# BMI SANDWICH WING BOX ANALYSIS AND TEST

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ABSTRACT

A composite sandwich single bay wing box test article was developed by Northrop Grumman and tested recently at NASA Langley Research Center. The objectives for the wing box development effort were to provide a demonstration article for manufacturing scale up of structural concepts related to a high speed transport wing, and to validate the structural performance of the design. The box concept consisted of highly loaded composite sandwich wing skins, with moderately loaded composite sandwich spars. The dimensions of the box were chosen to represent a single bay of the main wing box, with a spar spacing of 30 inches, height of 20 inches constant depth, and length of 64 inches. The bismaleimide facesheet laminates and titanium honeycomb core chosen for this task are high temperature materials able to sustain a 300°F service temperature. The completed test article is shown in Figure 1. The tests at NASA Langley demonstrated the structures ability to sustain axial tension and compression loads in excess of 20,000 lb/in, and to maintain integrity in the thermal environment. Test procedures, analysis failure predictions, and test results are presented.
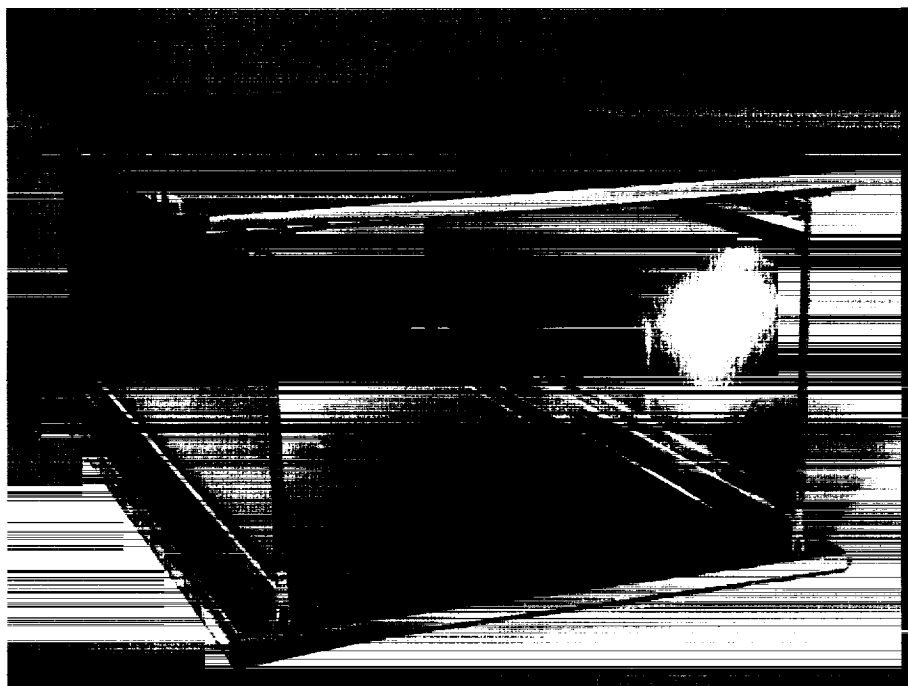
Figure 1. Composite Sandwich Single Bay Wing Box Test Component

BACKGROUND

Wing box design and fabrication (Reference 1) were performed under NASA contract NAS1-18842. The sandwich skin panels consist of IM7/5260 toughened BMI facesheets over titanium honeycomb core. The 1.0 inch thick core material is 8.0 pound-per-cubic-foot hexagonal core with a 3/16 inch cell size, 0.002 inch wall thickness, and spot welded nodes. The skin panel facesheets were sized based on notched compressive strain allowable of 4,600 micro-strain for the IM7/5260 material. The facesheet thickness distribution varies from 20 plies at the tip (50% 0° plies) to 28 plies at the root (64% 0° plies). Local facesheet doubler laminates were incorporated in the splice regions to increase laminate bearing strength.

The sandwich spar webs also consist of IM7/5260 toughened BMI facesheets over titanium honeycomb core. The 0.50 inch thick core material is 6.0 pound-per-cubic-foot square cell core with a 3/16 inch cell size, .002 inch wall thickness, and laser welded nodes. The facesheets are 12 plies (±45/0/±45/90)s laminates. The core ramps down at an angle of 27° to close out the core at the top of the spar, and provides a solid laminate region for attachment to the T cap. A 12 ply, 0.0624 inch thick doubler laminate was placed between the facesheets for enhanced laminate bearing strength in the bolted spar cap region.

A bonded composite H-section configuration was chosen for attachment of the spars to the lower wing skin, while a bolted composite T-joint was selected for the upper attachment. The bonded joint was preferred on the lower surface to reduce fuel penetrations in the skin. The bolted joint was chosen over a bonded design for the upper attachment to avoid bonding together a closed section. A fully bonded construction raised issues of inspectability and producibility for a large commercial transport, and was therefore dropped as a design option. Both the bonded and bolted joint selections were supported by element test data from work performed by Northrop Grumman under NAS1-19347, Tasks 5 and 6.

The wing design loads were based on the ultimate loads for a subsonic 3.75g pull-up maneuver at maximum gross takeoff weight. The axial skin loads selected for the wing box component were 10 kips/in at the tip and 20 kips/in at the root. The average shear loads in the spar webs for the same load condition were 2.5 kips/in.

LOADS AND TEST PLAN

The skin splices between the wing skins and the load introduction structure were a critical design concern due to the applied load of 20,000 lb/in. A three row bolted joint was designed, with a solid block of titanium replacing the core in the splice region and protruding from the end of the test box. The metal load introduction boxes reacted loads into outer splice plates and the center block, resulting in multiple load paths. Titanium tabs were added to the outer facesheets to further increase the bearing margins.

The test fixture was designed to sustain loads of 20,000 lbs/in in-plane and 2,500 lbs/in beam shear. The existing backstop at NASA LaRC was considered inadequate to support a cantilevered test of the box, and a self-reacting four point bending fixture was therefore designed for the test. The new wing box test fixture design, shown in Figure 2, was based on load introduction by four-point bending. The design and analysis for the fixture were conducted at the Northrop Grumman. The test fixture fabrication was a joint effort between NGC and NASA team. Details for the test fixture design and wing box test results were documented under NASA contract NAS1-20220 (Reference 2). The applied loads at each cylinder, and corresponding shear and moment diagrams for the wing box subcomponent are shown schematically in Figure 3. The load introduction structure was sized for a margin of safety of 2.0 at ultimate load for the majority of the fixture, with local areas designed to M.S.=1.5 (30,000 lb/in) at the skin splice regions.

The test plan included four test conditions. Condition 1 consisted of taking the box to 300°F temperature with no applied mechanical loads. In Condition 2 the box was subjected to limit design load (67% of ultimate mechanical loading) at room temperature. The Condition 3 loading consisted of design limit load with 300°F temperature. In Condition 4, the box was loaded to failure at room temperature. Wing box instrumentation included a total of 115 axial strain gages including shear gages, 18 rosette gages, 72 thermocouples, 4 LVDT, 4 displacement gages (DCDT), 3 ambient temperature monitoring sensors and 8 load measuring instruments were employed for the test.
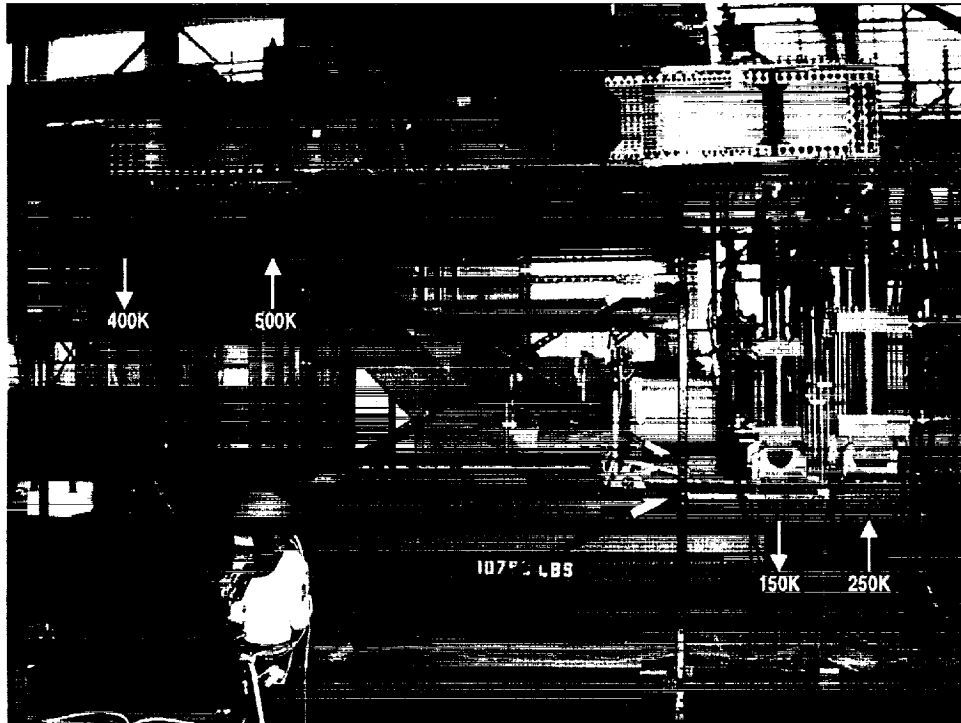
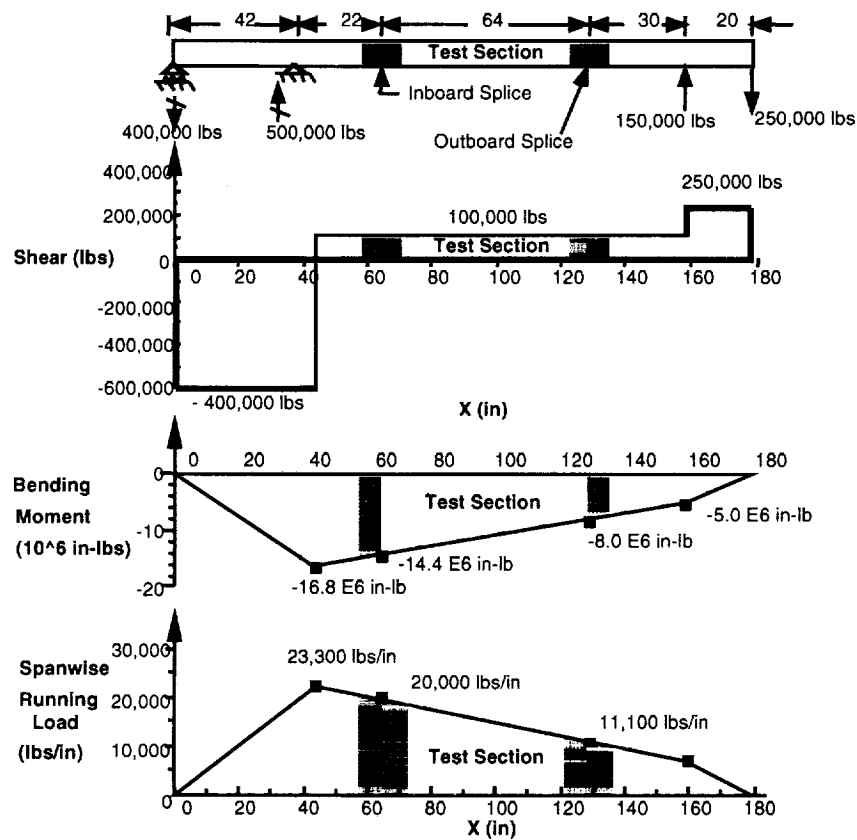**Figure 2. Composite Sandwich Single Bay Wing Box Test Setup at NASA Langley**

Figure 3. Test Loads and Ultimate Loading of Wing Box Subcomponent

FINITE ELEMENT ANALYSIS RESULTS

A NASTRAN finite element model was used to correlate the strain and deflection of the wing box. The wing test box and metal load introduction box assemblies were characterized using a symmetric half model with 1,819 plate elements and 11,205 degrees of freedom. The model, shown in Figure 4, used brick and bar elements at the support locations to simulate the support flexibility of the load reaction fixture. The beam deflection predicted by the NASTRAN model, shown in Figure 5, matched those measured during the test within 8%. The actual deformations for the test article were higher than predicted values due to flexibility of the load introduction bolted splices, which were not included in the model. Deformations within the test section matched predicted values within 5%.

Failure predictions were based upon the notched compressive strain allowable for the IM7/5260 facesheets at 5400 microstrain, which resulted in a zero margin of safety at the bolted spar attachment near the reaction end of the wing box at ultimate load. A summary of the margins of safety for the test article at ultimate load is shown in Figure 6. The measured test strains in the critical region, plotted in Figure 7, reached a maximum value of 5300 microstrain. The final failure occurred at 119% ultimate load, and initiated in this region.

Analysis correlation for the finite element model predictions versus the actual measured axial strain data are shown in Figures 8 and 9. Figure 8 shows the compressive strain contours at ultimate load for the lower skin panel with the bolted spar cap attachment. The measured strain data at ultimate load are included in boxes linked to the gage locations on the subcomponent. In general, the measured strains correlated within 6% of the predicted values. Axial strain concentrations were observed in the skins near the spar attachments due to shear lag across the wing box.

The centerline strain values were approximately 20% less than those predicted near the spars at each end of the test component. The axial strains were uniform across the width of the box near the mid-span of the test article, and measured ~5000μ at ultimate load.

The tensile load analysis correlation shown in Figure 9 exhibited similar behavior to the lower compressive skin. The highest measured strain of 5680μ occurred at the reaction end of the box, near the bonded spar web joint, and was 6% higher than the analysis prediction. There are no local stress concentrations from fasteners in this region because the bonded H-joint incorporates a uniform slot through the inner moldline facesheet to accommodate the pre-cured H-section. The presence of the continuous slot allows the inner and outer facesheets of the tensile skin to operate near the unnotched allowable strain (~11,000μ), and the calculated margin of safety for this skin was therefore high.

WING BOX FAILURE MODE

The failure initiated as a compression failure in the bolted sandwich skin near the reaction side as predicted by the NASTRAN analysis. The failure region is shown in Figure 10. The local compressive failure of the facesheet at the boltline propagated across the lower skin panel, disbonding the severed outer moldline facesheet in the process. As the bolted sandwich skin failed, the box lost its bending capability. This caused the sandwich spar web to fail in shear. The shear failure propagated at 45 degrees across the spar web, until reaching the upper tension sandwich panel. This bonded sandwich skin failed in a local bending mode approximately 20 inches inboard from the load reaction splice. The spar sandwich and bonded sandwich skin were both secondary forms of failure. The remainder of the wing box test article is structurally intact, and no further damage propagation was observed.
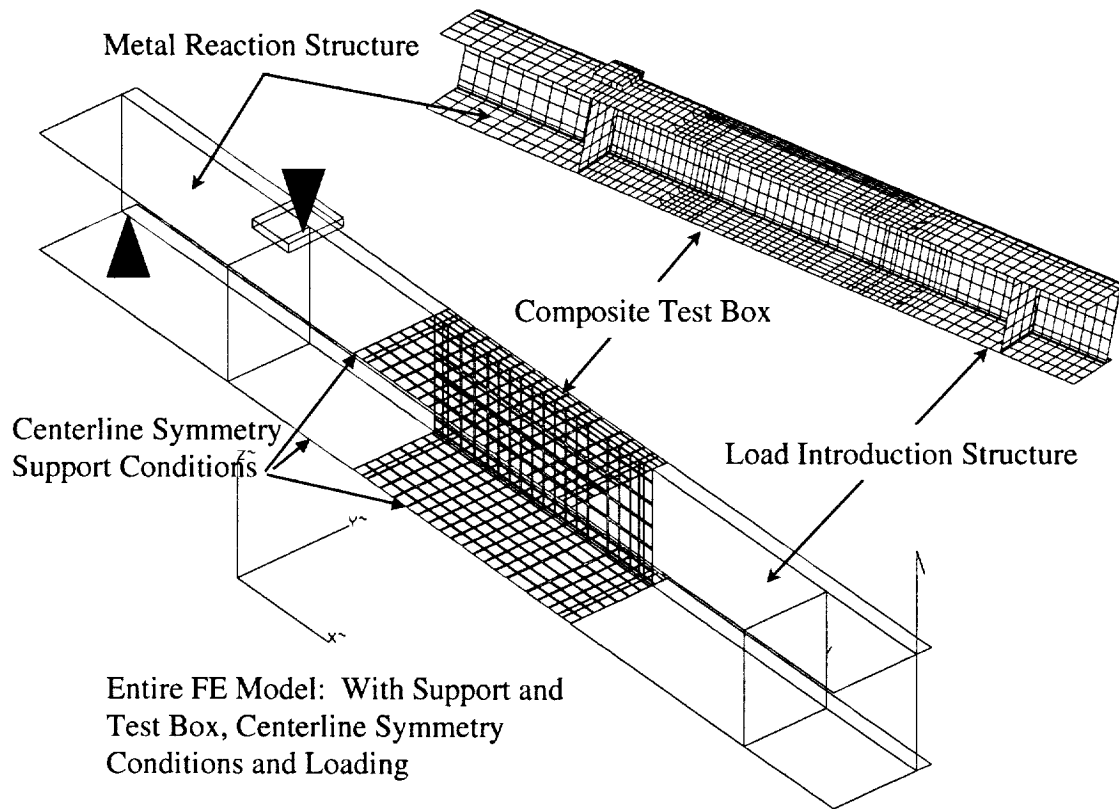
Figure 4. Finite Element Model Used for Mechanical Loading Prediction
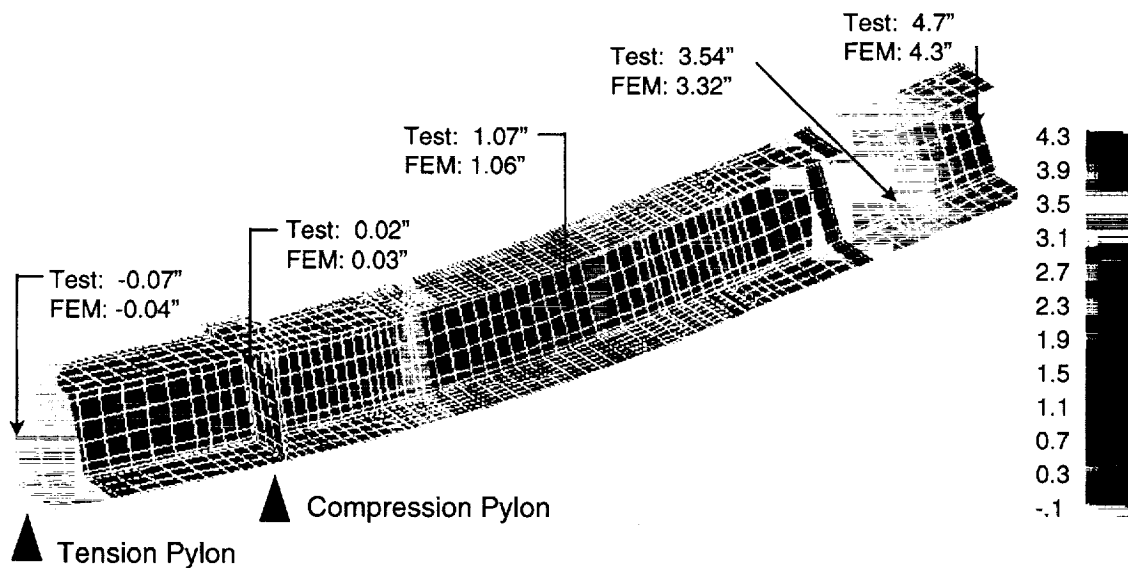
Figure 5. Wing Box Model Deformation Comparison With Test Data

Location	Critical Failure Mode	Design Allowable	Minimum Margin	Laminate Thickness
Lower Skin	Facesheet Tensile Rupture	OHT Strain = 7,500 µin/in	0.17	28 Plies
Upper Skin	Facesheet Compression Failure	FHC Strain = 5,400 µin/in	0.00	28 Plies
Spar Web	Facesheet Shear Failure	FHC Strain (Transformed) = 5,000 µin/in	0.32	12 Plies
T- Joint	Laminate Crushing Due to Bolt Bending		0.14	36 Plies
H-Joint	Shear Failure of Inboard Angle		0.14	6 Plies (woven)

Figure 6. Summary of Minimum Margins of Safety at Ultimate Load
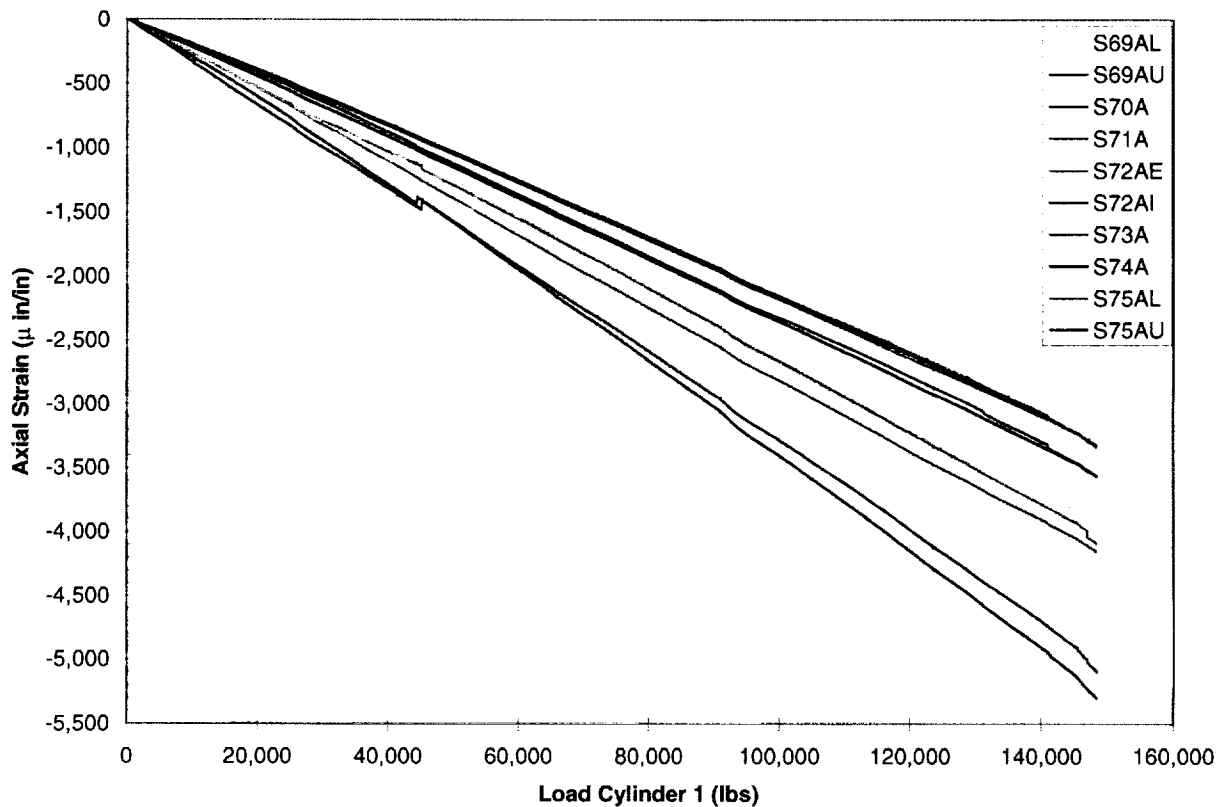

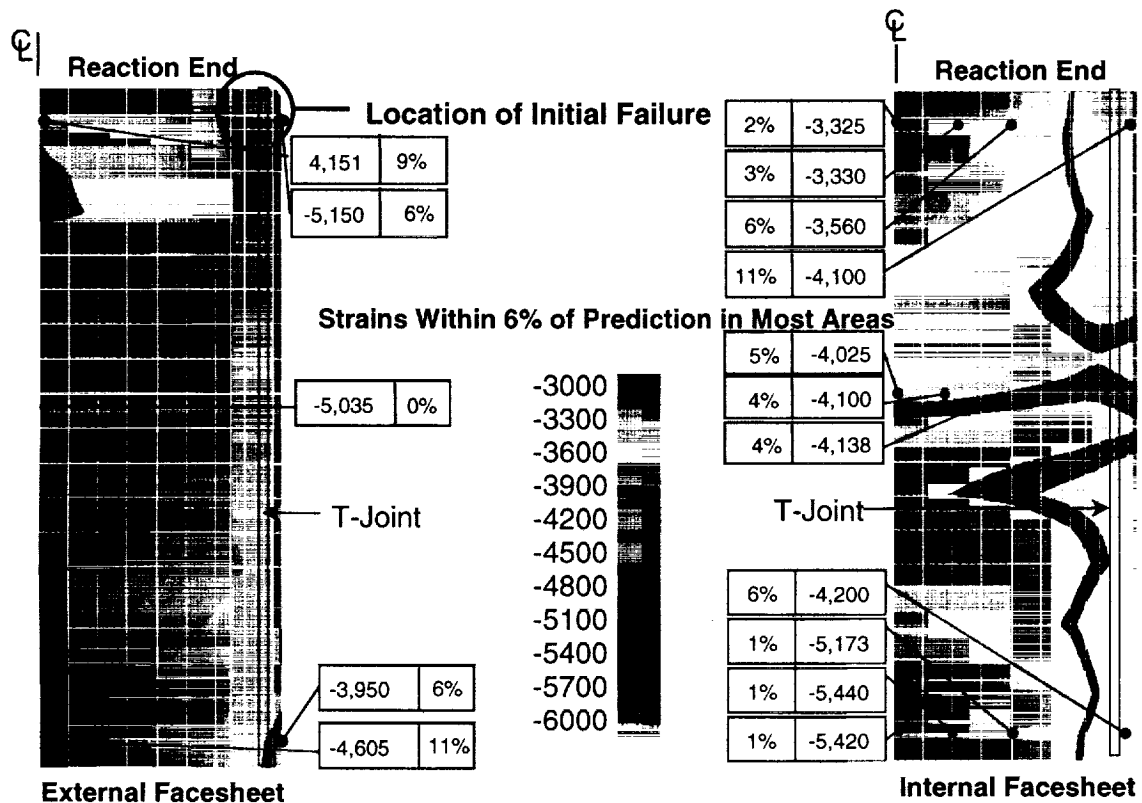**Figure 7. Compressive Strains at Reaction End of Wingbox**

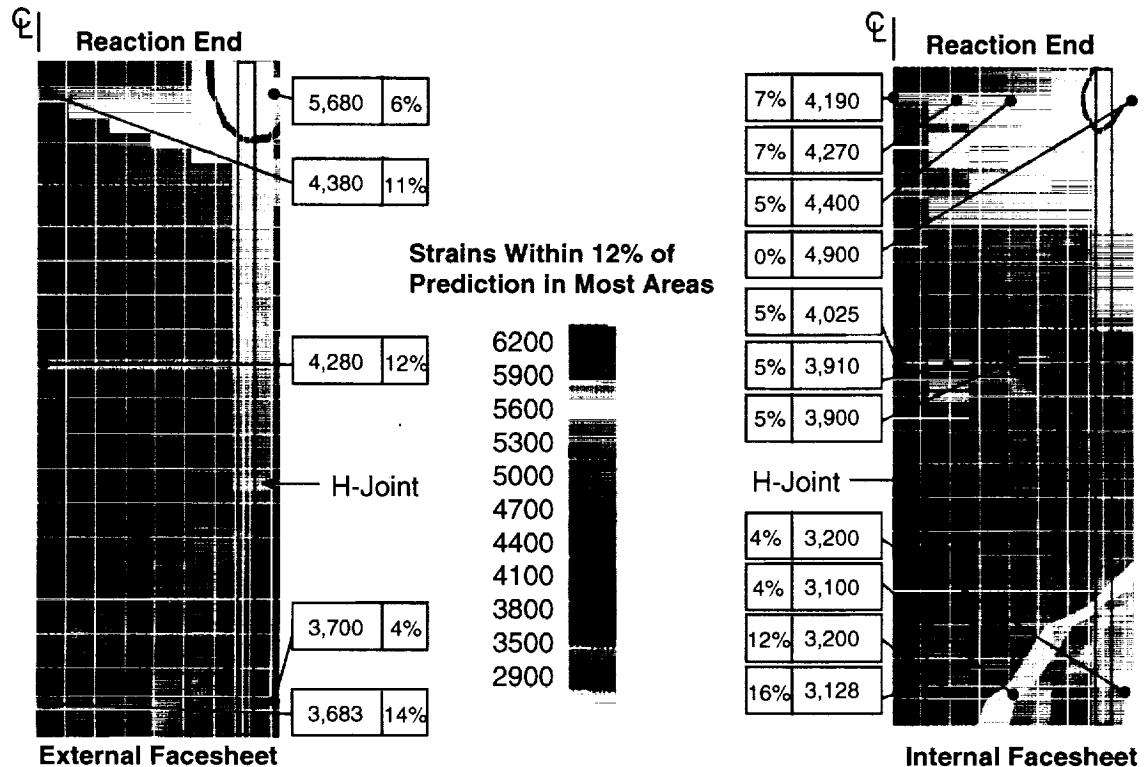**Figure 8. Correlation of Compressive Strains on Lower Skin Panel at Ultimate Load**

Figure 9. Correlation of Tensile Strains on Upper Skin Panel at Ultimate Load
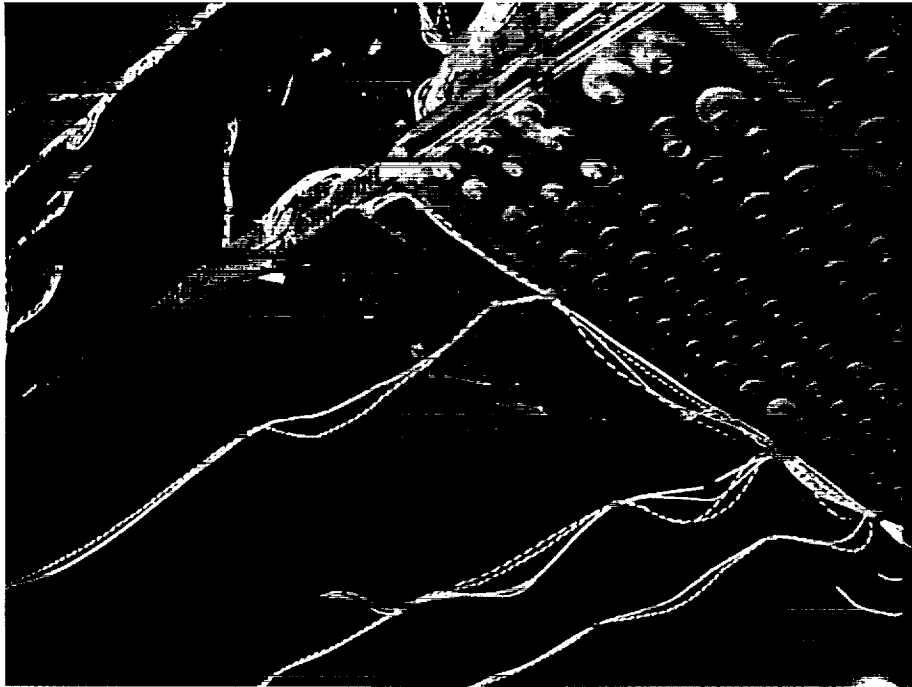
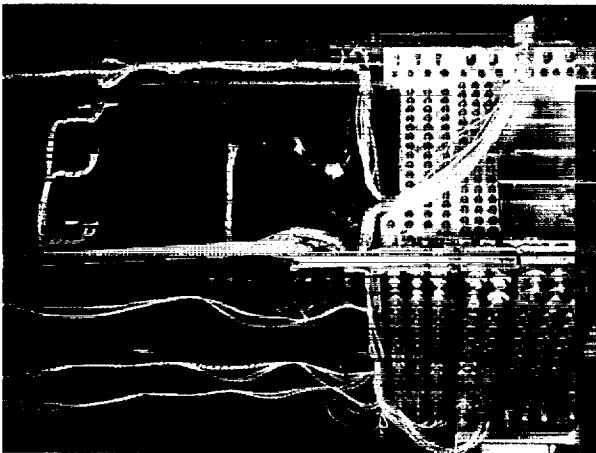**Figure 10. Wing Skin Compression Failure at 119% Ultimate Load**

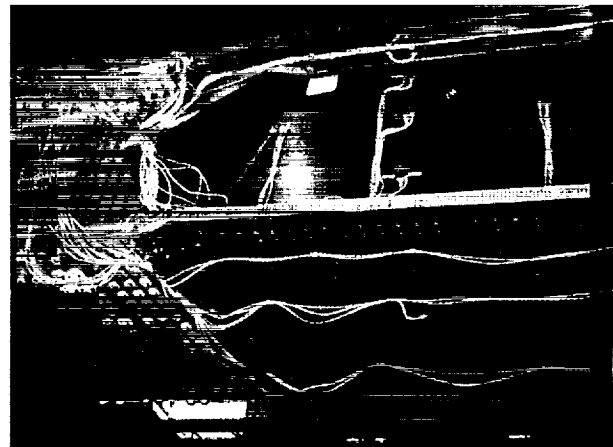**Figure 11. Wing Spar Secondary Shear Failure, Right Side**

Figure 12. Wing Spar Secondary Shear Failure, Left Side

SUMMARY

The wing box test program was a complete success, and validated the performance of an all composite sandwich wing box subjected to extreme bending loads typical of those for high speed transport wings. Thermal results also indicated that the selected materials were capable of sustaining flight limit loads in the 300°F environment. The failure at 119% of design ultimate load proved the capability of composite sandwich structures to sustain high magnitudes of structural load, and the ability to accurately model and predict these loads and failure modes. The close coordination between the NASA and Northrop Grumman participants on the test fixturing produced a system capable of producing the extreme loads required for this test. The high magnitude of load in the wing skins resulted in greater levels of complexity for the load introduction boxes than originally anticipated in meeting the factor of safety requirements for the fixture. In future programs it is highly recommended that conservative estimating practices be used for any fixtures requiring

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over 10,000 lb/in of running load capability. This conclusion was reinforced by other related experience on the HSR program (Reference 3).

REFERENCES

1. Deo, R.B., Russell, S.G., and Palm, T.E., "Innovative Structural Concepts for Supersonic Transports," NASA Contract NAS1-18842 Final Report, March 1997.
2. "BMI Sandwich Single Bay Wing Box Test Bismaleimide Matrix Composite Honeycomb Sandwich Concept," Boeing Subcontract RD-27263-C, Deliverable Item 24-10-02, December 1998.
3. "High Speed Research Program, Summary Report for Task 16 Materials and Structures Damage Tolerance and Task 25 Fuselage Structures," 30 September 1999.

First Chamber Costs

Materials	Quantity	Cost/Unit	Total
Raw Stainless Steel	4400	0.003	13.200
Machining Interface Chc	1	11.000	11.000
Transfer Line Materials	5	0.850	4.250
C-Seals	5	1.000	5.000
Consumables	1	2.000	2.000
Heat Exchanger Tubing	200	0.015	3.000
Heater Exchanger Shells	6	0.200	1.200
Boiler Pod Shell		0.350	0.000
Boiler Pod Heads		0.275	0.000
Heat Exchanger Caps	6	0.220	1.320
Fan Assembly	12	0.300	3.600
Fan Motors	12	0.200	2.400
Plumbing Component	1	3.000	3.000
Misc. Machining	1	5.000	5.000
Boiler Vessel		0.300	0.000
Boiler Heads		0.250	0.000
Finger Seals	5	0.500	2.500
Internal Heat Exchanger	1	3.000	3.000
Second Channel Machir	1	7.500	7.500
Second Channel Stock	1	1.200	1.200
Travel	1	5.000	5.000
Total	**12.500**	**Total**	**74.170**
			61.670

Labor Category	Projected	Actual
Project Management	82.23	34.208
Engineering	65.790	33.816
Design	49.840	23.574
Technician	44.860	12.209
Welding	44.860	15.342
		7.680
Total	**287.580**	**126.829**

New Chamber Costs

Materials	Quantity	Cost/Unit	Total
New Chamber			
Raw Stainless Steel	4400	0.004	17.600
Machining Interface Channe	1	11.000	11.000
Consumables	1	2.000	2.000
Heat Exchanger Tubing	150	0.015	2.250
Fan Assembly	12	0.500	6.000
Fan Motors	12	0.300	3.600
Plumbing Component	1	2.000	2.000
Misc. Machining	1	3.000	3.000
Internal Heat Exchanger	1	3.000	3.000
Second Channel Machining	1	7.500	7.500
Second Channel Stock	1	1.200	1.200
Transfer Line Materials	3	0.850	2.550
		Subtotal	**61.700**
2nd GHe Exchanger			
Transfer Line Materials	3	0.850	2.550
Heater Exchanger Shells	3	0.200	0.600
Boiler Pod Shell	1	0.350	0.350
Boiler Pod Heads	1	0.275	0.275
Heat Exchanger Caps	3	0.220	0.660
Heat Exchanger Tubing	100	0.015	1.500
Plumbing Component	1	2.000	2.000
Misc. Machining	1	2.000	2.000
Boiler Vessel	1	0.300	0.300
Boiler Heads	1	0.250	0.250
		Subtotal	**10.485**
Engineer, Design & Fab.			
Project Management	1	30.000	30.000
Design	1	40.000	40.000
Technician	1	20.000	20.000
Welding	1	30.000	30.000

Material	14.794		88.964
Material Burden			25.533
Fee (7%)	241.326		16.893
			258.219

KEY	
Not Required	
Boiler Pod Only	
Applicable to Both	

Fab. Material			25.000	25.000	1
Misc. Fab.			20.000	20.000	1
Contingency	17.409		15.000	15.000	1
Subtotal			**180.000**	
TOTAL			**252.185**	

KEY	
Applicable to Both	