CHALLENGES IN DEVELOPMENT OF STRUCTURAL REPAIR MANUALS FOR COMPOSITE FUSELAGE

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Abstract

There is wide application of composite materials in commercial transport fuselage structure. With extensive usage of composite material systems there is requirement for incorporation of maintainability and repairability requirements of composite primary structure into the design. Such issues need to be addressed to meet regulatory requirements and ensure that life-cycle costs are competitive with current metallic structure. Development of Structural Repair Manual (SRM) needs to account for maintenance issues early in the design cycle and provide multiple repair options. Furthermore, proposed repair solution must have tradeoff between ease of installation, damage resistance/tolerance (repair frequency), and inspection burdens. To support SRM development analysis methods are developed to assess structural strength, repair life, residual strength in the presence of damage, and to evaluate repair design concepts. This paper summarizes the experiences in development of repair options for composite fuselage components due to complexities of material system and additional analysis requirements.

***Keywords**: CFRP Repair, Composite Airframe Repair, Bearing Bypass, Bolted Repair*

Introduction

Composite materials were introduced into the commercial aircraft industry during the early 1960's and used mostly glass fiber. Development of more advanced fibers such as boron, aramid, and carbon offered the possibility of increased strength, reduced weight, improved corrosion resistance, and greater fatigue resistance than aluminum. These new material systems, commonly referred to as advanced composites were introduced to the industry very cautiously to ensure their capabilities [1]. In current generation of aircrafts the usage of advanced composite material systems has grown to 50%, resulting in weight saving of around 20% compared conventional aluminum airframe [2]. The important consideration from lifecycle and fleet cost was to enable the repair of airframe in similar manner as airlines would repair existing airplanes with - bolted repairs. Also ensuring those repairs are permanent and damage tolerant as they are on metal structure [2]. This will ensure airframes can be repaired in-field and increase airframe availability. Also bonded repair is an option providing better aerodynamics and aesthetic finish.

In this paper a typical case study on a repair configuration and analysis methods that can be employed for qualifying repairs of composite skin in particular and other composite structures in general is presented. Repair configurations proposed in SRM has to be generic and applicable to maximum possible locations on the fuselage. This task is complicated by varying lay-ups and loads in different bays. Repair of zone near large cutouts, windows and major attachment region is either covered by location specific repair or are not covered within the scope of generic SRM.

For demonstration of concept, a typical location in forward fuselage is chosen. The location chosen represents the most generic bay dimension and lay-up. The location is shown in Fig.1. This paper also addresses the issues encountered in development of SRM for composite fuselage and procedures to overcome such issues. Various issues related to repair material and fastener selection is discussed. Additional knockdown factors in strength for replacement of countersunk fasteners in composite fuselage, local thermal conditions and inspection requirements for various repair configurations are also discussed.

Structure Details and Damage Scenarios

The structure under consideration is basic acreage skin bounded by Stringers in circumferential direction and frames in the longitudinal as shown in Fig.2. The average bay size for section of fuselage is considered for analysis. The maximum size of damage that can be covered within SRM is limited by fail safety requirements of the airplane.

For damage within a bay, based on the proximity of damage to stringer or frame many damage scenarios are possible. Each of them warrants a specific repair solution. For discussion in this paper, damage centered between undamaged stringers and frame is chosen for analysis.

Repair Plate Loading

On any given skin panel bounded by frames and stringers, the primary loading are Axial, Hoop and Shear loads. Global Finite Element model can be used to obtain the load on skin panels. All applicable load cases (Aerodynamic and Internal Loads) need to be addressed to ensure strength capability of the repair. This calls for a down selection of the load cases enveloping all possible loading scenarios. In order to cover all zone an envelope of maximum Axial, Hoop and Shear Load for the given skin gage is used for qualifying the repair. Loading on the repair is shown in Fig.3. From the load flows, the load on bolt row is calculated based on pitch and the bolt load for a line of bolt is distributed. Based on the expected load distribution the fastener with highest bearing load (F1) and highest by pass load (F2) are chosen for detailed bearing bypass analysis.

Load in the repair plate is expected to be at maximum close to the damage. Thus these locations (like fastener F3) are most critical for fatigue analysis and are investigated in detail.

Analysis Using Classical Methods

Repair Plate Sizing

The repair plates are sized to restore the stiffness lost due to damage and repaired configuration to have a similar stiffness when compared to undamaged structure. This prevents redistribution of load and internal loads generated for undamaged configuration remains valid.

$$E_{composite} \times t_{composite} \le E_{repair} \times t_{repair} \le 1.3 \times E_{composite} \times t_{composite}$$

$$G_{composite} \times t_{composite} \le G_{repair} \times t_{repair} \le 1.3 \times G_{composite} \times t_{composite}$$

The load increase is limited to 1.3 in order to restrict load redistribution and to keep the loads models valid. Repair plate is sized to be at least being of equal stiffness as the base structure to avoid overloading of surrounding structure.

Bolt Load Distribution

The method used for bolt load distribution [3] is a simple way of analyzing redundant structure like doublers reinforcement using shear lag parameter,

$$K = \sqrt{\left(\frac{Gt_e}{Eb}\right)\left(\frac{1}{A_1} + \frac{1}{A_2}\right)}$$

and equivalent thickness $t_e = \frac{bk}{sG}$. Here k is stiffness of the joint and joint stiffness for a composite-metal joint is discussed by Alexander Rutman et al. [4]. The axial stress in repair plate in case of fastener rows A2 and H1 is calculated as $f_x = \frac{P}{A_t}\left(1 - \frac{\cosh Kx}{\cosh KL}\right)$ and for case of fastener row A1 is calculated as

$$f_x = \frac{P}{A_T A_2 \sin h\,K\,L}\left[A_2 \sin h\,KL - A_2 \sin hKx + A_1 \sin hK\,(L-x)\right].$$

The load in repair plate is calculated as $P_x = f_x \times A_2$ and bolt load is $P_{bolt} = P_{x+s} - P_x$. The bypass load in the composite skin is $P_{by-pass} = P - P_x$. The explanation of the terms used in equation and result for joints A1, A2 and H3 are presented in Fig.4 and Fig.6 respectively.

Thermal Loads

As the composite and metals have different coefficient of thermal expansion (CTE, α), the thermal cycling experienced by repair during the operation cycle (ground -air-ground cycle) of aircraft adds to load experienced by the joint. The total thermal load on the joint is distributed across the joint similar to mechanical load. Thermal Load on the joint is calculated as $P_{thermal} = E_{repair} A_2 (\alpha_{repair} - \alpha_{composite}) \Delta TL$. The thermal loading has significant impact joint load especially with Aluminum repairs and restricts the size of monolithic repairs. Thermal loading also plays a vital role in fatigue life of the repair and needs to be accounted for as additional load.

Analysis Using FEM

Finite Element model contains representation of composite fuselage skin and stringer idealized as shell with composite properties. The repair plate is idealized as metallic shell and connected to composite skin using Ruttman fastener representation [4] and same is shown in Fig.5. Bolt loads are calculated based on load transfer factors for Axial, Hoop and Shear loads obtained from separate unit load cases. The Axial Load is applied as point load at center of fuselage, Hoop load is simulated with application of pressure while Shear Load is simulated with applied torque at center of fuselage. The idealization and load application is shown in Fig.5. Results from FEM are compared against classical analysis and is presented in Fig.6.

Qualification of Bolted Joint

Static analysis of repair, involves qualification of bolted joint, ensuring bolt pattern as designed ensure complete effectiveness of repair and the repair is capable of carrying the induced loads. Bolt load is obtained either from classical analysis or FEM. While the assumption of classical analysis is that repair is completely effective and carries complete load of the skin panel it replaces. In FEM a more realistic loading is obtained and bolt loads are lower near to damage as seen in Fig.6 (A1). Away from the cutout the bolt load distribution is similar from both classical analysis and FEM. Based on the Bolt loads and by pass strains the joint is qualified. Fatigue and Damage tolerance analysis is done for critical fasteners.

Bearing Bypass Check

In multi-row joints, fastener holes are subjected to bearing and bypass loads that are reacted elsewhere in the joint. The effect of that type of loading on the stress distribution of a frictionless hole loaded by rigid pin was investigated by Naik and Crews [5]. Using similar principles the failure envelop is developed based on the strain allowable of the laminate and laminate bearing allowable. From the bypass load, axial, hoop and shear strains are calculated using the equations

$$\varepsilon_x = \frac{N_x}{t \times E_x} + \frac{N_y \times -\upsilon_{xy}}{t \times E_y}, \quad \varepsilon_y = \frac{N_y}{t \times E_y} + \frac{N_x \times -\upsilon_{xy}}{t \times E_x}$$

and $\gamma_{xy} = \dfrac{N_{xy}}{t \times G_{xy}}$.

Based on these equations the strains for other ply directions (+45 and -45) are calculated using the equations

$$\varepsilon_{45} = \frac{\varepsilon_x + \varepsilon_y}{2} + \frac{\gamma_{xy}}{2} \text{ and } \varepsilon_{-45} = \frac{\varepsilon_x + \varepsilon_y}{2} - \frac{\gamma_{xy}}{2}.$$

The bearing stress on the laminate is calculated as $S_b = \dfrac{P_b}{d \times t}$. Alternatively bearing stress and by-pass strains are directly recovered from finite element analysis results. Typical bearing bypasses envelop and fastener loading for the two critical fasteners is presented in Fig.7. Margin of Safety is graphically calculated from the bearing by-pass envelop.

Fatigue Analysis

As the repairs are intended to be permanent, fatigue analysis is carried out to ensure the durability of the repair location. Loads for all applicable location can be extracted from fatigue load model. Based on duty cycle for given mission/duration of flight, fatigue stress spectrum is developed for location. The critical location was identified as location having the least fatigue life. As loads model represent the baseline structure, correction factors are applied to the calculated stress. Using the worst corrected fatigue stress at the critical location fatigue stress for bolted joint is calculated. Margins are computed based on stress life approach and after applying the applicable modifying factors [6]. If the repair's fatigue life is greater than the design service objective (DSO = 44000 flight cycles) defined, repair is classified as permanent. For repair configuration not meeting the fatigue life requirement, safe life is calculated.

Damage Tolerance Analysis

Damage tolerance analysis is done to arrive at inspection requirements for the repair. An edge crack of minimum detectable crack length is assumed as the most critical fastener location along with secondary corner cracks (penny cracks). All the cracks are grown simultaneously under equivalent fatigue spectrum. The total life to failure (crack length reaches critical crack length) determines the crack propagation life. Based on various stages of growth of primary and secondary crack growth suitable inspection interval is determined. Stress intensity factors are calculated based on relations presented in NASA TM-X-73305 [7]. Crack propagation life is calculated based on Paris Equation and method discussed by

Barsom et al. [8]. Various phases of crack propagation are presented in Fig.8.

Based on severity of fatigue spectrum and geometric parameters effecting crack growth, most critical crack growth for repair plate yields the following safety-by inspection program in accordance with Federal Aviation Regulations (FAR) 25.571 airworthiness standard [9].

Inspection Type : Detailed Visual
Design Service Objective : 48000 flight cycles,
Repeat inspection interval : 20000 flight cycles, cut off
Threshold inspection interval : 36000 flight cycles,
cut off

Issues Related to Bolted Repairs (Fig.9)

The repair material selection for different regions of aircraft fuselage have to meet the requirements for corrosion, lightning protection and flammability. The keel region of the aircraft is wet zone and aluminum repair is not preferred as permanent repair due to occurrence galvanic corrosion. In the cargo storage region Aluminum repairs are no allowed due to flammability requirements as thin aluminum sheet can cause a blowout. The lightning threat is high in the crown region of the fuselage and minimum thickness requirements have to meet zone identified as high threat regions. Titanium is preferred material for repair for keel and forward crown region. Both titanium and aluminum meet all the laid out requirements for aft crown regions.

Replacement of existing fastener in composite skin requires special analysis. All components are connected to skin using counter sunk fasteners. If there is damage in vicinity of these attachments and this existing countersunk fastener needs replacement, special washers are required to fill the countersunk hole. The countersunk hole reduces the bearing area and adversely impacts the bearing-bypass allowable. For analysis of such fasteners a suitable modifications have to be done to bearing bypass allowable to account for the reduced bearing and bypass area.

In repairing skins of large ply count multiple plates are required to meet the thickness requirement based on stiffness criteria. This nested repair configuration has possibility of having hidden damages. The inspection requires removal of external plate and frequency of inspection gets enhanced for detection of such hidden damages.

Conclusion

The composite fuselage for large commercial airliners can be repaired similar to metallic fuselages using similar techniques. The development of repair configuration requires additional analysis to address specific issues related to composite material system. There are restrictions in material selection, fastener types and patterns due to composite material system but there is enough flexibility available for airframe to be repaired in field, with downtimes comparable to metallic airframes. The new repair techniques will evolve as usage of composite material system becomes more widespread and operation of all composite airframes will be more cost effective and reliable in future.

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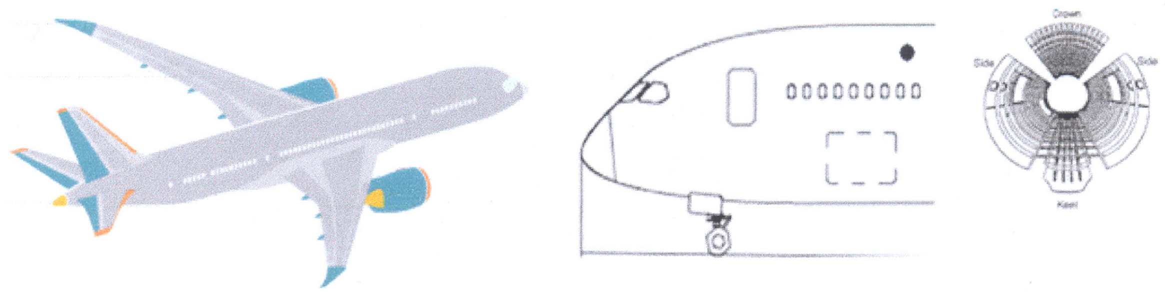

Fig.1 Usage of Composite [2] and Repair Location

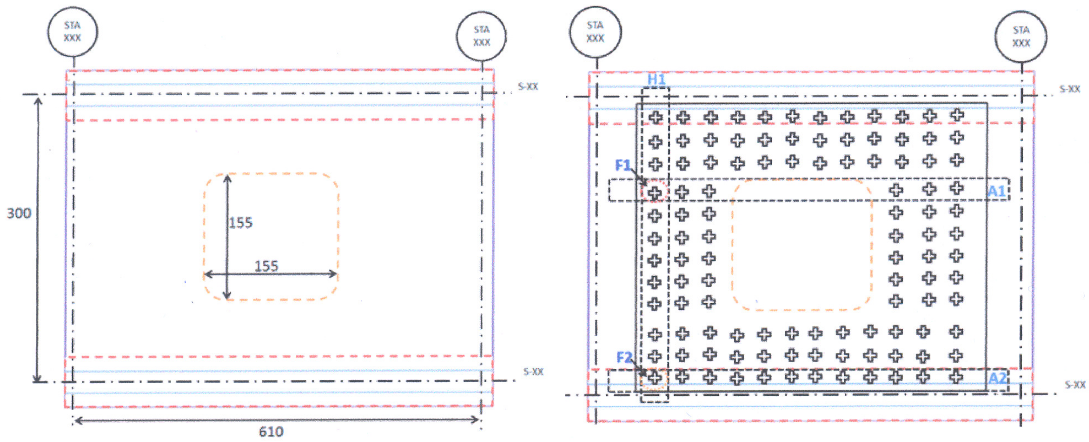

Fig.2 Damage Details and Repair Layout (All dimensions are in mm)

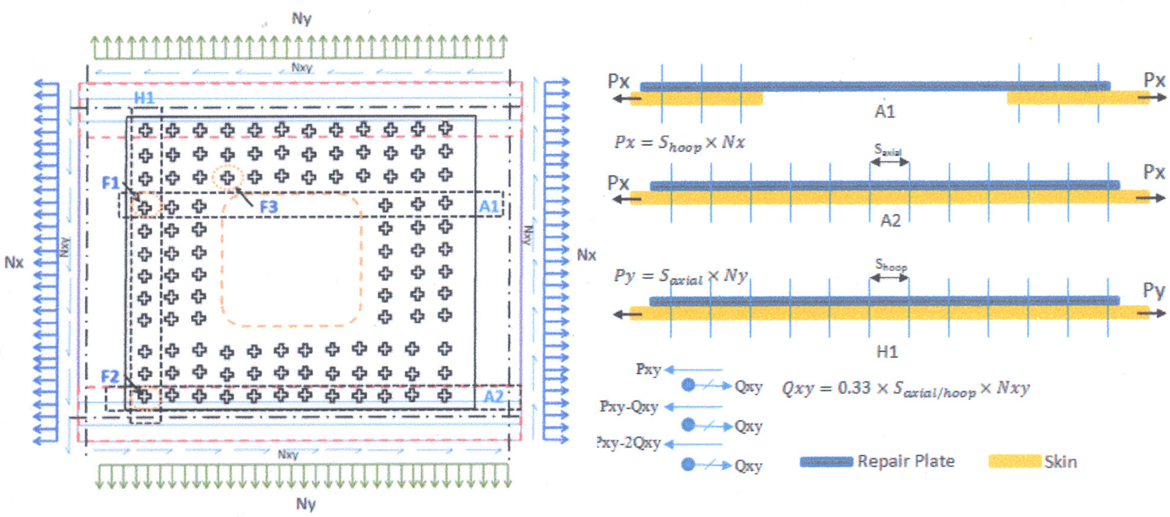

Fig.3 Critical Fasteners and Distribution of Load

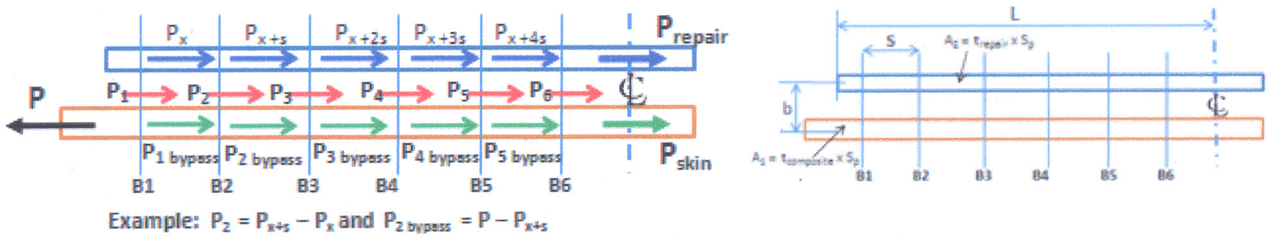

Fig.4 Bolt Load Distribution and Explanation of Terms in the Equation

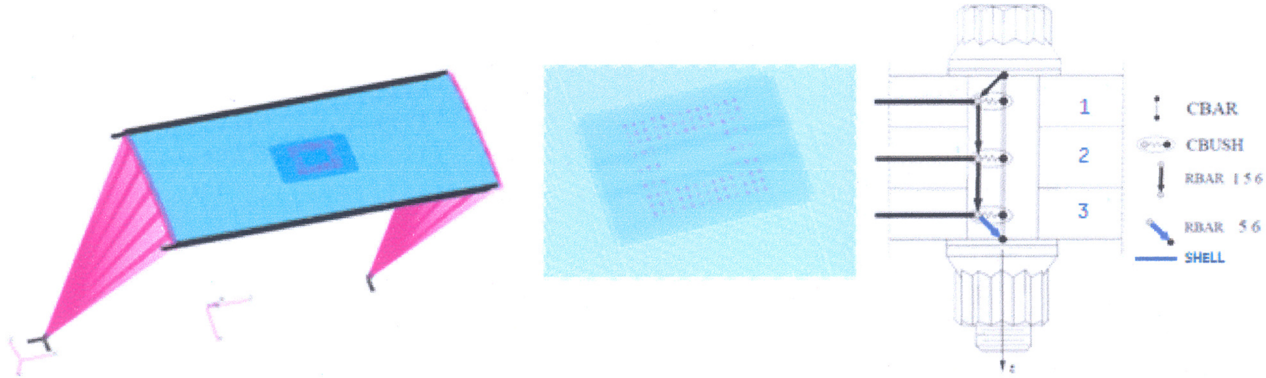

Fig.5 Finite Element Model and Bolt Idelization [4]

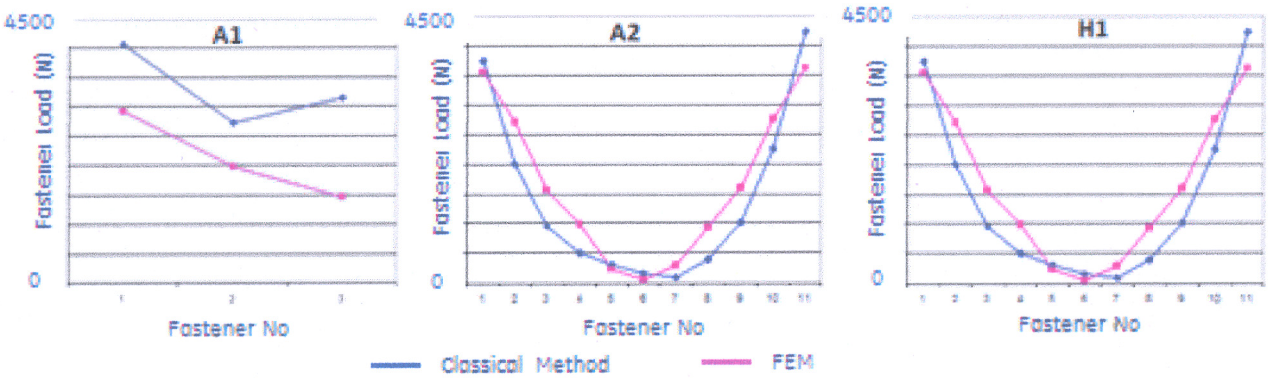

Fig.6 Comparison of Bolt Load Distribution Using Classical Approach and FEM

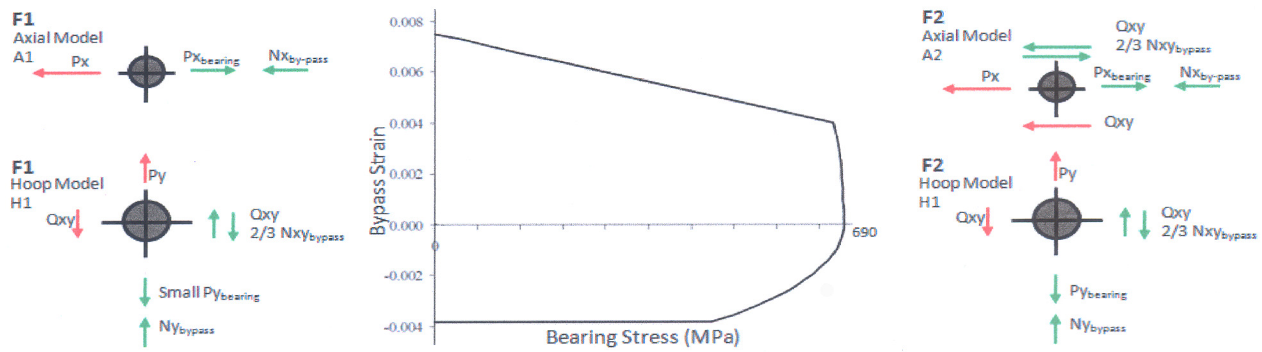

Fig.7 Bearing Bypass Envelop and Critical Fasteners Loading

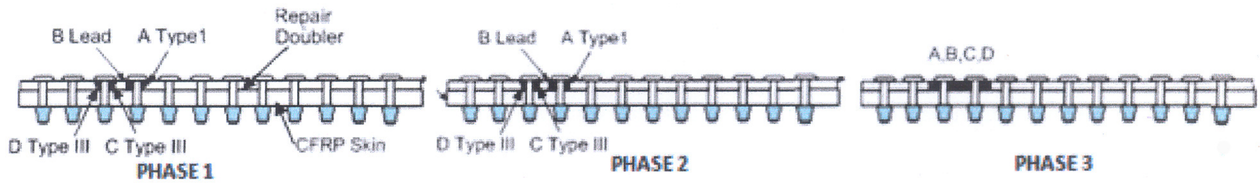

Fig.8 Phases of Crack Propagation

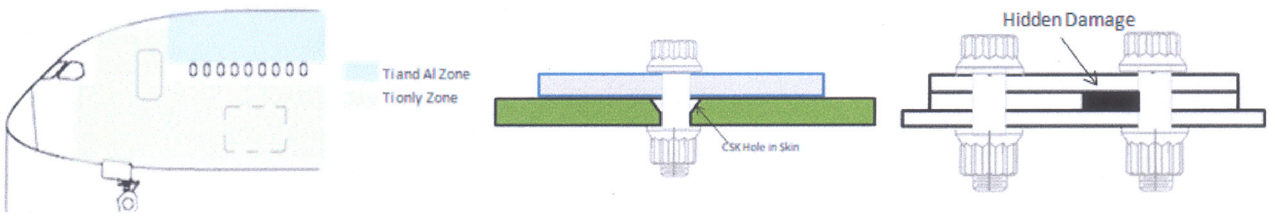

Fig.9 Issues Related to Bolted Repairs