disadvantages:

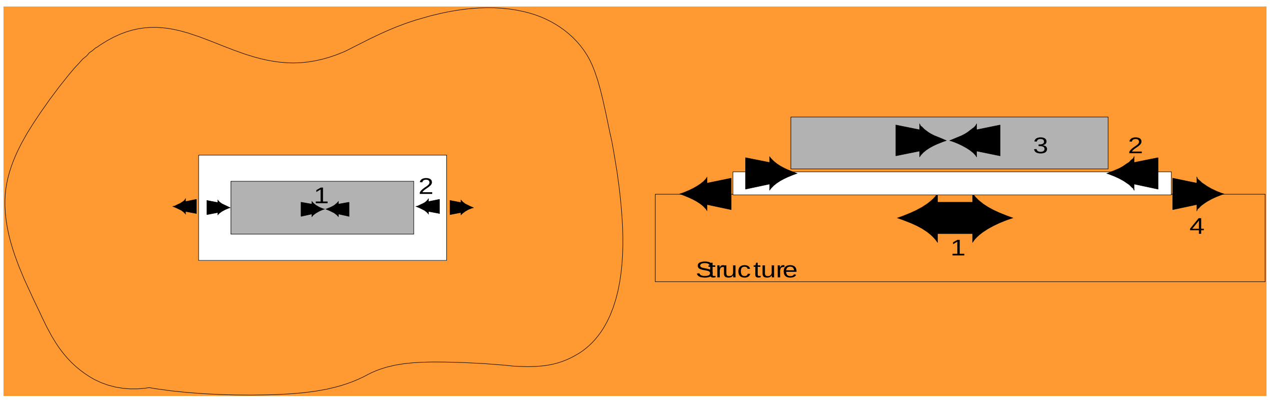
- More complicated
- Potential issues with fasteners through the patch, fibers will be cut

Research in the past has shown that the application of boron-epoxy patches in a compression dominated situation can lead to possible disbonding, starting at the patch tip. This is due to the fact that there is a significant CTE (coefficient of thermal expansion) mismatch between the repair material and the aluminum structure. Using a high temperature curing adhesive, there will be a thermal residual stress state at the patch run-outs that is compressive, whereas the thermal residual stresses at the center of the patch will be tensile. Adding mechanical compressive loading to this thermal residual stress state can therefore lead to disbonding of the patch or delamination in the patch itself, starting at the patch tips.

By combining the best properties of both aluminum and boron-epoxy, it is possible to create beneficial thermal stresses and beneficial fatigue properties in the locations where they are desired [3].

Figure 8 [3] shows the residual thermal stress state for a hybrid patch concept that is most beneficial for a compression dominated loading condition. The arrows do not imply externally applied loads, only thermal residual stresses.

Note that the repair is external but the principle for the grind-out remains the same. In case of the grind-out, the boron-epoxy part of the hybrid patch is bonded directly to the substrate at the bottom of the grind-out your picture shows the aluminum next to the substrate., after which the aluminum top layer (which is longer than the boron-epoxy part of the patch) is bonded on top of the boron-epoxy. By first bonding the boron-epoxy plies to the substrate (bottom of the grind-out), it is more difficult for a possible crack in the substrate to grow into the aluminum layer of the patch.



- White layer is aluminum
- Gray layer is boron-epoxy

Figure 8: Hybrid (external) patch concept for a compression dominated structure

By making the aluminum layer longer than the boron-epoxy layer, the tip of the repair is situated away from the boron-epoxy plies of the patch. Since the effective CTE of the structure is lower than the free CTE of the aluminum patch layer, the aluminum patch layer will create tensile stresses at the patch tip (4) in the structure. These tensile stresses are beneficial when the structure is predominantly loaded in compression, reducing the problems of patch tip disbonding. Very nice!

The hybrid repair used in this project consisted of the following:

- 7 plies of 5521/F4 boron-epoxy, shortest ply is 76.2 mm (3”), longest ply is 94 mm (3.7”), taper ratio of 1:10, ply thickness 0.136 mm (0.005”)
- 1 top layer of 2024-T3 bare aluminum with a thickness of 2 mm (0.08”)
- Hi-Loks countersunk at the bottom of grind-out
- Stiffness ratio of the repair-substrate combination is 1.31

The boron-epoxy part of the patch is bonded to the substrate using 1 adhesive layer of Hysol EA9696 0.06 lbs/ft^2. The aluminum top layer of the patch is bonded to the boron-epoxy/aluminum substrate using 1 adhesive layer of Hysol EA9696 0.06 lbs/ft^2 as well. The combined thickness of all these layers filled up the grind-out depth completely. Figure 9 through 11 show the hybrid repair before and after bonding.

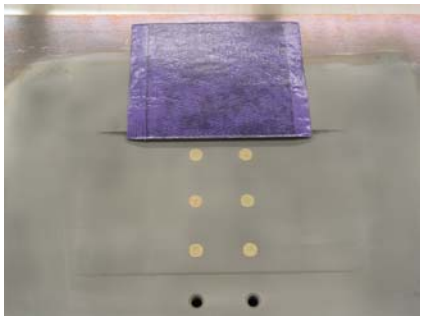


Figure 9: Boron-epoxy plies of the patch with 1 layer of EA9696 adhesive. Note the hi-loks at the bottom of the grind-out

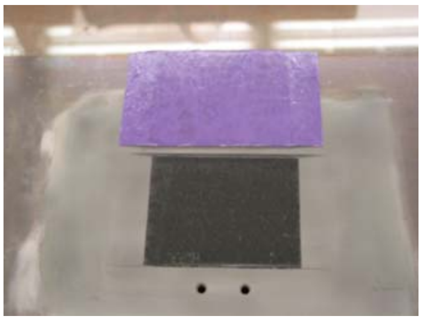


Figure 10: Boron-epoxy plies at bottom of grind-out, aluminum ply with EA9696 adhesive ready to be fitted. Note that the aluminum layer is longer than the boron-epoxy.

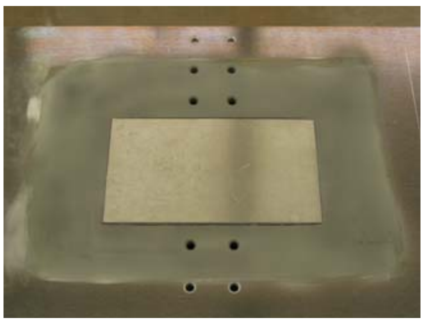


Figure 11: Hybrid patch installed

Bonding
All aluminum bonding surfaces (specimen, aluminum patch and aluminum part of hybrid patch) were surface prepped using solgel AC-130 and Cytec BR6747-1 water based primer.

The adhesive that was used for all bonding was Hysol EA9696, with a specific weight of 0.06 lbs/ft$^{2.}$ A Heatcon vacuum table was used to simulate in-field bonding conditions with only vacuum pressure and a 250 F cure cycle.

Boron-epoxy and hybrid patches were co-cured during the adhesive cure-cycle.

Fatigue Testing

In order to determine the most efficient and durable type of repair, it was decided to look at crack growth rates under the repairs. Crack growth rates will give valuable information on the stress state under the bonded repair. If no crack would be present one would be looking at a possible crack nucleation event, which typically have large amounts of scatter.

Note that the repairs were not designed with a crack in the substrate, the substrate was assumed to be undamaged. The crack growth rates are merely a measure of the stress reduction of the different types of repairs. Also, testing the repairs in fatigue will give a good insight in the durability of the repairs under spectrum loading and could bring to light other failure modes such as patch tip failures, crack nucleation in the patches itself or disbonding or delaminations in the repairs.

Since the starter crack was covered by the repair, and the backside of the crack was covered by the 1 mm (0.04”) thick half sheet, an eddy current surface scan was used to measure crack growth. Measurements were taken from the backside of the repair, through the thin half sheet. It proved to be an accurate way to determine crack size without taking the fasteners out and removing the thin sheet.

Since disbonds and delaminations were a concern during the compression loading, specimens were taken out of the test frame at regular intervals, disassembled and c-scanned (Figure 12) to look for any signs of disbonding.

Table 1 and Table 2 show the specimens that were tested. As can be seen, the testing was split up in a constant amplitude part and a spectrum test part.

The constant amplitude testing was split up into a tension and compression part:

- Tension load of 120 MPa (17.4 ksi), R=0.05 to determine the repair with the best crack stopping capability. Since no tensile load in the spectrum was over 115.8 MPa (16.8 ksi), using this load should be conservative and makes it possible to compare the results to other CAStLE test data from the past.
- Compression load of -150 MPa (-21.8 ksi), R=0.05 to determine if disbonds or delaminations are a problem. The spectrum has only 278 valleys that are below -150 MPa (out of 937,441 segments per block, 10 blocks per lifetime).

Spectrum fatigue testing:

- Spectrum testing was done to verify the performance under realistic fatigue loading and to check for unexpected failure modes
- Maximum tensile stress of 115.8 MPa (16.8 ksi)
- Maximum compressive stress of -193 MPa (-28 ksi)
- Although only twice the remaining life time has to be tested, it was decided to test two life times.

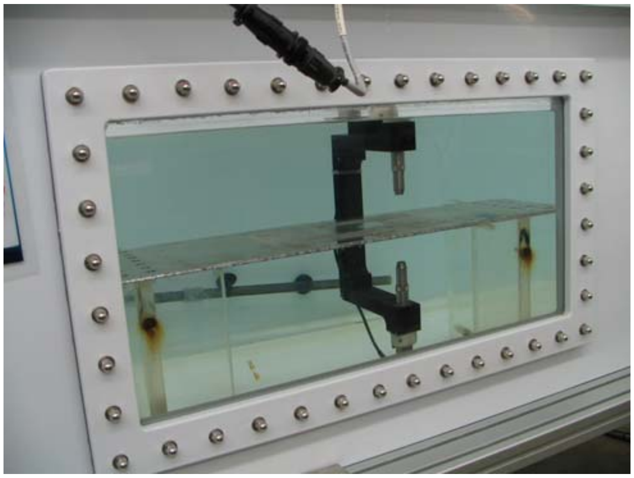


Figure 12: Bonded Repair in C-Scan Tank
Figure 13: MTS 810 110 kip

Testing was done using a MTS 810 110 kip (Figure 13) and 55 kip test frame. The constant amplitude tests were tested at a frequency of 4 Hz, the spectrum testing was ramp (load) rate controlled at 250 kN/sec. Humidity and temperature were monitored during testing.

Specimen Number	Repair Type	Loading Type	Stress (MPa)
1	No Repair	Tension	120, R=0.05
2	No Repair	Tension	120, R=0.05
3	No Repair	Tension	120, R=0.05
4	No Repair	Tension	120, R=0.05
5	Aluminum Patch	Tension	120, R=0.05
6	Aluminum Patch	Tension	120, R=0.05
7	Aluminum Patch	Tension	120, R=0.05
8	Aluminum Patch	Tension	120, R=0.05
9	Aluminum Patch	Compression	-150, R=0.05
10	Aluminum Patch	Compression	-150, R=0.05
11	Boron-epoxy Patch	Tension	120, R=0.05
12	Boron-epoxy Patch	Tension	120, R=0.05
13	Boron-epoxy Patch	Tension	120, R=0.05
14	Boron-epoxy Patch	Tension	120, R=0.05
15	Boron-epoxy Patch	Compression	-150, R=0.05
16	Boron-epoxy Patch	Compression	-150, R=0.05
17	Hybrid Patch	Tension	120, R=0.05
18	Hybrid Patch	Tension	120, R=0.05
19	Hybrid Patch	Tension	120, R=0.05
20	Hybrid Patch	Compression	-150, R=0.05
21	Hybrid Patch	Compression	-150, R=0.05

Table 1: Constant Amplitude Test Specimens (table captions go above table)

Specimen Number	Repair Type	Unit load in Spectrum
DR-1	Boron-epoxy Patch	1.0=115.8 MPa
DR-2	Boron-epoxy Patch	1.0=115.8 MPa
DR-3	Boron-epoxy Patch	1.0=115.8 MPa
DR-4	Hybrid Patch	1.0=115.8 MPa
DR-5	Hybrid Patch	1.0=115.8 MPa
DR-6	Hybrid Patch	1.0=115.8 MPa
DR-7	Aluminum Patch	1.0=115.8 MPa
DR-8	Aluminum Patch	1.0=115.8 MPa
DR-9	Aluminum Patch	1.0=115.8 MPa

Table 2: Spectrum Test Specimens (table captions go above table)

Test Results and Discussion

CA Tension Tests

Figure 14 shows crack growth curves for the specimens tested under constant amplitude fatigue loading.

As can be seen from the graph, the unrepaired specimen showed rapid crack growth compared to the repaired specimens. The final crack measurement was taken before the crack linked up with the adjacent rivet-holes, which are 25.4 mm (1”) away from the originating rivet hole.

As was to be expected, the hybrid and boron-epoxy repairs showed the slowest crack growth. This is due to the fact that the stiffness ratios of these repairs was higher than the stiffness ratio of the aluminum repair, which is limited by the requirement that the repairs must be flush with or within the contour of the skin. Using the boron-epoxy fibers that have an almost three times higher Young’s modulus than the aluminum repair, allows higher stiffness ratios.

Figure 14 also shows that the boron-epoxy and the hybrid repair show a similar crack growth curve, despite the fact that the thermal residual tensile stresses under the boron-epoxy repairs are higher than for the hybrid repairs. This can be explained by the fact that the hybrid patch is thicker than the boron patch. The advantage of a smaller CTE mismatch for the hybrid patch is compensated by shear lag in the thicker hybrid patch, which is most noticeable in the absence of bending [4].

No unexpected failure modes were seen during the tension tests. None of the aluminum repairs showed any crack nucleation in the repair itself, nor did any of the other rivet holes other than the test hole show any sign of crack nucleation. This is important since cracking at any of the other locations would change the load transfer through the thin sheet into the specimen. C-scans were made of all specimens after completion of the tests and none of the repairs showed any signs of disbonding or delaminations. This was not expected but is important to verify.

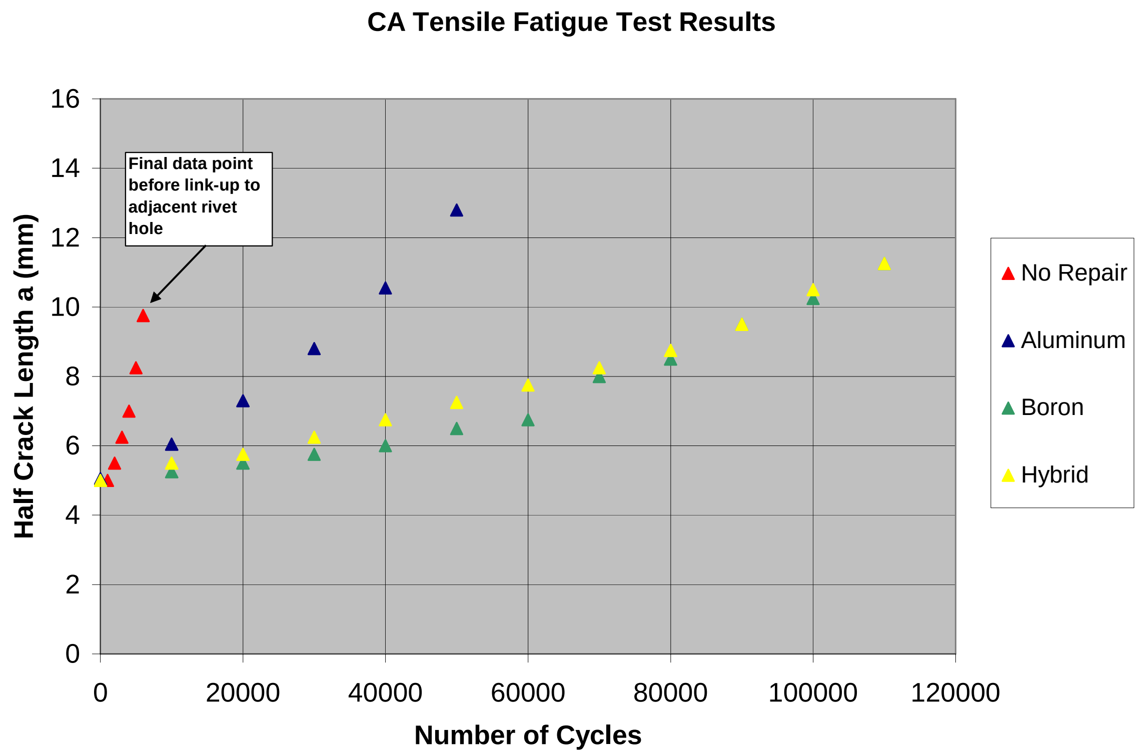


Figure 14: Crack Growth Curves for Constant Amplitude Specimens

CA Compression Tests

The constant amplitude compression tests showed some interesting results. Testing these panels at 4 Hz for 1.5 million cycles resulted in a total test time of slightly over 5 days per specimen, not counting inspection time.

Specimens with aluminum repairs showed no disbonding after 1.5 million compression cycles. Also no crack growth was seen for both specimens with an aluminum repair loaded in compression. Crack growth while loading the specimen in compression without bending is not expected when there are no thermal residual stresses in the specimen. Since the panels were bonded using a vacuum table, the entire specimen is heated and therefore there should be no thermal residual stresses in the specimens with an aluminum repair.

Note that this situation can change for an on-aircraft repair, where the structure is only locally heated. The cold surrounding structure will act as a restraint for the heated part of the structure, therefore restricting the expansion of the heated structure, resulting in a lower effective CTE. When applying an aluminum repair in this situation, there will be compressive residual thermal stresses in the skin at the center of the repair, and residual thermal tensile stresses in the skin at the patch tips, which will be beneficial for a compression dominated loading condition.

Both the hybrid and boron-epoxy repairs showed crack growth after being loaded in compression.

This is due to the residual thermal tensile stresses being cycled between their maximum, present after curing, and their minimum level, at the maximum applied compressive load.

The crack growth observed for the hybrid repair loaded in compression was approximately a factor five smaller compared to the same specimen loaded in tension, but is nevertheless significant. Boron-epoxy repairs loaded in compression showed even higher crack growth rates than the hybrid repairs loaded in compression, but there is not sufficient data available to make a final determination at this point. Will there be enough data at the end of the test program to make a final conclusion?

C-scanning the hybrid repairs after 1.5 million compression cycles showed no signs of disbonding or delaminations. When c-scanning the boron-epoxy repairs during the test, one of the repairs showed delaminations after as early as 250,000 cycles. Figure 15 shows an example of a c-scan for a boron-epoxy repair before and after testing. As can be clearly seen, more than half the patch has delaminated. It was determined that this was an interlaminar failure, not a failure of the bond line. It may be clear that this is a very unwanted failure mode, which would not have been noted in a static test.

Figure 15: C-Scan of Boron-Epoxy Repair Before and After Compression Test

Spectrum Tests

Table 2 shows the specimens that were tested under spectrum loading. Figure 16 shows the crack growth curves for the three different types of repairs. For readability, only one of the curves for each type of repair is shown although it should be mentioned that the results were repeatable. Very nice!

As can be seen, the hybrid repair is clearly performing the best, showing the smallest crack growth over 2 life times. None of the three hybrid repairs showed any signs of disbonding or delaminations after completion of the test. Also, no signs of patch cracking in the aluminum layer, no patch tip failure in the skin or crack nucleation anywhere else in the test specimen were observed.

Specimens with boron repairs showed crack growth that was irregular. Delaminations were found between 1.2 and 1.6 life times. They were initially found using coin tapping, and then confirmed using c-scan. Crack growth did not go up dramatically, supporting the conclusion that only part of the patch had delaminated and some layers were still bonded to the specimen and therefore still transferring part of the load over the crack.

The difference between the aluminum repair and the hybrid repair is slightly smaller to what was seen in the results from the constant amplitude fatigue tests. It appears that the aluminum is benefiting from the lack of thermal residual stresses, the cracks under the aluminum repairs do not grow while loaded in compression. However, crack growth is still very minimal considering that two life times were tested, and keep in mind that only twice the remaining life time of the aircraft has to be demonstrated. Also, the assumption is that no crack (other than an initial flaw of 0.005" why 0.005 instead of 0.05? radius corner crack) is present when the repair is bonded into the grind-out.

The aluminum repair showed no signs of unexpected failure modes, such as crack nucleation in the patch itself or in the specimen at the patch tip, or disbonding.

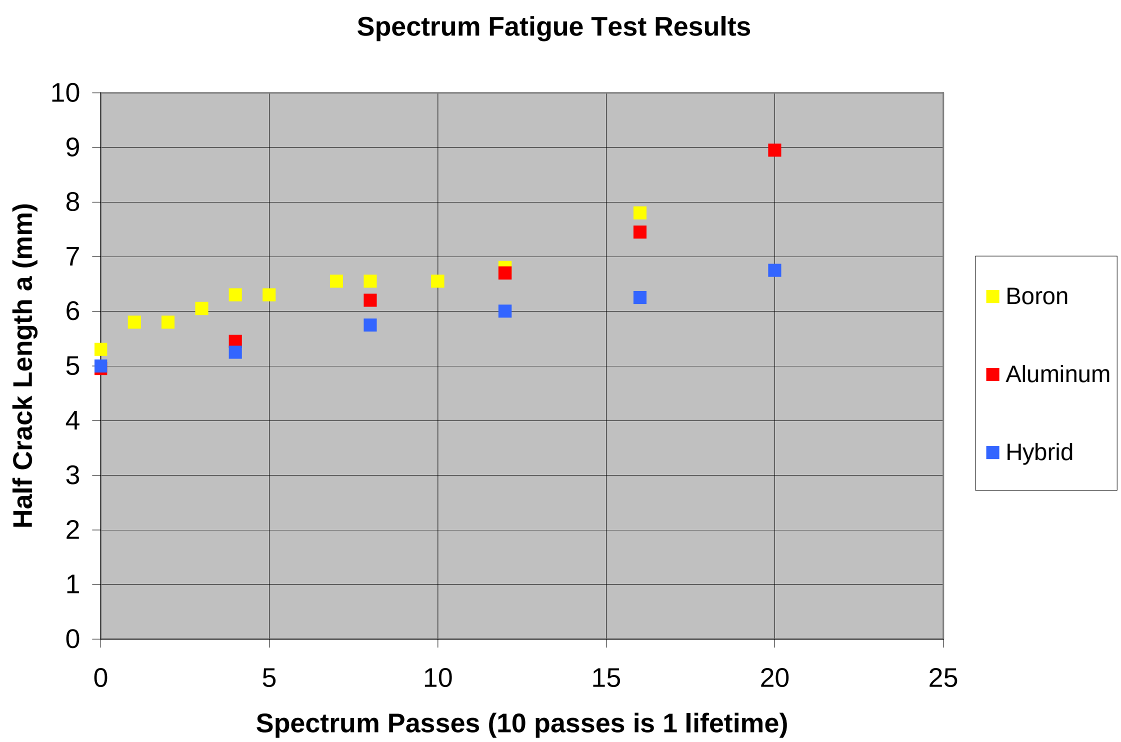


Figure 16: Crack Growth Curves for Spectrum Specimens

Conclusions

- The application of a flush repair seems to be a viable repair option for corrosion damage that is exceeding the current T.O. limits in a compression dominated structure.
- Fatigue tests, both constant amplitude and spectrum loading, showed that from a durability standpoint the best options are either a hybrid repair or an aluminum repair.
- Hybrid repairs are preferred based on:
  - slowest crack growth
  - damage tolerant patch
  - no compression problems under realistic flight loading conditions
- Aluminum patches can be a good alternative if residual thermal stresses and thermal stresses under operating conditions become a problem.
- Aluminum repairs showed no patch cracking problems.
- Boron-epoxy should be avoided due to delamination problems under compressive fatigue loading.
- Keeping the grind-outs as small as possible does put pressure on design guidelines for bonded repairs. Depending on location/loading/operating temperature, patches could need re-sizing.
- Cracks do grow in compression-compression fatigue cycle due to thermal residual stresses (boron and hybrid)

Recommendations and Future Work

- Repair different grind-out depths (in progress). This becomes important when trying to repair a grind-out where a crack is present in the substrate. When grinding out less material, there is less depth available for the repair, while keeping the repair flush with the contour. It might not be possible to control this crack, since the repair will have a reduced stiffness ratio (especially since the high stiffness boron-epoxy has been ruled out as a repair material).

- Bonded repairs on top of actual exfoliation damage (in progress). Instead of removing the exfoliation, it might be possible to bond over the exfoliation, therefore avoiding making a grind-out. However, good NDI techniques will be needed in order to determine the depth of the exfoliation and/or possible crack. This is especially important for sizing the repair. If no good NDI is available, the repair must be sized based on full depth exfoliation or a through-crack, therefore making it necessary to transfer all load from the skin into the repair.

- DUL (what is DUL? Design Ultimate Load?) test at operating temperature. Besides proofing the durability of the repair under realistic fatigue loading, as was done during this program, structures with a bonded repairs need to be able to withstand DUL at the appropriate operating temperature.

- Fine-tuning of hybrid repair to optimize performance at operating temperature. Since the hybrid repair still has a considerable amount of boron-epoxy plies, there are residual thermal stresses present after bonding.

- Practical implementation of making grind-outs on aircraft

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Stephan, very nice paper! My comments above are trivial in my view. Thanks for the great piece of engineering work!

References
[1] Composite Repairs to Metallic Structures Handbook (CRMS), AFRL/ML/TR-1998/4113
[2] Composite Materials and Adhesive Bonded Repairs, RAAF Engineering Standard C5033
[3] Guijt, C.B., Verhoeven, S., Greer, J.M., Van Galen, R.M., United States Air Force Academy, Center for Aircraft Structural Life Extension (CAStLE), DFEM/HQ USAFA, USA, *"A New Approach to Manipulate Thermal Stresses in Bonded Repairs"*, Aging Aircraft 2002, San Francisco, California, 16-19 September 2002.
[4] Verhoeven, S., Guijt, C.B., Greer, Jr., J.M., Dinnebier, H., United States Air Force Academy, Center for Aircraft Structural Life Extension (CAStLE), DFEM/HQ USAFA, USA, Van Galen, R.M., Delft University of Technology, Faculty of Aerospace Engineering, Delft, The Netherlands, *Comparison of the Effectiveness of Glare and Boron-epoxy Patches in Repairs of Thin and Thick Structures, With and Without Secondary Bending*, Aging Aircraft 2001, Orlando, Florida, 10-13 September 2001.