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ADVANCED COMPOSITE STABILIZER FOR BOEING 737 AIRCRAFT

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FIFTH QUARTERLY TECHNICAL PROGRESS REPORT
19 JULY 1978 THROUGH 18 OCTOBER 1978

PREPARED FOR:
NATIONAL AERONAUTICS AND SPACE ADMINISTRATION
LANGLEY RESEARCH CENTER
HAMPTON, VIRGINIA 23665

IN RESPONSE TO:
CONTRACT NAS1-15025
DRL LINE ITEM NUMBER 01

BOEING COMMERCIAL AIRPLANE COMPANY
P.O. BOX 3707
SEATTLE, WASHINGTON 98124

Boeing Commercial
Airplane Company
Contract NAS1-15025

This Report is Submitted in Compliance
With DRL Line Item Number 018

FIFTH QUARTERLY TECHNICAL PROGRESS REPORT
19 July 1978 through 18 October 1978

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Boeing Commercial
Airplane Company
Contract NAS1-15025

FOREWORD

This report was prepared by the Boeing Commercial Airplane Company, Renton, Washington, under Contract NAS1-15025. It is the fifth quarterly technical progress report covering work performed between 19 July 1978 and 18 October 1978. The program is sponsored by the National Aeronautics and Space Administration, Langley Research Center (NASA-LRC). Dr. H. A. Leybold is the Project Manager for NASA-LRC.

The following Boeing personnel are principal contributors to the program during the reporting period: G. Ohgi, Design; R. Johnson, Structural Analysis; M. Garvey, Manufacturing Specialist; D. Grant, Production Manager; L.D. Pritchett, Technical Operations Coordinator; and D.V. Chovil, Business Support Manager.

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SUMMARY

Activities related to development of an advanced composites stabilizer for the Boeing 737 commercial transport are reported.

Activities include discussion of the design and weight status, stiffness requirements, the finite element model, test programs, quality assurance, and manufacturing producibility studies.

Design details of the graphite/epoxy components are virtually complete. The major effort is now being expended on the metal and fiberglass trailing-edge components. The bending and torsional stiffness properties were satisfactory for both stability/control and for flutter requirements. The finite element model input geometry was revised to reflect the latest changes to production drawings. The program is progressing as scheduled.

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SECTION 1.0

INTRODUCTION

The escalation of jet-fuel prices is causing a reassessment of technology concepts and trades used in designing and building commercial airplanes. The task is to incorporate fuel-saving concepts into commercial aircraft design.

The potential weight savings and fuel reduction resulting from the use of advanced composites in aircraft structure, especially primary structure, are significant. However, the lack of technical confidence and cost data has delayed their use in commercial aircraft.

Hardware programs conducted in a production environment are required to establish and demonstrate the safety, operating-life characteristics, and manufacturing cost of advanced composites primary structure.

Boeing's approach to the problem is to obtain reliable production, technical, and cost data bases by the integration of advanced composites technology development under NASA contracts, which, when combined with company effort, will accelerate the application of composites. This approach addresses these data bases, and develops realistic production costs in a commercial transport manufacturing environment. Program emphases are directed toward developing the information needed to obtain an early production commitment decision by management, and will be conducted in a production environment.

Preliminary developments, as covered in the first quarterly report, were devoted to conceiving, developing, and analyzing alternative design

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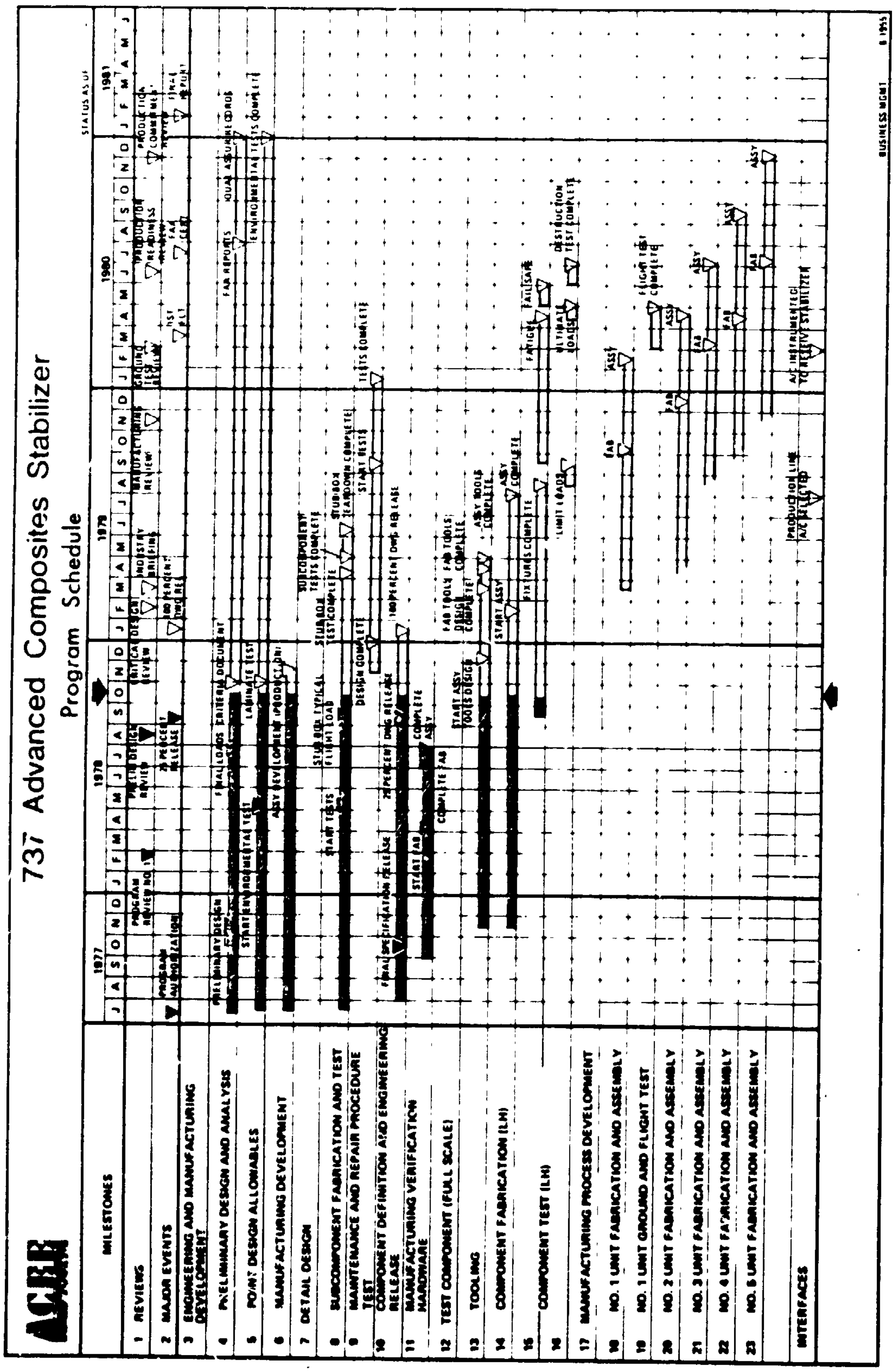

Figure 1-1. Program Master Schedule

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SECTION 2.0

DESIGN AND ANALYSIS

2.1 DESIGN STATUS

The design effort is continuing, with the detail design of graphite/epoxy components virtually completed. The major effort is now being expended on the metal and fiberglass trailing-edge components.

2.1.1 Skin Panel Stiffener Material

The graphite/epoxy tapes previously used in the skin panel stiffener caps have been replaced by graphite/epoxy fabric of equivalent stiffness. The current design and the previous design are shown in Figure 2-1. This change resulted from warpage experienced in the skin panels for the stub box (Test No. 21). A Boeing-funded study and verification program showed that removal of graphite/epoxy tape from the stringer caps will minimize warpage. The stub box skin warpage is described in Section 4.0.

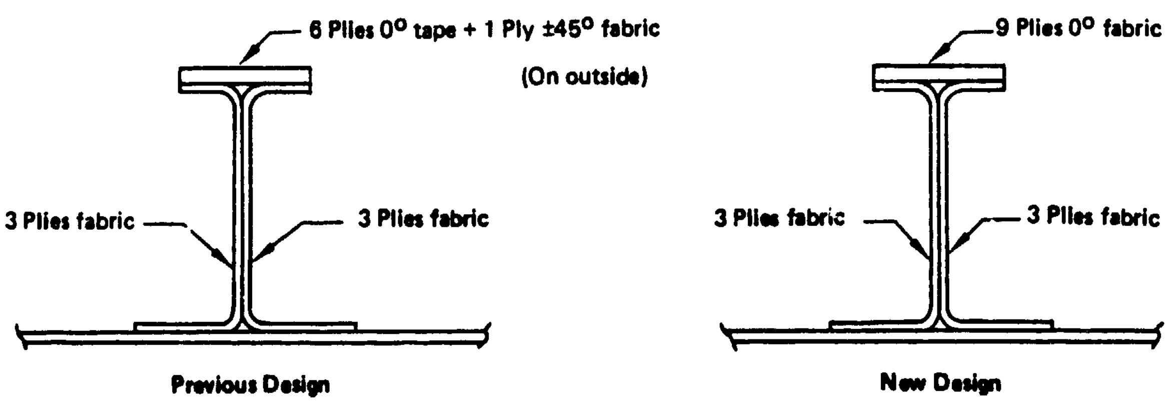

Figure 2.1. Skin Panel Stiffener Section

presented in Reference 1 have been evaluated by the Stability/Control and Flutter Technology groups. Analysis results have shown that the stiffness properties are satisfactory.

The ATLAS finite element model input geometry is being reviewed in detail. In the region of the rear spar and closure rib, the model is being revised to reflect the latest changes in production drawings.

2.3 WEIGHT STATUS

Table 2-1 presents the current weight status of the stabilizer. The only weight change, from the previous quarterly report, is the inclusion of omitted doubler material to the outboard section of the upper and lower skin panels. The upper panel increases by 0.4 kg (0.9 lb), the lower panel by 0.9 kg (1.9 lb), and the overall weight reduction decreases from 27% to 26%.

Weight evaluation of the stub box (Test No. 21) is completed and stabilizer production verification initiated.

SECTION 3.0

DEVELOPMENT TEST PLAN AND STATUS

3.1 ANCILLARY TEST PROGRAM

During this reporting period, the Test No. 1 bolted-joint moisture-conditioned specimens were tested. This recent wet test data and the previously published dry data (see Reference 2) are presented in Tables 3-1 and 3-2. These data are graphically displayed in Figures 3-1 and 3-2. Three typical failed specimens are shown in Figure 3-3. Results indicate the hot-wet or cold-dry conditions can affect joint strengths approximately ± 12%, compared to room temperature dry conditions.

Tension tests of impact-damaged coupons have been conducted. Impact levels used in the test program were selected based on the following:

- Visually undetectable damage
  2.82 N-m (25 in-lb) impact on 16-ply laminate
  5.65 N-m (50 in-lb) impact on 24-ply laminate
- Visually barely detectable damage
  5.65 N-m (50 in-lb) impact on 16-ply laminate
  8.36 N-m (74 in-lb) impact on 24-ply laminate

The specimen descriptions and test results are presented in Tables 3-3 and 3-4, and results are plotted in Figures 3-4 through 3-7. For comparison purposes, the undamaged room-temperature dry laminate ultimate tension stress is 413.7 MPa (60 ksi). A typical specimen is shown in Figure 3-8, and a typical test setup is shown in Figure 3-9. These test results indicated that:

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Table 3-1. 100% Load Transfer Joint Test Results (Cont)

Drawing No. Assembly No. 65C17768	Environmental condition [2]	Test temperature °C	Test temperature (°F)	Fastener diameter mm	Fastener diameter (in)	Specimen geometry L mm	Specimen geometry L (in)	Specimen geometry W mm	Specimen geometry W (in)	W/D	Failure load [1] N	Failure load [1] (lb)	Average end load N/mm	Average end load (lb/in)
-7	Wet	21	(70)	4.76	(3/16)	282.4	(11.12)	14.2	(0.56)	3	10 074	(2 265)	738	(4 205)
		21	(70)								10 942	(2 460)		
		21	(70)								10 408	(2 340)		
		82	(180)								9 251	(2 080)	689	(3 929)
		82	(180)								9 719	(2 185)		
		82	(180)								10 386	(2 335)		
-8	Wet	21	(70)			321.1	(12.64)	23.9	(0.94)	5	13 678	(3 075)	575	(3 277)
		21	(70)								13 522	(3 040)		
		21	(70)								13 900	(3 125)		
		82	(180)								13 122	(2 950)	541	(3 082)
		82	(180)								13 255	(2 980)		
		82	(180)								12 276	(2 760)		
-9	Wet	21	(70)	6.35	(1/4)	317.5	(12.5)	19.0	(0.75)	3	15 390	(3 460)	781	(4 453)
		21	(70)								13 967	(3 140)		
		21	(70)								15 212	(3 420)		
		82	(180)								15 946	(3 585)	828	(4 717)
		82	(180)								16 057	(3 610)		
		82	(180)								15 212	(3 420)		
-10	Wet	21	(70)			368.3	(14.5)	31.8	(1.25)	5	18 980	(4 267)	595	(3 394)
		21	(70)								19 064	(4 286)		
		21	(70)								18 562	(4 173)		
		-53	(-65)								19 794	(4 450)	614	(3 498)
		-53	(-65)								19 660	(4 420)		
		-53	(-65)								18 904	(4 250)		
		82	(180)								17 814	(4 005)	545	(3 108)
		82	(180)								17 925	(4 030)		
		82	(180)								16 102	(3 620)		

[1] Failures in countersunk splice plate

[2] Specimen conditioned at 60°C (140°F), 100% relative humidity until 0.23-cm (0.09-in) rider specimen reaches 1.1% moisture content.

- Static tension
- Fabric layup: [(0/90)(±45)]
- Material: Narmco 5208, 7-mil fabric

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Table 3-2. 50% Load Transfer Joint Test Results (Cont)

Drawing No. Assembly No. 65C17768	Environmental condition [2]	Test temperature °C	(°F)	Fastener diameter mm	(in)	Specimen geometry L mm	L (in)	W mm	W (in)	W/D	Failure load [1] N	(lb)	Average end load N/mm	(lb/in)
-11	Wet	21	(70)	4.76	(3/16)	359.1	(14.14)	23.9	(0.94)	5	22 044	(4 956)	916	(5 229)
		21	(70)								22 284	(5 010)		
		21	(70)								21 261	(4 780)		
		82	(180)								22 342	(5 023)	935	(5 338)
		82	(180)								22 031	(4 953)		
		82	(180)								22 587	(5 078)		
-12	Wet	21	(70)			396.7	(15.62)	33.3	(1.31)	7	26 817	(6 027)	814	(4 646)
		21	(70)								27 933	(6 280)		
		21	(70)								26 466	(5 950)		
		82	(180)								27 444	(6 170)	791	(4 517)
		82	(180)								25 131	(5 650)		
		82	(180)								26 377	(5 930)		
-13	Wet	21	(70)	6.35	(1/4)	419.0	(16.5)	31.8	(1.25)	5	27 600	(6 205)	868	(4 954)
		21	(70)								26 999	(6 070)		
		21	(70)								26 045	(6 305)		
		82	(180)								28 623	(6 435)	899	(5 134)
		82	(180)								28 267	(6 355)		
		82	(180)								28 756	(6 465)		
-14	Wet	21	(70)			469.9	(18.5)	44.4	(1.75)	7	33 938	(7 630)	748	(4 273)
		21	(70)								32 359	(7 275)		
		21	(70)								33 493	(7 530)		
		82	(180)								35 428	(7 965)	788	(4 499)
		82	(180)								34 806	(7 825)		
		82	(180)								34 828	(7 830)		

[1] Failures in countersunk splice plate

[2] Specimen conditioned at 60°C (140°F), 100% relative humidity until 0.23-cm (0.09-in) rider specimen reaches 1.1% moisture content.

- Static tension
- Fabric layup: [(0/90)(±45)]
- Material: Narmco 5208, 7-mil fabric

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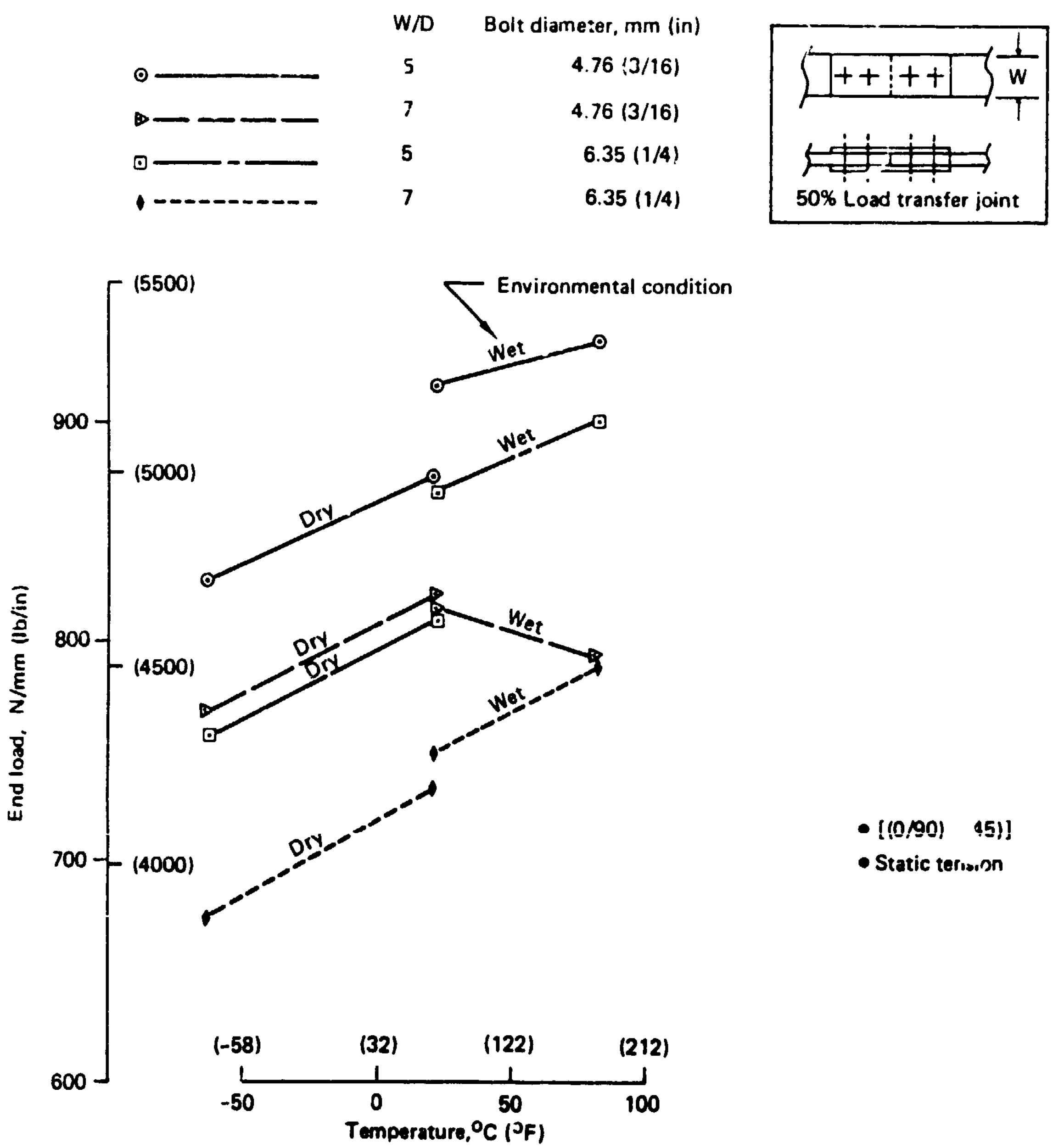

Note: Wet specimen conditioned at 60°C (140°F), 100% relative humidity until 0.23-cm (0.09-in) rider specimen reaches 1.1% moisture content.

Figure 3-2. Effect of Environment on 50% Load Transfer Joint

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Table 3-3. Impact Test Results—Dry

Dwg. No. Assy. No. [2] 65C17768	Test temperature °C	Test temperature (°F)	Impact level N-m	Impact level (in-lb)	Specimen geometry W mm	W (in)	t mm	t (in)	Fabric layup [1]	No. of plies	Failure load N	Failure load (lb)	Average failure stress MPa	Average failure stress (psi)
-3 and -5	21	(70)	2.825	(25)	58.4	(2.3)	3.05	(0.12)	[(0/90)(±45)]	16	60 315	(13 560)	347.2	(50 367)
	21	(70)									60 982	(13 710)		
	21	(70)									64 185	(14 430)		
	-53	(-65)									56 178	(12 630)	324.8	(47 101)
	-53	(-65)									59 381	(13 350)		
	-53	(-65)									57 913	(13 020)		
	82	(180)									63 784	(14 340)	374.7	(54 348)
	82	(180)									69 656	(15 660)		
	21	(70)	5.65	(50)							42 567	(9 570)	247.1	(35 833)
	21	(70)									44 302	(9 960)		
	21	(70)									45 103	(10 140)		
	-53	(-65)									42 434	(9 540)	227.6	(33 007)
	-53	(-65)									38 431	(8 640)		
	-53	(-65)									40 699	(9 150)		
	82	(180)									44 613	(10 030)	247.7	(35 930)
	82	(180)									43 324	(9 740)		
	82	(180)									44 391	(9 980)		
-4 and -6	21	(70)					4.57	(0.18)		24	90 472	(20 340)	341.9	(49 589)
	21	(70)									93 408	(21 000)		
	21	(70)									90 072	(20 250)		
	-53	(-65)									85 001	(19 110)	313.4	(45 459)
	-53	(-65)									80 731	(18 150)		
	-53	(-65)									85 402	(19 200)		
	21	(70)	8.36	(74)							76 061	(17 100)	277.0	(40 169)
	21	(70)									71 390	(16 050)		
	21	(70)									74 459	(16 740)		
	-53	(-65)									68 455	(15 390)	262.0	(37 995)
	-53	(-65)									68 722	(15 450)		
	-53	(-65)									72 725	(16 350)		

[1] Material: Narmco 5208, 7-mil fabric
[2] Reworked per Drawing 65C17797

- Environmental condition—dry
- Static tension

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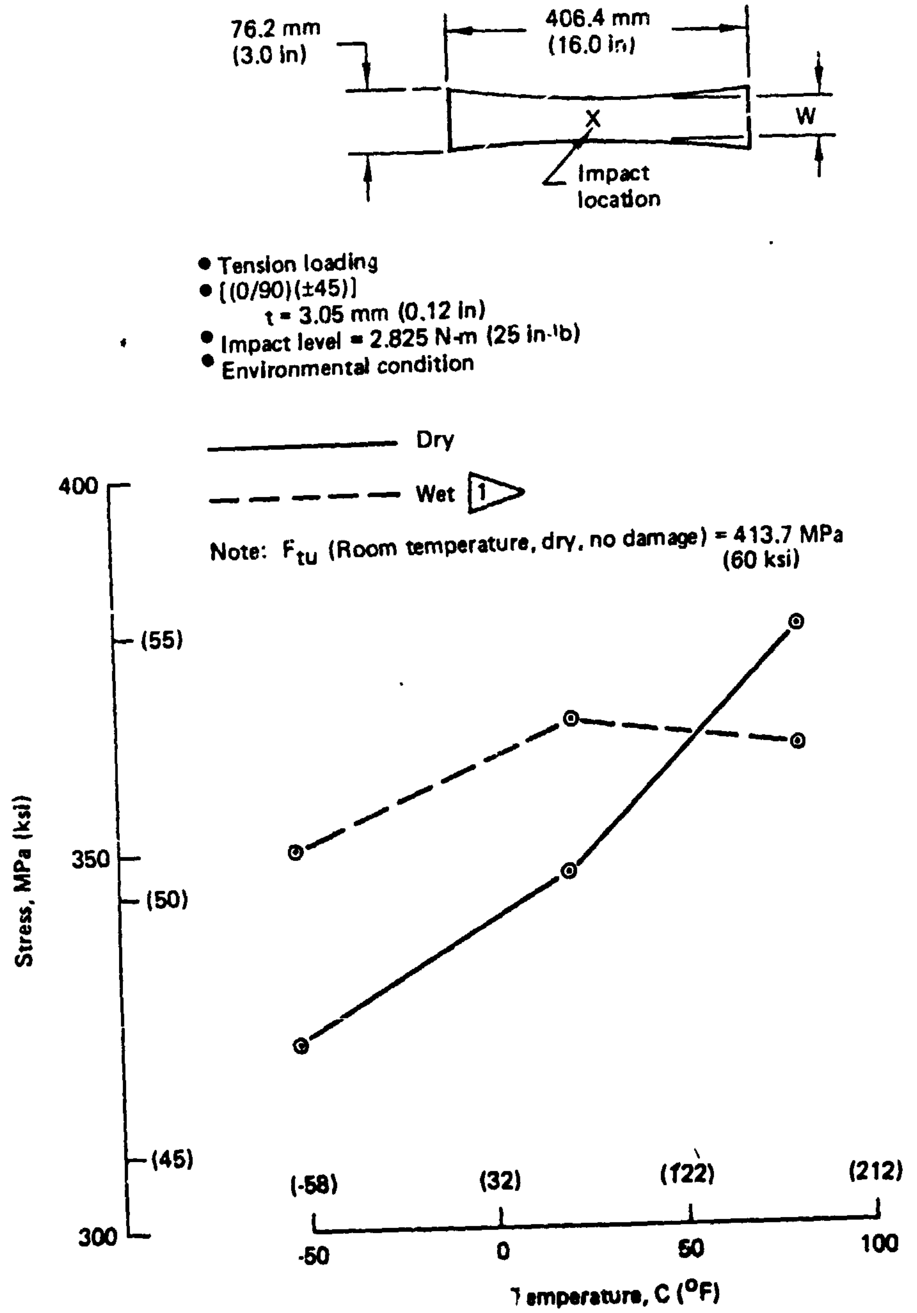

1 Specimens conditioned at 60°C (140°F), 100% relative humidity until 0.23-cm (0.09-in) rider specimen reaches 1.1% moisture content.

Figure 3-4. Effect of 2.82 N-m Impact on Laminate [(0/90)(±45)] (16 Plies)

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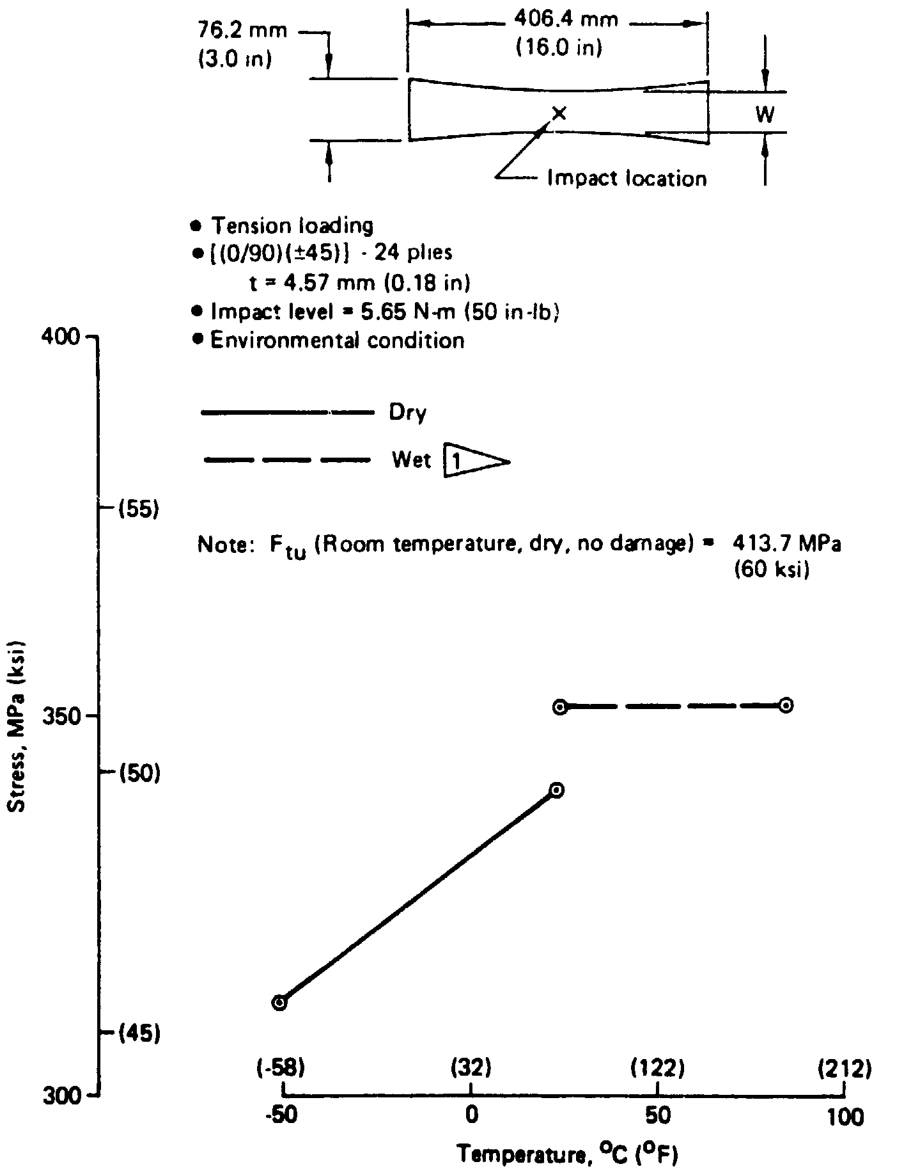

1 Specimens conditioned at 60°C (140°F), 100% relative humidity until 0.23-cm (0.09-in) rider specimen reaches 1.1% moisture content.

Figure 3-6. Effect of 5.65 N-m Impact on Laminate [(0/90)(±45)] (24 Plies)

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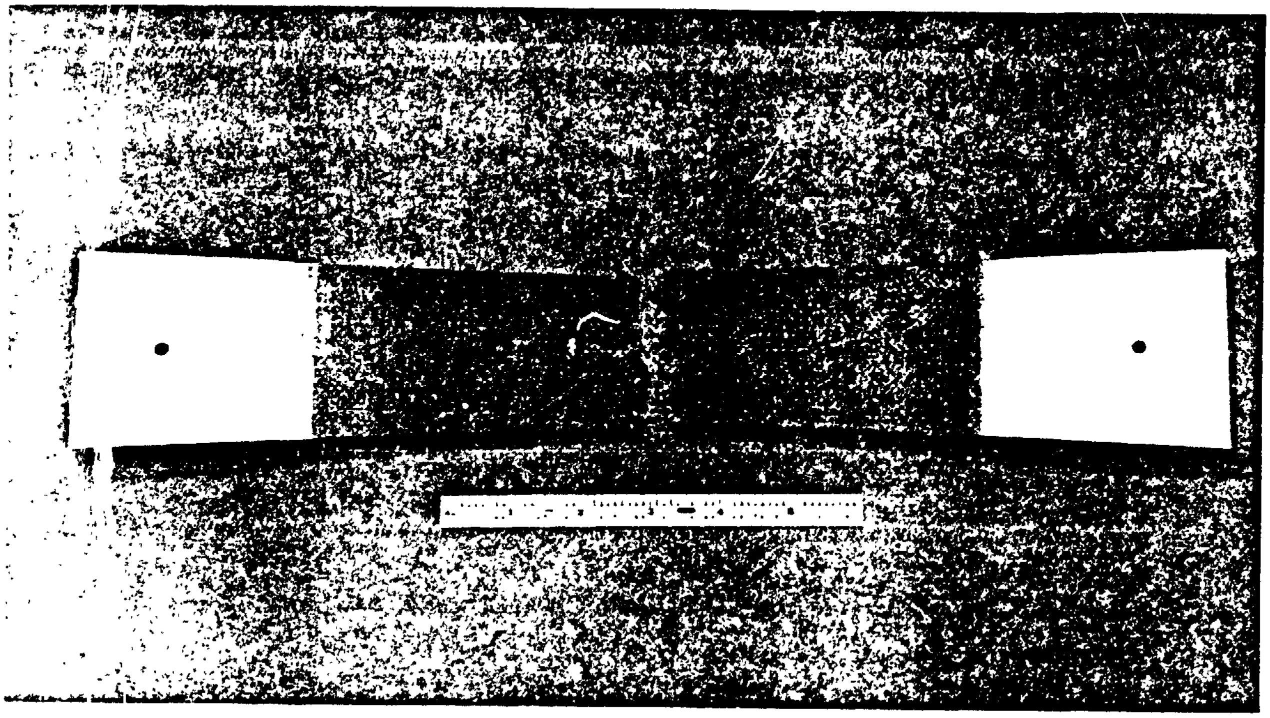

Figure 3-8. Typical Tension Impact Specimen

Figure 3-9. Tension Impact Test Setup

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Table 3-5. Spar Chord Crippling Test Data

Drawing No. Assembly No. Identification No. 65C17791		Test temperature		Environmental condition [1]	(P_{cr}) ultimate		(P_{cr}) elastic		(ϵ_{cr}) elastic
		C°	(°F)		N	(lb)	N	(lb)	$\mu\epsilon$
-1	-1A	21	(70)	Dry	54 933	(12 350)	31 136	(7 000)	1 850
↓	-1B	↓	↓	↓	53 376	(12 000)	31 803	(7 150)	1 920
↓	-1C	↓	↓	↓	52 486	(11 800)	30 469	(6 850)	1 820
↓	-1D	-53	(-65)	↓	51 152	(11 500)	33 360	(7 500)	1 930
↓	-1E	↓	↓	↓	48 038	(10 800)	29 802	(6 700)	1 870
↓	-1F	↓	↓	↓	46 704	(10 500)	31 136	(7 000)	1 900
↓	-1G	21	(70)	Wet	39 587	(8 900)	28 022	(6 300)	1 730
↓	-1H	↓	↓	↓	47 594	(10 700)	26 688	(6 000)	1 780
↓	-1I	↓	↓	↓	44 035	(9 900)	30 246	(6 800)	1 970
↓	-1J	82	(180)	↓	39 142	(8 800)	28 467	(6 400)	1 880
↓	-1K	↓	↓	↓	29 802	(6 700)	29 802	(6 700)	1 860
↓	-1L	↓	↓	↓	35 584	(8 000)	28 022	(6 300)	1 790
-2	-1	21	(70)	Dry	169 914	(38 200)	145 005	(32 600)	5 800
↓	-2	↓	↓	↓	169 914	(38 200)	142 336	(32 000)	5 007
↓	-3	↓	↓	↓	152 566	(34 300)	136 109	(30 600)	4 858
↓	-4	21	(70)	Wet	142 781	(32 100)	130 326	(29 300)	5 100
↓	-5	↓	↓	↓	149 008	(33 500)	137 888	(31 000)	5 275
↓	-6	↓	↓	↓	170 358	(38 300)	137 888	(31 000)	5 080
↓	-7	82	(180)	↓	153 901	(34 600)	126 768	(28 500)	4 940
↓	-8	↓	↓	↓	123 654	(27 800)	117 427	(26 400)	4 520
↓	-9	↓	↓	↓	128 547	(28 900)	125 434	(28 200)	4 790

[1] **Wet specimens conditioned at 60°C (140°F), 100% relative humidity until 0.23-cm (0.09-in) rider specimen reaches 1.1% moisture content.**

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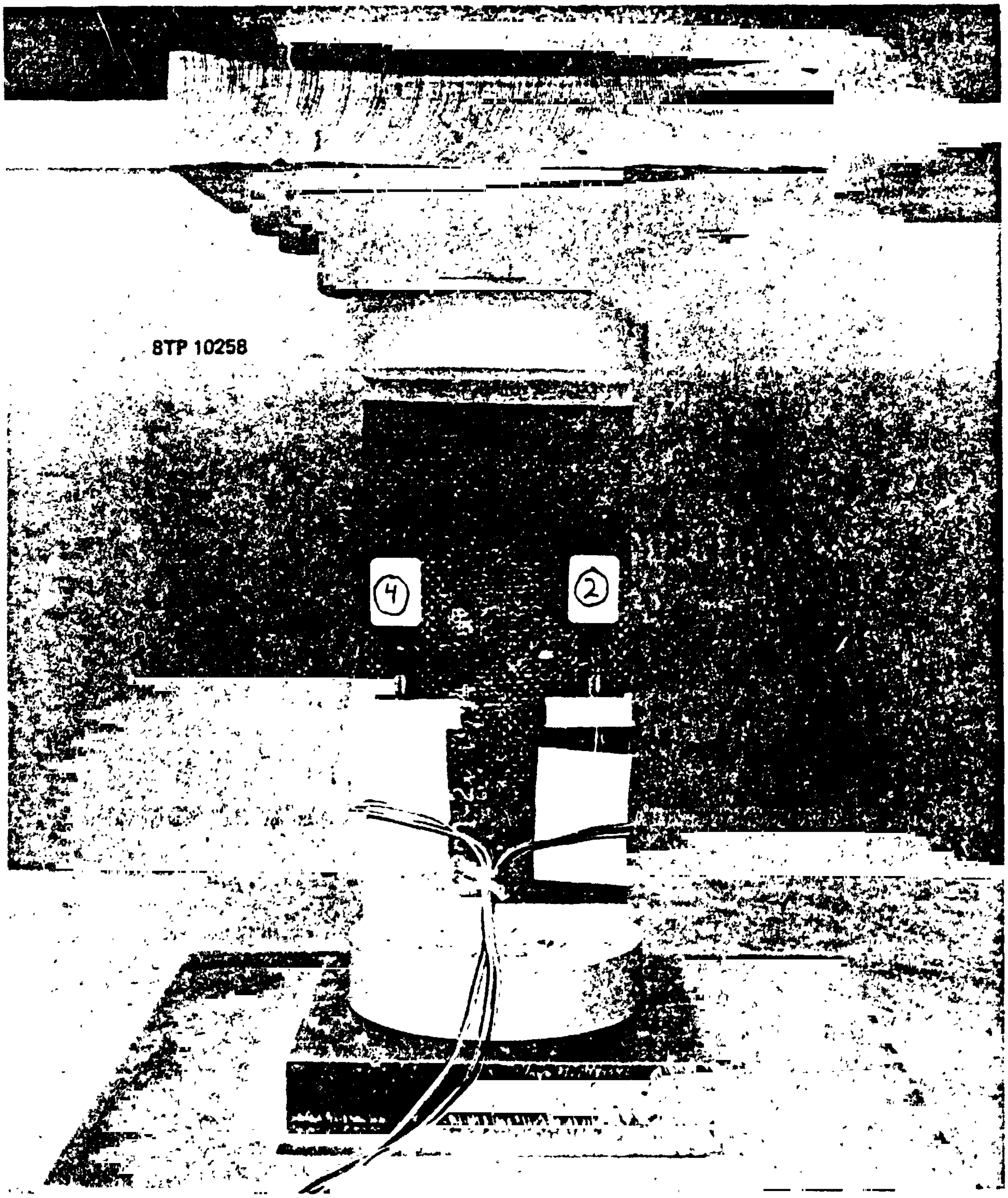

Figure 3-11. Typical Spar Chord Crippling Test Setup

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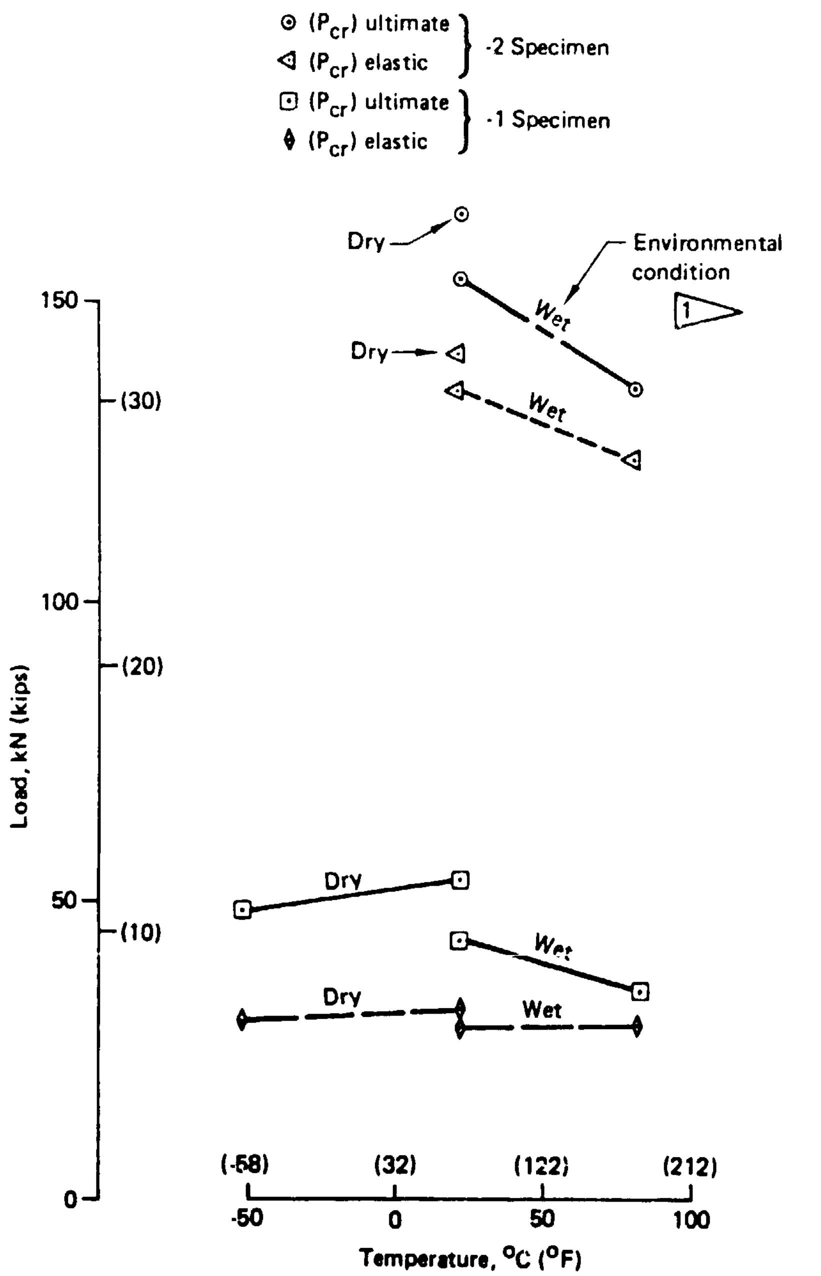

1 Specimens conditioned at 60°C (140°F), 100% relative humidity until 0.23 cm (0.09-in) rider specimen reaches 1.1% moisture content.

Figure 3-13. Effect of Temperature and Environmental Conditioning on Spar Chord Crippling Specimens

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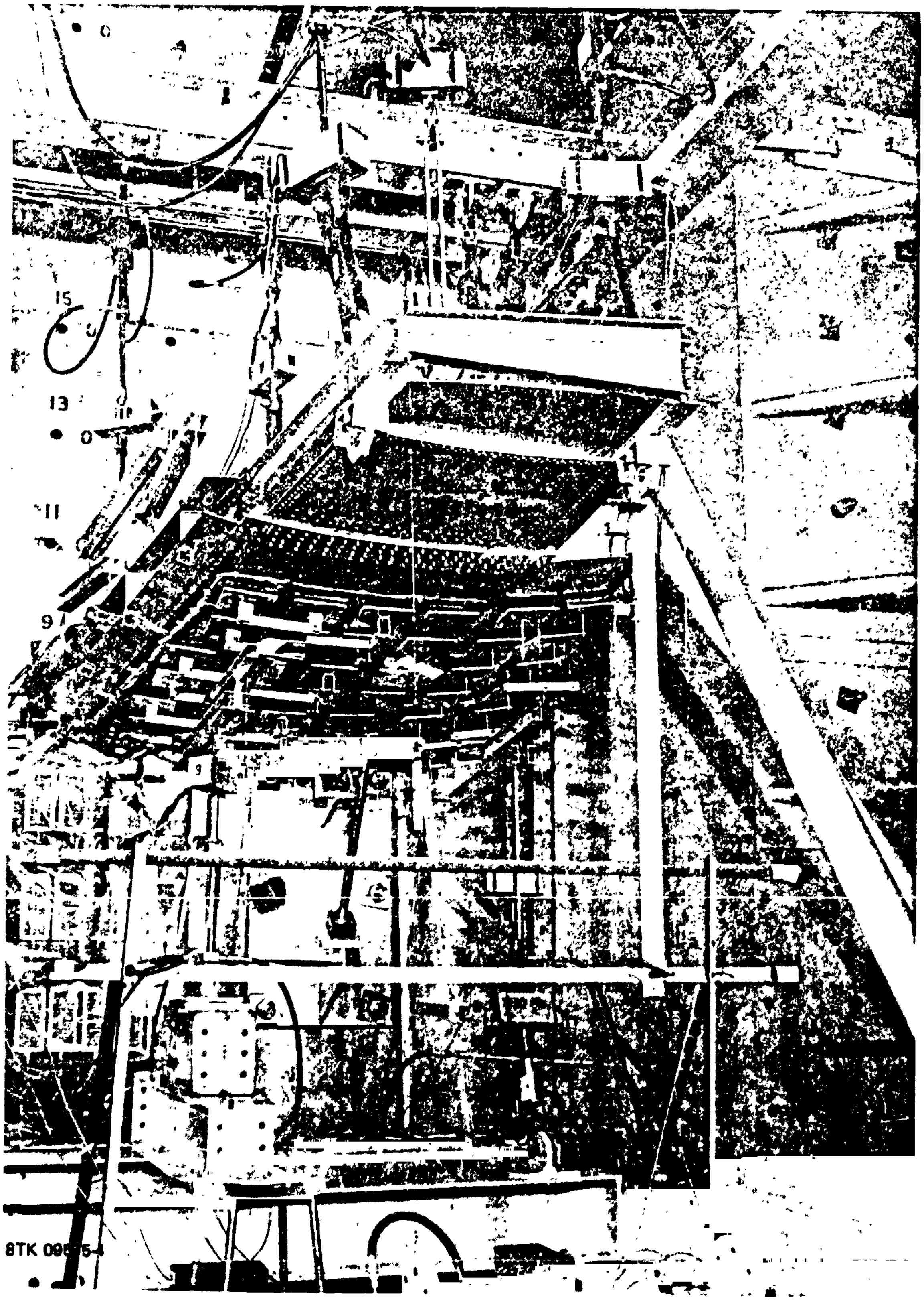

Figure 3-14. Stub Box Test Setup

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SECTION 4.0

OPERATIONS DEVELOPMENT

This section discusses results of the manufacturing producibility studies, ancillary test component fabrication and assembly, quality assurance development, and manufacturing verification hardware.

4.1 PRODUCIBILITY STUDIES

Two producibility studies were conducted during this reporting period. One addressed minimizing porosity in the spar radii, the other addressed minimizing warpage in the "I"-stiffened configurated design.

Spar Radius Study-Both the front and rear spar details for the stub box had porosity in the tool side radii (Figure 4-1). In order to minimize and possibly eliminate the porosity, a study was conducted to determine the effect of reducing the corner radius of the layup tool. A test spar/lug aluminum tool was modified with three different radii: 0.48 cm (0.19 in), 0.25 cm (0.10 in), and 0.08 cm (0.03 in). Two parts from each radius were fabricated. The preliminary results indicate that the 0.25-cm (0.10-in) radius gave the best results; however, the improvement was not sufficient to warrant the cost and schedule impact for rework of the spar tools.

"I"-Stiffened Panel Warpage Study-The completed "I"-stiffened stub box skin panels had up to 1.59-cm (0.627-in) warpage. This warpage did not affect the assembly of the stub box. However, a Boeing-funded program was conducted in an effort to minimize the warpage. Under this program, an "I"-stiffened configurated design, using the identical skin layup as the stub box upper surface panel, was fabricated by replacing the unidirectional (0^{o}) tape on the top of the stringer-free flange with graphite/epoxy fabric. The overall warpage of this panel was less than half the warpage of the stub box panels. The warpage was reduced to 0.672 cm (0.302 in).

Engineering has reviewed the results and is proceeding with this configuration for the production stabilizer skin panels.

4.2 ANCILLARY TEST COMPONENT FABRICATION

The ancillary test plan includes allowables and environmental, and concept verification. The following describes the fabrication status of each effort as of September 28, 1978.

Allowables and Environmental—This part of the ancillary test program includes material allowables (Test No. 1), mechanical joints (Test No. 5), and environmental specimens (Test No. 4). All ancillary test specimens have been fabricated and assembled.

Concept Verification—This part of the ancillary test program includes spar chord crippling (Test No. 7), skin-to-rib joints (Test No. 9), skin panel (Test No. 10), spar shear web (Test No. 11), spar/lug (Test No. 12), sonic test box (Test No. 20), stub box (Test No. 21), discontinuous laminate critical strain (Test No. 22), skin panel-to-rib joint (Test No. 24), production verification specimens (Test No. 25), and manufacturing feasibility spar test coupons (Test No. 26). The following describes the fabrication and assembly status:

- Spar chord crippling (Test No. 7)—Detail fabrication and assembly complete
- Skin-to-rib joints (Test No. 9)—Detail fabrication and assembly complete
- Skin panel (Test No. 10)—Detail fabrication 40% complete (Figure 4-2)
- Spar shear web (Test No. 11)—Detail fabrication and assembly complete
- Spar/lug (Test No. 12)—Detail fabrication and assembly complete (Figures 4-3 and 4-4)

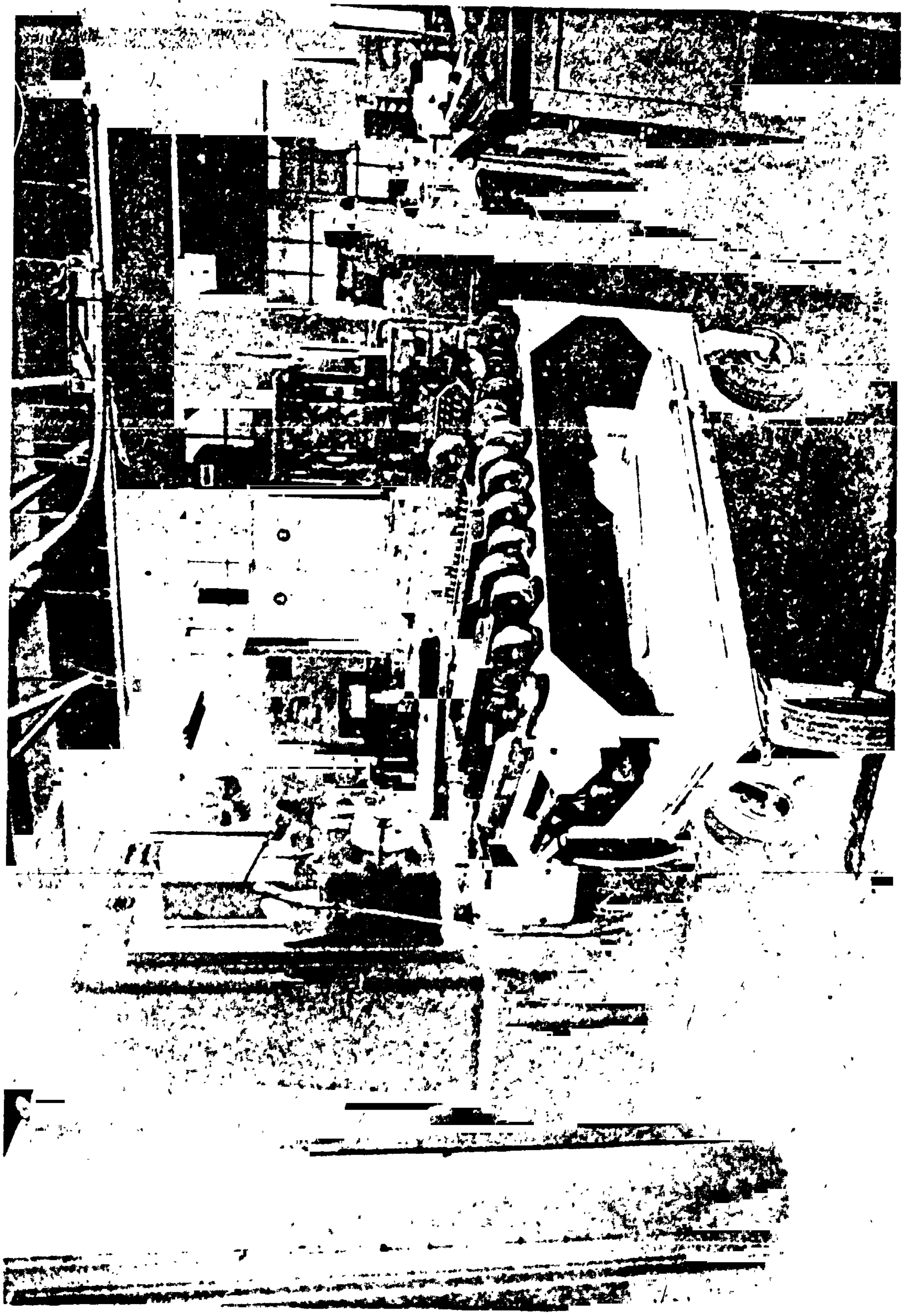

Figure 4-4. Test No. 12, Spar/Lug—Showing Spar/Lug Tension Specimens Ready for Drilling

As reported, the two major items derived from the fabrication of the stub box were the corner build-up on ribs, and the warpage of the "I"-stiffened skin panel. The production ribs have been redesigned to eliminate the corner build-ups. During the assembly of the "I"-stiffened skin panels, the warpage could be removed with hand pressure. However, on the production skin panels, the graphite/epoxy tapes used in the caps of the stiffeners will be replaced by graphite/epoxy fabric of equivalent stiffness in order to reduce the warpage.

Figures 4-5 through 4-16 show the assembly and completed stub box.

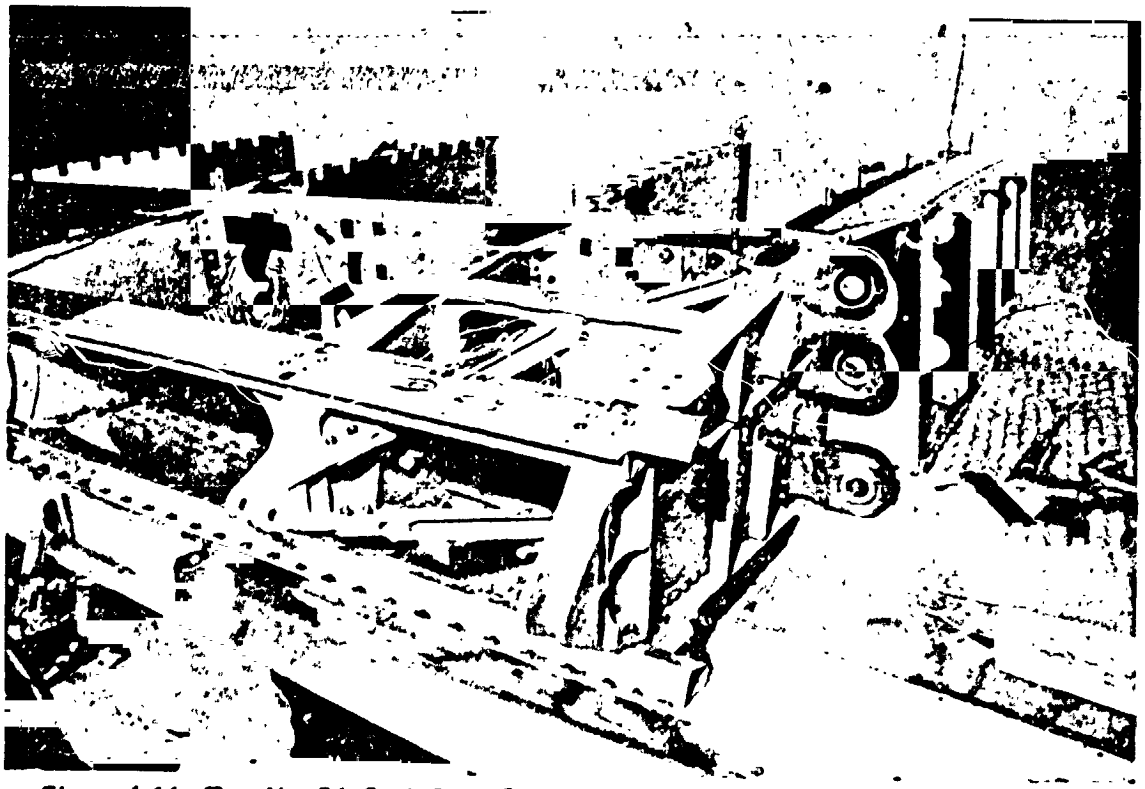

Figure 4-11. Test No. 21, Stub Box—Showing Trailing-Edge Beam, Rear Spar, and Graphite/Epoxy Ribs in Place.

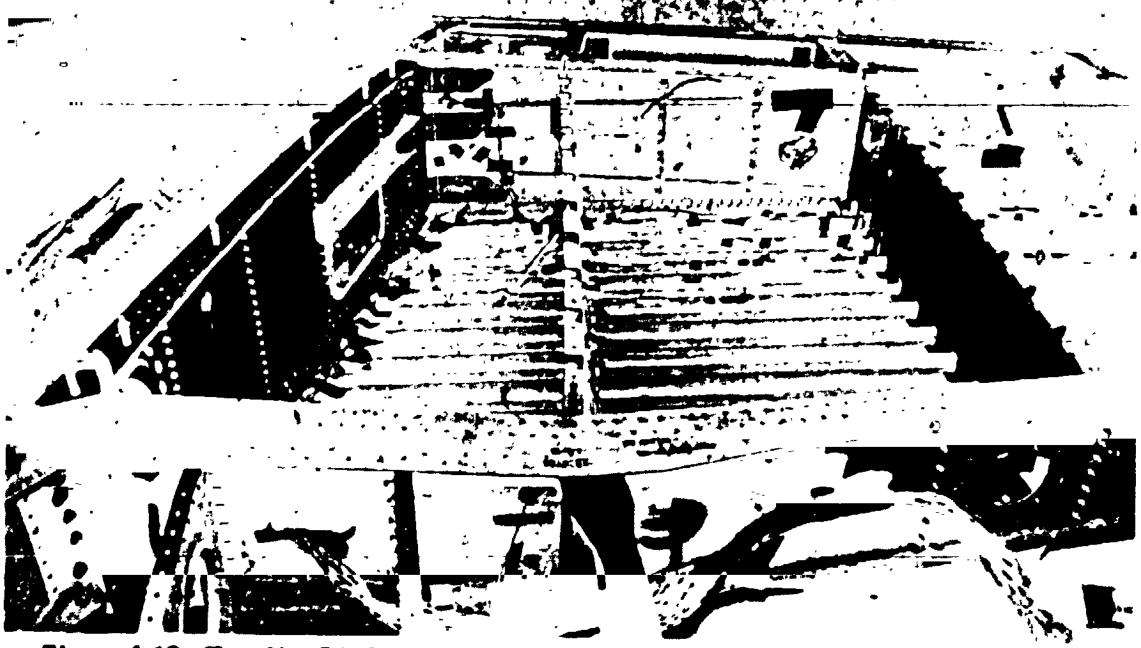

Figure 4-12. Test No. 21, Stub Box—Showing Front and Rear Spar, Lower Skin Panel and Ribs with Instrumentation

4-8, 4-9, 4-10

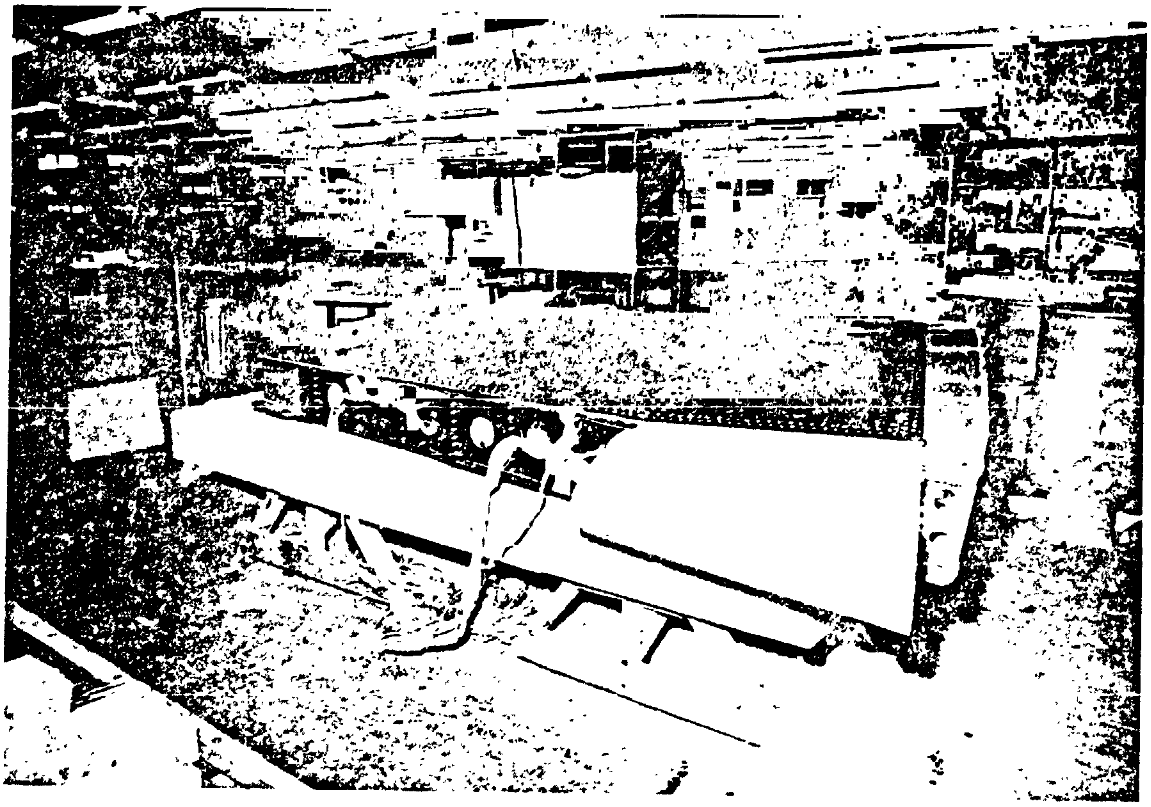

Figure 4-15. Test No. 21, Stub Box—Showing Front View of Completed Assembly

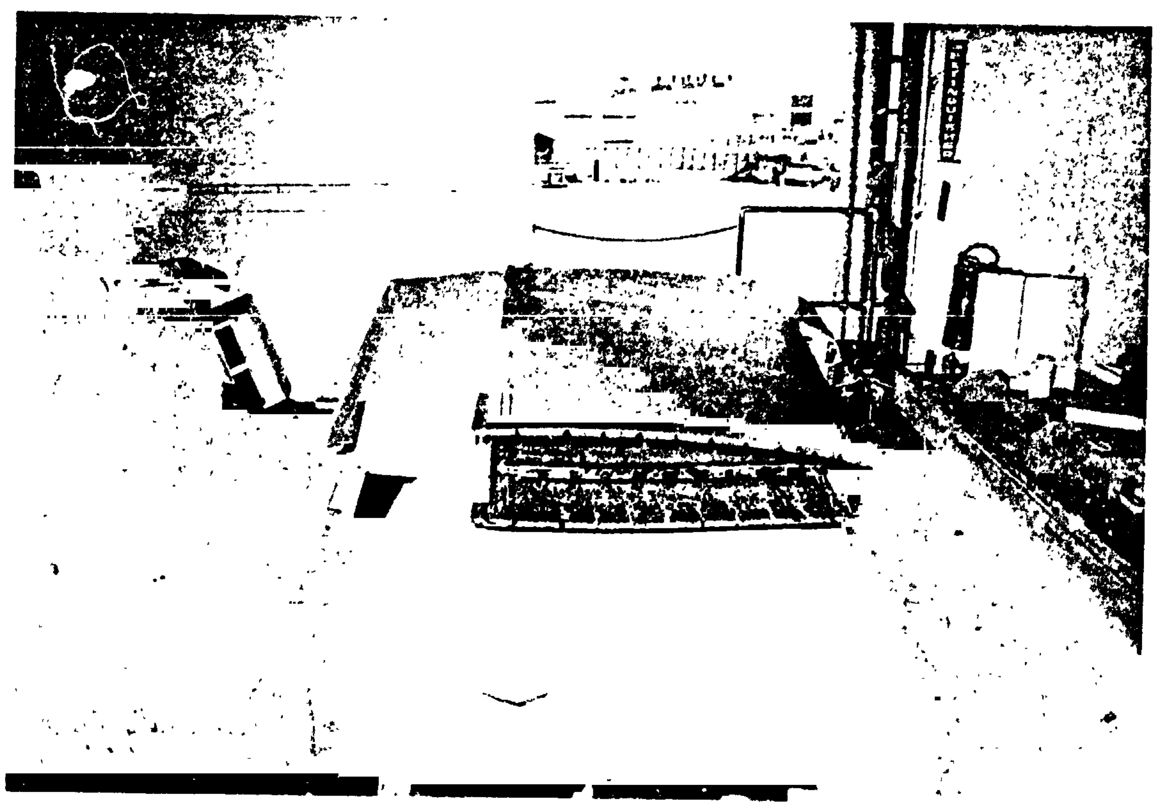

Figure 4-16. Test No. 21, Stub Box—Showing Side View of Completed Assembly

SECTION 5.0

REFERENCES

1. "Advanced Composite Stabilizer for Boeing 737 Aircraft," Fourth Quarterly Technical Progress Report, NASA Contract NAS1-15025, July 1978

2. "Advanced Composite Stabilizer for Boeing 737 Aircraft," Third Quarterly Technical Progress Report, NASA Contract NAS1-15025, April 1978

3. "Advanced Composite Stabilizer for Boeing 737 Aircraft," Second Quarterly Technical Progress Report, NASA Contract NAS1-15025, January 1978