STRUCTURAL ANALYSIS OF THE MINIMAX 91 ULTRALIGHT/EXPERIMENTAL AIRCRAFT

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INTRODUCTION

This analysis has been prepared for the purpose of verifying the adequacy of the structural design of the 1991 model of the Minimax ultralight/ experimental aircraft design. Since the aircraft is built by the purchaser using the materials kit furnished by TEAM, Inc., no claim can be made by the company as to the quality of construction. However, as is explained in the appendix, the original basic structure as designed and properly constructed by the company has been static tested to design loads without failure. Structural adequacy of modifications of the basic design have been analytically verified, and where necessary, additional component tests have been performed.

TABLE OF CONTENTS

A. AIRCRAFT SPECIFICATIONS AND DESIGN LIMITS

Minimax is a conventional configuration single engine monoplane with a strut braced shoulder wing. The pilot is seated between the main and rear spar. A conventional fixed landing gear with steerable tailwheel is used. The lower ends of the wing struts attach to the outboard end of the landing gear axle. Construction is of Sitka Spruce, Northern White Pine, Mahoghany plywood, and Birch plywood, with aluminum and steel fittings and struts.

Specifications:

Design gross weight:	560 lbs.
Wing area:	112.5 sq. ft.
Span:	25 ft.
Airfoil:	NACA 4414 (Modified) $C_{Lmax}=1.5$, $C_{Mac}=-.1$
Horizontal tail area:	21.56 sq. ft.
Horizontal tail span:	7.5 ft.
Vertical tail area:	7.5 sq. ft.
Length:	15 ft.
Power:	28 to 45 hp.
Design load factor:	+4.4 G -1.8 G
Minimum safety factor:	1.5

Design Performance at gross weight (See Figure A-1):

V_s:	35 mi./hr.
V_a:	67 mi./hr.
V_d:	111 mi./hr.
V_{ne}:	100 mi./hr.

Conditions of Analysis:

1. Design positive and negative load factors at V_a and V_d.
2. Loads with full aileron deflection at 2/3 of positive design load factor at V_a.
3. Loads with 1/3 aileron deflection at 2/3 of positive design load factor at V_d.
4. Center of gravity range 21 to 30 percent mac.

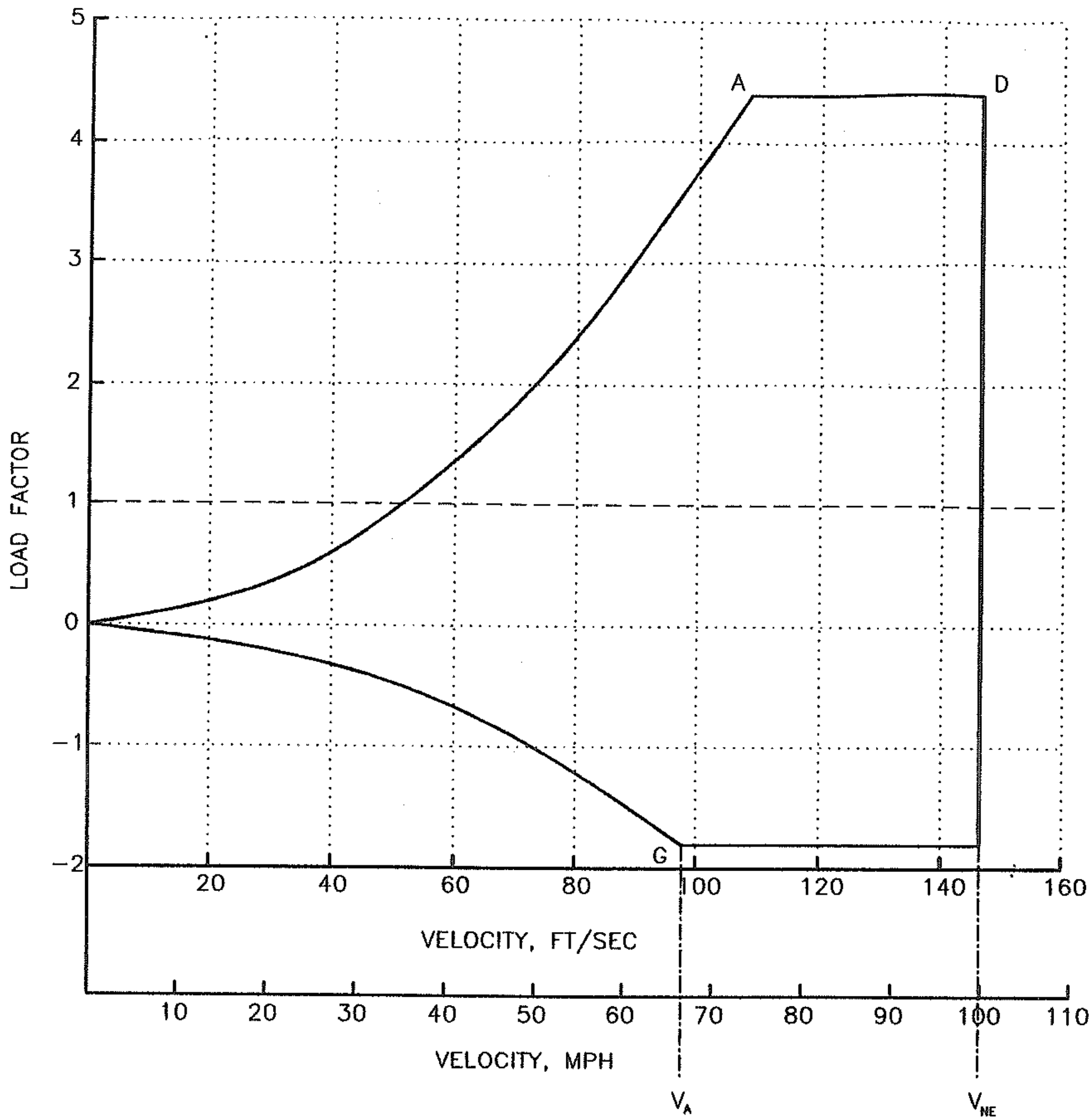


FIGURE A−1
FLIGHT ENVELOPE

B. MATERIAL SPECIFICATIONS

References 2, 3, 4, 7, 9

1. Western White Pine (Canadian):

Mod. of Rupture	= 9300	psi.
Mod. of Elasticity	= 1,460,000	psi.
Ult. Compression, par. to Grain	= 5240	psi.
Yield Compression, perp. to Grain	= 470	psi.
Ult. Shear, par. to Grain	= 920	psi.

2. Sitka Spruce:

Mod. of Rupture	= 10,200	psi.
Mod. of Elasticity	= 1,570,000	psi.
Ult. Compression, par. to Grain	= 5610	psi.
Yield Compression, perp. to Grain	= 580	psi.
Ult. Shear, par. to Grain	= 1150	psi.

3. Birch Plywood:

Properties of Yellow Birch:

Mod. of Rupture	= 16600	psi.
Max. Compression, Par. to Grain	= 8170	psi.
Max. Compression, Perp. to Grain	= 970	psi.
Max. Tension, Perp. to Grain	= 920	psi.
Max. Shear, Par. to Grain	= 1880	psi.

3-Ply, 1/8 Thickness:

Ultimate Shear Strength, 0-90 deg.	= 1940	psi.
Ultimate Shear Strength, 45 deg.	= 2800	psi.
Max. Bearing, par. to face grain	= 5750	psi.
Max. Bearing, perp. to face grain	= 3350	psi.

3-Ply, 1/16 Thickness:

Ultimate Shear Strength, 0-90 deg.	= 2400	psi.
Ultimate Shear Strength, 45 deg.	= 2800	psi.
Max. Bearing, par. to face grain	= 5750	psi.
Max. Bearing, perp. to face grain	= 3350	psi.

5-Ply, 1/4 Thickness:

Ultimate Shear Strength, 0-90 deg.	= 1940	psi.
Max. Bearing, par. to face grain	= 5290	psi.
Max. Bearing, perp. to face grain	= 3850	psi.

4. Aluminum, 6061-T6:

Ultimate Tensile Strength	=45000	psi.
Ultimate Shear Strength	=30000	psi.
Ultimate Bearing Strength	=69000	psi.

5. Steel, 4130-N:

Ultimate Tensile Strength	=95000	psi.
Yield Strength, Tension	=75000	psi
Ultimate Shear Strength	=55000	psi.
Ultimate Bearing Strength	=140000	psi.

C. WING STRUCTURE

1. General:

The wing is a two spar, strut braced, all wood structure. Lift loads are distributed proportionally between the front and rear spar, thence transmitted to the fuselage and lower strut attach point (landing gear axle) via two aluminum struts. Torsion is resisted by the distribution of loads between the spars. A D-section leading edge provides redundant torsion resistance. Drag loads are resisted by wood diagonals. Spars are constructed of Northern White Pine caps separated by Birch plywood webs. Ribs are conventional truss construction. The wing is covered with 1.6 ounce dacron fabric. Figure C-1 shows the wing structure general arrangement.

2. Conditions of Analysis

The following specific loading conditions were examined to determine the most critical wing loads. Loads indicated are design loads. A minimum safety factor of 1.5 was applied to all calculations. Calculated safety factors are shown for critical components.

(1) 4.4 G positive symmetrical load factor at points A and D (figure A-1).
(2) 1.8 G negative symmetrical load factor at point G.
(3) 2.93 G pos. load factor with full aileron deflection at V_a.
(4) 2.93 G pos. load factor with 1/3 aileron deflection at V_d.

3. Load Distribution:

Total positive lift (perpendicular to the relative wind) is assumed to be 1.05 times the gross weight times the design positive load factor. Total negative lift is equal to the weight times the negative design load factor. Schrenk's approximation is used to determine lift distribution, modified by aileron deflection for maximum torsion load determination. Increased moment coefficient due to aileron deflection is assumed to be (Ref. 1):

$$C_{Mac}=C_{Mac}-.01\text{ del}$$ (del is down aileron, degrees)

A numerical method is used to calculate moment, shear, and torsion. The wing semispan is divided into 80 segments (1.875 inches per segment). The total loads are determined by adding the incremental loads per section from the tip inward to the specific spanwise location. For the analysis, the resultant of lift and drag is resolved into components perpendicular to the airfoil centerline for shear and moment (normal), and parallel to the centerline for drag (axial). Note that the axial component is then positive (forward acting), relative to the wing structure.

The basic lift distribution is modified by the wing weight (Distributed load), the strut loads, and the fuselage mount loads (Concentrated loads). The additional bending moment, shear, and torsion is added or subtracted from the basic aerodynamic loads as necessary.

4. Spar Analysis:

Spar analysis is done in accordance with Reference 8. In order to determine the maximum allowable compressive stresses in the spar caps, the material modulus of rupture is modified according to the spar cross section (per Reference 8) to determine the maximum ultimate stress in the compression flange. The opposite flange is then checked for the negative compressive load, and dimensions modified if necessary. These ultimate stresses are multiplied by 2/3 to determine the maximum allowable stress. The calculated stresses are then compared with these maximum allowables. A doubler was added to the rear spar as a result of preliminary calculations. Figure C-2 and C-3 show the calculated spar bending loads versus allowable loads.

Maximum shear load for the front spar occurs at the +4.4 G symmetrical load, and is calculated to be 293 pounds in the web just inboard of the strut attach position (72.75 inches from aircraft centerline). From Ref. 10, section 2.7, the shear web is calculated to support a shear load of 262 pounds without buckling. Since shear buckling of thin plywood is an inexact science, a test was performed (in addition to the original structural static test) on a representative shear sample identical in dimensions to the spar web. This sample resisted a total shear load of 375 pounds before the test fixture failed. The sample itself did not fail. The leading edge D-section is not loaded in torsion, and is held essentially rigid (in twist) relative to the spar by the wing and strut structure. Therefore, the D-section can be assumed to support part of the shear load of the main spar. Based on the deflection of the spar web for a given load, the D-section is calculated to support 46 percent of the load of the shear web , or 172 pounds, for a total of 547 pounds. This provides a safety factor of 1.87.

The rear spar shear web dimensions preclude buckling failure, therefore, the maximum shear load depends on the basic shear strength of the 1/16 inch plywood. The maximum shear load for the rear spar occurs at V_d with 1/3 aileron deflection. This load is calculated to be 350 pounds just inboard of the strut attach position. Ultimate shear strength of this section is 637 pounds, for a safety factor of 1.82.

5. Struts and Fittings:

All attachments to the wood spars are essentially identical aluminum strap fittings, made of 3/16 X 3/4, 6061-T6 Aluminum alloy attached to the wood spars via two AN4 bolts. The spar cross section is 3/4 inch pine, with one 1/16 inch and two 1/8 inch birch ply doublers. Ultimate strength of each fitting is therefore 2863 pounds. Figure C-4 shows the typical fittings and the associated maximum loads. Note that the front spar butts up against the fuselage, thus transferring the load directly to the fuselage carry-through in compression.

Figure C-4 also shows the strut fittings and minimum safety factors. Safety factors in all cases exceed the minimum requirements.

The strut system does not include jury struts, therefore the maximum allowable compressive load is determined by buckling of the struts. The maximum compressive load in the front strut is 821 pounds. Classical long column theory yields a buckling load for the strut of 1266 pounds, for a factor of safety of 1.54. The rear strut is not loaded significantly in compression for the required conditions.

6. Drag Bracing and Compression Ribs:

The drag/anti-drag loads are resisted by wood diagonal members, and compression ribs as shown on figure C-5. Also shown is a typical compression rib. Maximum loads in each member are indicated on the figure. For most members, the weakest point is the glue joint between the member and the front or rear spar. Gluing area for the diagonal drag members is 1.8 square inches, for a nominal strength of 1656 pounds.

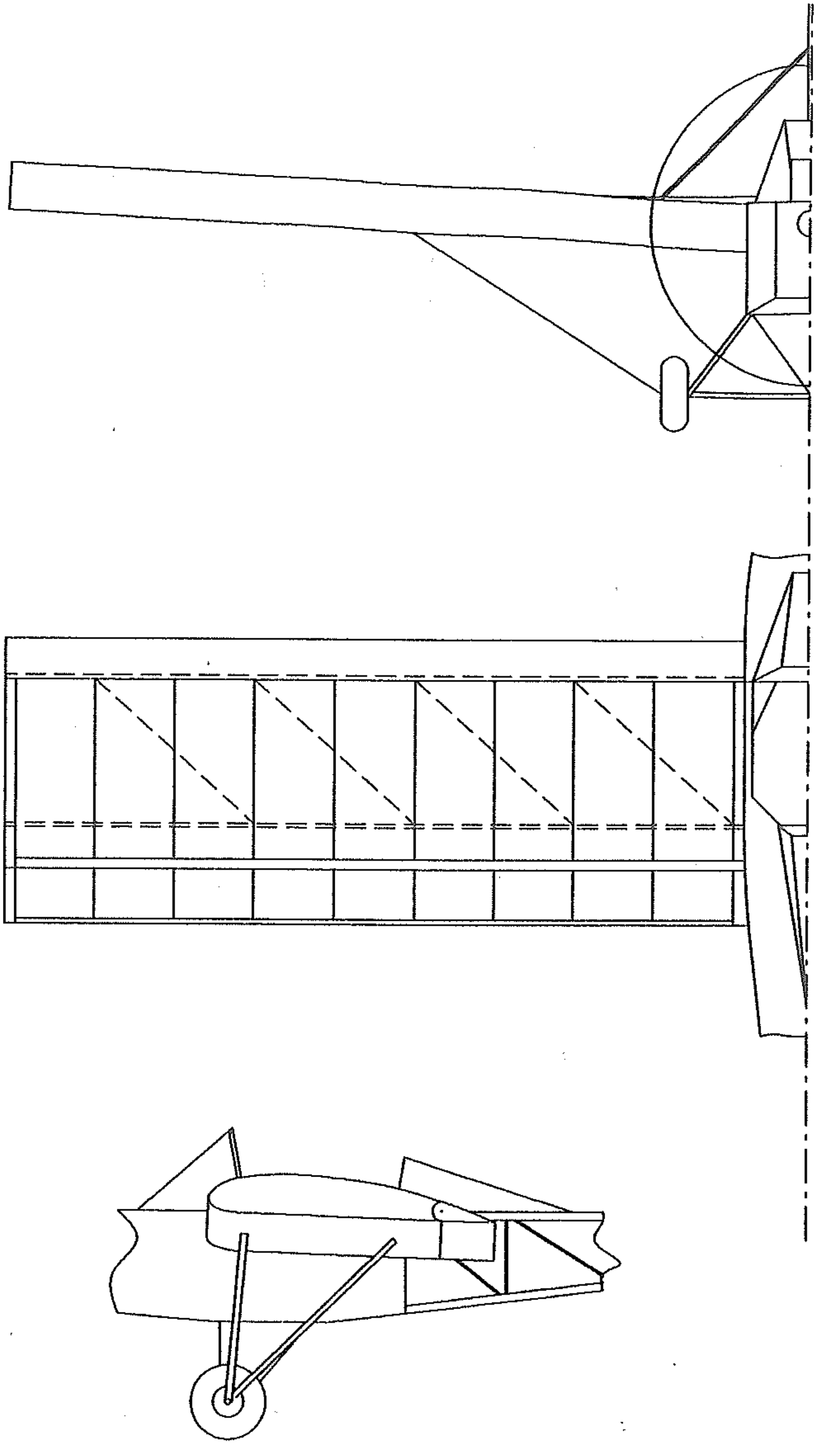


FIGURE C-1

WING AND STRUT LAYOUT

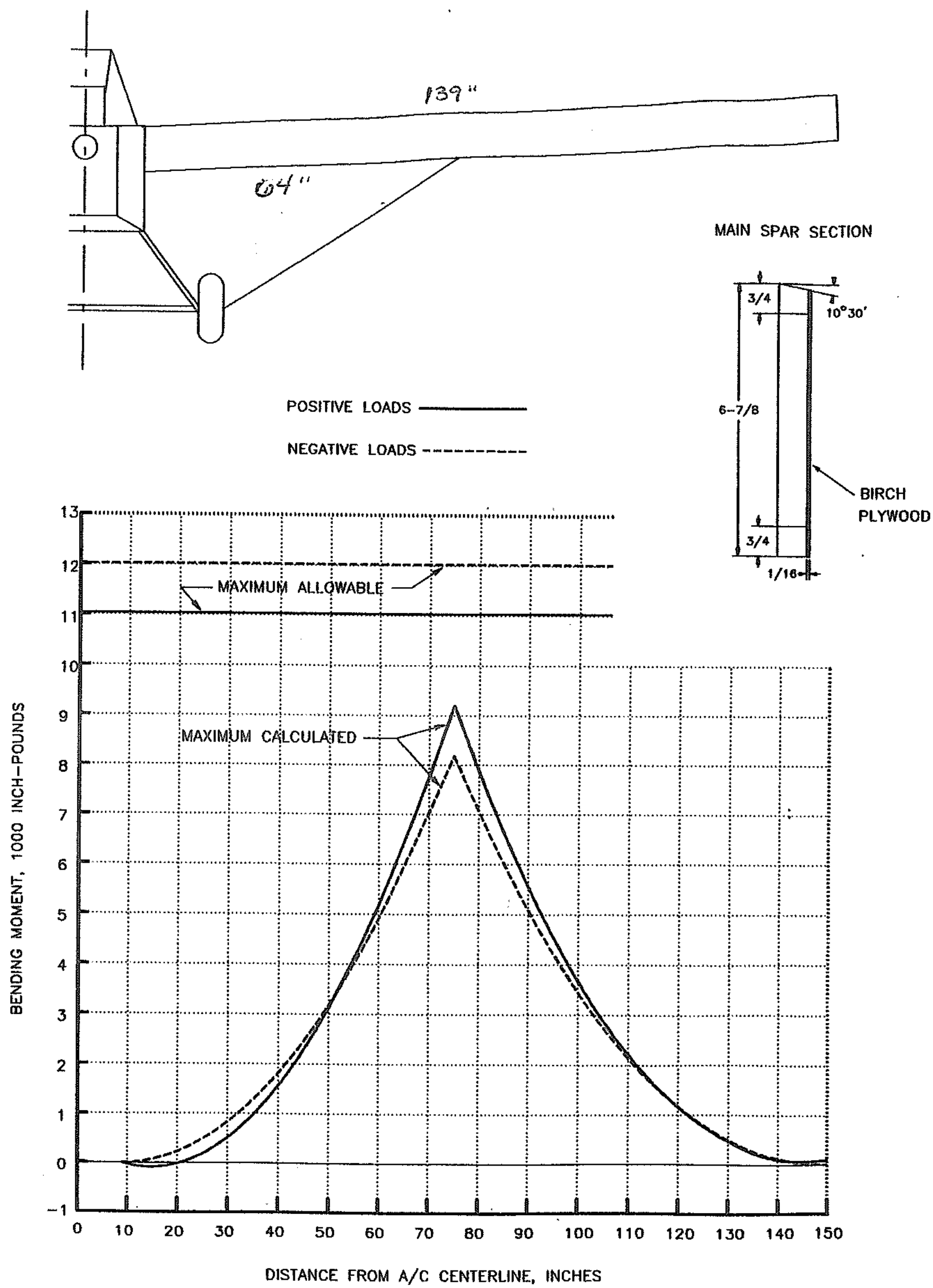


FIGURE C-2
MAIN SPAR BENDING LOADS

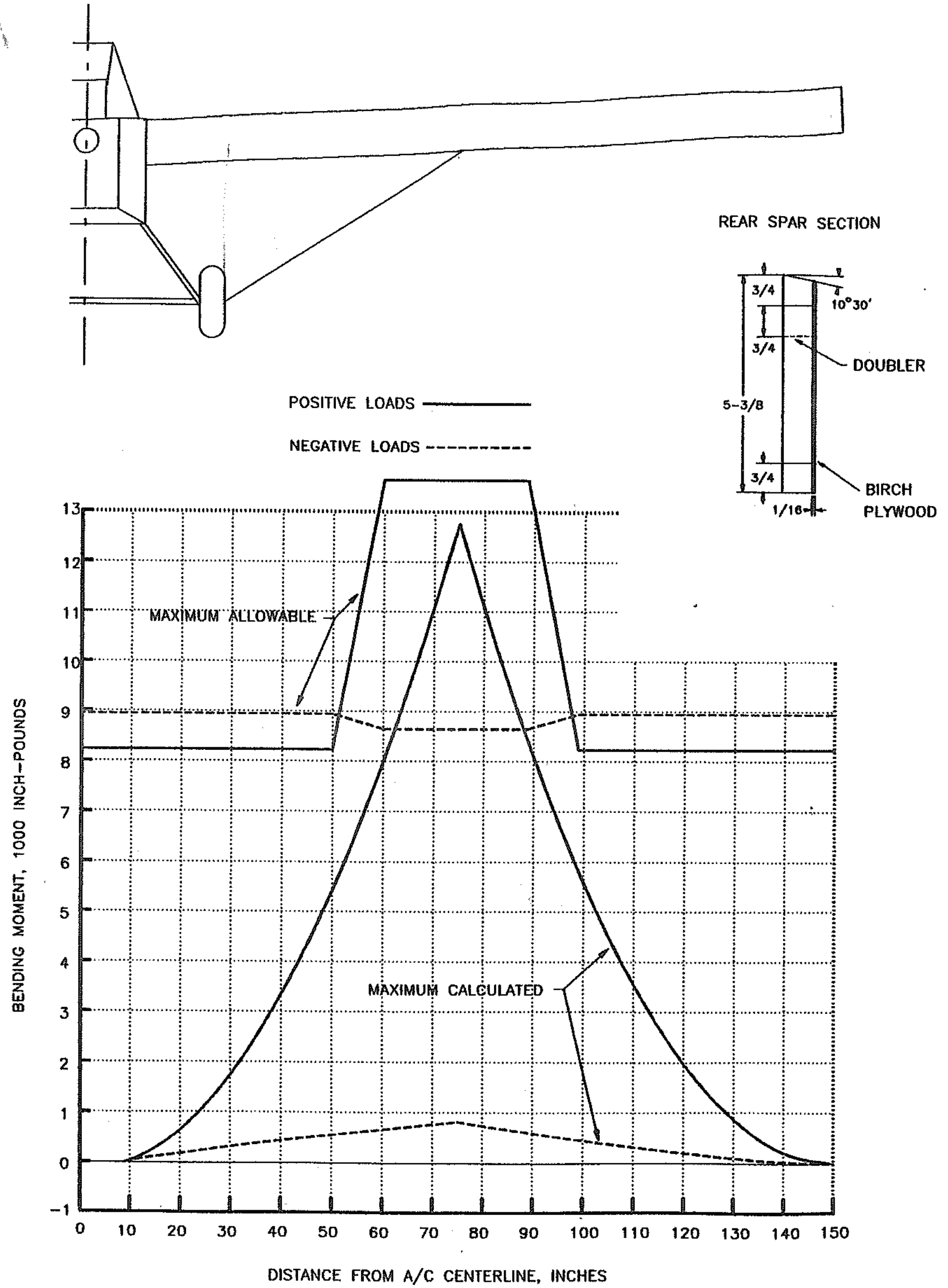
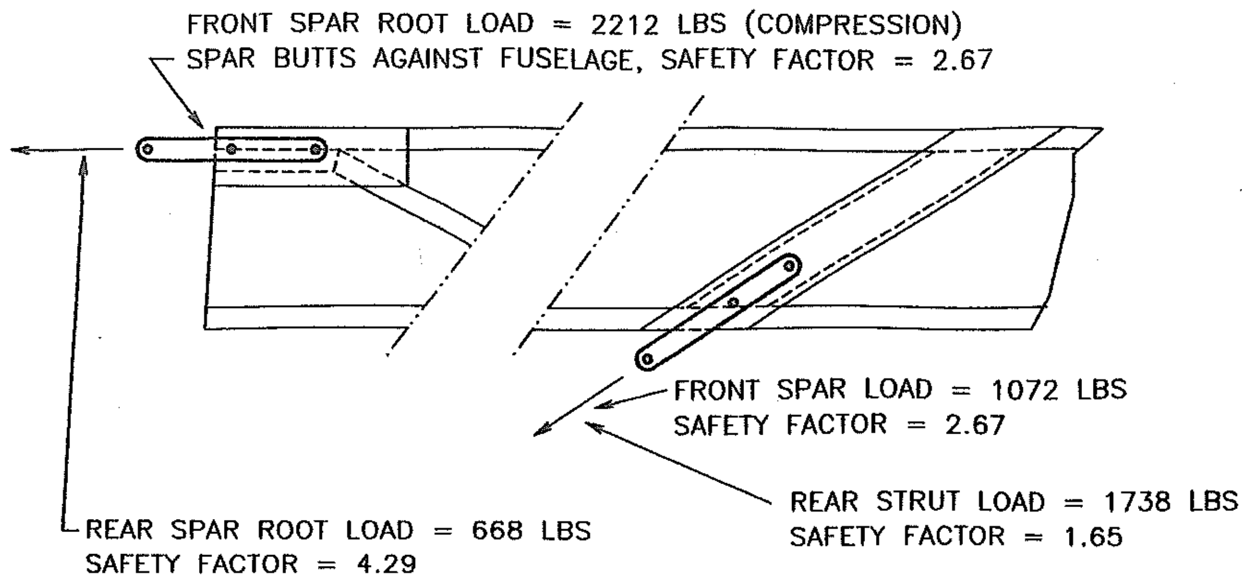


FIGURE C-3

REAR SPAR BENDING LOAD



NOTE: EACH FITTING IS ATTACHED TO SPAR VIA TWO AN4 BOLTS.
STRENGTH OF EACH FITTING IS 2863 POUNDS.

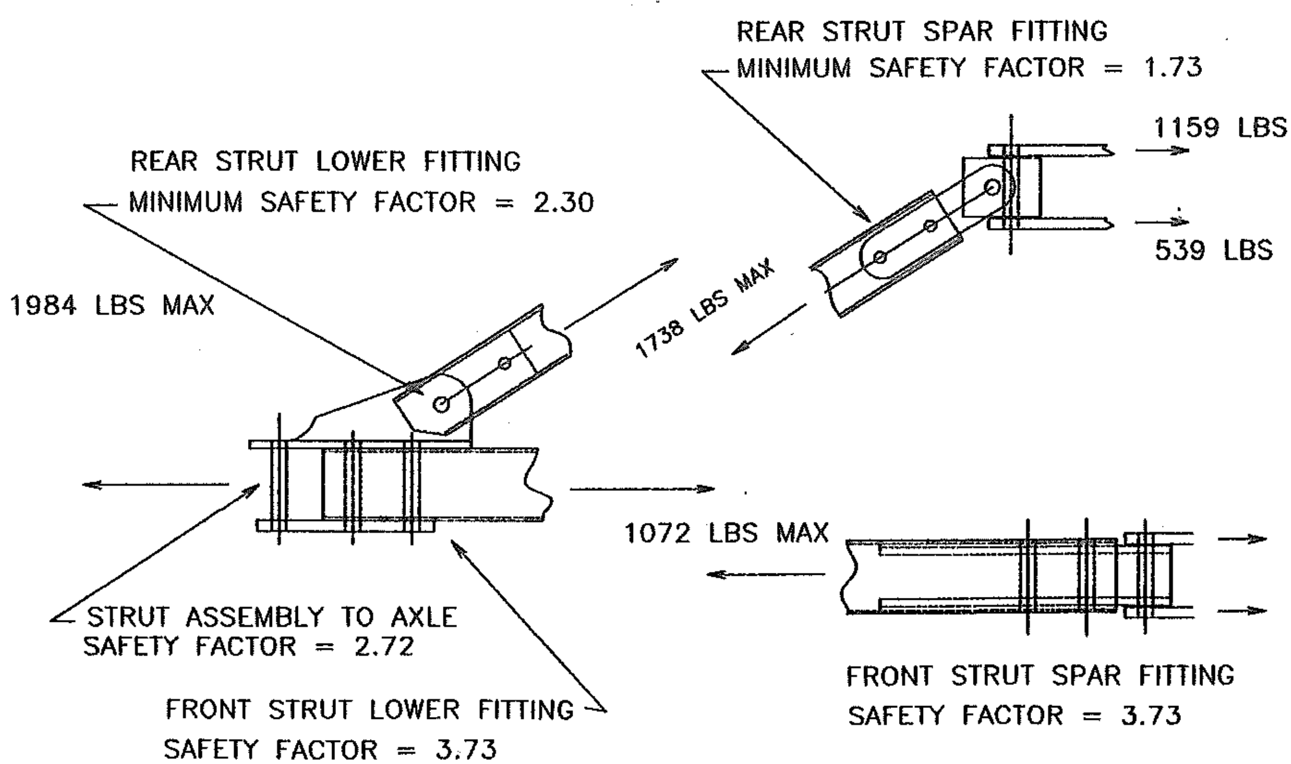
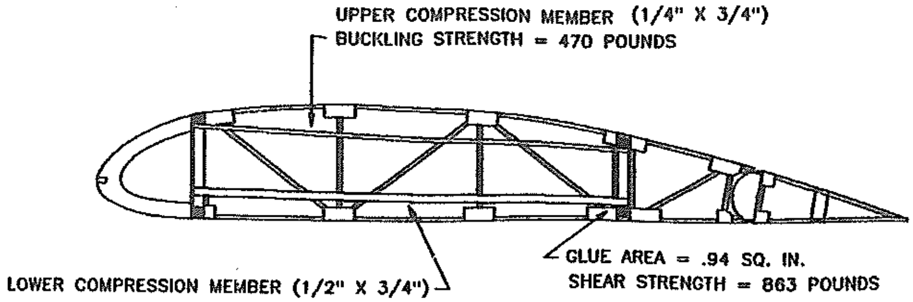
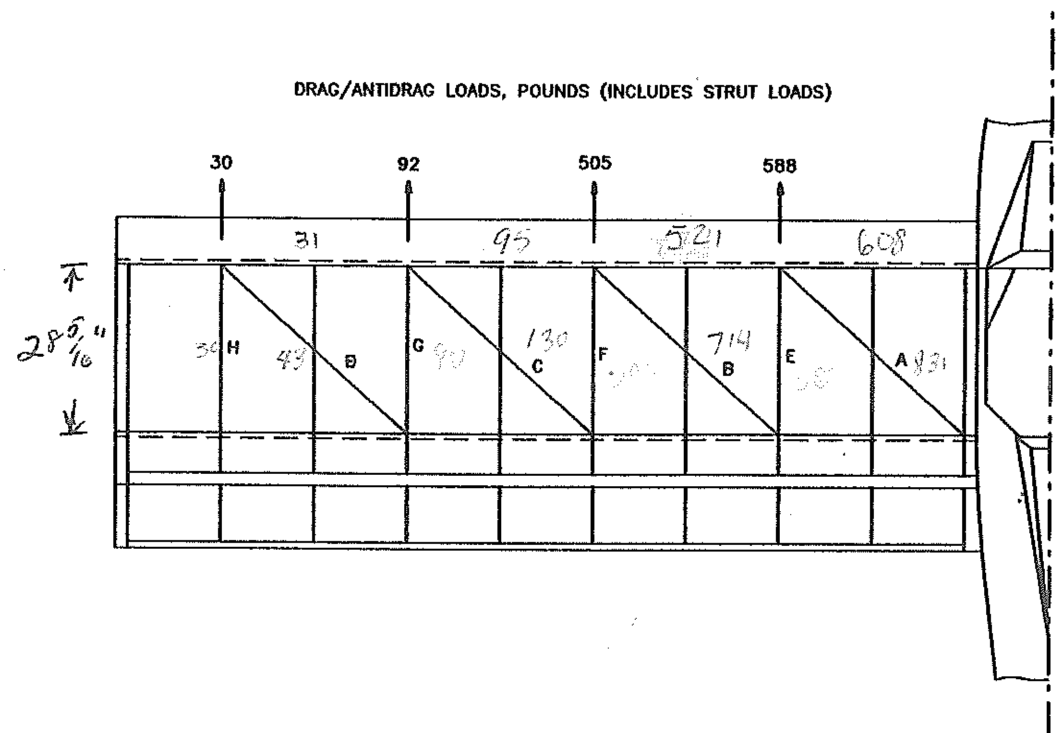


FIGURE C-4
WING AND WING STRUT FITTINGS



TOTAL COMPRESSION RIB STRENGTH = 1333 POUNDS
MAXIMUM LOAD IN RIB #3 = 588 POUNDS
SAFETY FACTOR = 2.27

TYPICAL COMPRESSION RIB



DRAG/ANTI-DRAG MEMBER	LOAD (POUNDS)	SAFETY FACTOR
A	831	1.99
B	714	2.32
C	130	12.7
D	43	38.5

COMPRESSION MEMBER	LOAD (POUNDS)	SAFETY FACTOR
E	588	2.27
F	505	2.64
G	92	14.5
H	30	44.4

FIGURE C-5

WING DRAG/ANTI-DRAG BRACING

D. HORIZONTAL STABILIZER STRUCTURE

The stabilizer spar is sized for the maximum load experienced at the design load factor of 4.4 G at the maximum dive speed V_d, with a center of gravity location at 21 percent of the chord. Bending is resisted by the spar and bracing struts. Calculated load at this condition is 220 pounds (downward). In accordance with reference 1, the stabilizer should be designed for an asymmetric load of 100 percent on one side and 50 percent on the other. In addition, the bracing geometry is such that vertical stabilizer/rudder loads are transferred to the horizontal stabilizer structure via the struts, and then to the fuselage attach points by the spar and leading edge.

This last condition applies the maximum loading to the spar. Figure D-1 shows the distribution of bending loads along with the calculated maximum allowable load (2/3 of the ultimate strength) for the condition in which the maximum symmetric download of 220 pounds is applied, along with the maximum calculated fin/rudder load of 76 pounds.

E. VERTICAL FIN STRUCTURE

The vertical tail structure is identical to that of the horizontal tail. Maximum loads are much less, therefore, no further analysis is considered necessary.

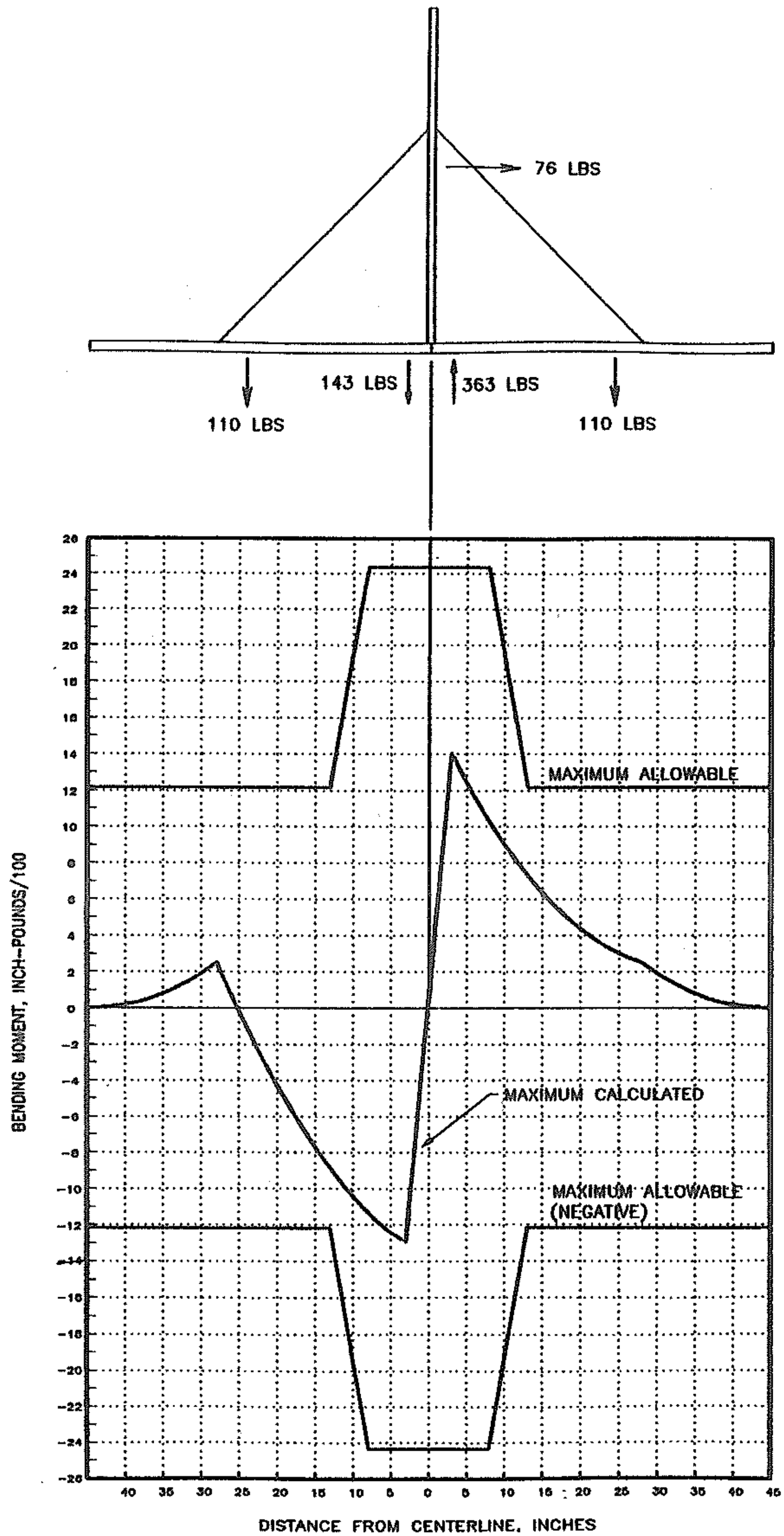
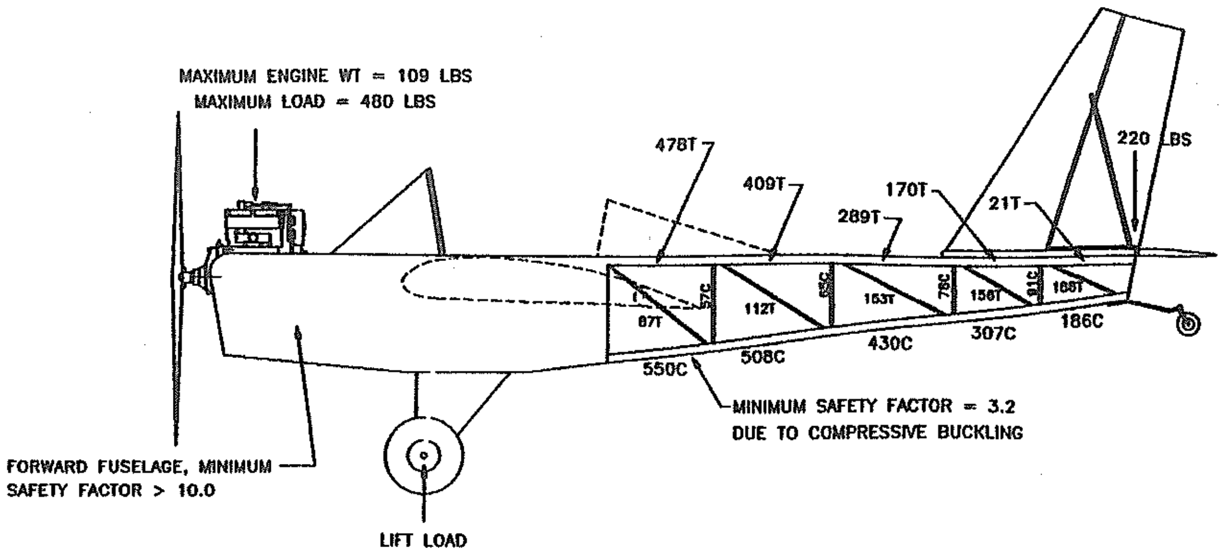


FIGURE D-1
HORIZONTAL TAIL SPAR LOADS

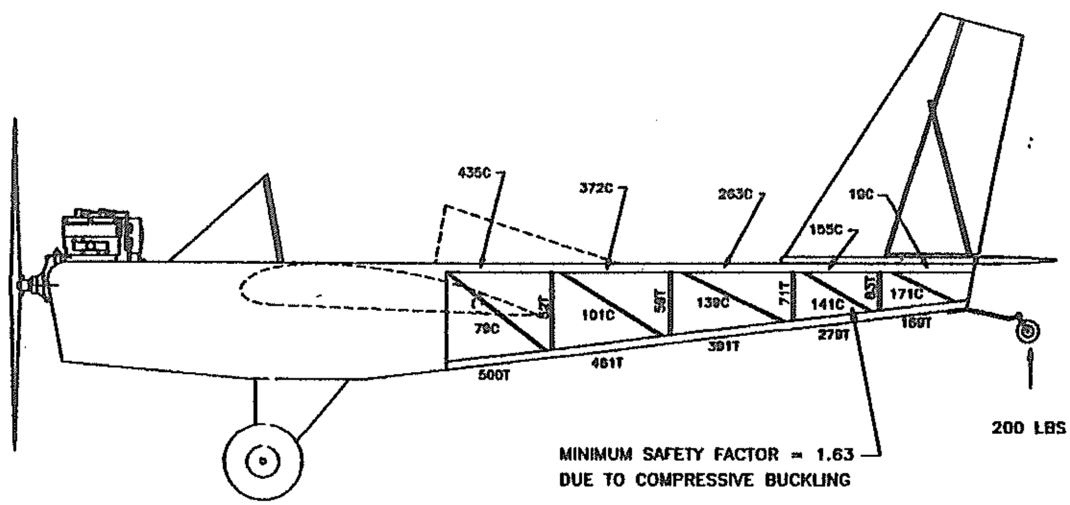
F. FUSELAGE STRUCTURE

The forward fuselage is a plywood box with pine longerons at each corner. The rear fuselage aft of the seat is a truss structure with pine longerons, uprights, and diagonals. Plywood doublers stiffen the rear longerons to resist compressive and fabric bow-in loads. Critical loads considered were the 220 pound maximum tail download at 4.4 G load factor and CG at 21 percent; tailwheel landing load of 200 pounds at aft CG; and torsion load due to the maximum rudder load of 1368 inch-pounds (which is higher than the torsion due to the asymmetrical tail load). Figure F-1 and F-2 show the calculated worst case loads and strength.

The forward wing carry through structure is shown on plans sheet 16. Maximum compressive load is 2212 pounds, applied at the wing pin joint. This results in a maximum tensile stress in the upper wood member of 5945 psi, for a safety factor of 1.56.



MAXIMUM FLIGHT LOADS, @ V_a , 4.4 G LOAD FACTOR



MAXIMUM TAILWHEEL LOAD

FIGURE F–1

FUSELAGE LAYOUT, MAXIMUM LOADS

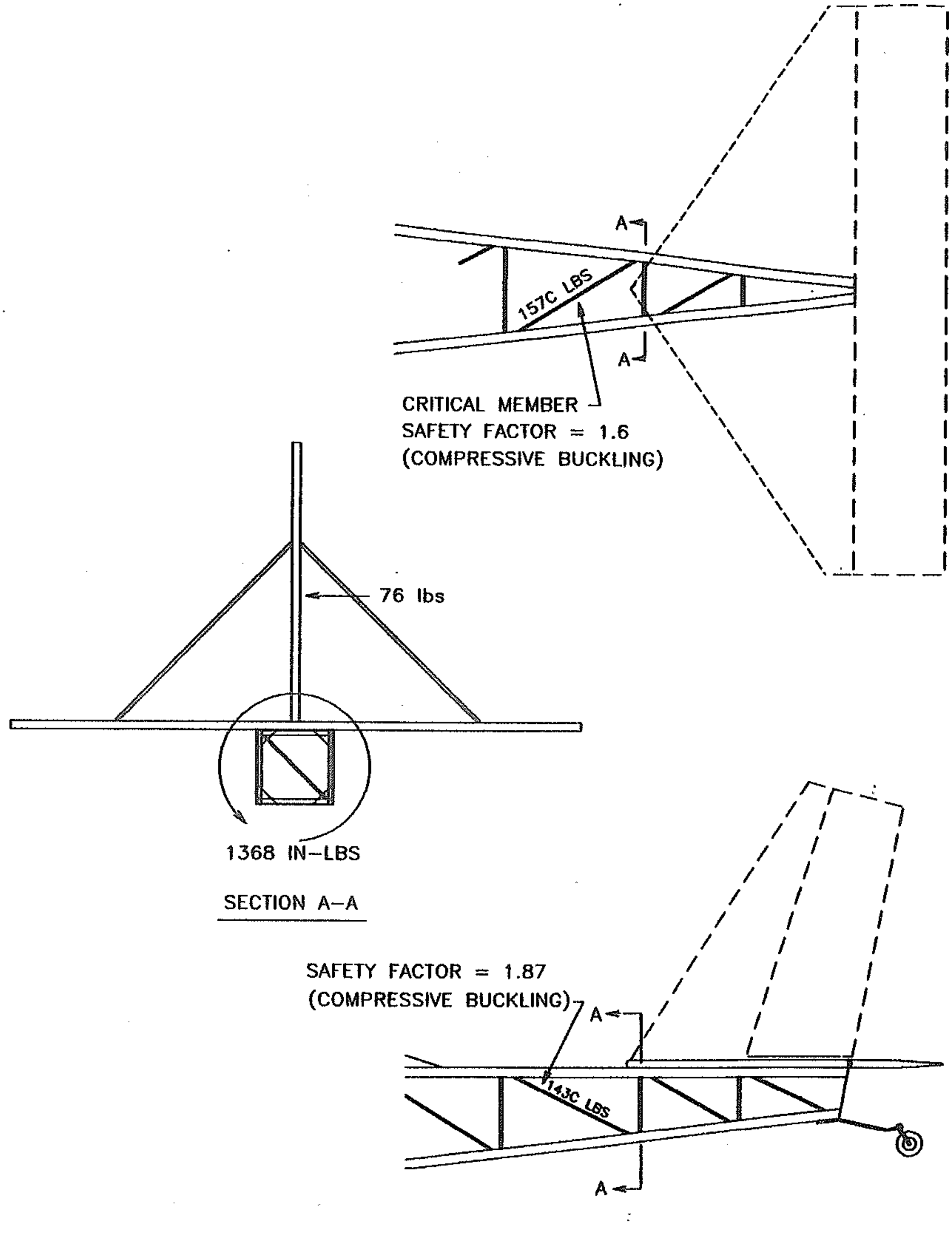


FIGURE F-2

REAR FUSELAGE TORSION LOAD

G. LANDING GEAR

The main landing gear assembly consists of built up wood landing gear legs attached to the fuselage via piano hinges (to facilitate easy removal) and a 4130-N steel axle. The primary gear structure is rigid, all shock absorbing being accomplished by the low pressure tires. The primary flight loads are supported by the gear structure via the strut attachments at the outboard end of the axle. Since the gear has been stressed for the flight conditions, which are more severe than the landing loads by far, no analysis of the gear for landing loads is deemed necessary. Figure G-1 shows the maximum landing gear component loads compared with calculated strength.

H. ENGINE MOUNTS

The engine is attached directly to the wood/plywood engine mounting bulkhead via rubber "lord" mounts, which is built integral with the forward fuselage. Loads consist of the engine weight times the design load factors and engine dynamic loads as determined from reference 1. The maximum shear load in the engine mount bulkhead is approximately 140 psi, and the bending stress in the bulkhead cross member is approximately 1710 psi.

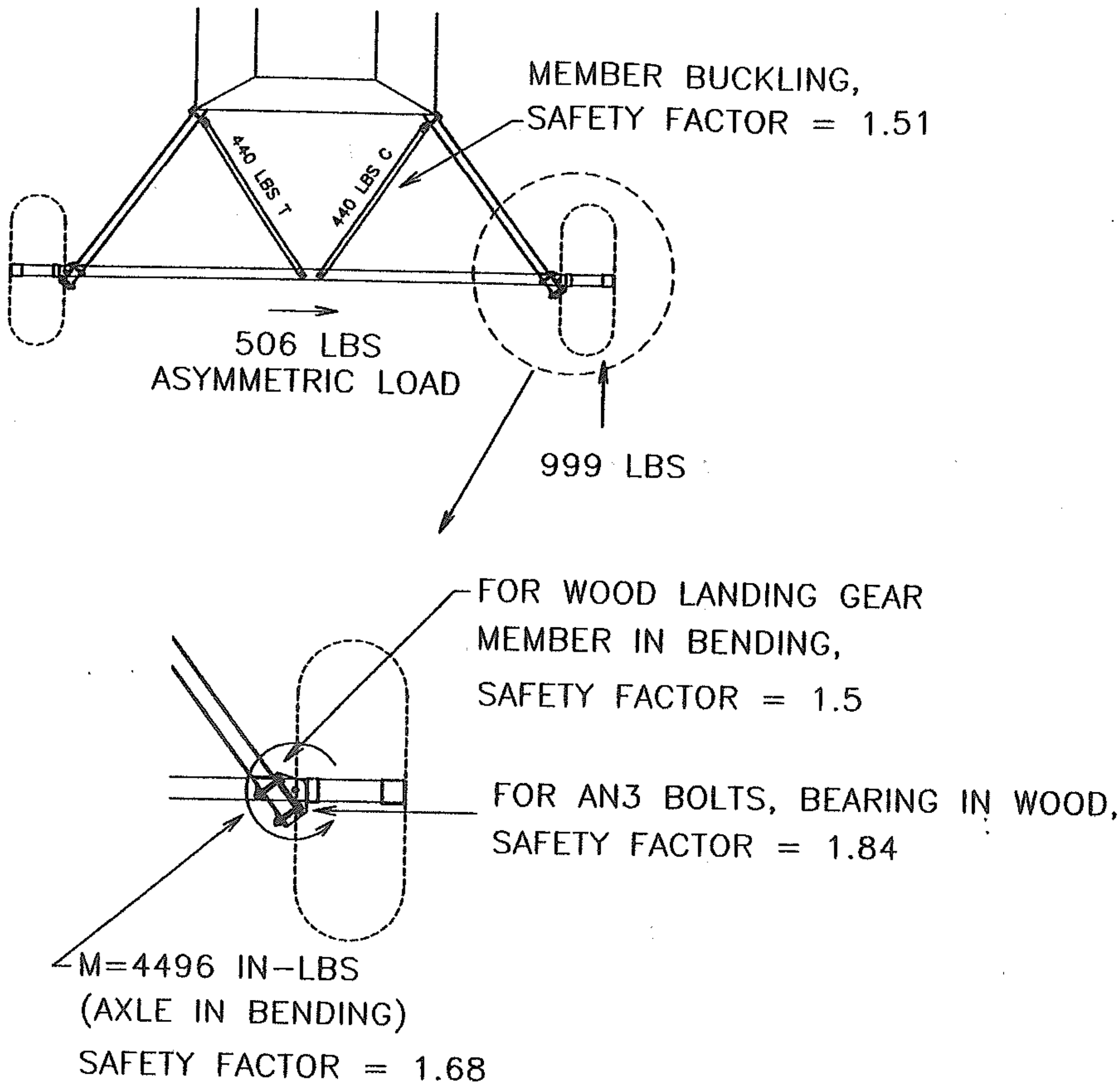


FIGURE G-1
LANDING GEAR LOADS

I. CONTROL SYSTEM

Maximum hinge moments for each control were determined from in-flight stick force measurements for the elevator and from reference 1 for the aileron and rudder loads. Loads thus determined are:

Elevator	:	207 in. lbs.
Ailerons(each)	:	253 in. lbs.
Rudder	:	167 in. lbs.

1. Elevator:

The elevator is actuated via a heavy duty push-pull armored "Morse" cable, connected directly to the control stick. The elevator horn is 2.25 inches from the hinge line to the pushrod, giving a total force (compression) of 92.6 pounds. The manufacturer's rating for the cable is 100 pounds compressive. Static tests were performed on the elevator control assembly. The cable was loaded to a total of 278 pounds where failure occurred. This corresponds to a safety factor of 3.0

2. Rudder:

The rudder and steerable tailwheel are actuated by separate 1/8 inch braided cable. Maximum calculated load in the cables is only 56 pounds, far less than the cable strength.

3. Flaperons:

The aircraft has combined full span ailerons and flaps (flaperons). The ailerons are actuated by push-pull armored cables rated at a usable load of 80 pounds compression and 100 pounds tension by the manufacturer. The aileron end of the cable housing is attached to the flap actuating arm which controls simultaneous deflection of the left and right surface to achieve flap deflection. Maximum calculated load in the cables is 73 pounds in tension. Compression loads are substantially less, since the normal aerodynamic load on the aileron tends to maintain the control cable in tension. To ensure adequate strength, a static test of the flaperon system was done. The flaperons were loaded so as to load the cables to 1.5 times the maximum calculated load of 73 pounds, while the ailerons were exercised to ensure proper operation.

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APPENDIX

STATIC TEST RESULTS

In 1986, an original MiniMAX structure was subjected to static test loading to verify the analytical stress analysis and to augment flight load testing.

Specific tests performed were:

1. Wing structure for lift and drag loads at 4.4 G positive, symmetrically loaded, at a simulated gross weight of 470 pounds.

2. Wing structure for lift and drag loads at 3.3 G positive, with the aileron deflected fully down. This represents the maximum load condition for the rear spar and strut.

3. Horizontal stabilizer, elevator, and control cable for the maximum load obtained at 4.4 G with the center of gravity at 24 percent (maximum forward).

4. Horizontal stabilizer and rear fuselage for asymmetric loading of the stabilizer.

5. Elevator control cable for ultimate failure load.

For the wing structure tests, a total of 1760 pounds of sandbags were used to simulate the airloads. For condition 1, the aircraft was inverted with the nose angled 7 degrees down relative to the horizontal to properly simulate lift and drag vectors. For condition 2, the angle was reduced to 4 degrees and the load was applied further aft on the wing to simulate the increased load due to aileron deflection. Results of these loadings were satisfactory with no failure of the properly built structure.

No other failures were experienced at design conditions. The elevator control push-pull cable was tested to failure, which occurred at greater than 3.3 times the design load.

This test, performed on the original MiniMAX structure, verified the analytical methods used in the original design, therefore, no further static tests of the improved 1991 airframe were considered necessary. As mentioned in the previous text, some tests were performed on specific components where analytical methods were considered inadequate.