N95- 28480

Application of Damage Tolerance Methodology in Certification of the Piaggio P-180 Avanti

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ABSTRACT

The Piaggio P-180 Avanti, a twin pusher-prop engine nine-passenger business aircraft was certified in 1990, to the requirements of FAR Part 23 and Associated Special Conditions for Composite Structure.

Certification included the application of a damage tolerant methodology to the design of the composite forward wing and empennage (vertical fin, horizontal stabilizer, tailcone and rudder) structure. This methodology included an extensive analytical evaluation coupled with sub-component and full-scale testing of the structure.

The work from the Damage Tolerance Analysis Assessment was incorporated into the full-scale testing. Damage representing hazards such as dropped tools, ground equipment, handling, and runway debris, was applied to the test articles. Additional substantiation included allowing manufacturing discrepancies to exist unrepaired on the full-scale articles and simulated bondline failures in critical elements.

The importance of full-scale testing in the critical environmental conditions and the application of critical damage are addressed. The implication of damage tolerance on static and fatigue testing is discussed. Good correlation between finite element solutions and experimental test data was observed.

INTRODUCTION

Advancements in the use of composites as an improved performance material in primary aircraft structure has been steadily increasing. The primary advantages of composites usage include improved fatigue life and corrosion resistance, as well as lower weight. Several business/commuter FAR 23 aircraft have been certified with composite primary structure, as well as FAR 25 transport aircraft such as the Airbus A320 and the ATR 72. The Piaggio P-180 Avanti Program is an example of a FAR 23 aircraft with extensive use of composite primary structure.

Figure 1 illustrates the composite structural components on the P-180. As the industry moves forward into the twenty-first century, composites usage will increase dramatically on aircraft like the Boeing 777/787 and the McDonnell-Douglas MD12X.

To date, all aircraft have been certified to their requisite FARs in addition to a series of Special Conditions which apply directly to the extensive use of the composites in the airframe design. Although there are different certification criteria applied to each category of aircraft, the criteria for use of primary composite structure have one aspect in common: they involve the application of a damage tolerance methodology.

DAMAGE TOLERANCE

Damage Tolerance is basically the application of known damage threats to the aircraft structure during its typical lifetime usage and demonstration that this damage does not alter the safe operation of the aircraft. Failsafe Analysis should not be confused with Damage Tolerance because it deals with demonstrating adequate redundancy with critical load paths severed or incapacitated. Natural threats include runway debris, lightning strike, engine wash, bird strike, or even hail. Accidental threats typically encompass dropped objects such as tools, aircraft parts, luggage/cargo, or other maintenance related damage such as saw cuts or punctures.

These types of damages can be further quantified into the potential level of damage caused as a result of the incident. Threats which cause barely visible impact damage (BVID) may not be easily recognized and therefore not usually repaired. The application of larger threats causes visible or highly visible impact damage (VID) which would typically be repaired when discovered. Since BVID may not be easily detected, and repaired, the aircraft structure must be capable of ultimate and repeated loads, with BVID. VID is repaired but since there may be some time before the structure is inspected, aircraft structure must be capable of limit and repeated loads, with VID.

Damage threats may also exist in the form of manufacturing defects such as: cut or missing laminate plies, bondline voids or cured laminate voids (delaminations/porosity). These defects must also be evaluated in the design of composite aircraft primary structure.

Finally, the environmental effects on composite structure must also be considered in conjunction with the application of damage tolerance since material behavior is typically affected by the environment.

DAMAGE IDENTIFICATION AND EVALUATION

In the case of the Piaggio P-180 program, the application of damage tolerance can be separated into three distinct phases:

1. The identification and susceptibility of perceived threats.

2. An analytical evaluation of the effect of damage threats.

3. Test evaluation of the structure including the damage threats.

By nature of composite structure design, the aircraft structure is inherently damage tolerant. Most primary composite structure is designed to a strain cut-off value indicative of the most critical environment and strength property. Additionally, the use of composite laminate analysis programs in most applications assumes part failure after one lamina ply has failed ("first ply failure").

In the identification and susceptibility analysis phase of the damage tolerance program, potential damage from any outside source was considered. The potential types of damage for the Piaggio P-180 program are tabulated in Figures 2 and 3. The types of damage can be separated into two categories: 1) Manufacturing Damage and 2) Flight Operational Damage.

Manufacturing damage was typically evaluated analytically. The analytical substantiation included a damage tolerance analysis assessment which consisted of Hazard Analysis, Damage Susceptibility Evaluations and Failure Mode Analysis. The Hazard Analysis was performed to identify and quantify the frequency and severity of the probable hazards to which the aircraft was expected to be exposed to during its service lifetime. Impact tests were performed on actual structure to quantify the levels of damage experienced from the probable hazard (Damage Susceptibility Evaluation). An analytical evaluation of potential failure modes, caused by both manufacturing defects and in-service damage, was conducted using the NASTRAN finite element model. Damage was represented in the finite element model by either removal of specific elements or removal of laminate plies in the material property cards. Strength checks were made for the new load distributions as a result of damage.

Flight operational damage, typically in the form of impacts and punctures to the aircraft structure, was analytically modelled as missing NASTRAN elements. In this case, damage representing BVID was applied to ultimate load and VID was applied at limit load.

Other analytical substantiation work included evaluating "fail-safe" design features by removing critical elements and showing load redistribution within the remaining structure, with positive margin of safety at limit load.

CERTIFICATION TESTING

Usually, in a typical damage tolerance program, the testing phases of a "building-block" approach divide into coupon testing, followed by larger scale element testing, and concluding with the full-scale testing. This method is used to adequately evaluate all "unknowns and structural concerns" so surprises do not occur during the final full-scale testing phase. The "building-block" approach is not a specific FAA requirement. The original P-180 composite structure certification program included the "building-block" approach. However, after completion of the coupon testing phase, the certification program was changed to emphasize the full-scale testing.

With the changes in the certification testing, the program was re-scoped to include environmental condition of the full-scale test article. The testing of fully-saturated structure eliminates many of the analysis headaches associated with trying to correlate RTD predictions to RTD testing results and substantiating the ETW predictions by the coupon and element testing at their critical environments. The revised testing program is summarized in Figure 4.

FATIGUE TEST SPECTRUM

The original fatigue spectrum proposed by Piaggio to the certification authorities was based on a FAR 25 transport category spectrum. This was not accepted because most FAR 23 aircraft fly in a more rigorous spectrum than a 767 flight from New York to Los Angeles, for example. The spectrum was revised to take into account the work NASA had developed from actual flying time in other FAR 23 aircraft. The resulting spectrum increased the numbers of take-offs and landings and increased the "G-loadings" and frequencies on many of the flight maneuvers.

The next issue to tackle was how to convert the fatigue spectrum into a full-scale test spectrum. The problem was that the aircraft was designed with a metal fuselage and wing structure and a composite canard and empennage. In metal aircraft structure, loads typically in excess of 60-

70% limit load are clipped from the spectra. This is due to the plastic behavior of metals where high loads tend to blunt the tip of any fatigue crack that may be growing as a result of the applied loads. Composites are more brittle and are affected by high end loads. Contrary to the behavior of metals, composite spectra include the high end loads but truncate the low end loads, since loads typically around 30% of limit load show no effect on the fatigue life. An example of an S/N curve for metals and for composites is depicted in Figure 5.

The composite forward wing and empennage also included several metal fittings. For FAR 23 aircraft, a fatigue analysis can be used for metal parts in lieu of testing if a life scatter factor of 8 is applied to the analysis. This analysis coupled with fail-safe redundancy features in the fitting designs was sufficient to certify the metal fittings. Therefore, a composite derived spectrum could be applied to the empennage and forward wing tests, since the fittings were certified by other means.

Using in-house coupon test data and other published literature, a truncation level of 35% of limit load was established for all the P-180 composite structures. The resulting test spectra included all high end lift loads up to and including limit load.

FULL-SCALE STATIC AND REPEATED LOADS TESTING

The prototype flight articles as well as the structural test articles were fabricated in Sikorsky Aircraft's Composite Development Center. The only difference between the flight and test articles was that a higher level of manufacturing flaws were permitted on the test articles. These flaws were documented in inspection reports, and eventually allowed the inspection criteria to be updated after successful completion of all certification testing. Other flaws, in the form of simulated adhesive bondline delaminations were also introduced in the manufacturing sequence.

The manufacturing "tool-proof" articles were utilized to perform the impact testing evaluation to help distinguish between Barely Visible Impact Damage (BVID) and Visible Impact Damage (VID), and the associated energy levels and impactor geometries. BVID was considered to be at the threshold of detectability where the damage could be seen with an unaided eye at a distance of approximately 2 feet. Prior to any static testing, all BVID was applied to all the test articles. The different types of BVID are tabulated in Figure 6.

Prior to clearance of the flight test airplanes, it was necessary to successfully demonstrate Room Temperature Dry (RTD) static ultimate load. There were some initial surprises in the static testing. The forward wing failed prematurely at 120% limit load, due to the honeycomb core being installed in the wrong ribbon direction. Another test article, with the core in the correct ribbon direction, successfully demonstrated ultimate load-RTD. There was also some "teething" problems with the empennage static test. The tailcone skins, which were designed to post-buckle above limit load, started buckling at about 90% limit load. A repair was made to the skin, and the test article made it to 120% limit load where the buckled skins caused a bulkhead to buckle and fail. The tailcone design was then changed to be shear resistant to ultimate load.

A revised design empennage assembly then successfully demonstrated RTD ultimate load with BVID. Flight testing of the aircraft was allowed to commence. All BVID on the test articles was re-measured to insure that no flaw growth had occurred during ultimate load. The test articles were then disassembled and placed into environmental conditioning chambers. The chambers were heated to 180°F and 87% relative humidity (RH) to accelerate environmental conditioning. Rider coupons representative of the thinnest and thickest laminate were periodically weighted until all specimens showed a minimum moisture uptake of 1.1% (considered saturated based upon using Fick's Law of moisture absorption).

After environmental conditioning was complete, the test articles were loaded back into the test frames. Environment tents were placed around the test structure and the test environment was brought up to 160-180°F and 82-87% RH. A life time of repeated loads was applied to the test articles. One lifetime represented 30,000 flight hours and the spectrum loads were applied in blocks of 3000 hours, with inspection at the end of each block. At the conclusion of the 30,000 equivalent flight hours of repeated loads testing, the structures were loaded back up to ultimate load at elevated temperature wet environment. Both the forward wing and empennage successfully demonstrated ETW ultimate load with no evidence of growth from the BVID.

FULL-SCALE DAMAGE TOLERANCE TESTING

After careful inspection of all the structure, VID was then applied. The types of VID imposed on the test articles is tabulated in Figure 6. The VID was instrumented with strain gauges and acoustic emission sensors. Other types of damage in the form of skin cuts/tears were applied prior to commencement of damage tolerance testing. These particular locations had been selected using the finite element model output, and were indicative of the most highly loaded external structure. Skin tears/cuts were considered to be highly visible between .5 and .75 inches depending on the applicable structure. Like the VID, the skin tears were instrumented for close monitoring throughout damage tolerance testing. Skin tear damage areas are tabulated in Figure 7.

The damage tolerance testing was relatively uneventful for the VID and skin tears. All impacts and cuts were periodically inspected every 9000 equivalent flight hours, and no evidence of flaw growth was experienced. There were instances where adhesive bondlines would make noise which was picked up by adjacent AE sensors. The bondline noise was believed to be the result of the brittle behavior of the adhesive and cracks may have initiated due to previous static ultimate and fatigue testing. Since all the joints were designed with fasteners carrying ultimate load, no attempt was made to repair bondlines, and they were passively monitored.

At the conclusion of damage tolerance testing, the test articles were successfully loaded to ETW limit load to demonstrate residual strength after two lifetimes. The residual strength demonstration was conducted to comply with the FAR Special Conditions, even though the structure had seen ETW limit load three additional times during each lifetime. All VID and skin cuts were re-inspected and again no evidence of flaw growth had appeared.

Following damage tolerance testing, Severe Damage Demonstrations were conducted to further determine the damage tolerance of the structure as designed. More skin cuts/tears, as tabulated in Figure 8 were imposed on the test structure (without repairing any of the other damage) and ETW limit load was applied. The load case applied to each new sever damage was indicative of the most critical load case for that portion of structure. The new damage locations were strain gauged and acoustically monitored. No evidence of flaw growth was exhibited during the phase of testing and strain gauge trends remained basically the same.

After Severe Damage Demonstrations followed the Repair Substantiation phase of testing. In this phase, all the different types of possible field repairs were evaluated. Most of the VID and Severe Damage were repaired using repair techniques approved in the field repair service manual. These repairs included: prepreg repairs, wet lay-up repairs, and bolted/bonded joint repairs. They were typical of what could be expected in the field, whether at an authorized repair station or "in the middle of nowhere".

With the repairs in place (with added strain gauges and AE instrumentation), the structures were successfully loaded to ETW ultimate load. No evidence of any acoustic noise from the repairs was exhibited and no change in stiffness was measured as a result of these repairs. The final test phase was the most interesting. In these final tests, the structure was to loaded in its most severe load environment until failure.

Since several different load cases affect the criticality of the empennage structure, the plan provided for multiple load applications until failure. The first load case application was horizontal stabilizer gust down. The test was taken to 190% limit load (limitation of test facility) and reacted load in excess of the minimum three second requirement. Since acoustic emission noise was recorded during this load excursion, the structure was re-inspected. A crack was found in the tailcone center bulkhead flange near the top of the fuselage at approximately Butt Line 0. Since it was obvious that the test results would be facility limited, it was decided to apply a hybrid load case. The hybrid load case was a combination of Engine-out side load on the vertical fin and horizontal stabilizer gust down (two singular load cases which can never occur together). The structure was loaded to facility capacity at 190% limit load. The load was kept to see if a failure would eventually precipitate. Approximately two minutes into the load application, some "oil-canning" noises were distinctly heard. The center bulkhead structure had apparently buckled, forcing load to redistribute into the aft spar. Then, the right hand side (RHS) spar cap immediately failed in column compression redistributing load into the left hand side (LHS) cap and web severing the latter from the cap and completely failing the vertical fin assembly. A post-test teardown of the ground test article helped determine the failure scenario. The failure of the aft spar cap was as anticipated. Results of post-test strain gauge surveys indicated reasonable correlation (usually within 20%) between analytical predictions and measured strain.

The results of the damage tolerance testing were utilized to determine an inspection interval for the production fleet. A life scatter factor of 3 was to be applied to the test results and was accepted by the certification authorities. Another scatter factor of 3 was established to allow an inspection two inspection intervals to miss potential damage during visual inspections. The damage tolerance testing demonstrated 30,000 hours of life with no flaw growth. After application of the various factors, the permitted inspection interval became 3300 hours (= 30,000/9) which was further reduced to 3000 hours to be consistent with the projected service intervals for the P-180 aircraft.

CONCLUSION

The application of damage tolerant methodology helped design an aircraft structure which demonstrated better than adequate safety margins in the presence of manufacturing defects, impact flaws, skin cuts, and bondline inclusions.

An emphasis was placed on full-scale testing demonstrations, in the most critical environment with moisture saturated structure. The certification effort was international in scope, because the type certificate holder Piaggio had to be certified first by Registro Aeronautico Italiano (RAI), and then by bilateral agreement with the FAA.

Overall, the program was finally successful because compliance with the applicable FARs and Special Conditions was achieved with the successful completion of the full-scale testing results. As several issues regarding composite certification were addressed during the program, it is hoped that the results of the P-180 program will serve as a precedent for future composite aircraft certification programs because many issues have already been addressed.

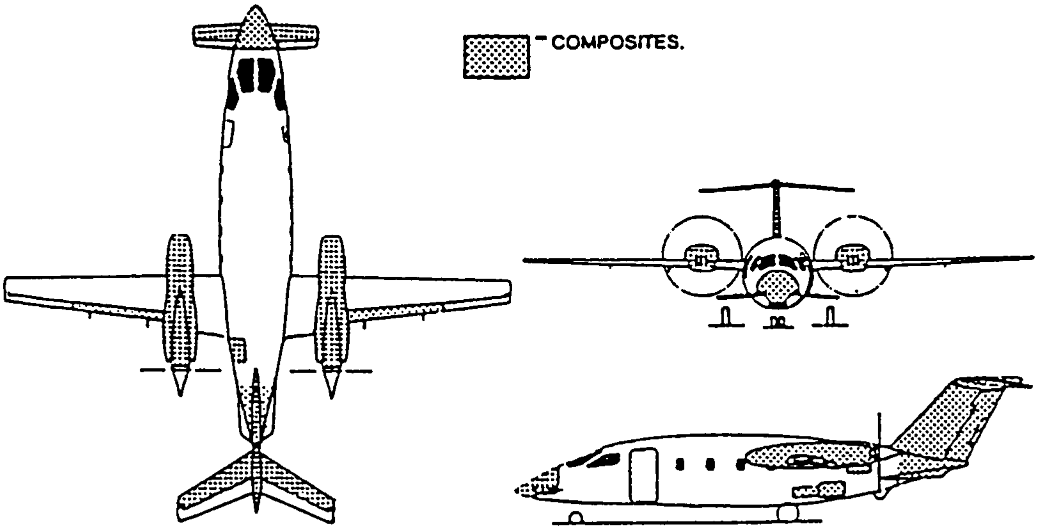


FIGURE 1. - COMPOSITE COMPONENTS ON THE PIAGGIO P-180 AVANTI.

ELEMENT	DEFECT
SANDWICH SKINS	BOND VOIDS FOREIGN OBJECTS CONTAMINATION INCORRECT CORE INCORRECT RIBBON DIRECTION LAMINATE DIMPLING DARTING/MISALIGNED PLIES ON CORE RAMP
SOLID LAMINATE	VOIDS WRINKLES FOREIGN OBJECTS CONTAMINATION
ASSEMBLY/JOINTS	BOND VOIDS FOREIGN OBJECTS CONTAMINATION VOIDS/BAD CURE/DELAMINATIONS/ DAMAGE DURING ASSEMBLY

FIGURE 2. - IN-PROCESS MANUFACTURING HAZARDS.

FIGURE 3. - FLIGHT OPERATIONAL HAZARDS.

STRUCTURE	HAZARD
FORWARD WING HORIZONTAL STABILIZER VERTICAL FIN/TAILCONE RUDDER	DROPPED TOOL DROPPED PART FOOT TRAFFIC GROUND EQUIPMENT HANDLING RUNWAY DEBRIS HAIL ENGINE WASH LUGGAGE

FIGURE 5. - S/N CURVE (ALUMINIUM VS. COMPOSITE).

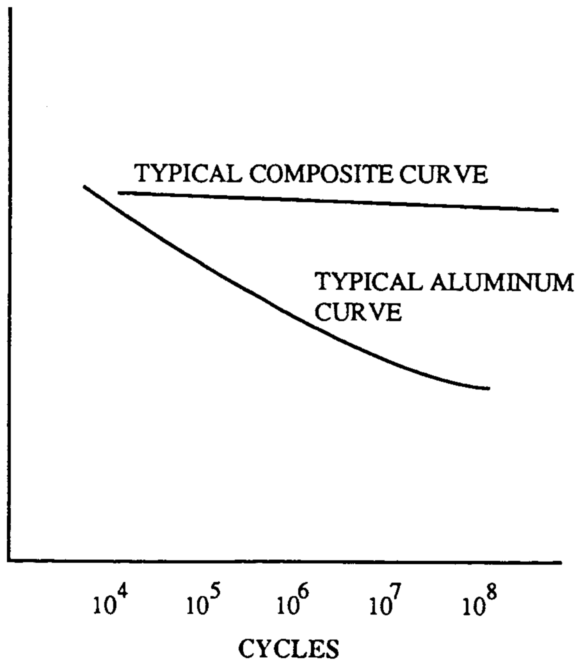
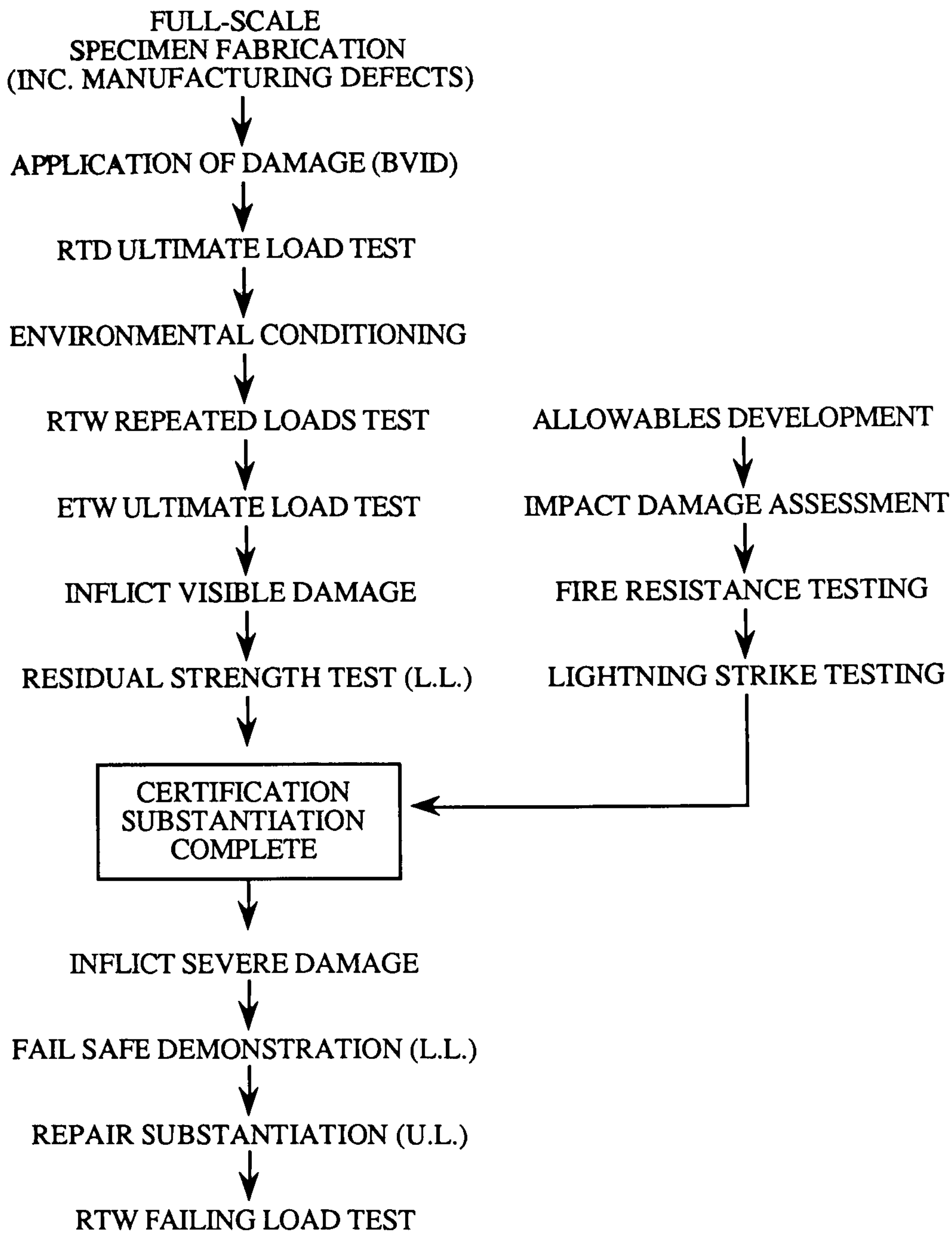


FIGURE 4. - REVISED CERTIFICATION TEST ACTIVITIES FLOW CHART.



STRUCTURE	IMPACT HAZARD	IMPACTOR DIA.	IMPACT ENERGY	BVID	VID
FORWARD WING					
MAIN BOX	DROPPED TOOL	1.5	180/108	NO/NO	
	DROPPED PART	.4	108		YES
	GROUND EQUIPMENT	.25	180		YES
TRAILING EDGE	DROPPED TOOL	.5	180		YES
LEADING EDGE	HAIL	.88/.25	24	NO	
	GROUND EQUIPMENT	.5	180		YES
HORIZONTAL STABILIZER	DROPPED TOOL	1.5	180		
MAIN BOX	DROPPED PART	.5	108	NO	
	GROUND EQUIPMENT	.25	180	YES	YES
LEADING EDGE	HAIL	.88	24		
	HAIL	.25	24	NO	
				YES	
VERTICAL FIN	DROPPED TOOL	1.5	180		
	GROUND EQUIPMENT	.25	180	YES	YES
	ENGINE WASH	.88	144		YES
	HAIL	.88/.25	24		
				NO/NO	
TAILCONE	DROPPED TOOL	1.5	480		YES
	DROPPED PART	1.5	288		YES
	GROUND EQUIPMENT	1.5/2.5	180		YES/YES
	ENGINE WASH/RUNWAY DEBRIS	.88/.25	144		YES/YES
	LUGGAGE	1.5./.25	37		
	HAIL	.88/.25	24	NO/YES	
				NO/NO	
RUDDER	GROUND EQUIPMENT	1.5	180/108		YES
	HANDLING DAMAGE	.25	180/108	YES	YES/YES

FIGURE 6. - SUMMARY OF APPLIED IMPACT DAMAGE.

STRUCTURE	LOCATION
FORWARD WING	UPPER SKIN BL 2.2 LHS
	LOWER SKIN BL 10.25 RHS
HORIZONTAL STABILIZER	UPPER SKIN BL 10.5 RHS
	LOWER SKIN BL 10.5 LH
VERTICAL FIN/TAILCONE	LHS VERTICAL FIN SKIN BETWEEN SPARS 1 AND 2 8" BELOW UPPER CLOSEOUT RIB
	VERTICAL FIN LEADING EDGE 36" BELOW CLOSEOUT RIB
	LHS VERTICAL FIN SKIN BETWEEN SPARS 3 AND 4 21" BELOW UPPER CLOSEOUT RIB
	TAILCONE - LHS 3" FORWARD OF AFT BULKHEAD NEAR STRINGER 4
RUDDER	LHS SKIN - APPROX. 2" ABOVE MID-HINGE FITTING

FIGURE 7. - SUMMARY OF DAMAGE TOLERANCE SKIN CUTS.

FIGURE 8. - SUMMARY OF SEVERE DAMAGE SKIN CUTS.

STRUCTURE	LOCATION
FORWARD WING	RHS AFT SPAR WEB BL 22 LHS UPPER SPAR CAP/SKIN BL 6.5
HORIZONTAL STABILIZER	RHS AFT SPAR WEB BL 44 RHS UPPER FORWARD SPAR CAP/SKIN BL 6
VERTICAL FIN/TAILCONE	LHS TAILCONE STRINGER #5 3.5" AFT OF FORWARD BULKHEAD RHS TAILCONE 1.9" ABOVE STRINGER #4 3.2" FORWARD OF AFT BULKHEAD LHS VERTICAL FIN #4 SPAR CAP/SKIN CUT AT APPROX. WL 49 VERTICAL FIN #1 SPAR WEB BL 0.0 AT APPROX. WL 52
RUDDER	LHS SPAR CAP APPROX. 34" ABOVE BOTTOM OF L.E. FORWARD SPAR WEB APPROX. 33" ABOVE BOTTOM OF L.E.