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AGING OF COMPOSITE AIRCRAFT STRUCTURES

TEARDOWN OF A DECOMMISSIONED B737-200 HORIZONTAL STABILIZER

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ABSTRACT

As more commercial and military airplanes are required to maintain operational capability beyond their original design service objectives, it has become necessary to assess the structural health of aging aircraft to ensure their airworthiness and structural integrity for continued service. Previously, most of the aging aircraft studies have focused on metallic structures; however, as more composite components are certified and used on primary and secondary aircraft structures, it is crucial to address this aging concern for composite components as well.

To use advanced composite materials in aircraft primary structures, it is necessary to demonstrate equivalent levels of safety, durability, and damage tolerance to metallic structures. The use of these materials also improves profitability to the airlines in terms of operating costs. With these concerns in mind, the objective of this paper is to summarize the findings on a Boeing-737-200 composite horizontal stabilizer teardown after 18 years of service.

INTRODUCTION

As more commercial and military airplanes are required to maintain operational capability beyond their original design life objective, it has become necessary to answer the questions of their continued airworthiness and structural integrity. Most research conducted on aging structures thus far has focused on metallic components; however, with the increasing use of composite materials in primary and secondary structures, it has become necessary to address the long-term structural health of aging composite components as well.

For advanced composite materials to be used in aircraft primary structure, it is necessary to demonstrate equivalent levels of safety, durability, and damage tolerance with respect to metal structures. These materials also improve profitability to the airlines in terms of initial and maintenance costs. Composites offer great advantages over metals: improved specific strength and stiffness, the ability to be tailored to design requirements in various directions,
enhanced cost advantages (especially assembly), operating and maintenance costs and the potential of tremendous weight savings, which is closely coupled to fuel savings. This paper examines the structural health of a composite Boeing-737-200 horizontal stabilizer after 18 years of service and presents data that substantiates aging of composite materials.

The B-737-200 right-hand (R/H) horizontal stabilizer, which is the subject of the current investigation, is one of the oldest medium primary structures built using composite materials. The structure was built as part of the National Aeronautics and Space Administration (NASA) Aircraft Energy Efficiency (ACEE) program using carbon-reinforced graphite epoxy composites and was certified by the Federal Aviation Administration (FAA) and entered service in August 1984. The structure was in service for 18 years and was retired in 2002 after completing 48,000 flights.

This paper provides an overview of the ACEE program, the investigative plan, details of the teardown activity, and results of the nondestructive inspection (NDI) as well as the physical, thermal, and mechanical test results. A nondestructive evaluation, in accordance with the recommended field methods, verified the current state of the retired stabilizer. Additional sophisticated NDI techniques, such as shearography and laser ultrasonic (UT) were also used to characterize any damage present in the structure. Destructive evaluation using the original test methods used for certification was performed to establish the end of service life capabilities of the structure. Current NDI, mechanical, and physical test methods are compared with those used in the development program to assess differences in capabilities between 1982 and today. The ultimate goal is to understand the aging mechanisms and characterize their effects on the composite structure.

As an attempt to characterize the structural health of the horizontal stabilizer after 18 years of service, the National Institute for Aviation Research (NIAR) acquired the aged structure in 2005 and conducted several nondestructive and destructive tests to assess its structural health. Generated data can be used to understand aging mechanisms on composite parts currently in service and to reveal main differences between damage mechanisms and damage accumulation in metallic versus composite components. This data could aid in future inspection and maintenance plans for composite structures to ensure their continued airworthiness and safety.

Nondestructive inspection (NDI) methods were conducted on the structure, prior to disassembly. The objective was to use current state-of-the-art inspection methods and evaluate their accuracy in detecting flaws in the structure. All inspections prior to teardown were conducted at the Sandia National Laboratories and at The Boeing Company. Methods used included thermography, Rapidscan™, laser UT, and pulse echo UT.
Destructive evaluation included thermal analysis, image analysis, and physical and mechanical tests according to the appropriate ASTM standards and/or the standards used to generate the baseline data as summarized in the NASA report [1].

Thermal analysis was conducted using both Dynamic Mechanical Analysis (DMA) and Differential Scanning Calorimetry (DSC) to determine the aged material’s glass transition temperature (Tg) as well as its degree of cure. Image analysis was conducted to characterize the state of the structure at the microscopic level and to detect possible flaws induced during manufacture or service. Physical tests were conducted to establish moisture and porosity levels in the composite structure and compare them to the design values. Mechanical tests were performed using the same standards applied to generate allowable values. Strength and stiffness values of coupons/elements extracted from the aged structure were compared to those obtained during the design phase.

The ultimate goal of the investigation was to assess the overall structural health of the composite stabilizer after 18 years of service, to identify possible changes in the material properties due to environmental effects and/or flight service, to provide data to help understand aging mechanisms in composite structures, and to gain confidence in the long-term durability of composite materials.

ACEE PROGRAM OVERVIEW

The B-737-200 graphite/epoxy horizontal stabilizer used for this investigation was built as part of the NASA ACEE, which was initiated in late 1975. This comprehensive program was motivated by the escalation of jet fuel prices and, thus, the necessity of developing new technology concepts in designing and building commercial aircraft components.

The ACEE comprehensive program was subdivided into four development areas: laminar flow systems, advanced aerodynamics, flight controls, and composite structures [2]. The ACEE composite program focused on using advanced materials in existing aircraft structural components thus offering a high potential for weight savings and thereby operating costs of commercial aircraft. A building block approach was adopted where composite structure development would start with lightly loaded secondary components followed by medium primary components, and finally conclude with wing and fuselage design. The program was terminated before the accomplishment of its major goals; i.e., before the implementation of composite materials in wing or fuselage primary structures.

Under the ACEE program, The Boeing Commercial Aircraft Company, Douglas Aircraft Company, and Lockheed Corporation contracted to develop the following secondary and medium primary components:
the upper aft rudder of the DC-10, the inboard ailerons of the L-1011, the elevators of the B-727, the vertical stabilizers of the L-1011 and DC-10, and the horizontal stabilizer of the B-737-200 as shown in Figures 1 and 2. All of these components yielded weight savings of at least 21.6% with a maximum of 28.4% weight reduction for the L-1011 vertical stabilizer.

Specific objectives of the B-737-200 composite horizontal stabilizer program included achieving a minimum of 20% weight reduction with respect to the metal structure, fabricating at least 40% by weight of its components using composite materials, demonstrating cost effectiveness of the structure, obtaining FAA certification for the structure, and closely monitoring its performance in service.

As part of the program, Boeing redesigned and manufactured five shipsets of the B-737-200 horizontal stabilizer using graphite/epoxy composites. The OEM adopted a fail-safe damage tolerant approach to certify the structure where the structure’s capability to sustain ultimate loads was validated by analysis supported by appropriate test evidence [3]. This complies with the code of
Federal Regulations Part 25 certification requirements for commercial transport aircraft.

Analysis was conducted to establish the most critical environment in which the stabilizer will be operating. It was found that a combined moisture level of 1% with temperatures extremes of 180°F and -65°F would simulate the worst environmental conditions to which the structure would be exposed. Allowable laminate, element, and subcomponent data was generated at these extreme conditions. Subsequent statistical reduction factors were implemented and provided high-confidence design values to use in the substantiating analysis.

Certification was completed in August 1982 and all five shipsets were deployed in 1984. The OEM closely monitored the performance of the stabilizers for 7 years. Outstanding performance was demonstrated with no in-service incidents attributed to the composite structure.

Figure 2. Composite Medium Primary Components Developed under the ACEE Composites Program, (a) Lockheed L-1011 vertical fin, (b) Douglas DC-10 vertical stabilizer, (c) Boeing 737 horizontal stabilizer [2]

COMPOSITE STABILIZER DESCRIPTION

The B-737 horizontal stabilizer shown in figure 3 consists of a structural box, a leading edge, a stabilizer tip, a fixed fiberglass honeycomb trailing edge and ribs, an elevator and body gap covers. Each stabilizer is secured to a metal center...
structure with three lugs at the rear spar and two lugs at the front spar. The structural box was redesigned using graphite epoxy composites such that maximum commonality is achieved with the existing metal configuration and that both structures are interchangeable in terms of geometry and aerodynamic shape. The bending and torsional stiffness of the composite stabilizer as well as its aerodynamic shape and plan form were made comparable to the metal stabilizer to meet control effectiveness and flutter requirements [1]. Furthermore the composite stabilizer was designed to be damage tolerant and its strength, durability, inspectability and serviceability equivalent to that of the metal structure.

Structural details of the existing metal and composite stabilizer designs are shown in figure 4. Several composite structural box arrangements were evaluated during the design phase of the composite stabilizer: a multiple rib concept, a honeycomb concept and a stiffened skin concept. The stiffened skin concept as shown in figure 4 was chosen due to its structural efficiency and minimal cost [1]. The structural arrangement consists of a single co-cured skin/ I stiffener combination, 191 inches long and 50.5 inches wide at the inboard end with stringers spaced 3.85 inches apart. Details of the stiffened skin are shown in figure 5. Mechanical fasteners were used to attach the stabilizer skins to the spars and ribs. Titanium Hi-Loks with corrosion resistant steel (CRES) collars and washers were used to assemble the lower skin. Inconel monogram blind fasteners, or “Big Foots” were used to assemble the upper skin. Honeycomb ribs were used because of the simplicity of the concept in terms of tooling, fabrication and cost and were fastened to the skins using graphite-epoxy shear ties. Details of the honeycomb rib cross sections are shown in figure 5. The spars were I beams consisting of two precured C-channels and two precured caps subsequently bonded together. Bolted steel spar lugs were used to attach the structural box to the fuselage center section. The spar lugs used two steel plates bonded and bolted externally to a graphite epoxy chord. The composite design yielded 21.6% weight savings with respect to the metal configuration with a final weight of approximately 206 lbs.
Figure 2. B737 Horizontal Stabilizer General Arrangement [1]
MATERIAL SELECTION

Various materials were evaluated and manufacturing producibility was assessed by fabricating test panels from each candidate material to determine factors such as drape tack, work time, and degree of difficulty in lay-up. In addition all materials were expected to comply to specific requirements and tolerances on prepreg and cured laminate properties. Finally additional factors such as available industrial database, demonstrated resin durability in various environments, supplier production experience, capacity and control, ability to produce all material forms were all considered. The systems evaluated included Narmco T300/5208 and T300/5235, Fiberite T300/934 and T300/976, Hercules AS/3501-5A and Hexcel T300/F263 and T300/F288. Narmco T300/5208 was the material of choice because it satisfied most of the evaluation criteria [1].
The predominant material form was fabric with selected use of tape. The structural details used hand-layup procedures throughout.

CORROSION PROTECTION SCHEME

The corrosion protection system used on the composite stabilizer aimed at isolating the graphite-epoxy areas from the aluminum structure to avoid galvanic corrosion of the aluminum as shown in figure 6. Corrosion protection was achieved by co-curing a fiberglass ply onto the graphite-epoxy structure in areas where the graphite epoxy structure interfaces the aluminum surface or painting the surface with primer and epoxy enamel. In addition to protecting all graphite-epoxy surfaces from galvanic corrosion, all aluminum surfaces were anodized or alodine treated, primed and enameled. During the assembly, wet polysulfide sealant was used between the graphite-epoxy and the aluminum parts. All fasteners through the aluminum surface were installed with wet polysulfide sealant.

![Figure 6. Corrosion Protection Scheme [1]](image)

LIGHTNING PROTECTION SCHEME

The lightning protection system developed for the advanced composite stabilizer provided an electrical path around the entire perimeter of the structural box by means of bonding straps connecting the aluminum leading edge, the aluminum rib cap of the outboard closure rib and the aluminum elevator spar. All these components were electrically grounded to the fuselage by means of the spar lugs and the leading and trailing edge ribs.
An aluminum flame spray was applied to the composite stabilizer's critical strike area, 18 inches outboard on the upper and lower skins in the event of a lightning strike. These outboard skin panels were insulated using a layer of fiberglass co-cured to the skin panels. The flame spray was applied to the outboard sections of the skin after fabrication. It was then electrically connected to the metal cap of the outboard closure rib using 4 mechanical fasteners. The conductive surface was then alodine coated primed and painted as shown in figure 7.

![Lightning Protection Scheme](image)

Figure 7. Lightning Protection Scheme

**SUPPORTING DATA FOR CERTIFICATION**

Extensive static, dynamic, sonic environments, electrodynamic effects and environment analyses were conducted to substantiate the structure for airline flight use but also to provide the necessary data for certification. A detailed finite element analysis was developed for the stabilizer and the most critical load cases were considered. Ultimate strains were obtained by combining the mechanical load strains and the thermal and moisture analysis strains. Margins of safety were obtained by comparing the calculated ultimate strains with the allowable strains for the different environments.

A thorough test program was carried out to support the data generation necessary to certify the structure. A building block approach was followed were coupons, elements, large panels and test boxes were tested [1,4]. Coupons and elements were used to generate mechanical properties, interlaminar properties, identify stress concentration effects, environmental factors and to characterize impact damage properties and demonstrate material durability.
Large panels and test boxes were used to validate design concepts, verify analysis methods and correlate test and analytical data.

MAINTENANCE PLANNING PROGRAM

A maintenance program was developed by the OEM to assist the operators in ensuring the airworthiness and safety of the horizontal stabilizers. The plan was accepted by the FAA and is presented in Table 1 below:

Table 1. Maintenance Planning Schedule [5]

<table>
<thead>
<tr>
<th>Check</th>
<th>Inspection Interval (Flight Hr)</th>
<th>Description</th>
</tr>
</thead>
<tbody>
<tr>
<td>Preflight-Transit</td>
<td>Walk Around</td>
<td></td>
</tr>
<tr>
<td>A</td>
<td>75</td>
<td>• Visual Inspection of exterior surface, from ground level</td>
</tr>
<tr>
<td>B</td>
<td>300</td>
<td>• Visual Inspection of external surfaces</td>
</tr>
<tr>
<td>C</td>
<td>1200</td>
<td>• External Visual Inspection</td>
</tr>
<tr>
<td></td>
<td></td>
<td>• Exposed Rear Spar Area</td>
</tr>
<tr>
<td></td>
<td></td>
<td>• Exposed Hinge Fittings and thermal linkage</td>
</tr>
<tr>
<td></td>
<td>2400</td>
<td>• Front and Rear Spar to center section attachment lugs</td>
</tr>
<tr>
<td></td>
<td></td>
<td>• Inboard Edge of Rear Spar Web</td>
</tr>
<tr>
<td></td>
<td></td>
<td>• Trailing Edge Cavities</td>
</tr>
<tr>
<td>Structural</td>
<td>14000</td>
<td>• External Visual Inspection</td>
</tr>
<tr>
<td></td>
<td></td>
<td>• NDT Inspection upper and lower skin from the rear spar forward to stringer 3 between the side-of-body and the rib at stabilizer station 111.1</td>
</tr>
<tr>
<td></td>
<td></td>
<td>• Front and rear spar attachment lugs, pins, bushings and fittings</td>
</tr>
<tr>
<td></td>
<td></td>
<td>• Internal trailing-edge structure</td>
</tr>
<tr>
<td></td>
<td></td>
<td>• Internal structure, spars, stiffeners, closure ribs; access by removing gap covers, access hole covers, removable leading edge, removable lower trailing edge panels and removable tip</td>
</tr>
</tbody>
</table>

B737 HORIZONTAL STABILIZER FLEET STATUS

Table 2 below summarizes the status of all five composite stabilizer shipsets as of January 2008. As shown in the table, as of the beginning of this year, four shipsets have been removed from service after accumulating 47000, 19300,
55000 and 48000 flights respectively. The first shipset has been re-introduced in service and is being operated by a foreign carrier.

Table 2. B737 Composite Stabilizer Fleet Status as of December 2008

<table>
<thead>
<tr>
<th>Shipset / Production Line #</th>
<th>Entry into Service</th>
<th>Carrier</th>
<th>Status as of December, 2008</th>
</tr>
</thead>
<tbody>
<tr>
<td>1 / 1003</td>
<td>2 May 1984</td>
<td>F</td>
<td>(60000 hours, 45000 flights), sold to a foreign carrier</td>
</tr>
<tr>
<td>2 / 1012</td>
<td>21 March 1984</td>
<td>A</td>
<td>Removed from Service (62000 hours, 47000 flights)</td>
</tr>
<tr>
<td>4 / 1036</td>
<td>17 July 1984</td>
<td>B &amp; C</td>
<td>Stabilizers removed from service 2002 (approx. 39000 hours, 55000 flights); partial teardown of R/H unit at Boeing</td>
</tr>
<tr>
<td>5 / 1042</td>
<td>14 August 1984</td>
<td>B &amp; D</td>
<td>Stabilizers removed from service 2002 (approx. 52000 hours, 48000 flights); teardown of L/H unit at Boeing; teardown of R/H unit at NIAR, Wichita State</td>
</tr>
</tbody>
</table>

TEARDOWN OBJECTIVE

The objective of the teardown of the R/H B737 graphite epoxy horizontal stabilizer was to evaluate the aging effects on the performance of the composite structure after 18 years of service. The main goal was to evaluate the structural health of the aged composite structure after 48000 flights equivalent to about 2/3 of its design service objective.

To accomplish this task, the research was subdivided into several subtasks: non-destructive and destructive tasks. The goal of the non-destructive inspections was to characterize the state of the structure after 18 years of service, to investigate the existence/ extent of flaws introduced during manufacture or service using current methods used in the field but also more sophisticated methods including Thermography, Laser UT, Rapidscan™, etc…

The objective of the destructive inspection/ evaluation was to confirm the existence of flaws detected using NDI, to conduct mechanical tests, thermal analysis, physical tests, image analysis, etc…

VISUAL INSPECTION/ NON-DESTRUCTIVE INSPECTION PRIOR TO TEARDOWN

The general visual inspection conducted on the outside of the structure prior to teardown revealed numerous cracked and peeled paint areas in the upper and lower skins. In most fastener areas, the paint was cracked and loose. The
surface blemishes were caused by excessive paint/surfacer, aging, and exposure to weather.

Non-Destructive Inspection scans were conducted at the Sandia National Laboratories using pulse echo ultrasonics (RAPIDSCAN™) as shown in figure 8. The upper skin scans showed extensive regions of porosity or small delaminations. The Rapidscan™ inspections also showed a disbond at the inboard section of the upper skin at stringer 7 as shown in the figure. The areas circled in red showed excessive porosity from the Rapidcan pulse echo inspection but were identified as disbands after teardown.

The lower skin scans showed less porosity in comparison to the upper skin but a few areas were identified as disbands, at the inboard section of the lower skin at stringer 1 runout, at the outboard section of the skin at rib 6, stringers 2 and 3 and between ribs 5 and 6, at stringer 3 as shown in figure 8.

The main findings of the NDI inspection were the excessive porosity levels found in the upper skin but also the stringer disbands in the inboard section of the upper skin.
Figure 8. Rapidscan™ analysis (pulse echo time of flight data) of the R/H of the B737 stabilizer “Courtesy of Sandia National laboratories and NDT solutions Ltd. UK”
DISASSEMBLY

The R/H B-737 horizontal stabilizer as shown in figure 9 was disassembled as follows: the upper skin was first demated by drilling out the blindfasteners. A combination of carbide and cobalt drill bits was used to drill out the “big foot” blindfasteners head as shown in figure 10. Once the fastener’s head was drilled out, the fastener’s shank was accessible, and could be easily driven out of the structure.

Figure 9. B737 CRFP R/H Horizontal Stabilizer prior to disassembly

Figure 10. “Big Foot Blind fastener Removal"
As shown in figure 11, all blind fasteners attaching the upper skin to the spars and the ribs were carefully drilled out in order not to introduce additional damage to the structure. The dismantled upper skin is shown in figure 12. The lower skin was dismantled next at the Titanium Hi-Lok attachment points to the spars and ribs. The lower skin, after disassembly is shown in figure 13.

![Figure 11. Upper Skin Disassembly](image1)

As shown in figure 12 below, the R/H upper skin appeared to be flat with no evidence of curvature or residual strain. Furthermore both skins appeared to be in extremely good condition with no evidence of detrimental aging or deterioration.

![Figure 12. B-737 R/H Composite Stabilizer Upper Skin, After Disassembly](image2)
Once both skins were demated, the structural box, shown in figure 14 was further disassembled demating the spars from the ribs as shown in figure 15.

Figure 13. B-737 R/H Composite Stabilizer Lower Skin, After Disassembly

Figure 14. R/H B-737 Composite Stabilizer Structural Box
Figure 15. R/H B-737 Composite Stabilizer Spars
VISUAL INSPECTION FINDINGS

As shown in figures 12 through 15, the first striking observation that could be made after disassembly is that the stabilizer appeared to be in really good condition with no evidence of pitting or corrosion as would be observed in a metal structure with similar service history. All fasteners seemed to be in really good condition with a few exceptions as shown in figure 16.

![Corroded Fasteners Due to Sealant Deterioration](image1)

Visual inspection also showed evidence of Tedlar degradation due to environmental exposure as seen in figure 17. Tedlar was used as a moisture protective film, was co-cured to parts of the structure mainly ribs and spars as a protection/ barrier from moisture ingestion.

![Degradation of TEDLAR, Moisture Barrier Film](image2)
After a careful inspection of the disassembled structure, evidence of both liquid and phenolic solid shims was found and is shown in figures 18 and 19. The shims used seemed to be very effective and held up very well in service. The bonding of the liquid shim was effective, yet the release agent used on one side allowed separation at disassembly. There were no signs of the liquid shim breaking up, softening, or deteriorating.

**Phenolic Shims**

![Figure 18. Phenolic Shims used to fill the gaps between the upper skin and ribs 2 and 8](image18)

![Figure 19. Liquid Shims used to fill gaps between the upper skin and the stabilizer ribs](image19)
The visual inspection/ findings of the B-737 R/H stabilizer teardown showed that the composite structure held extremely well with minor signs of degradation or deterioration due to environmental exposure. Tedlar and fastener sealant degradation were the only signs of deterioration due to environmental exposure found on the structure. Disassembly also showed evidence of shimming to seal the gaps between the stabilizer’s skins and ribs. All shims used held extremely well with no evidence of softening, or cracking. Furthermore, a visual inspection of the upper skin repair found between rib stations 2 and 3 confirmed that the repair was a surface repair only not and did not extend to the stringers.

Visual Inspection/ Destructive evaluation of disbonded Areas

Visual inspection confirmed the existence of the delaminated stringers at the inboard section of the upper skin at stringer 7 runout. Visual inspection also showed the existence of additional delaminations at the upper skin stringer runouts for stringers 3 and 4 as shown in figure 20.
Similar to the delaminations found in the inboard section of the upper skin at stringer runouts, a delamination was detected in the inboard section of the lower skin at stringer 1 runout and was confirmed after disassembling the structure. Destructive inspection was conducted on all other disbonds identified using RapidsCan™ in the lower skin and were confirmed. Some of those disbonded stringers are illustrated in figure 21 below.

**Figure 20. Upper Skin Delaminations at Stringer 3, 4 and 7 Runouts**

**Figure 21. Lower Skin Disbonds confirmed after teardown**

**NON DESTRUCTIVE INSPECTION AFTER TEARDOWN**

Pulse-echo and through-transmission non-destructive inspection methods were used to characterize the stabilizer’s damage state after 18 years of service. Both methods confirmed the large amounts of porosity in the upper skin as previously identified in the Rapidscan™ inspection results. Results summary is shown in figure 22 below. A skin repair is also shown in the same figure, in the upper skin between rib stations 2 and 3 and stringers 1 and 8.

Pulse-echo results using today’s equipment capabilities showed the increase accuracy/sensitivity of the current inspection methods compared to those used in the 1970’s. 5 Mhz transducers were used to inspect the aged structure according to current inspection requirements. 1 Mhz transducer was used to obtain a scan according to the 1970’s requirements. This is shown in figure 23 below.
Figure 22. Niar Pulse-Echo Scans of the B737 Horizontal Stabilizer
THERMAL INSPECTION

Thermal analysis was conducted on coupons excised from both the upper and lower skins to investigate possible changes in the resin chemistry of the material. The thermal tests conducted included Dynamic Mechanical Analysis (DMA) and Differential Scanning Calorimetry (DSC). DMA tests measure the response of a material to a periodic stress and provide information about the modulus and the damping of the material. The modulus is characteristic of the material fiber stiffness whereas the damping is a characteristic of the matrix of the material.

DMA curves provide two values of glass transition temperature, a value based on the onset storage modulus or material fiber stiffness loss and a value based on material damping/maximum viscosity which is the peak of tanδ. The glass transition temperature (Tg) value based on the onset of storage modulus is always more conservative than the value obtained using the peak of tanδ.

All DMA curves obtained for coupons extracted from both skins exhibited a shallow storage modulus transition and a narrow tanδ which is an indication of a highly cross-linked material therefore a material fully cured as shown in figure 24.

DSC was also conducted on specimens extracted from both skins to evaluate the degree of cure of the material. As shown in figure 25, the low heat of reaction values support the results obtained using DMA, i.e., that the material is fully cross linked and fully cured.
Figure 24. DMA of a Coupon Extracted from the B-737 Upper Skin

Figure 25. DSC of a Coupon Extracted from the B-737 Upper Skin
DMA was conducted using the parameters specified in the ASTM standard E1640 but also according to the Boeing method and both sets of results are summarized in figures 26, 27 and 28 below. All values are compared against the baseline dry Tg obtained using TMA supplied by NARMCO.

When comparing the Tg values obtained for the front and rear spars to the upper skin Tg, it was found that the glass transition temperature obtained from coupons extracted from the spars was at least 15°F than the upper skin values. The increased Tg is most probably due to the additional curing that occurred during the secondary bonding of the spars. The skins were only subjected to one single cure whereas the spars were initially cured as separate C-channels and subsequently bonded using film adhesive.

![Figure 26. DMA Results for Coupons Excised From the Upper skin of the B-737 Horizontal Stabilizer (ASTM Standard)](image-url)
Figure 27. DMA Results for Coupons Excised From the Upper skin of the B-737 Horizontal Stabilizer (Boeing Method)

Figure 28. DMA Results Comparison for Coupons excised from the Upper Skin and the Front and Rear Spars of the B-737 Horizontal Stabilizer

Figure 29 is a comparison between new material Tg with respect to upper and lower skin Tg. As shown in the figure, the new material glass transition temperature correlates very well with the average skin Tg which is indicates that the new 5208 resin has comparable thermal properties to the original NARMCO 5208 formulation.
Figure 29. DMA Results Comparison for Coupons excised from the skins and coupons manufactured from the new T300/5208 (ASTM Standard)

Figure 30 illustrates the degree of cure for the new T300/5208 material versus the aged material extracted from the skins and spars. A DSC test on uncured T300/5208 was used to generate the heat of reaction value necessary to yield a fully cured part. Heat of reaction values obtained from DSC tests on spar and skin coupons were then used to generate a cure conversion % indicative of the degree of cure of the part.

From the figure, it can be seen that spars have reached an almost fully cured status, as additional post-curing occurred during the secondary bonding process (4% cure conversion increase with respect to the new material). The upper skin has a lower Heat of Reaction value than the new material as additional curing has occurred most probably during the life span of the structure. This is shown in a 2% cure conversion increase with respect to new material.
MOISTURE CONTENT EVALUATION

Moisture Content in the aged structure was quantified using ASTM D5229. Figures 31 and 32 show the moisture content distribution in both skins based on coupons excised from both locations and subsequently dried. All moisture levels obtained were lower than the design moisture level of 1.1%.

Figure 30. % Cure Comparison between New Material, Spars and Upper Skin

Figure 31. Moisture Distribution in the upper skin

Figure 32. Moisture Distribution in the lower skin
Physical tests were conducted per ASTM D3171 and ASTM D2734 to determine the resin and fiber volume fractions in the structure but also to quantify the different levels of porosity in the structure. As shown in figure 33, the maximum void content in the upper skin was in the order of 7.14% whereas the maximum void content found in the lower skin was in the order of 3%.
Figure 3. Void Content Summary

<table>
<thead>
<tr>
<th>Void Content %</th>
<th>0-0.5</th>
<th>0.5-1.0</th>
<th>1.0-1.5</th>
<th>1.5-2.0</th>
<th>2.0-2.5</th>
<th>Above 5</th>
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<tr>
<td>0-0.5</td>
<td></td>
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<tr>
<td>0.5-1.0</td>
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<td>Above 5</td>
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</table>
MICROSCOPY/ IMAGE ANALYSIS

Image analysis was conducted on samples extracted from both the upper and lower skins to inspect the structure at the microscopic level for voids, microcracks or any evidence of aging or material degradation. Extracted samples were potted, polished and viewed under the microscope to detect any possible evidence of material aging. Figures 34 and 35 show evidence of porosity but also micro-cracking emanating from the voids/ porous areas.

Figure 34. Cross Section of Stringer Flanges 2 AND 3, rib station 1 at a magnification of 50x (Upper Skin)

Figure 35 shows cross sections from two different stringer webs at a magnification of 50x. Both cross sections show evidence of porosity between the laminate plies.

Figure 35. Cross Section of stringer 2 rib 2 and stringer 3 rib 3 webs at a magnification of 50x (Upper Skin)

MECHANICAL TESTING

Mechanical testing was conducted using 1980’s specimen configurations and standards. Because of lay-up schedule variation between the different rib
stations, thickness variation, curvature in the structure, mechanical test specimens were carefully extracted from the structure, carefully and individually tabbed when applicable and carefully machined and tested. All tension specimens were individually tabbed as shown in figure 36.

![Figure 36. Tabbing fixture for the tension specimens](image)

As shown in figure 37, the compression specimens are flat 1” wide by 5.5” long rectangular specimens. The fixture used for testing consists of two anti-buckling plates supporting the entire length of the specimen with a 0.2” opening to allow specimen deformation during loading. The compressive loads are transferred to the specimens through end loading primarily. Two edge extensometers were used to generate stiffness values for the different configurations tested.

![Figure 37. Compression test fixture and test specimens](image)
A 22-kip sevohydraulic actuator mounted on a 22-kip MTS load frame was used for loading. The MTS test equipment was calibrated and verified according to the ASTM E4 standard to ensure the accuracy of the load and displacement readings. The coupons were loaded at a rate of 0.05in/min. The actuator was controlled with the MTS Flextest-GT system and the MTS Testworks software was used to program the parameters for controlling the test and acquiring data. Recorded data included time, actuator displacement, force, and extensometer readings.

Figure 38 summarizes the compression results obtained for coupons extracted from the upper skin, tested as extracted. As shown in the figure, the normalized compression strength of US-R1-STR9_10-C2 and C3 is very comparable to the baseline strength of 102-1 and 102-2. This proves that the mechanical properties of the aged structure have not degraded due to aging and are still comparable to baseline allowable data. Furthermore it should be noted that only 2 baseline data points were available to compare the data to.

![Figure 38](image_url)

**Figure 4. Upper skin compression strength test data, as extracted**

Similar to the upper skin values, the lower skin normalized data obtained from LS-R4-STR2-3-C1/C2/C3 correlated very well with the baseline data with no evidence of deterioration or strength degradation attributed to aging. Figure 39 shows a good correlation between the stiffness data obtained from the sample specimens extracted from the aged structure LS-R4-STR2-3-C1/C2/C3 and the baseline modulus data. The compression strength and stiffness data did
not show any evidence of strength degradation that can be attributed to the aging of the stabilizer.

Figure 39. Lower skin compression modulus data, as extracted

Tension testing of extracted coupons from the aging stabilizer was also conducted per the 1980’s requirements and specifications. As shown in figure 40, the coupon configuration is a tabbed dog bone with a 1” width at the gage section and a 1.5” width at the ends.

A 22-kip sevohydraulic actuator mounted on a 22-kip MTS load frame was used for loading. The MTS test equipment was calibrated and verified according to the ASTM E4 standard to ensure the accuracy of the load and displacement readings. The coupons were loaded at a rate of 0.05in/min. The actuator was controlled with the MTS Flextest-GT system and the MTS Testworks software was used to program the parameters for controlling the test and acquiring data. Recorded data included time, actuator displacement, force, and strain gage readings.
Figure 50. Tension Test set-up and specimens

Figure 41 summarizes the tension test results obtained for coupons extracted from the lower skin. The normalized strength data for LS-R4-STR3-4-T1 is very consistent/ comparable to the baseline 101-1 and 101-2 tension baseline and does not show any evidence of degradation of the tensile properties due to material aging. LS-R4-STR3-4-T1 is the only coupon from the batch tested that has the same stacking sequence as 101-1 and 101-2 baseline coupons.

![Figure 41. Lower skin tension test results](image)

**Lower Skin Tension Test Results**

<table>
<thead>
<tr>
<th></th>
<th>Mechanical Data-Measured</th>
<th>Mechanical Data-Normalized</th>
</tr>
</thead>
<tbody>
<tr>
<td>101-1 Baseline</td>
<td>50</td>
<td>50</td>
</tr>
<tr>
<td>101-2 Baseline</td>
<td>50</td>
<td>50</td>
</tr>
<tr>
<td>LS_R2_STR4_5-T1</td>
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<td>40</td>
</tr>
<tr>
<td>LS_R2_STR5_6-T1</td>
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<tr>
<td>LS_R4_STR3_4-T1</td>
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<tr>
<td>LS_R6_STR0 1-T1</td>
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</tr>
<tr>
<td>LS_R6_STR1 2-T2</td>
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</tr>
</tbody>
</table>

![Figure 41. Lower skin tension strength data, as extracted](image)
Figure 42 shows a good correlation between the stiffness data obtained from the sample specimen extracted from the aged structure LS-R4-STR3-4-T1 and the baseline modulus data. The tension strength and stiffness data did not show any evidence of degradation that can be attributed to the aging of the stabilizer.

![Lower Skin Tension Test Results, Modulus](image)

Figure 62. Lower skin tension modulus data, as extracted

Three stringer crippling panel tests were conducted on a few elements extracted from the upper and lower skins. The test set-up is shown in figure 43 below.

![Three-stringer element crippling test set-up](image)

Figure 73. Three-stringer element crippling test set-up
A 55-kip servohydraulic actuator mounted on a 55-kip MTS load frame was used for loading. The MTS test equipment was calibrated and verified according to the ASTM E4 standard to ensure the accuracy of the load and displacement readings. The elements were loaded at a rate of 0.05 in/min. The actuator was controlled with the MTS TesTar IIm system and the MTS Testworks software was used to program the parameters for controlling the test and acquiring data. Recorded data included time, actuator displacement, force, and strain gage readings. A picture of a failed three stringer crippling element is shown in figure 44 below.

![Figure 44. Post-Test picture of a three stringer element](image)

**Figure 44.** Post-Test picture of a three stringer element

**Skin Panel Load vs Strain Readings**

<table>
<thead>
<tr>
<th>Load, lbf</th>
<th>Strain, micro in/in</th>
</tr>
</thead>
<tbody>
<tr>
<td>0</td>
<td>Failure Strain (5200 micro in/in)</td>
</tr>
<tr>
<td>5000</td>
<td></td>
</tr>
<tr>
<td>10000</td>
<td></td>
</tr>
<tr>
<td>15000</td>
<td></td>
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<td>20000</td>
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<td>25000</td>
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</tr>
<tr>
<td>30000</td>
<td></td>
</tr>
<tr>
<td>35000</td>
<td></td>
</tr>
</tbody>
</table>

Failure Load 30193 lbs
32000 lbs for baseline

![Figure 45. Crippling element load vs strain data](image)

**Figure 45.** Crippling element load vs strain data
CONCLUSIONS

The teardown of the right-hand Boeing-737-200 stabilizer revealed a composite structure that held extremely well after 18 years of service with no obvious signs of aging to the naked eye, such as pitting and corrosion, as would a metal structure with a similar service history exhibit.

Physical test results showed moisture levels in the structure after 18 years of service as predicted during the design phase. Thermal analysis results were very consistent with those obtained for the left-hand stabilizer and with the baseline data provided by the supplier. Thermal analysis showed that the degree of cure of the spars is close to 100%, that additional curing may have occurred during the secondary bonding of the spars. Furthermore, additional curing may have occurred in the upper skin due to environmental exposure (overall at least 95% cure was achieved in the structure).

For mechanical tests where baseline data was available (1970s certification and test data), the residual strengths met or exceeded the baseline values. No significant degradation was noted in any of the tests.

The teardown also showed that significant improvements in composite manufacturing processes, and nondestructive inspection methods have been made in the last 30 years.

Furthermore, the new T300/5208 material appeared to have comparable thermal properties to old material; i.e., comparable resin system but significantly higher tensile and compressive properties indicative of an improvement in the fiber strength in the last 30 years.

Finally, the teardown provides closure to a very successful National Aeronautics and Space Administration program and affirms the viability of composite materials for use in structural components. From all data generated, the margins were sufficient to warrant a “no significant degradation” conclusion.
REFERENCES

2. Dow, Marvin B., “The ACEE Program and Basic Composites Research at Langley Research Center (1975 to 1986).”