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SUPERSONIC ROCKET LAUNCHER

409 629

SYSTEMS DESIGN LABORATORY
MISSILE SYSTEMS DIVISION
HUGHES AIRCRAFT COMPANY
CANOGA PARK, CALIFORNIA

MARCH 1975

FINAL REPORT: JUNE 1974 - FEBRUARY 1975

Distribution limited to U. S. Government agencies only; this report documents test and evaluation; distribution limitation applied March 1975 . Other requests for this document must be referred to the Air Force Armament Laboratory (DLDG), Eglin Air Force Base, Florida 32542.

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Supersonic Rocket Launcher
2.75-Inch Folding Fin Aircraft Rocket (FFAR)

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A prototype, low cost, lightweight, supersonic rocket launcher was designed and fabricated. The launcher carries eighteen 2.75-inch folding fin aircraft rockets and is designed to withstand the loading caused by supersonic carriage on an aircraft. The launcher features a lightweight composite structure consisting of a foam encapsulated, integrally bonded aluminum launch tube matrix in combination with a fiber reinforced epoxy laminated structural suspension system and outer skin. The launcher has three separate

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20. ABSTRACT (Continued)

and removable sections to facilitate maintenance and replacement of worn or damaged components. The forward section consists of the launch tube matrix and aerodynamic nose shape. The middle section contains the detents, electrical contacts, and intervalometer. The aft section is an aerodynamic base fairing. This program was the first attempt within the United States to fabricate a supersonic rocket launcher, and the technology developed on this program will be applicable to future developmental programs for advanced rocket launchers.

Preliminary producibility studies are included. These studies show that future development of a mass produced, low cost, lightweight, all plastic launcher appears to be obtainable at a cost which is competitive with that of current subsonic launchers such as the LAU-61/A.

PREFACE

This report documents work performed during the period from 17 June 1974 to 21 February 1975, by the Systems Design Laboratory, Missile Systems Division, Hughes Aircraft Company, Canoga Park, California, under Contract F08635-74-C-0153 with the Air Force Armament Laboratory, Armament Development and Test Center, Eglin Air Force Base, Florida. Mr. David G. Uhrig (DLDG) managed the program for the Armament Laboratory.

This report has been reviewed and is approved for publication.

FOR THE COMMANDER

A. D. Brown Jr.

ALFRED D. BROWN, JR., Colonel, USAF
Chief, Guns, Rockets and Explosives Division

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SECTION I

INTRODUCTION

During the period beginning 17 June 1974 and ending 21 February 1975, an experimental supersonic rocket launcher (SSRL) was designed and fabricated by the Hughes Aircraft Company and delivered to the USAF for ground testing. The program included design, stress analysis, materials selection, and prototype fabrication.

The experimental launcher, shown on an F-4 Phantom jet in Figure 1, carries eighteen 2.75-inch folding fin aircraft rockets (FFARs) in individual aluminum launch tubes. The launcher consists of three major sections. The forward section contains the 18 aluminum launch tubes and provides the basic aerodynamic shape and main structural integrity and strength of the launcher. The middle section contains the electrical firing circuitry and rocket retention mechanisms. The tail section of the launcher is a hollow aerodynamic fairing designed to reduce base drag.

The launcher features a lightweight composite structure consisting of a foam encapsulated, integrally bonded aluminum tube matrix in combination with a glass fiber reinforced epoxy laminated structural system and outer skin as highlighted in Figure 2.

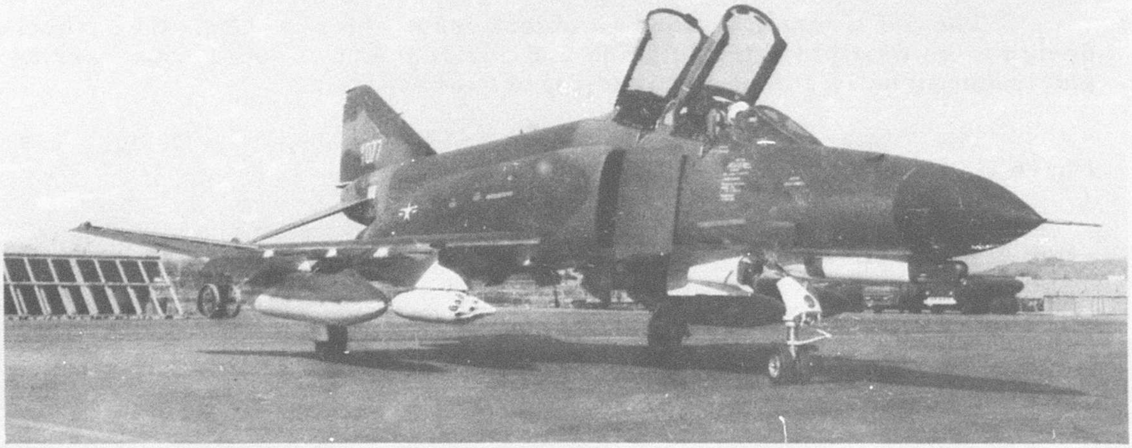


Figure 1. Supersonic Rocket Launcher Installed on F-4 Phantom Inboard Pylon

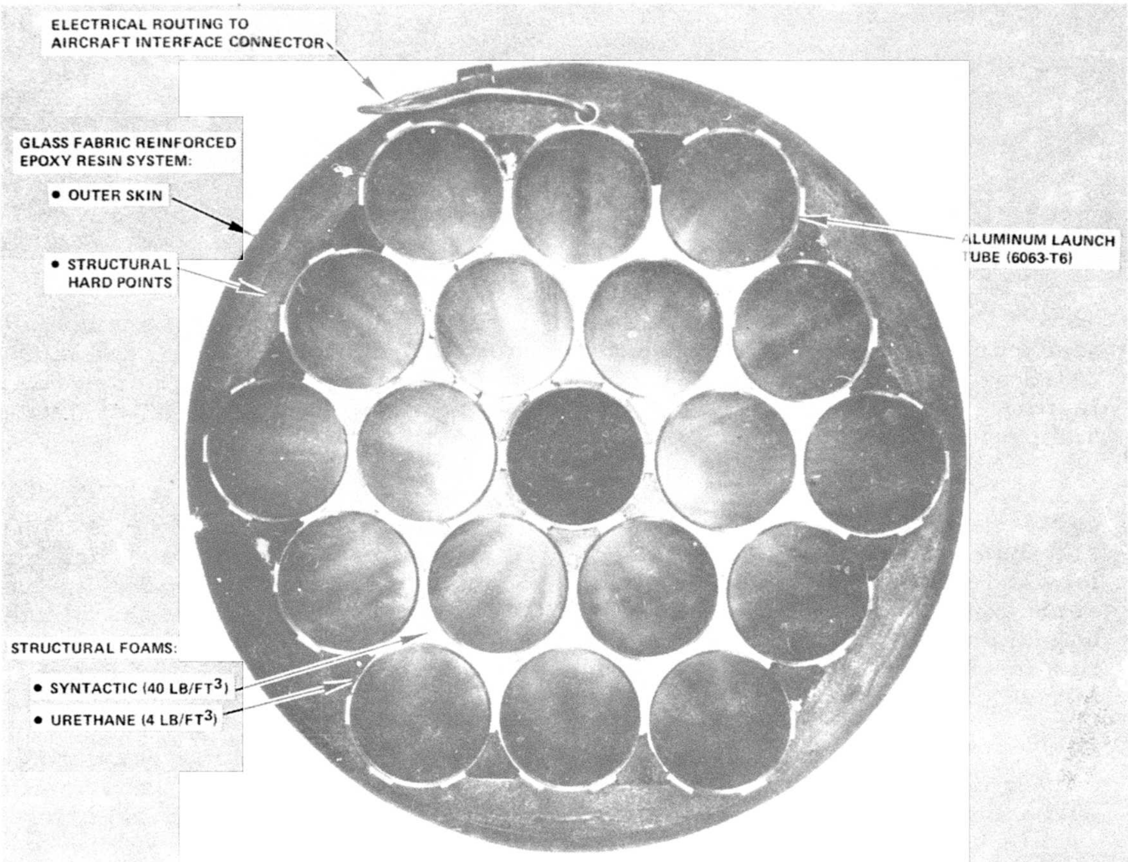


Figure 2. Supersonic Rocket Launcher Construction

The SSRL meets the design objectives of the program. The launcher design is compatible with the loads and environment of supersonic carriage and is functionally complete to facilitate ground testing.

The program was conducted in two phases: Phase I – Design, and Phase II – Fabrication.

A baseline design was developed during the design phase of the program. This activity included the establishment of a detailed design, aerodynamic and loads definition, inertial and structural analysis, and a materials tradeoff analysis and test effort to define selection parameters. This activity culminated in a design review at Eglin Air Force Base, Florida. Following a detailed review and approval cycle after the Phase I design review, the approved design was fabricated.

1) Experimental SSRL Design. A structural analysis was conducted to develop a preliminary design for the SSRL. The design was based on minimum physical properties of candidate fiber-reinforced, composite materials stressing low cost and lightweight construction.

2) Process and Material Evaluation. An initial screening program was performed to establish basic mechanical properties of candidate composite and adhesive materials. Manufacturers process recommendations and data were evaluated to obtain maximum physical properties optimizing their potential use relative to the SSRL design.

3) Material Selection. Studies were conducted to determine engineering properties of materials considered acceptable to meet program objectives. Developmental effort to verify design allowables resulting from their proposed processing was initiated. The composite material mechanical properties determined by verification testing included tensile, flexure, and bearing strengths and moduli. The adhesive materials were evaluated by lap shear measurements. The properties were obtained utilizing procedures, environments, and test conditions in a manner that would permit final selection of materials. The structural support rings and base fairing were fabricated of Ferro Corporation's CE-9000/7781 preimpregnated — partially cured and stabilized fiberglass reinforced resin system (prepreg) glass cloth per MIL-C-9084 impregnated with epoxy resin per MIL-R-9300. The launcher skin was made from CE-306/ 7781 prepreg. Two adhesive systems were selected, one for general bonding such as in process positioning and another for structural bonding of the launch tubes. These adhesive materials are a paste type EA 934 and a film type HT-424, respectively; both are qualified to MMM-A-132. Two foam systems were selected, a rigid urethane (4 lb/ft^3 density) for encapsulation of the launch tube matrix and a syntactic (40 lb/ft^3 density) per HP16-108 for fairing between tubes at the nose of the launcher and sealing at the aft end.

4) SSRL Fabrication. Following completion of the design phase the launcher was fabricated, using the selected materials and processes. The fabrication was divided into two basic elements consisting of composite materials fabrication and metal fabrication.

5) <u>Projected Design.</u> Conceptual design studies were conducted to indicate how the unit might evolve into a mass producible design. Several alternatives are available depending upon the expected technology growth of composite structures. These approaches range from a combination of aluminum and composite material to an all plastic design. One area which would significantly enhance producibility and maintainability involves a design improvement of the detent/electrical contact arrangement to eliminate a secondary bulkhead and structural joint. The basic launcher structure could be simplified a great deal with a reduction in cost. By simplifying the design, it is estimated that an SSRL can be produced which will be cost competitive with the current subsonic designs.

SECTION II

DESIGN AND CONSTRUCTION

The experimental supersonic rocket launcher (SSRL) shown in Figure 3 is capable of launching eighteen 2.75-inch folding fin aircraft rockets (FFARs) from individual tubes spaced symmetrically about a central axis. The launcher's primary structure consists of a foam encapsulated, integrally bonded aluminum launch tube matrix in combination with a fiber reinforced epoxy laminated structural suspension system and outer skin.

The launcher is an assembly of three major sections (see Figure 4); 1) an integral nose cone and tube section providing the aerodynamic shaping of the conical nose, 2) a removable base section housing the mechanical detents and the electrical firing contacts, and 3) a base section fairing to provide the necessary aerodynamic configuration required for subsonic and supersonic environments.

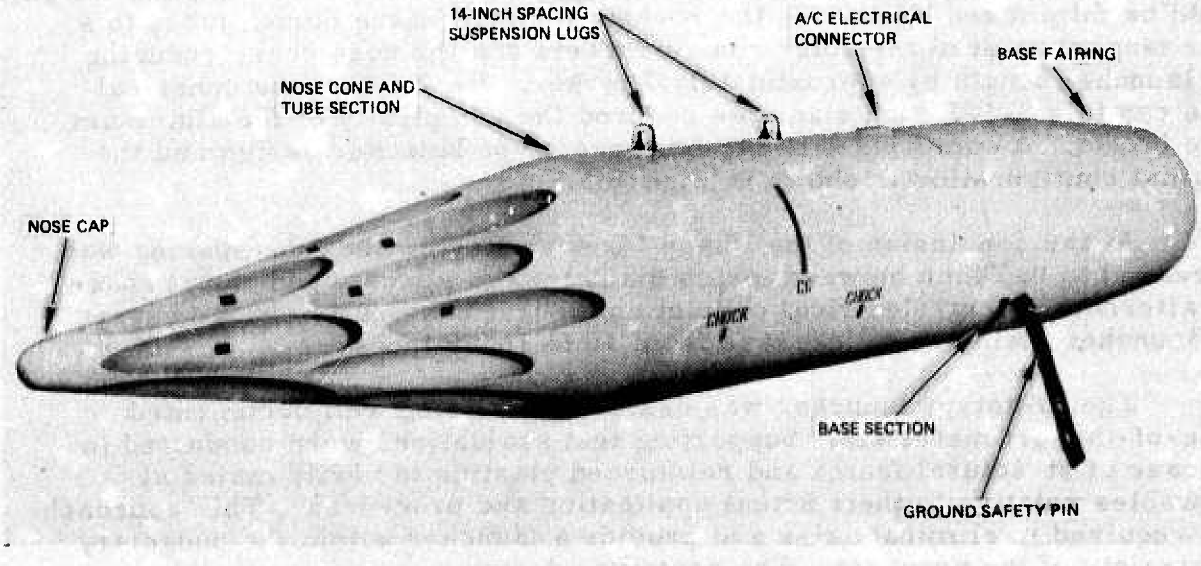


Figure 3. Experimental Rocket Launcher

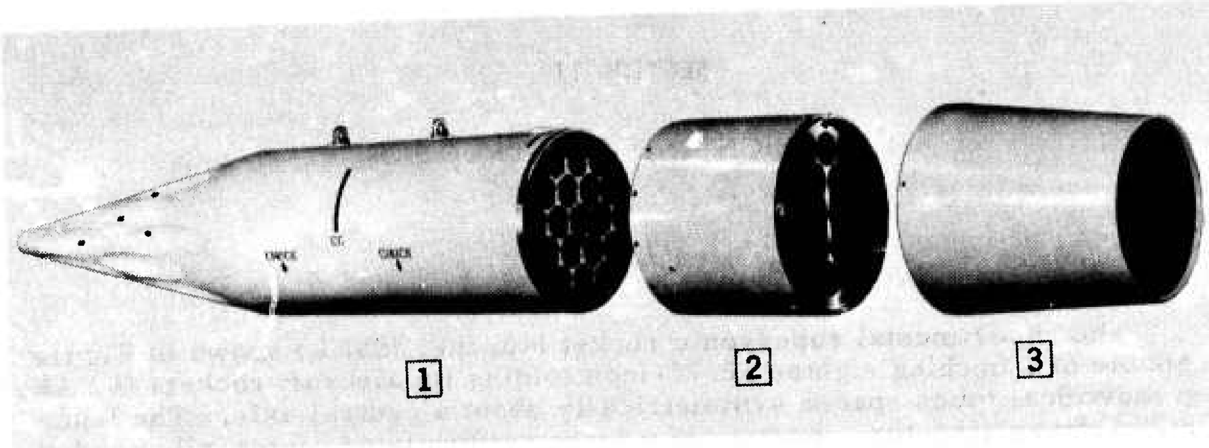


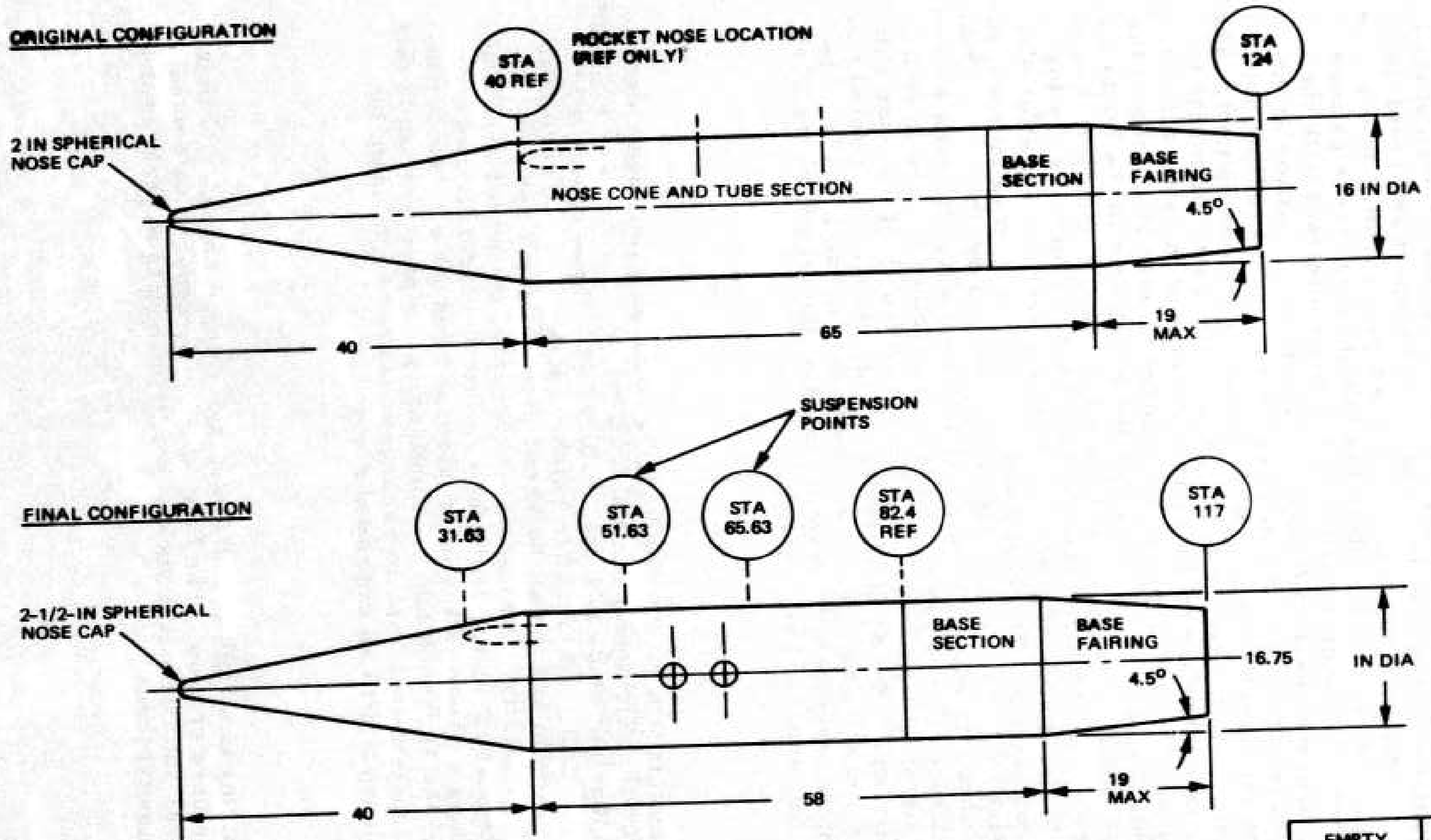
Figure 4. Launcher Sections

During the design phase, a number of alternate construction schemes were investigated. A dual approach evaluating a combination aluminum and plastic launcher as well as an all plastic launcher was pursued. The combination aluminum and plastic approach gave the greatest assurance of program success while remaining within the contract budget. The all plastic approach involved greater risk and could not be pursued within the original program budget. As a result, the aluminum-composite structure was selected.

It was determined that the launcher could be simplified somewhat from the original configuration specified in the contract. The launcher weight could be minimized by moving the rockets forward in the launch tubes to a near tangent point of the outer row of rockets and the nose cone, reducing the launcher length by approximately 7 inches. Revision of the spherical nose cap to a 2-1/2-inch diameter reduced the complexity of the aluminum nose insert. A comparison of the final prototype launcher design and the original configuration is shown in Figure 5.

At the conclusion of the Phase I Design Review the tube spacing was increased to 0.3 inch between inside diameters to allocate additional space for alternate detent/electrical contact configurations. This change caused the launcher diameter to increase from 16 to 16.75 inches.

The prototype launcher was designed, utilizing well documented state-of-the-art materials. Supporting test evaluations were conducted in the case of structural foams and reinforced plastics to verify material allowables relative to their actual application and processes. This approach was required to eliminate risk and provide a launcher within the budgetary constraints of the program. The prototype design is mass producible; however, several more competitive alternates also appear attractive for a production configuration.



	EMPTY	LOADED
WEIGHT (POUNDS) CG LOCATION LAUNCHER STAT	229 62.20	913 56.51

Figure 5. Comparison of Supersonic Rocket Launcher Original Configuration and the Final Prototype Design

NOSE CONE TUBE SECTION

Two fiberglass reinforced rings, one at each of the 14-inch spaced bomb lugs, are utilized to provide the main lateral support and react the swaybrace/lug suspension loads. The rings are tapered from 7 to 4 inches to minimize weight and essentially follow the inflight ring stress pattern around the launcher. The widest portion of the rings interface with the launch lug and swaybrace reaction footprint. The thinnest portion at the bottom of the launcher provides structural continuity and a cradling area to facilitate handling the launcher by standard ground support equipment. The rings are supported from the launch tube matrix using a molded insert to build up from the tube interface to the inside diameter of the structural rings. A machined aluminum lug well replaces the molded inserts at the top of the launcher to provide the mechanical interface for attachment of the bomb lug. Another aluminum fitting, located toward the aft end of the launcher, houses the electrical connector. An electrical conduit that interconnects the lug suspension point to the electrical connector and base section, satisfies electrical bonding requirements. The round body contour is formed by encapsulating the bonded tube matrix in a urethane foam covered by fiberglass skin. A high density syntactic foam is used for fairing and sealing the open spaces between the launch tubes to achieve the conical nose contour. The launch tube matrix consists of 19 tubes which run the entire length of the nose cone/tube section and are contoured at their forward edge to form the conical nose. The center tube is blocked by the nose cap, reducing the number of usable launch tubes to 18. A structural joint is provided on the aft end of the section for attachment of the launcher base section.

BASE SECTION

The base section is a removable housing containing the rocket detents and electrical contacts. Eighteen modified LAU-61 launch tubes are supported between two aluminum bulkheads and are enclosed by a replaceable outer fiberglass reinforced skin. The forward bulkhead joins the base section to the nose cone/tube section and the aft bulkhead provides for a bayonet-type attachment of the base fairing. An electrical connector on the forward bulkhead provides electrical continuity from the tube section aircraft connector to the 18 rocket firing contacts of the intervalometer and the electrical contacts within each of the 18 launch tubes. Configuring the base section in this manner allows alternate detent/contact arrangements to be evaluated by replacing any or all tubes with an improved design.

BASE FAIRING

The base fairing is a lightweight fiberglass reinforced epoxy hollow cone attached to the base section. The fairing is constructed to withstand the high temperatures and corrosive rocket exhaust products as well as the aerodynamic and inertial loads.

COMPARISONS WITH SUBSONIC ROCKET LAUNCHERS

In terms of design weight, the supersonic rocket launcher competes quite favorably with the current subsonic launchers. The production launcher is based on a productized version of the prototype design. The growth launcher is based on an advanced technology all plastic launcher.

The comparison of physical characteristics and configuration differences between the subsonic and supersonic rocket launchers are shown in Table 1 and Figure 6, respectively.

TABLE 1. COMPARISON OF SUPERSONIC AND SUBSONIC LAUNCHER WEIGHTS

Characteristic	LAU-61A	LAU-69A	Prototype SSRL	Production SSRL	Growth SSRL
Launcher weight, pounds	133	98	229	206	165
Number of rockets carried	19	19	18	18	18
Rocket weight, pounds	22	22	39	39	39
Total rocket weight, pounds	418	418	684	684	684
Loaded launcher weight, pounds	551	516	913	890	849
$\frac{\text{Total rocket weight}}{\text{Empty launcher weight}}$	3.1	4.3	3.0	3.3	4.1

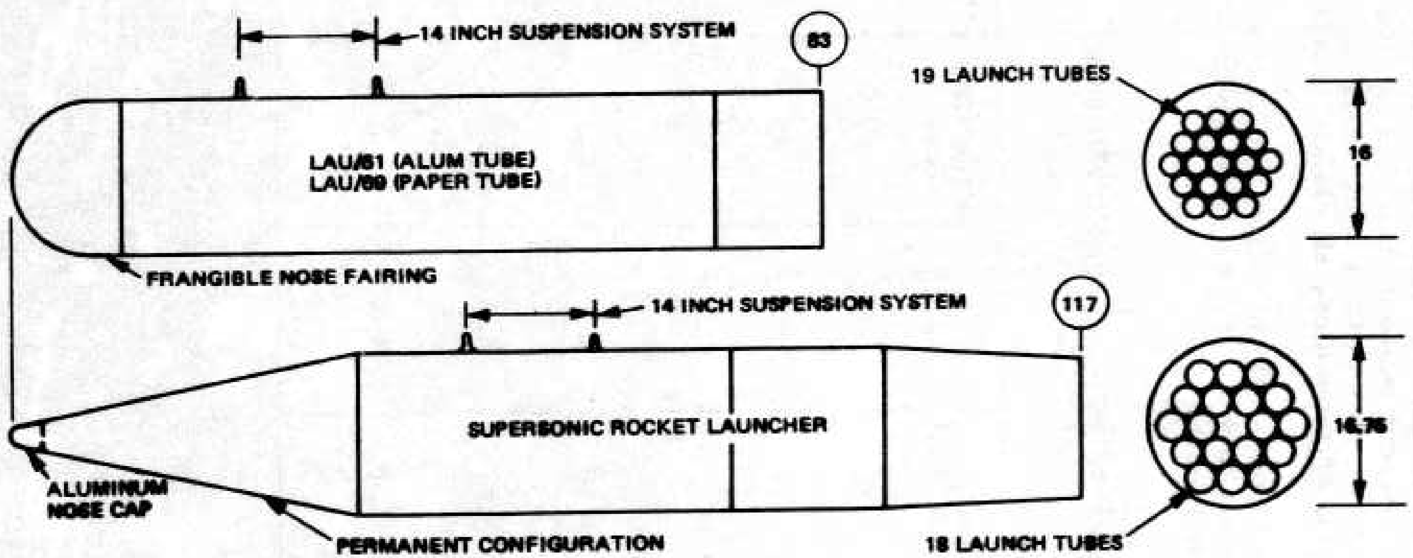


Figure 6. Comparison of Supersonic and Subsonic Rocket Launcher

SECTION III

MATERIALS SELECTION

PLASTIC COMPOSITE MATERIALS

The materials selection for the launcher was oriented to limit candidate materials to state-of-the-art, commercially available, and specification qualified materials. This approach was chosen since program goals were to demonstrate structural feasibility and manufacturing development rather than new materials application. Following this approach, material screening entailed only vendor contact and initial screening based on vendor recommendations and supplied data. Candidate materials recommended and discussed were considered first as to resin system and second as to reinforcement.

Three resin systems (polyester, phenolic, and epoxy) were considered. Polyester resin systems were eliminated since physical property retention at a stabilized 200°F temperature did not conform to structural requirements. Evaluation of phenolic resins for application at 200°F indicated the lack of processing versatility did not compensate for their property retention capability at maximum application temperature. Therefore, an epoxy resin system was selected for fabrication of the launcher.

Tradeoff studies were made for four reinforcing materials.

1) Epoxy resin/graphite fiber and cloth

2) Epoxy resin/Kevlar 49 fiber and cloth

3) Epoxy resin/"S" glass fiber and cloth

4) Epoxy resin/"E" glass fiber and cloth.

Tradeoff studies of the candidate composites comparing materials costs, manufacturing compatibility and launcher structural proper requirements narrowed those selected materials for evaluation. Table 2 is a comparison of physical properties, specific gravity, and cost of the major types of composites considered.

TABLE 2. COMPARISONS OF MAJOR CANDIDATE COMPOSITES

Property	Epoxy/ Graphite (Unidirectional Fiber)	Epoxy/ Kevlar 49 (Cloth)	Epoxy/ "S" Glass (Cloth)	Epoxy/ "E" Glass (Cloth)
Tensile strength (KSI)	110.0	72.0	185.0	71.0
Modulus (PSI x 10^6)	26.0	5.0	7.0	4.1
Flexural strength (KSI)	130.0	60.0	200.0	97.0
Modulus (PSI x 10^6)	30.0	4.0	6.0	3.8
Specific gravity	1.5	1.4	1.9	1.9
Cost ($/lb)	70-200	24-40	12-25	4-6

A summary of tradeoff studies follows.

1) Graphite reinforced epoxy. The graphite family of reinforcements, while structurally attractive, do not comply with the low cost objectives of the program.

2) Kevlar 49 reinforced epoxy. Tradeoff studies considering physical properties and weight against cost did not justify Kevlar 49 as compared to glass reinforced epoxy.

3) "S" glass reinforced epoxy. Comparisons of physical properties to cost between "S" glass and "E" glass eliminated "S" glass from further consideration.

Based upon the composite evaluations, epoxy resin/"E" glass fiber and cloth were selected.

A comparison of manufacturing methods, cost and resulting physical properties was performed. Major methods included filament winding of unidirectional fiber, layup of unidirectional fiber and glass cloth, and tape wrapping of glass cloth. Results indicated that tape wrapping of the main structural rings and layup of the skin with glass cloth were the most efficient and versatile procedures.

Vendor screening of epoxy resin/"E" glass cloth materials led to the selection of Ferro Corporation CE9000/7781 for the structural rings, aft skin, and aft fairing. Fabrication of the outer skin over the urethane foam limited the skin cure to a maximum temperature of 250°F. Based upon this requirement, Ferro Corporation FC306/7781 was chosen for the outer skin.

Selection of Ferro Corporation's materials was based on the following:

1) The materials are relatively new resins which are qualified to specifications and are in wide use in aircraft and aerospace applications.

2) Retention of mechanical properties at 200°F of these materials was excellent when processed within the parameters dictated by the launcher manufacturing requirements.

A complete material evaluation of main structural materials was performed to obtain the necessary data to establish design allowables. Table 3 summarizes the test plan parameters, and Appendix A contains a complete tabulation of test data and stress-strain curves. Also included are the -1 and -3 standard deviations and range of ultimate failures of the averaged stress-strain curves.

ADHESIVES

In an evaluation similar to that for the composites, two adhesives were selected. A film adhesive, American Cyanamid Company HT-424, was chosen to bond the tube matrix. A gap-filling paste adhesive, Shell Chemical Company EA-934, was selected for applications of required telescoping assembly, nonuniform bond thickness, and low temperature cures. Such applications included bonding of the filler blocks, structural rings, fittings, conduit, and nose cap. Adhesive selection was based on previous experience with these materials in similar structural applications.

Descriptions of the tests performed to establish design allowables for this specific application are summarized in Table 4, and Appendix B contains the tabulation of data and standard deviations.

FOAMS

Two types of foam were required to fulfill the launcher design requirements.

1) A foam to fill open areas between the tubes and to form the external configuration. This application required the ability to withstand high compression loads at 200°F and exhibit good resistance to weather. An epoxy/glass microballoon syntactic foam per Hughes Aircraft Company Specification HP16-108 was selected.

TABLE 3. FULL-SCALE EVALUATION TESTING AND DATA GENERATION FOR STRUCTURAL REINFORCED PLASTIC COMPOSITE MATERIAL

Test	No. of Specimens	Test Temperature	Test Method	Specimen Size (inches)	Remarks
Tensile	5 15	Ambient 200°F	ASTM D-638	8-1/2 x 3/4 x 0.1	Stress/strain curve
Flexure	5 15	Ambient 200°F	ASTM D-790	4 x 1 x 0.1	Stress/strain curve
Compressive	5 15	Ambient 200°F	FTMS 406 Method 1021	3 x 1/2 x 0.1	Failure stress
Bearing	5 15	Ambient 200°F	ASTM D-953	7 x 15/16 x 0.1	Failure stress

TABLE 4. FULL-SCALE EVALUATION TESTING AND DATA GENERATION FOR ADHESIVE SYSTEMS

Test	No. of Specimens	Test Temperature	Test Method	Specimen Size (inches)	Remarks
Lap Shear	5 15	Ambient 200°F	ASTM D-1002	9 x 1 x 0.1	Failure Stress

2) A foam to form the cylindrical configuration of the tube matrix and support the outer skin. Urethane Systems Corporation 230-4 and Minnesota Mining and Manufacturing (3M) CR-765 were considered, as both exceed the structural requirements at a reasonable density (4 lb/ft^3). Urethane Systems Corporation 230-4 was selected. The 3M's CR-765 required the tube matrix with the structural ring bonded in place be heated to 300-400°F. Because the thermal expansion of the aluminum tubes and the epoxy glass rings differ greatly (a factor of 3) damage to the bond integrity was possible and, therefore, the 230-4 system, which cures at 200°F, was chosen.

SECTION IV

STRESS ANALYSIS

A stress analysis was performed to verify that the launcher is structurally adequate under the critical captive flight loads. Based on the maximum ultimate design loads, positive margins have been calculated for the critical load paths in the structure.

The maximum loading condition was established after the review of the load factors and angles of attack and sideslips from MIL-A-8591C for the wing mounted store. They were:

n_x = ± 1.5

n_y = 7.5

n_z = 6.0

$\ddot{\theta}$ = ± 4 rad/sec^2

$\ddot{\psi}$ = ± 2 rad/sec^2

α_s = 19.5 degrees

β_s = 7.32 degrees

The above factors were then combined with the weight and aerodynamic distributions from Appendix Figures C-1 through C-4, and then converted to 30 lumped forces for input to the MARS program (Matrix Analysis Routine for Structure). Included were the elastic properties of the launcher structure such as bending, torsional and axial stiffnesses. Bending moments shown were then obtained and are presented in Appendix Figures C-5 and C-6. The moments were based on a total weight of 950 pounds; the suspension lugs were located at Stations 47.6 and 61.6. Ground handling was also considered in the selection of the critical condition. The worst loading condition for the launcher tube was taken as a longitudinal load of 750 pounds simulating a rocket hangfire and an internal pressure of 7 psi.

The stress analysis consisted of selecting the critical structural elements of the launcher. They were tube, outside wrap, joints at Stations 82.7, 84.6, 96.8, and 98.3, aft ring, and aft lug interface. The analysis assumed the selected materials to be homogeneous and isotropic. The calculated stresses were then compared to the allowables which were modified from the manufacturer's data sheets. A test program was conducted to confirm the mechanical properties of the selected fiberglass materials. Thermal effects have been considered in conjunction with establishing the material allowables at 190°F. This temperature is considered recovery temperature of the fiberglass launcher at M = 1.2, 5,000 feet.

The analysis indicates that the launcher assembly should be structurally adequate for the design loads and thermal environments.

SECTION V

LAUNCHER FABRICATION

NOSE CONE TUBE SECTION

The design and material selection assured fabrication with a minimum of tooling. The prototype launcher was essentially built on itself using the central axis tube as a tooling point.

The first operation in the fabrication cycle was to assemble the tube matrix from the extruded aluminum tubes. Aluminum alloy 6063-T6 was selected as the launch tube material to minimize fabrication problems and procurement time. Tolerances are less difficult to hold with an air quenched alloy and die tailoring is minimized.

The tube matrix was assembled utilizing a simple bond fixture by capitalizing on the self-nesting feature of the tube geometry and the basic hexagonal shape of the completed assembly.

Film adhesive supplied in the form of a tape was applied to the interfacing surfaces of the launch tubes as shown in Figure 7. The tubes were then stacked in the bond fixture and clamped as shown in Figures 8 and 9. Heat shrinkable tape was applied between each of the bond fixture positions and the consolidated unit cured in a conventional forced air oven for one hour to cure the adhesive system.

The molded inserts used to build up from the tube matrix to the structural ring were fabricated by a layup method. Three extruded launch tubes were used for tooling to provide the proper interface with the launch tube matrix as shown in Figure 10. After laying up the fiberglass reinforced epoxy laminate, the unit was vacuum-bag-cured at 350°F and 15 psi as shown in Figure 11. The cured inserts were then shaped and bonded to the tube matrix with the room temperature cure EA-934 adhesive.

The assembly was set up in a lathe centered on the central axis launch tube; the inserts turned to their final diameter and a rough cut taken to shape the nose as shown in Figures 12 and 13.

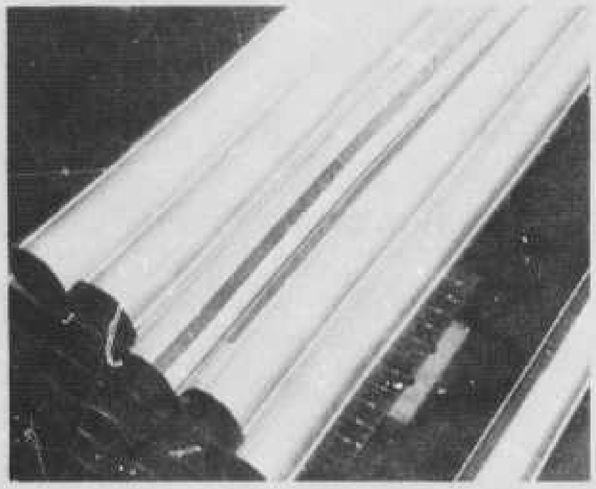


Figure 7. Application of Film Adhesive

Figure 8. Bonding of Launch Tubes

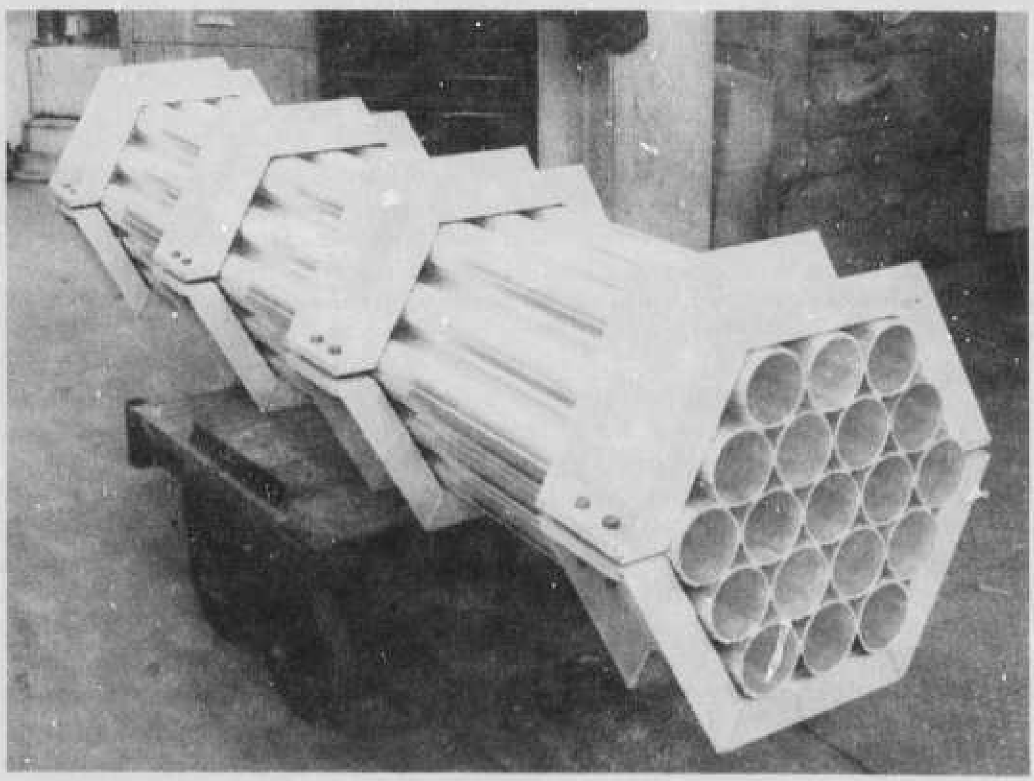


Figure 9. Launch Tube Bond Fixture

Figure 10. Insert Molding Tool

Figure 11. Insert Ready for Cure

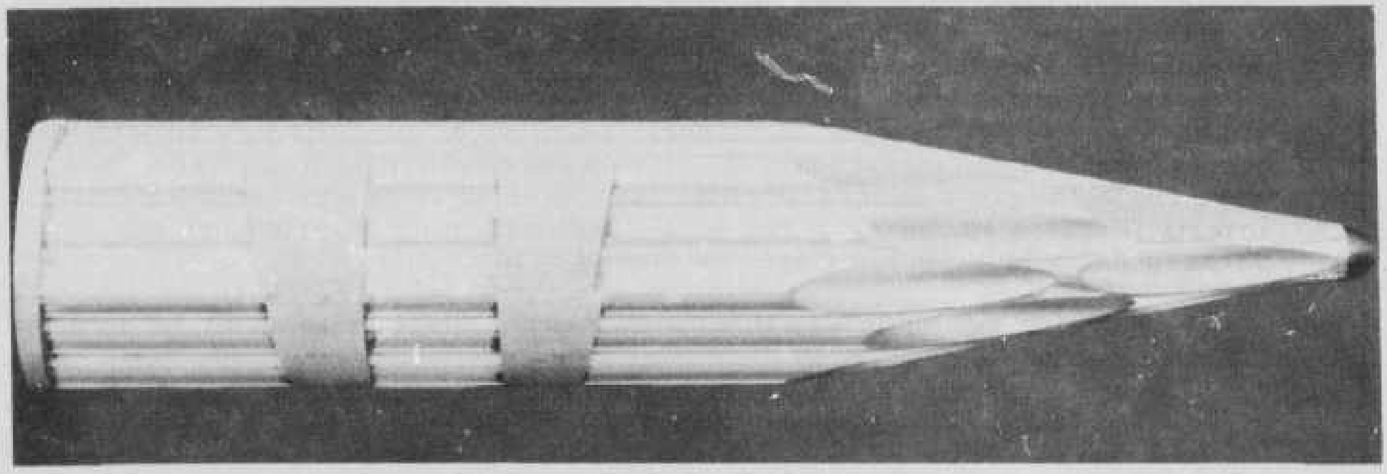


Figure 12. Machined Launch Tube Matrix

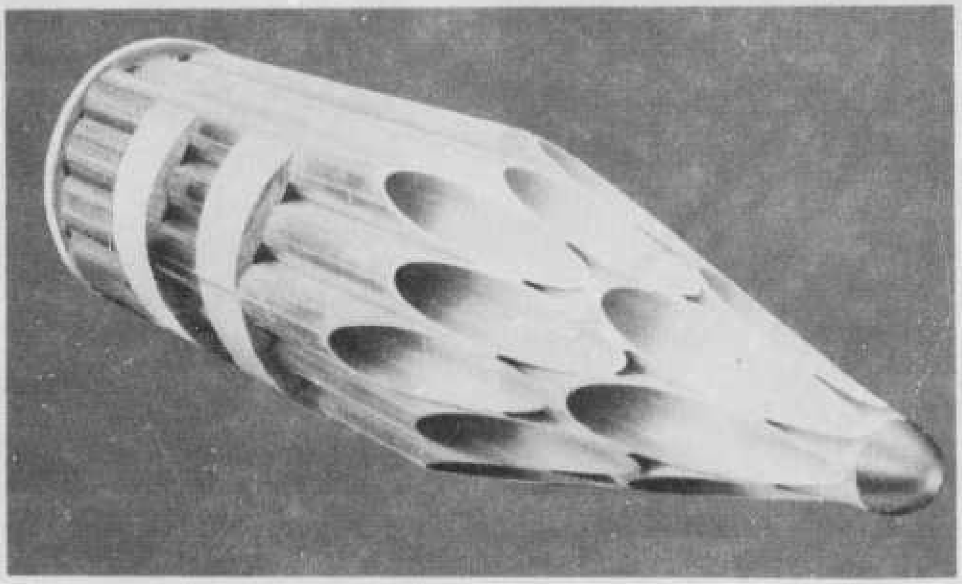


Figure 13. Machined Nose Contour

The structural rings were formed by a conventional tape wrapping operation on a mandrel shown in Figure 14, and cured in an autoclave at a temperature of 350°F and a pressure of 30 psi. Three rings were fabricated; two for suspension points and a third for the base section joint. The rings were machined to drawing specifications, as shown in Figure 15, utilizing diamond grit cutting tools to eliminate high tool wear and slow cutting speeds normally associated with the machining of glass reinforced components. A slight taper was included on the internal diameter of the rings to facilitate bonding and provide an approximate 90 percent effective bond surface. Holes were cut in the rings to accommodate the machined aluminum bomb lug inserts (Figures 16 and 17). The bomb lugs wells were inserted into the corresponding holes in the rings and bonded in place.

The ring assembly was then bonded to the tube matrix, with the bomb lug spacings held by fixturing.

The launcher contour was obtained by spraying the outside surfaces of the tube matrix with a 4 lb/ft^3 free rising urethane foam which was later overwrapped with a fiberglass epoxy laminate, using standard laminating procedures, to form a 0.07-inch thick reinforced skin. Figure 18 shows the launcher with the structural rings, inserts and urethane foam prior to installing skin.

The skin was cured at 250°F and 15 psi. The curing temperature was chosen to provide the required physical properties without thermally degrading or crushing the urethane foam.

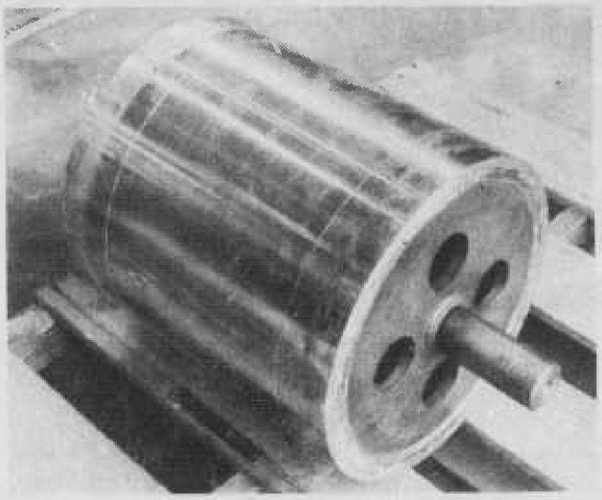


Figure 14. Structural Ring Tape Wrap Mandrel

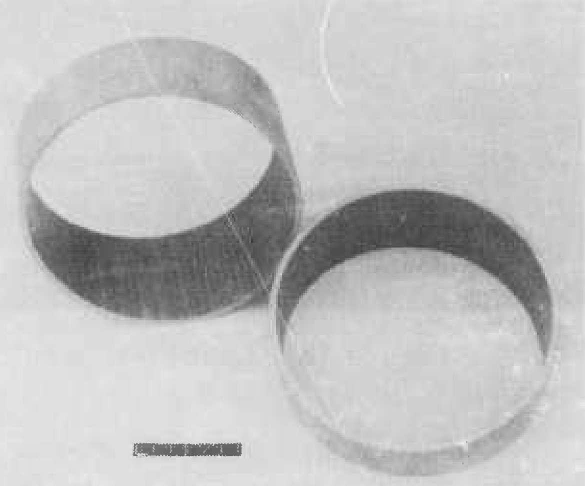


Figure 15. Structural Suspension Rings

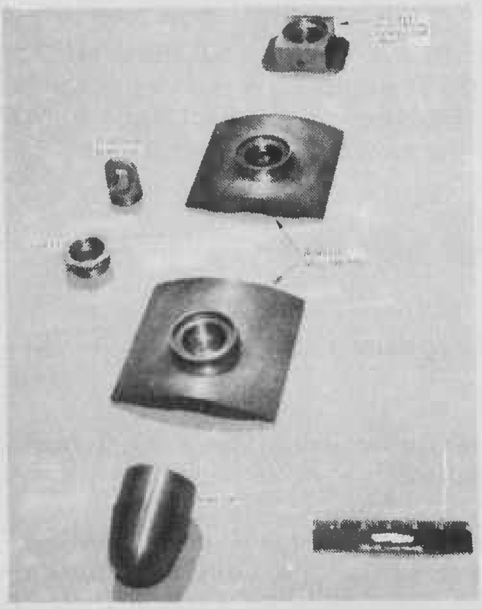


Figure 16. Machined Aluminum Inserts

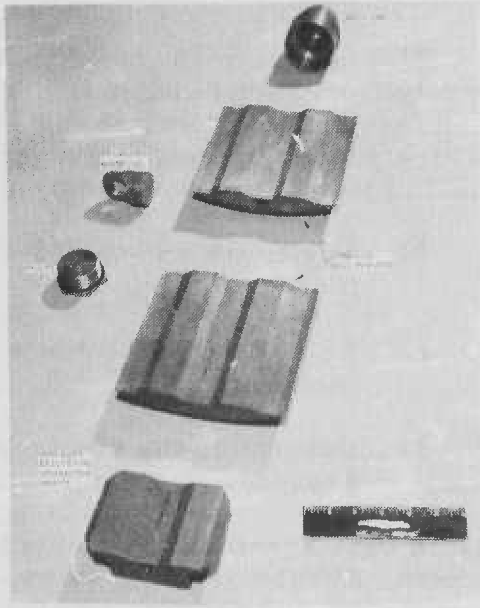


Figure 17. Machined Aluminum Inserts (Reverse Side)

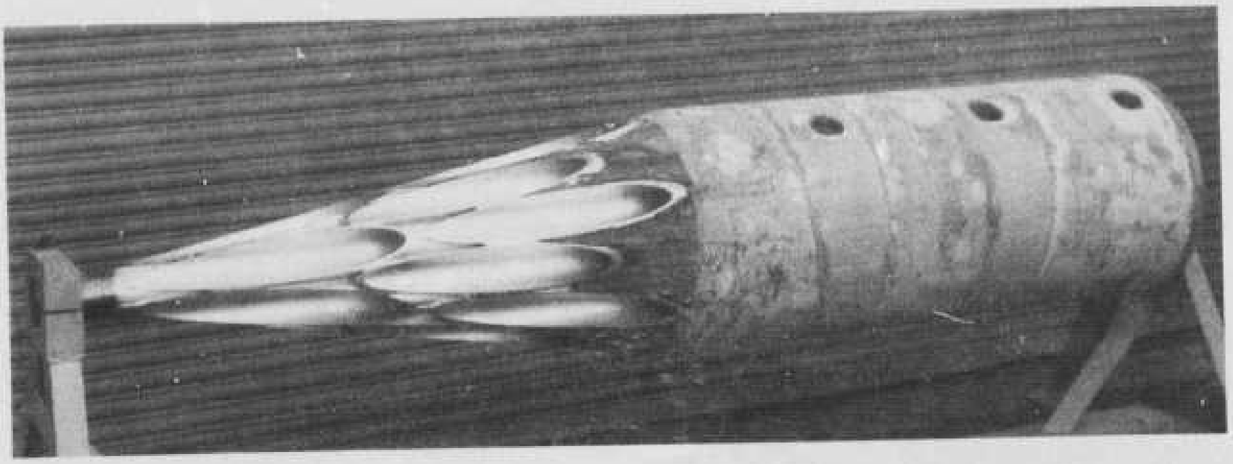


Figure 18. Launcher Nose Cone/Tube Section Subassembly

A syntactic foam was injected into the spaces between the tubes at the forward and aft end, and at the termination of the fiberglass skin around the perimeter of the tube matrix as shown in Figures 19 and 20. The outer skin was terminated about 1/4-inch from the perimeter of the tube matrix. The outside surfaces of the launcher nose cone/tube section were then faired and the sharp edges of the tubes were broken, completing the assembly.

BASE SECTION

The tube pattern was transferred from the nose cone/tube section to the two base section bulkheads. The used GFP supplied LAU-61 launchers were disassembled to utilize the mechanical detents and electrical contacts. To insure proper operation the detents and contacts were cleaned and tested. Specific objectives of these tests were to:

1) Verify the 290-to 325-pound no-release load on each detent mechanism

2) Establish a confidence level of repeatability for each detent mechanism

3) Determine the electrical resistance between the rocket motor ground/firing contact and connector.

Of the 38 tube assemblies obtained from the two GFE LAU-61/A rocket launchers, 23 tube assemblies met the objectives. The best 18 tube assemblies were cut and delivered to the shop for modifications required for final assembly in the prototype supersonic rocket launcher base section.

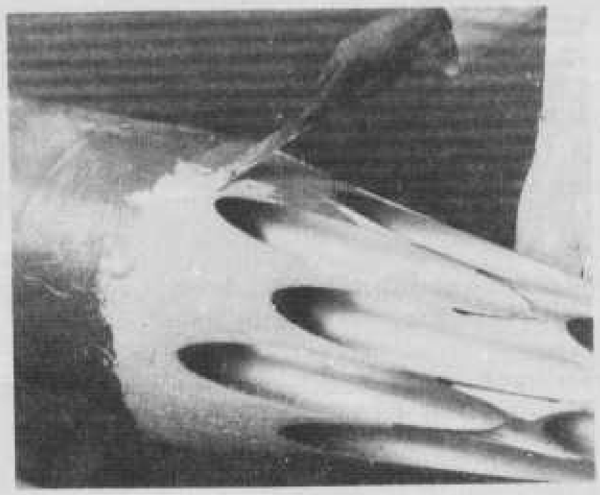


Figure 19. Fairing of Nose Contour

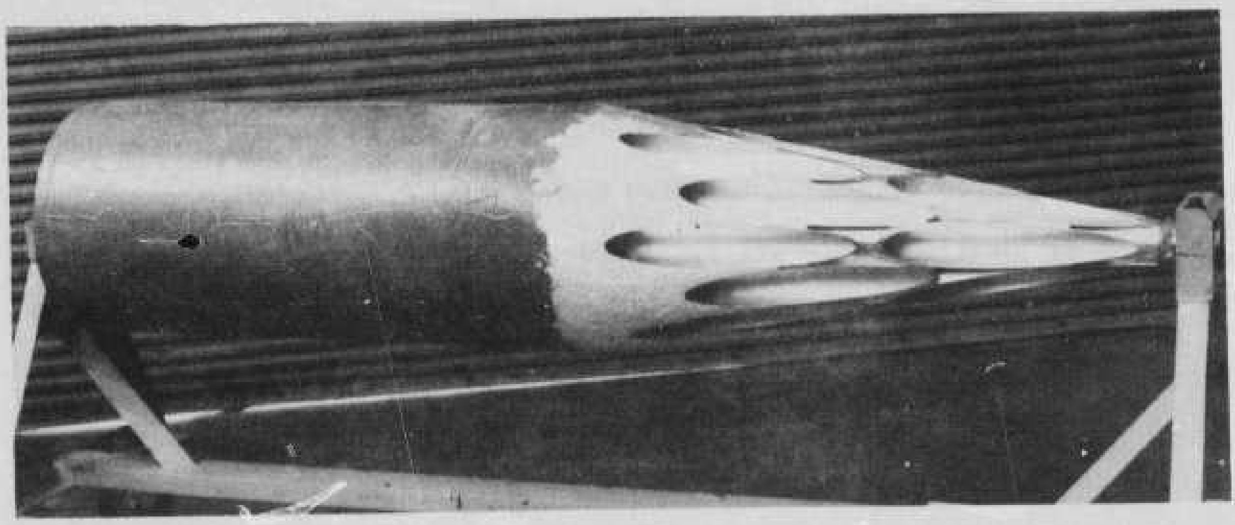


Figure 20. Launcher Nose Cone/Tube Section With Fiberglass Skin Installed

TEST ITEM DESCRIPTION

The following components were required to perform the detent pull test:

1) Two each rocket launchers, LAU-61/A, FSN V1055-065-3591 BBUA, Serial Nos. 6-92 and 6-96

2) Eighteen each inert 2.75-inch Folding Fin Aircraft Rockets, FSN 1340-00-038-8194-J103

3) Eighteen each Practice Warheads, WTU-1/B, FSN 1340-00-111-3432

4) Pneumatic test fixture, Hughes fabrication.

TEST INSTRUMENTATION AND PROCEDURE

The instrumentation is shown schematically in Figure 21 and hooked up to the test fixture as shown in Figure 22. Calibration of the test fixture was established by imposing a 125,000-ohm dummy load representing 322 pounds across the load cell. The output gain was set at 50 pounds per inch per displacement. The load cell accuracy was determined by calculating the pull force using the pneumatic pressure acting on the cylinder piston.

A detent pull test was conducted. Each rocket motor bourrelet was inspected and only those with sharp, well defined edges were used for this test. The specific test procedure, using the pneumatic test fixture assembly shown in Figure 22, is described below.

1) Load rocket in identified launch tube

2) Install and secure launch tube on fixture

3) Connect load cell

4) Run calibration check

5) Gradually increase regulator pressure

CAUTION

Do not exceed 325 pounds.

6) After pull test load has been recorded, close valve and bleed down system.

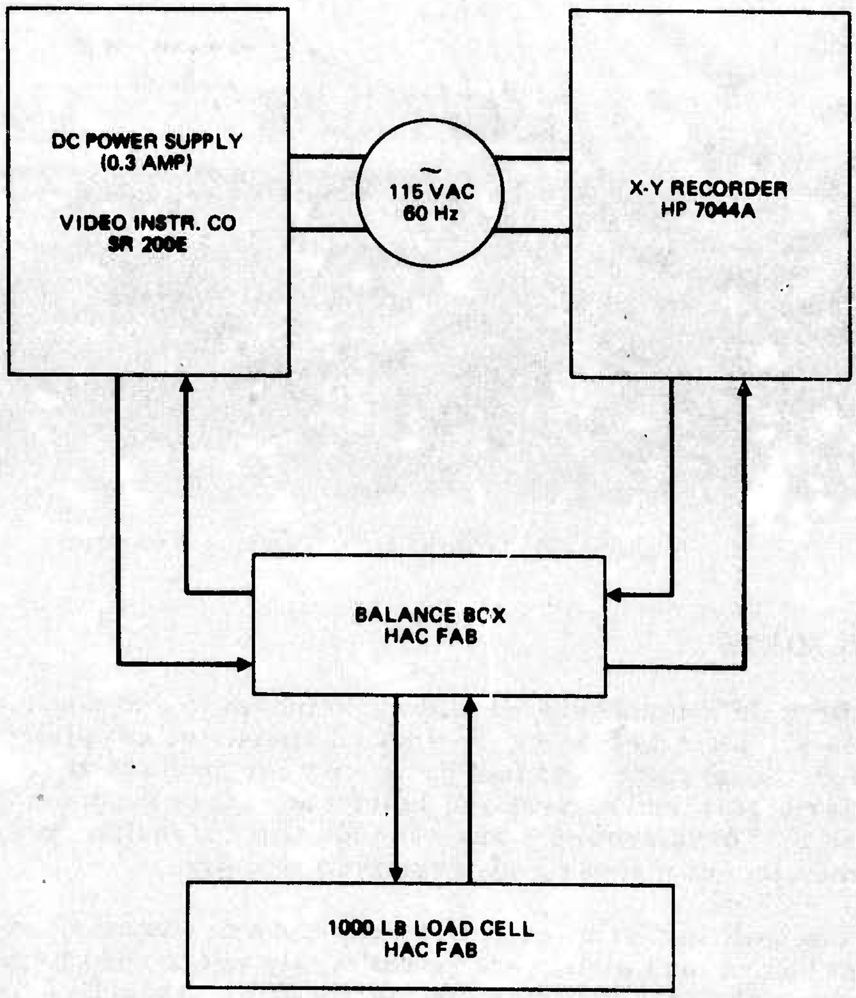


Figure 21. Instrumentation Schematic

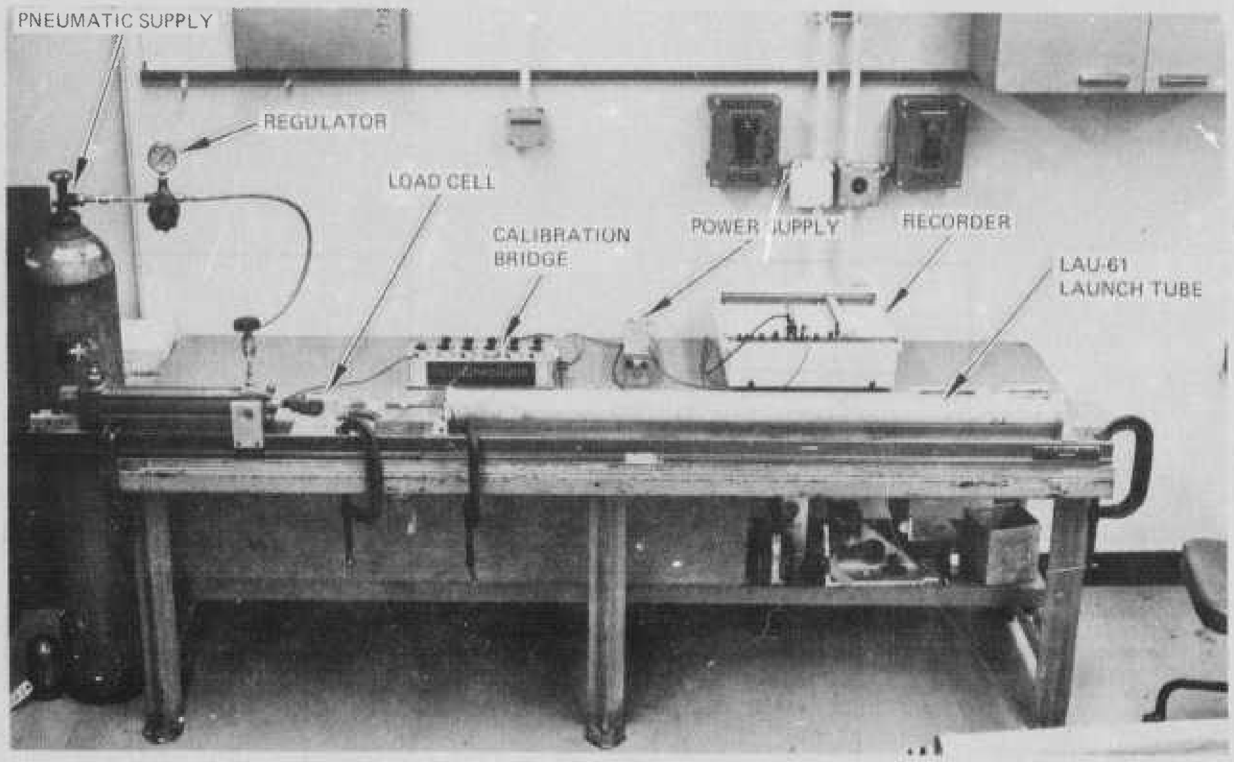


Figure 22. Pneumatic Fixture Assembly

TEST RESULTS

Both GFP launchers were disassembled to dismantle the launch tube assemblies. Launcher S/N 6-96 showed excessive corrosion on both the detent and rocket motor ground/firing contact mechanisms. The contact mechanisms required solvent and lubricant to free the moving components. Launcher S/N 6-92 showed signs of moderate corrosion; however, the detent and contact mechanisms were free to move.

The pull test data for both launchers are summarized in Table 5. The first and second tests were intentionally overloaded to determine the actual breaking force of the detent mechanism. Repeating the test on the same tube after the detent had released the rocket showed significantly lower release levels. Inspection of the detent revealed the notch had deformed to create a larger gap. As expected, all of the tube assemblies from launcher S/N 6-92 passed the detent pull test, whereas only 6 of 18 tube assemblies tested from launcher S/N 6-96 passed. One tube assembly from launcher S/N 6-96 was permanently frozen and was damaged in an attempt to loosen the contact mechanism.

TABLE 5. SUMMARY OF LAUNCH TUBE DETENT TESTING

Tube S/N	LAU-61/A (S/N 6-96) Retention Load (pounds)	Initial Release Load (pounds)	Subsequent Release Tests (pounds)	Tube S/N	LAU-61/A (A/N 6-92)** Retention Load (pounds)
1	-	390	213	19	293
2	-	356	238*	20	293
3	-	81	113	21	296
4	-	286	-	22	297
5	-	278	-	23	296
6	-	157	-	24	295
7	-	268	-	25	297
8	-	203	-	26	297
9	-	256	-	27	296
10	-	284	137	28	296
11	-	83		29	296
12	293	-		30	296
13	293	-		31	298
14	293	-		32	298
15	293	-		33	300
16	-	145		34	298
17	-	127		35	300
18	-	195		36	300
				37	299

*A third release test was conducted yielding a still lower value of 75 pounds.
**All detents passed initial tests.

DISCUSSION

The detent mechanism provides the longitudinal restraint of the loaded 2.75-inch Folding Fin Aircraft Rockets (FFAR) in the tube assembly during shipping, takeoffs, captive flights, and landings. The detent is notched to interface with the rocket motor bourrelet. It is held down in place by a leaf spring to preclude inadvertent release due to vibration or shock loads. At launch, the FFAR is released by the rocket motor thrust overriding the detent restraining force.

The overload tests (Nos. 1 and 2) served two purposes: 1) to gain further insight as to the function of the detent mechanism, and 2) to provide a known failure mode for analysis. The primary cause of failure (the inability to sustain a 290-pound load) is the amount of clearance between the detent and the rocket motor bourrelet. Those detents with large clearances (excessive notch wear) had little or no chance of meeting the load requirements. These detents with small clearances (minimum notch wear) were expected to meet the load requirement, provided well defined rocket motor bourrelets (no rounded edges) were used in the test.

After Test No. 1, the tube assembly was cut to provide access to the detent mechanism. The detent material had undergone permanent deformation such that the notch had elongated until only a small ridge was left. This ridge was unable to restrain the FFAR as the load was applied.

Although new motor contact assemblies and detent mechanisms would be preferred for use in the prototype base section, the 18 tube segments thus salvaged should function satisfactorily in the supersonic rocket launcher until such time as an improved base section is available.

The modified tubes were installed and electrical connections completed and checked out through the interface connector, intervalometer, safe arm device and electrical detents and the outer skin installed between bulkheads as shown in Figure 23. The electrical schematic is shown in Figure 24.

BASE SKIN AND AFT FAIRING

The base skin and aft fairing were fabricated using the CE 9000/7781 material. Processing was accomplished by standard hand-layup techniques on steel mandrels machined to the internal configurations of the fairings. Each ply was terminated with a splice joint of 1/2-inch minimum. Also, splices were staggered to assure no joints coincide in location sacrificing structural integrity.

The layup was wrapped with two layers of Mylar® shrink tape to provide pressure required for laminating. Curing was accomplished in forced air ovens at 350°F for 4 hours. The cured parts required a minimum of machining. Machining operations involved only turning to length on the base skin. The aft fairing required turning to length, machining of the outside diameter at the section equal to the body dimension, and machining of the undercut bayonet locking feature. The major outside surfaces required only light sanding to assure the proper surface for painting.

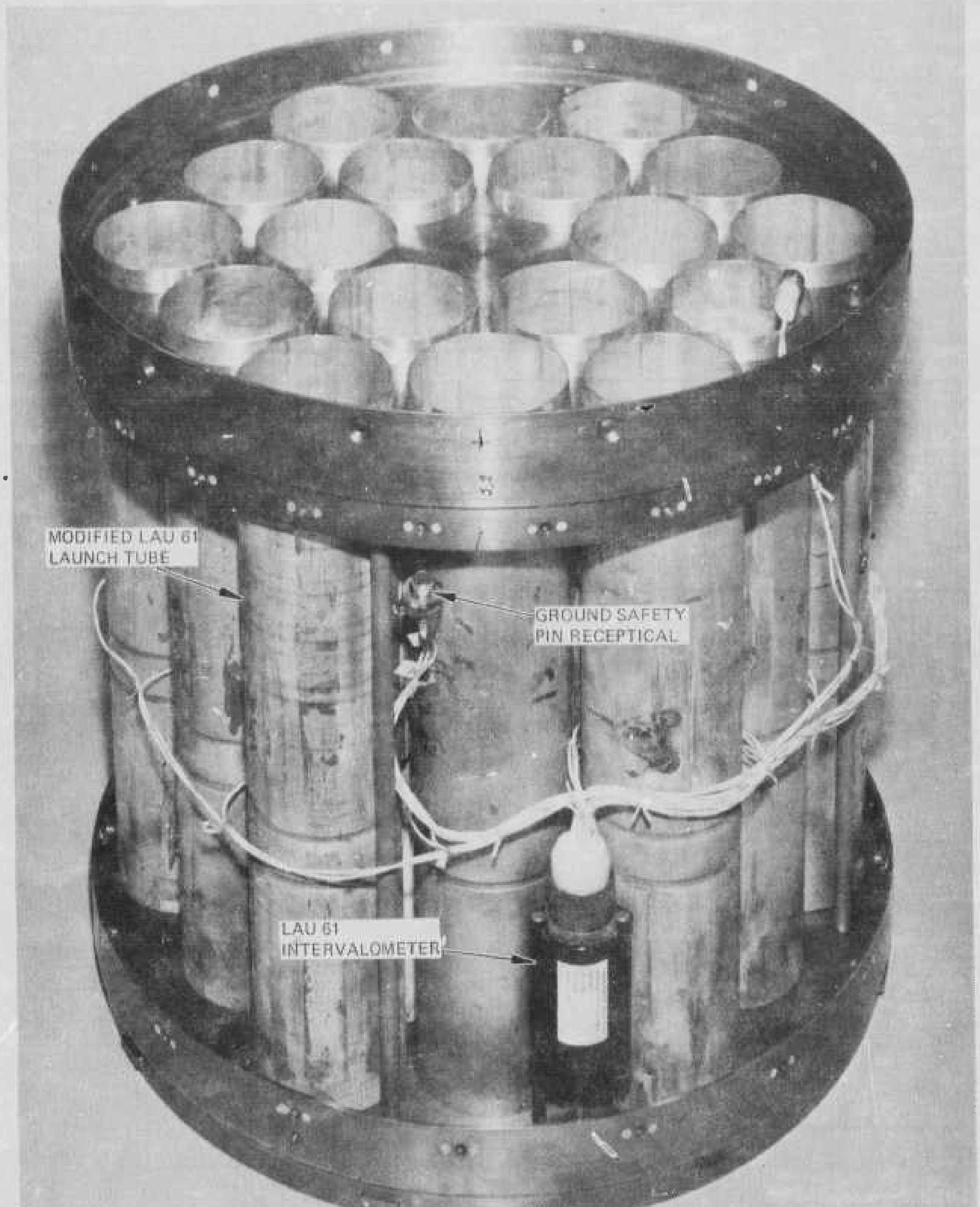
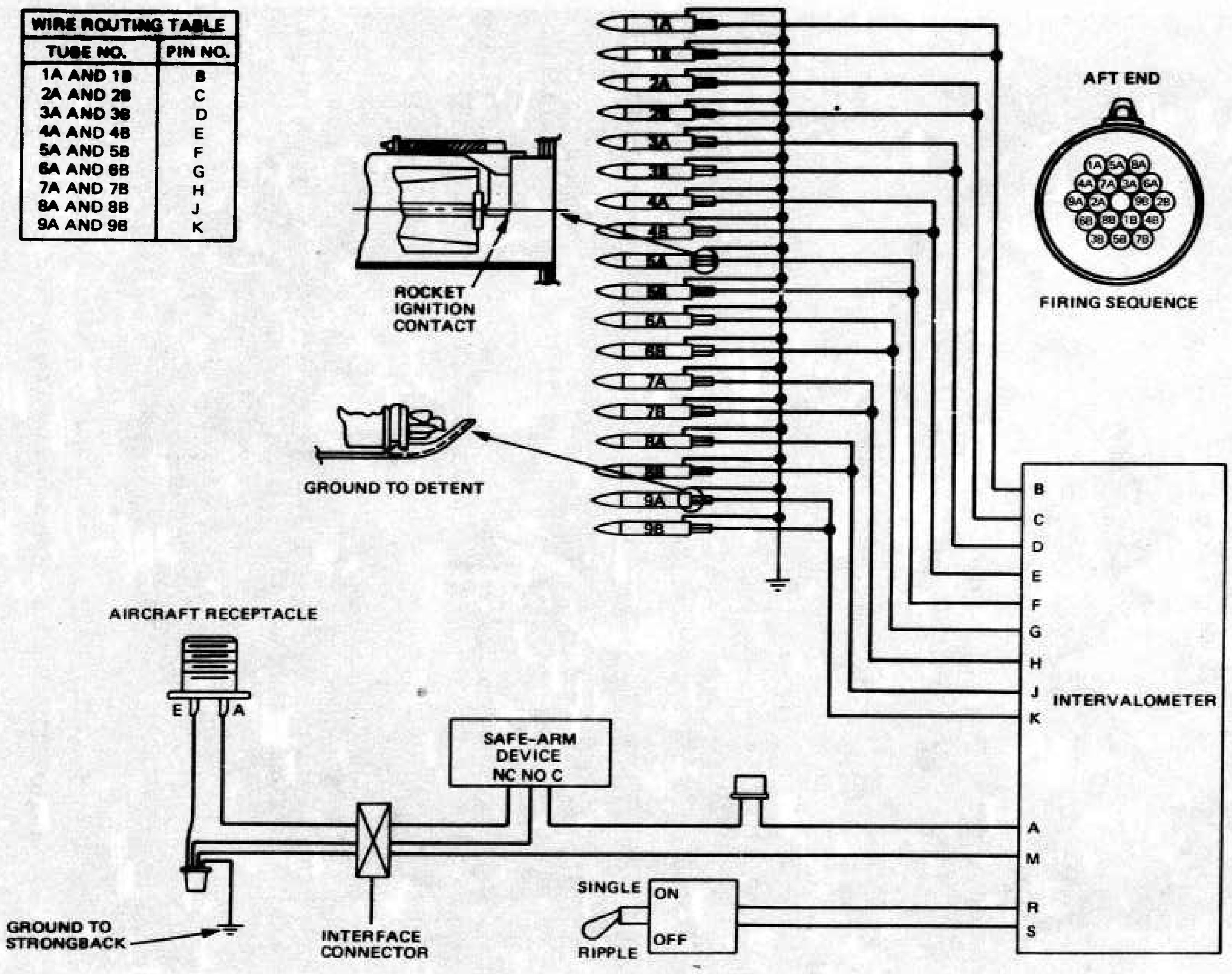


Figure 23. Base Section Assembly Less Fiberglass Skin



WIRE ROUTING TABLE	
TUBE NO.	PIN NO.
1A AND 1B	B
2A AND 2B	C
3A AND 3B	D
4A AND 4B	E
5A AND 5B	F
6A AND 6B	G
7A AND 7B	H
8A AND 8B	J
9A AND 9B	K

Figure 24. Supersonic Experimental Launcher Electrical System

SECTION VI

PROJECTED DESIGN

Several alternative construction schemes were identified during the course of this program. These approaches ranged from a combination of metal and composite, similar to the prototype, to an all-plastic structure. The most producible design approach in each case would include an improvement/ simplification of the rocket detenting and electrical mechanisms.

Based on the history of subsonic rocket launchers, the total quantity of parts produced is worthy of a significant nonrecurring tooling expenditure. Hughes experience demonstrates that total quantity rather than rate determines the quality of tooling. This type of approach opens the door to some rather new technology oriented fabrication techniques which would yield the lowest possible unit production price.

The first approach involves a systematic replacement of unit components with more weight efficient and lower cost structures. The second approach toward minimizing unit costs involves a conceptual design change.

The current launcher tube design is manufacturing limited rather than stress limited. A pultruded fiberglass tube would replace the extruded aluminum tube used in the prototype launcher. The tube could be a combination of braided and longitudinal fibers to increase the load carrying capacity. The weight of the launcher would be reduced by direct replacement with the less dense fiber reinforced material. Further reduction in weight could be accomplished by reducing the tube wall thickness to a minimum. Launcher fabrication cost could be still further reduced by the addition of an adhesive installation step directly in the pultrusion process eliminating subsequent cleaning and processing operations.

COMPOSITE SUSPENSION INSERT

To minimize weight and cost, the aluminum bomb lug well and swaybrace support inserts could be replaced with a molded composite material. Recent successful experience with a material used in a similar fashion was gained during a program at Hughes Fullerton, and indicates suitability of this approach.

AERODYNAMIC SKIN

Several tradeoffs remain to be conducted relative to final selection of the technique for applying the aerodynamic skin. The skin can be manufactured using a number of processes as indicated:

1) Lay-up

2) Filament wound

3) Tape wrap

4) Braided

5) Chopper spray-up

6) Molded preform

The selection of a particular system is dependent on the interrelationship between the launcher inner structure, cost, and environmental factors.

STRUCTURAL FOAMS

The feasibility of using a predominantly structural foam launcher should be explored more completely. Use of a structural foam of either the self-skinning or thermoplastic type alone or in combinations with the previously described approaches could significantly reduce the labor content associated with manufacturing the SSRL.

COST ANALYSIS

An objective of this program was to accumulate producibility data relative to a production launcher. Two approaches are offered; one based on the design of the prototype launcher, the other based on a technology oriented all plastic design.

The aluminum composite construction technique was selected as the estimating baseline, since the prototype launcher is the most completely defined. The baseline launcher is essentially an extension of the processes and materials used in fabricating the prototype as productized for mass production. Table 6 summarizes the budgetary manufacturing costs as a function of unit quantity which was estimated by the Hughes Aircraft Company, Tucson Manufacturing facility. Each column entry is a stand-alone quotation. Differences can be used as delta costs if previously implemented for the lower rate. The estimates are considered to be conservative.

TABLE 6. SSRL COST SUMMARY (1975 DOLLARS)

	Total Quantity (units)		
	10,000	20,000	100,000
Delivery rate (units per year)	1,200	2,400	6,000
Implementation $	508,000	819,000	1,451,000
Tooling (total) $	120,000	200,000	280,000
Material (unit) $	2,214	2,093	1,937
Labor (unit) $	1,118	825	556
Total manufacturing cost per unit $	3,332	2,918	2,493

A study was conducted to relate the cost of the SSRL to the current subsonic rocket launcher. Although the subsonic rocket launcher, LAU-61, has been out of production for approximately 4 years, manufacturing cost estimates in the current market range from $1000 to $1100 per unit.

The projected unit costs of the supersonic rocket launcher can be reduced considerably by further advancement in materials technology. Improvement/simplification of the rocket detenting and electrical mechanisms will have a significant positive effect on reducing unit costs. Preliminary estimates, based on an "all plastic" nose cone/tube section and improvement of the rocket detenting and electrical mechanisms, indicate that a unit cost of less than $1200 is obtainable.

In conclusion, the results of the preliminary producibility studies show that future development of a mass produced, low cost, lightweight, all plastic launcher appears to be obtainable at a cost which is competitive with that of current subsonic launchers such as the LAU-61/A.

SECTION VII

CONCLUSIONS AND RECOMMENDATIONS

The objective of the program was to design and fabricate a prototype supersonic rocket launcher while demonstrating the suitability of low cost, composite materials. Accomplishments were as follows:

- Demonstrated the ability to design an efficient composite SSRL structure utilizing low cost materials.
- Fabricated a functional prototype suitable for ground testing.
- Identified construction techniques which will yield a low cost, lightweight SSRL.

As a result of conducting this program, recommendations as to future work are proposed in the following areas:

- Explore alternate detent/electrical contact concepts to improve reliability and reduce system complexity.
- Evaluate new state-of-the-art materials such as structural foams to further reduce costs.
- Fabricate an all-plastic launcher.
- Conduct a structural and functional test program.

APPENDIX A

SUMMARY OF REINFORCED PLASTIC COMPOSITE EVALUATION

TABLE A-1. SUMMARY OF TENSILE PROPERTIES TESTING

Material: Ferro Corporation CE 9000/7781

Test Method: ASTM D-638

Average Specimen Dimensions:

ASTM D-638, Type I
Gauge: 2 inches
Gauge cross section: 0.1055 x 0.4843

Crosshead Speed: 0.05 inch/minute

Conditioning: 1/2 hour at test temperature

Specimen Number	Tensile Strength (KSI)		Tensile Modulus (PSI x 10^6)	
	74°F	200°F	74°F	200°F
1	57.2	55.2	4.8	3.8
2	59.1	50.3	3.8	3.9
3	65.7	48.4	4.2	4.0
4	61.7	48.8	4.0	4.2
5	60.4	52.8	4.1	4.0
6		56.7		3.9
7		54.9		3.8
8		50.8		3.7
9		51.6		3.8
10		50.0		3.8
11		49.7		3.6
12		54.9		3.8
13		55.0		4.3
14		56.1		3.8
15		54.7		4.0
Average	60.8	51.9	4.2	3.9
-1 S.D.	3.2	3.1	0.4	0.18
-2 S.D.	9.6	9.4	1.1	0.55

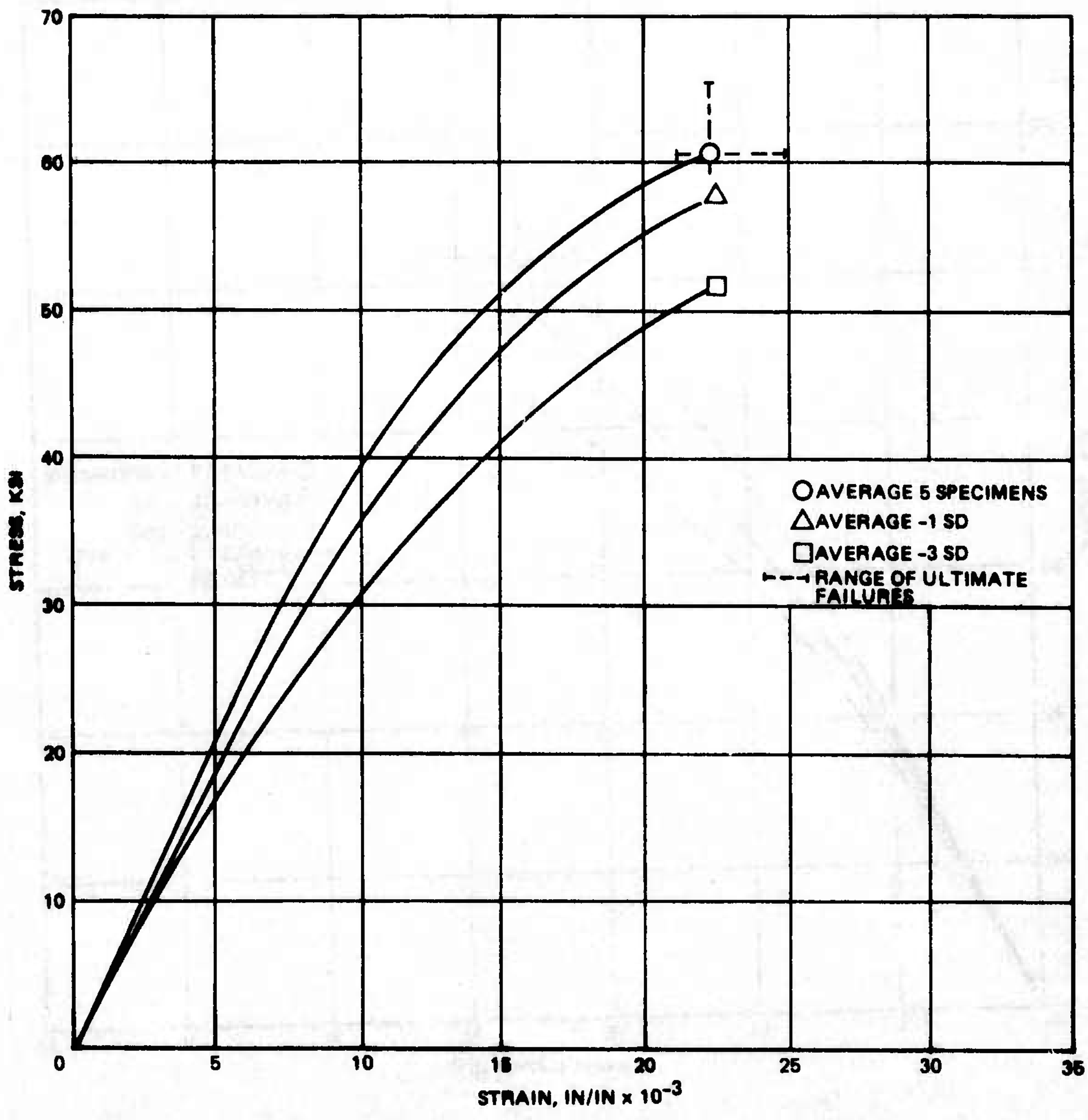


Figure A-1. Tensile Characteristics of CE-9000 at 74°F

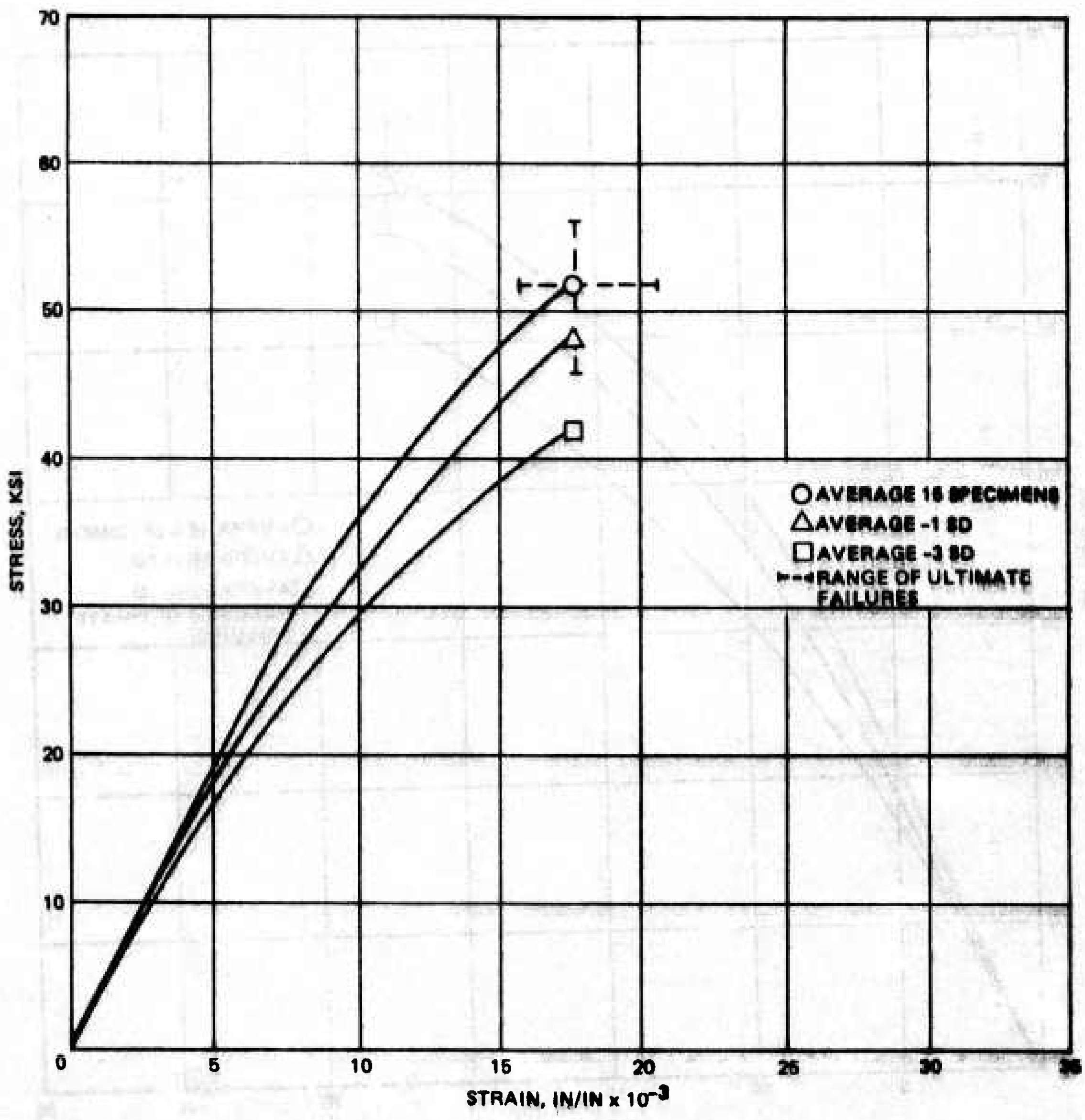


Figure A-2. Tensile Characteristics of CE-9000 at $200^{\circ}F$

TABLE A-2. SUMMARY OF FLEXURAL PROPERTIES TESTING

Material: Ferro Corporation CE 9000/7781

Test Method: ASTM D-790 (Procedure A)

Average Specimen Dimensions:

0.1057 x 1.0037 x 4.000

Crosshead Speed: 0.05 inch/minute

Conditioning: 1/2 hour at test temperature

Specimen Number	Flexural Strength (KSI)		Flexural Modulus (PSI x 10^6)	
	75^oF	200^oF	74^oF	200^oF
1	82.9	81.3	3.9	4.1
2	81.9	76.2	3.9	3.9
3	84.7	81.0	3.8	3.9
4	89.5	73.3	3.8	3.8
5	90.9	81.2	4.0	3.9
6		76.4		3.9
7		75.3		3.9
8		82.2		3.8
9		76.0		3.9
10		77.3		3.8
11		75.4		3.9
12		76.4		3.8
13		77.3		4.0
14		77.0		3.9
15		73.2		4.0
Average	86.0	77.3	3.9	3.9
-1 S.D.	4.0	2.8	0.08	0.08
-3 S.D.	12.0	8.6	0.25	0.25

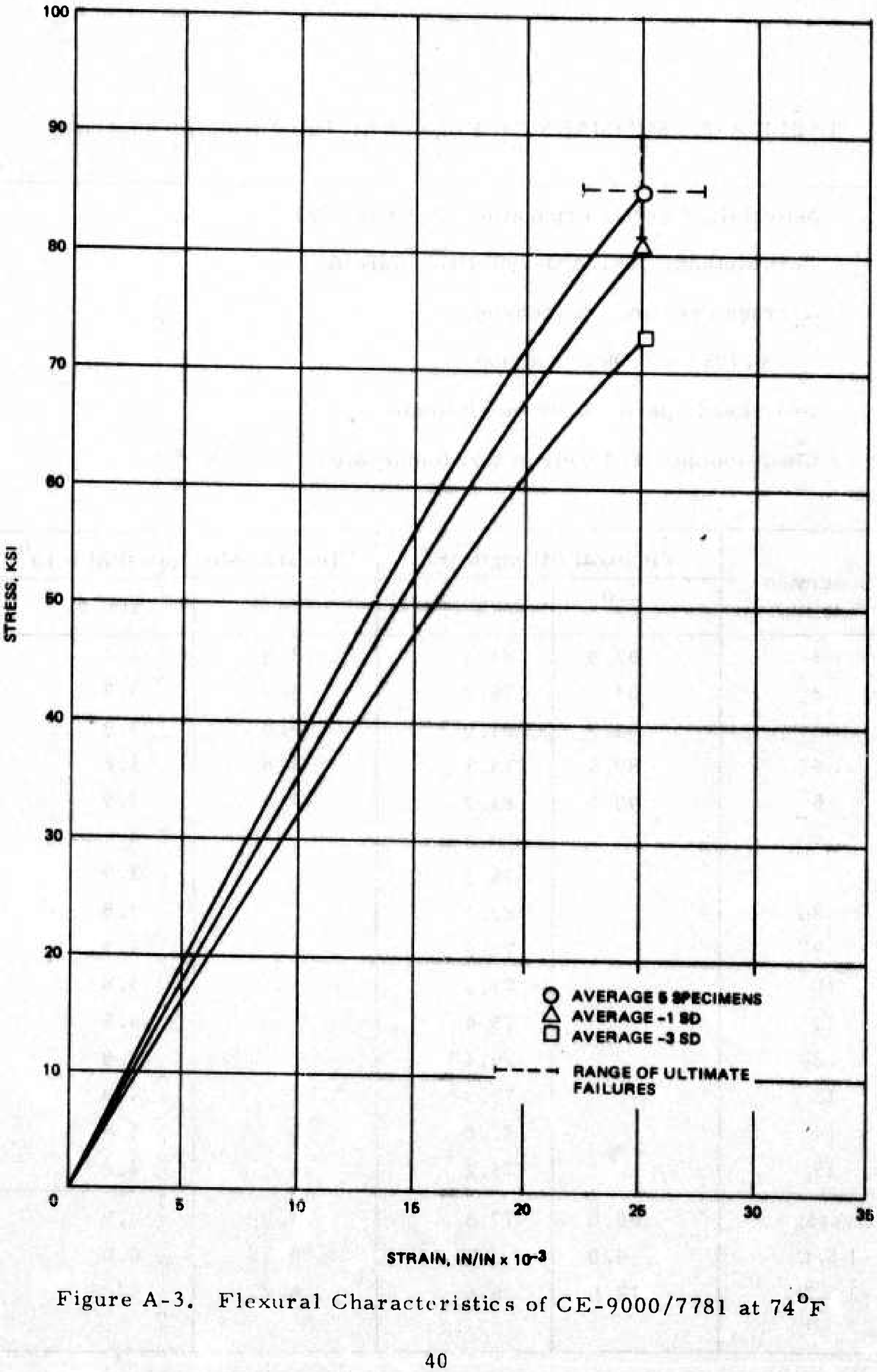


Figure A-3. Flexural Characteristics of CE-9000/7781 at 74°F

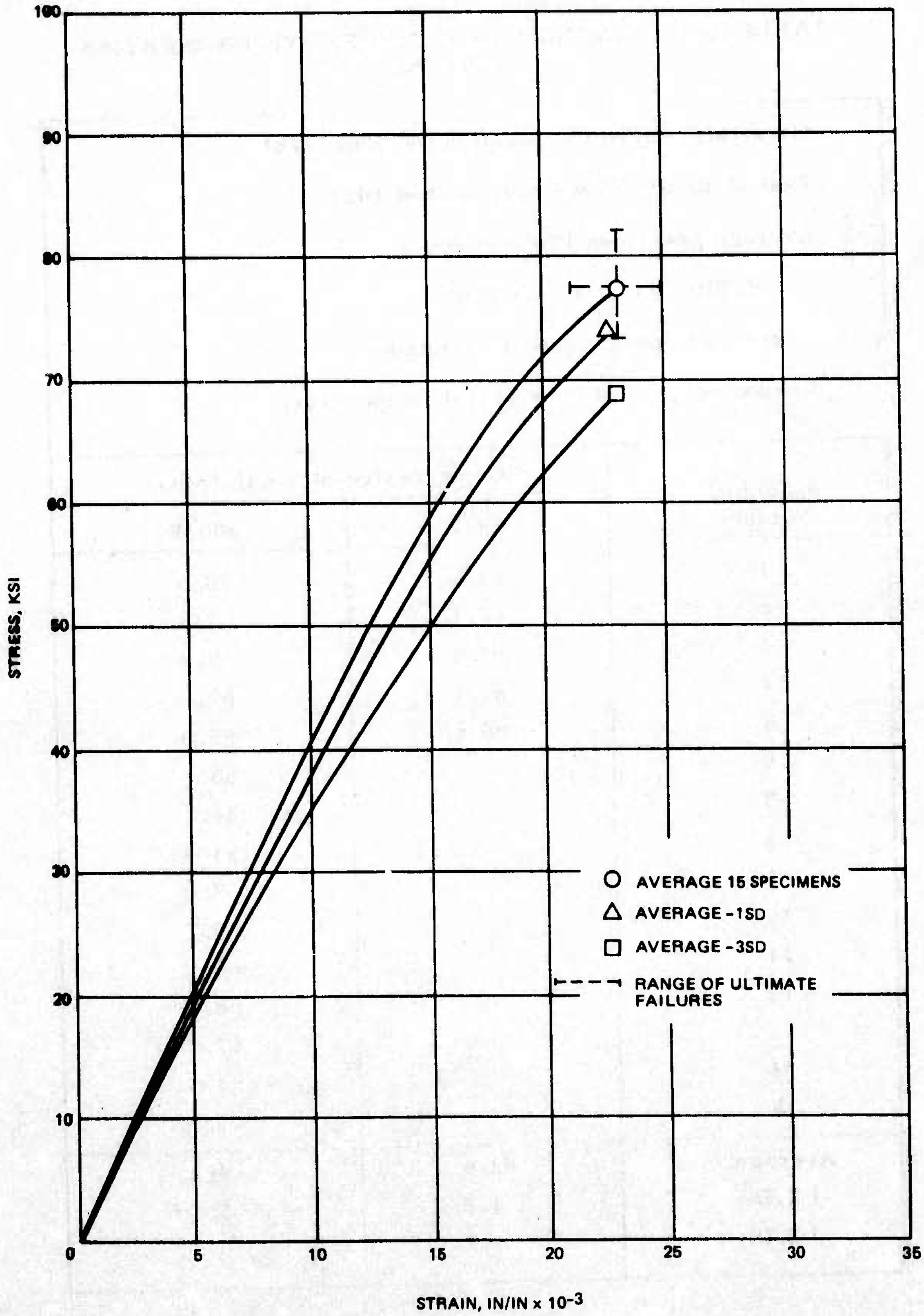


Figure A-4. Flexural Characteristics of CE-9000 at 200°F

TABLE A-3. SUMMARY OF COMPRESSIVE PROPERTIES TESTING

Material: Ferro Corporation CE 9000/7781

Test Method: FTMS 406, Method 1021

Average Specimen Dimensions:

0.1061 x 0.5013 x 3.0000

Crosshead Speed: 0.05 inch/minute

Conditioning: 1/2 hour at test temperature

Specimen Number	Compressive Strength (KSI)	
	$74^{o}F$	$200^{o}F$
1	83.7	70.6
2	82.4	67.8
3	83.4	69.8
4	83.1	69.4
5	80.6	65.0
6		68.6
7		58.1
8		51.8
9		67.0
10		59.7
11		68.2
12		58.4
13		67.1
14		63.3
15		67.5
Average	82.6	64.8
-1 S.D.	1.2	5.4
-3 S.D.	3.7	16.3

TABLE A-4. SUMMARY OF BEARING PROPERTIES TESTING

Material: Ferro Corporation CE 9000/7781

Test Method: ASTM D-953

Average Specimen Dimensions:

Hole: 0.250
Edge Distance: 0.500

Crosshead Speed: 0.05 inch/minute

Conditioning: 1/2 hour at test temperature

Specimen Number	Bearing Strength (KSI)	
	74^oF	200^oF
1	51.5	41.9
2	52.6	43.4
3	53.2	41.5
4	50.8	41.9
5	51.2	40.8
6		41.5
7		42.1
8		41.9
9		42.4
10		40.7
11		41.3
12		43.6
13		42.5
14		42.4
15		40.4
Average	51.9	41.9
-1 S.D.	1.0	0.91
-3 S.D.	3.0	2.73

APPENDIX B

SUMMARY OF ADHESIVE EVALUATION

TABLE B-1. SUMMARY OF LAP SHEAR PROPERTIES TESTING

Material: HT-424 Film Adhesive

Test Method: ASTM D-1002

Crosshead Speed: 0.05 inch/minute

Average Specimen Dimensions: Bond Area of 0.5 in^2

Substrate Material: Aluminum 6061-T6

Conditioning: 1/2 hour at test temperature

Specimen Number	Lap Shear Strength (PSI)	
	74°F	200°F
1	2430	1530
2	2350	1450
3	2500	1340
4	1890	1480
5	2290	1490
6		1400
7		1510
8		1440
9		1650
10		1130
11		1450
12		1390
13		1440
14		1540
15		1590
Average	2292	1455
-1 S.D.	238	120
-3 S.D.	715	359

TABLE B-2. SUMMARY OF LAP SHEAR PROPERTIES TESTING

Material: EA 934 Paste Adhesive

Test Method: ASTM D-1002

Crosshead Speed: 0.05 inch/minute

Average Specimen Dimensions:

Bond Area of 0.5 in^2

Substrate Material: Aluminum 6061-T6

Conditioning: 10 minutes at test temperature

Specimen Number	Lap Shear Strength (PSI)		
	Tested at 74°F Cured 24 hours at room temperature	Tested at 200°F Cured 24 hours at room temperature	Tested at 200°F Cured 1 hour at 25°F
1	1370	300	1120
2	1470	320	1440
3	1390	530	1200
4	1120	420	1150
5	1180	740	1500
6		870	
7		380	
8		490	
9		400	
10		550	
Average	1306	500	1282
-1 S.D.	149	183	175
-3 S.D.	446	548	526

APPENDIX C

MASS PROPERTIES AND STRUCTURAL
ANALYSIS

STRUCTURAL ANALYSIS

The following data is typical of the structural work accomplished and was used to size the preliminary SSRL structure. Differences between the initial and current design were verified by hand calculations which are not included.

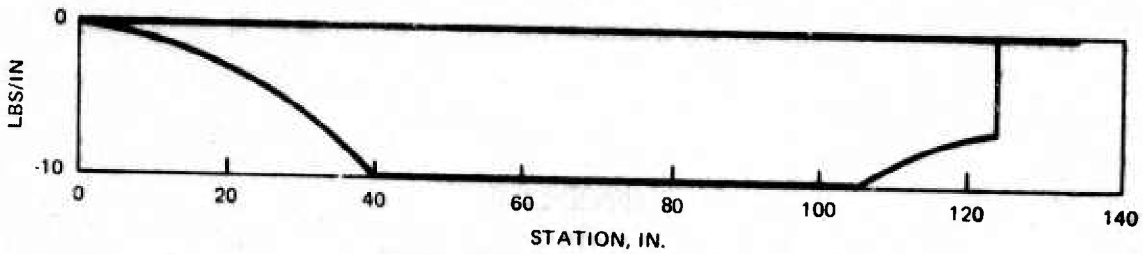


Figure C-1. Vertical Weight Distribution

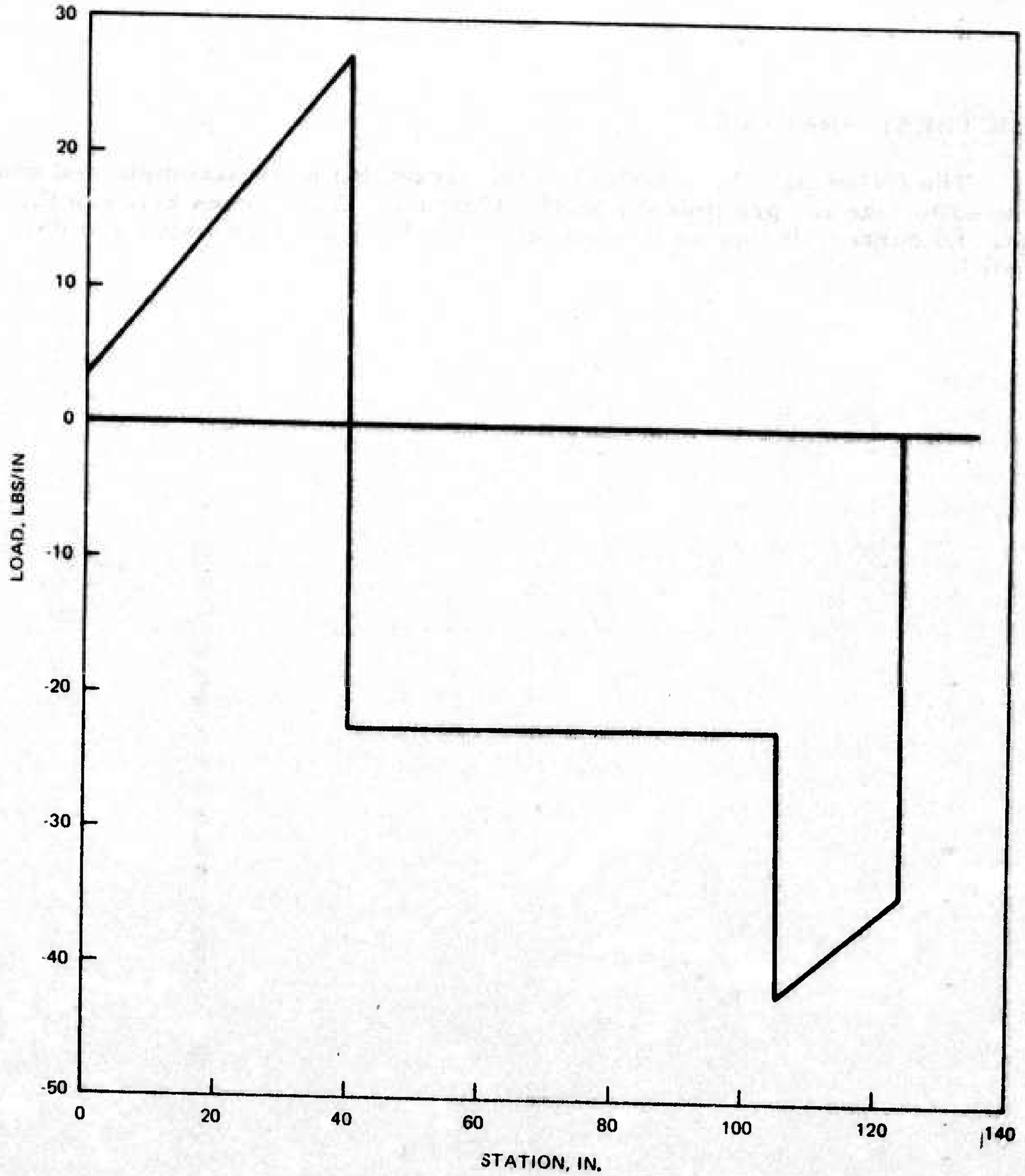


Figure C-2. Vertical Aerodynamic Distribution

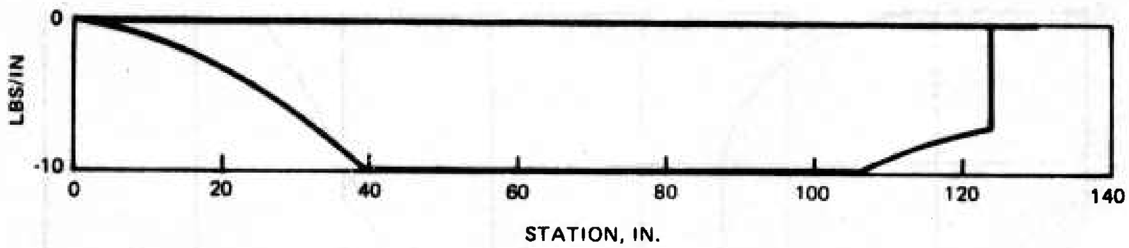


Figure C-3. Lateral Weight Distribution

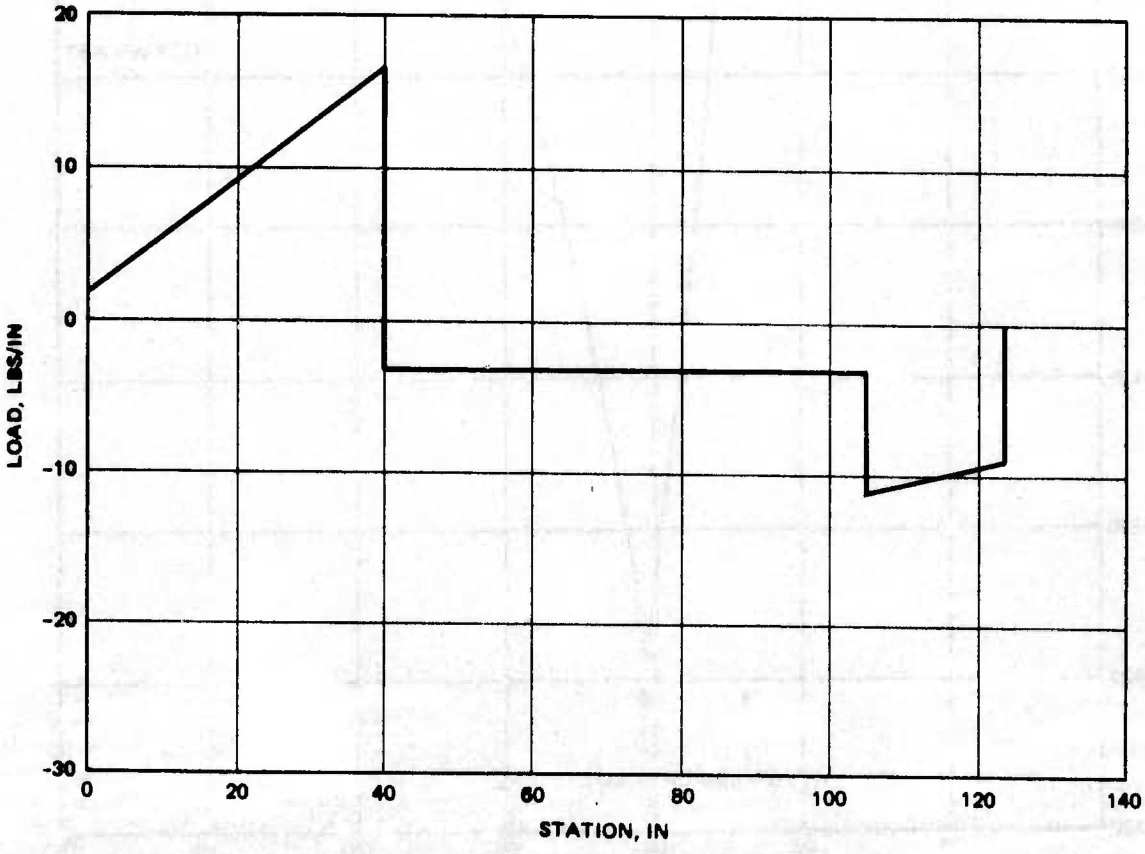


Figure C-4. Lateral Aerodynamic Distribution
(M = 1.2 at 5000 feet)

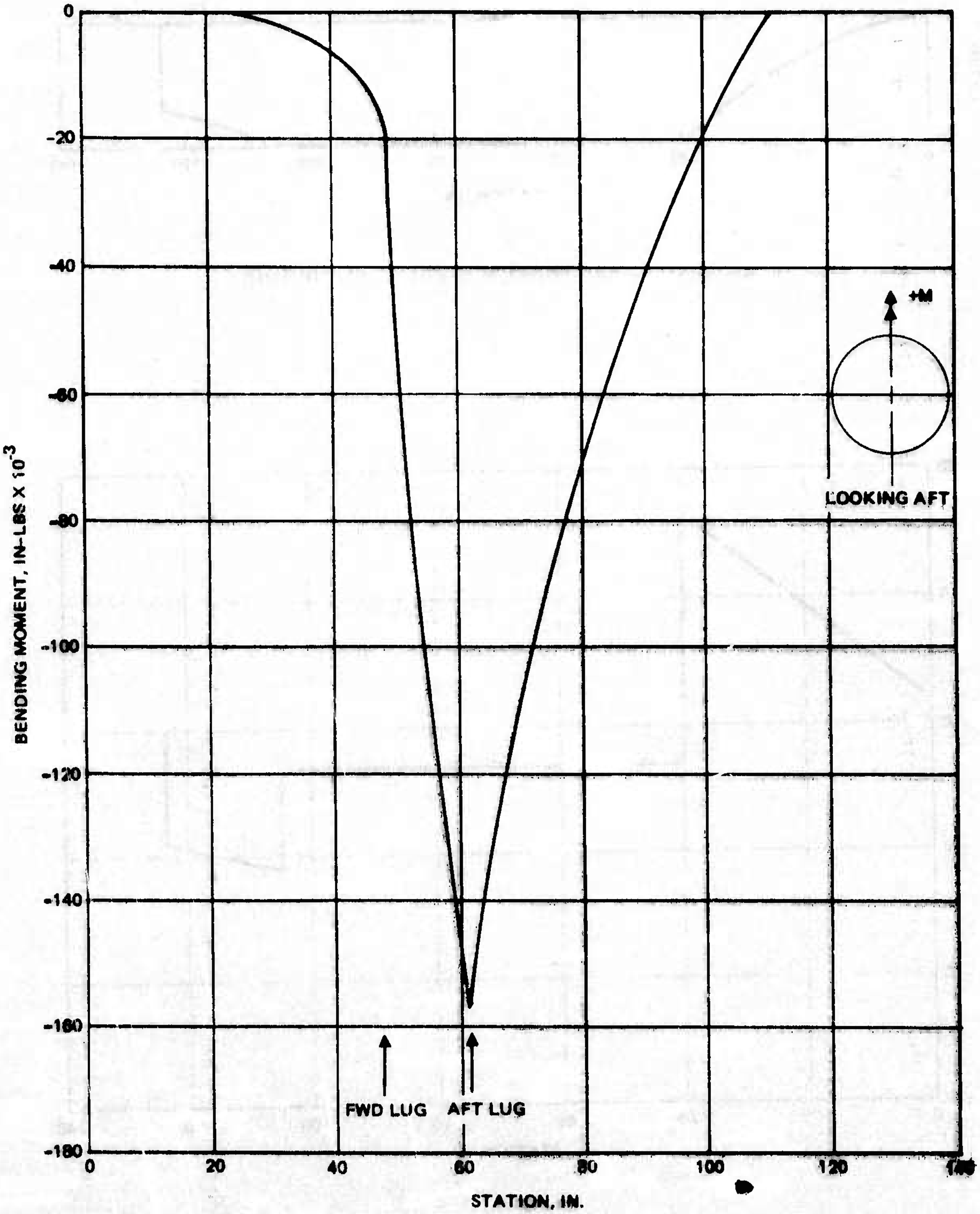


Figure C-5. Limit Bending Moment Diagram About Vertical Plane

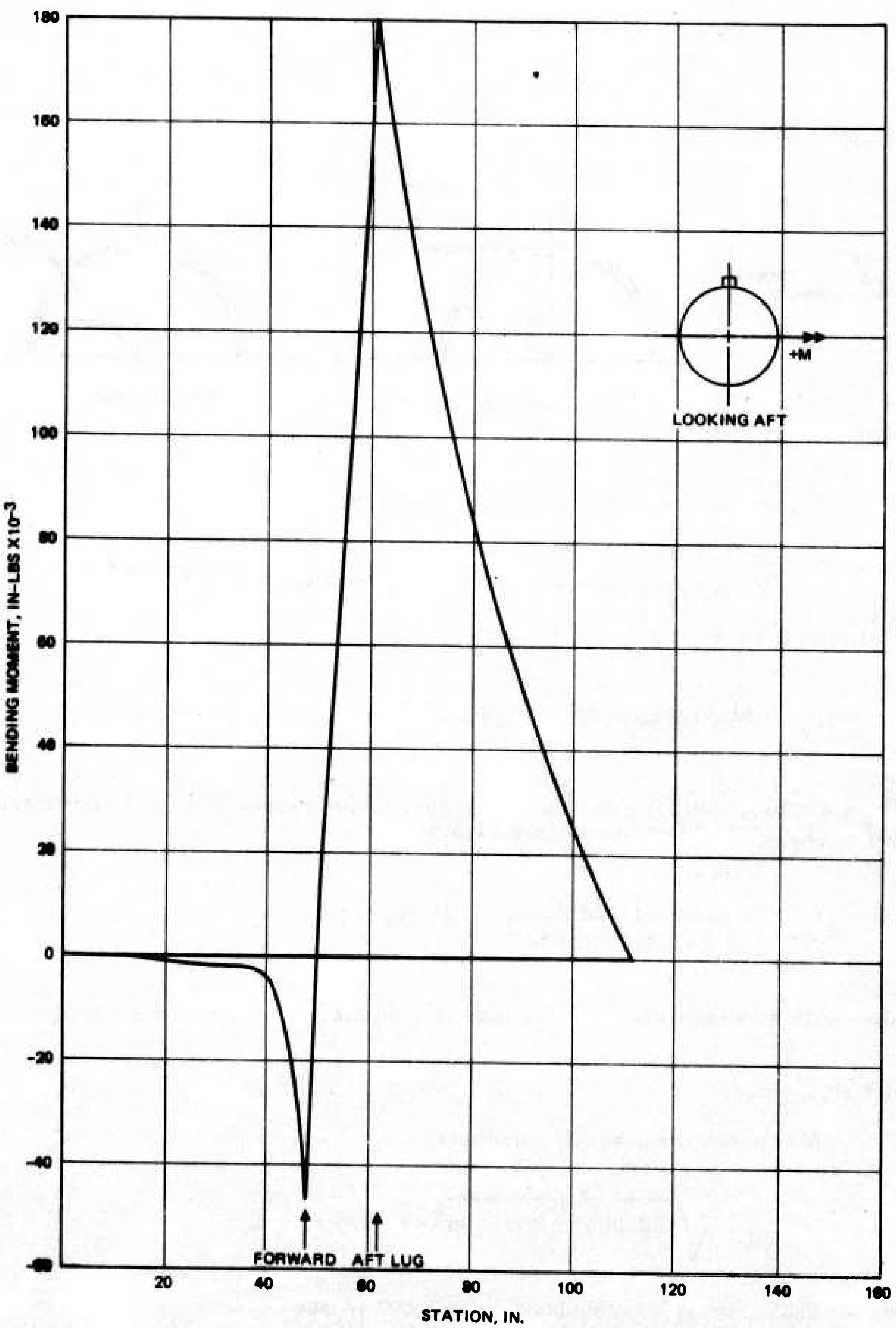
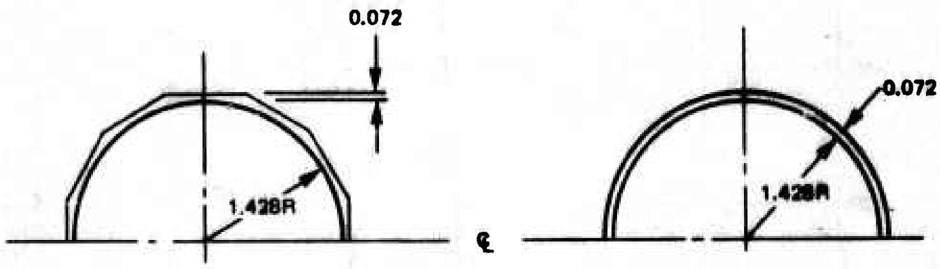


Figure C-6. Limit Bending Moment Diagram About Lateral Plane

TUBE

MATERIAL:
6063-T6 EXTRUSION
F_{tu} = 30,000 PSI



ACTUAL SECTION IDEALIZED SECTION

Worst loading condition during hangfire:

Maximum internal pressure	=	7 psi	Limit load
Longitudinal force	=	750 lbs	

Maximum hoop stress due to pressure:

$$f_u = \frac{(1.5)(7)(1.464)}{0.072} = 214 \text{ psi}$$

Maximum longitudinal stress due to force; compressive stress is considered only because all tubes are bonded together.

$$f_{cu} = \frac{(1.5)(750)}{(\pi)(1.464)(0.072)^2} = 1700 \text{ psi}$$

Since both stresses are low, the tube is considered structually adequate.

OUTSIDE WRAP

Maximum limit bending moment:

$$M_{LIM} = \sqrt{180,000^2 + 157,000^2} = 239,000 \text{ in-lbs}$$

$$M_{ULT} = (1.5)(239,000) = 358,000 \text{ in-lbs}$$

Maximum flexure stress:

$$f_{bu} = \frac{358,000}{(\pi)(8)^2(0.070)} = 25,000 \text{ psi}$$

Flexure allowable of fiberglass, CE-306/7781

$$F_{bu} = 45,000 \text{ psi}$$

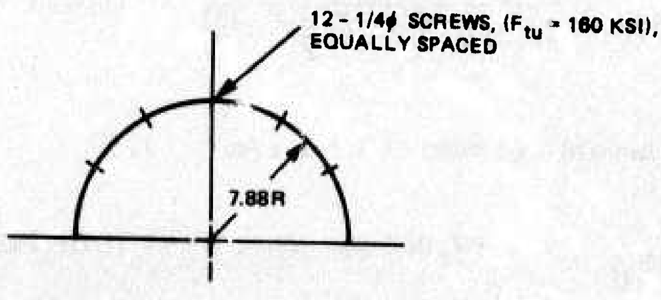
$$M.S._{ULT} = \frac{45,000}{25,500} - 1 = 0.76$$

JOINT at STATION 82.7

Maximum limit bending moment:

$$M_{LIM} = \sqrt{72,000^2 + 62,000^2} = 95,500 \text{ in-lbs}$$

$$M_{ULT} = (1.5)(95,000) = 142,500 \text{ in-lbs}$$



Maximum shear per screw:

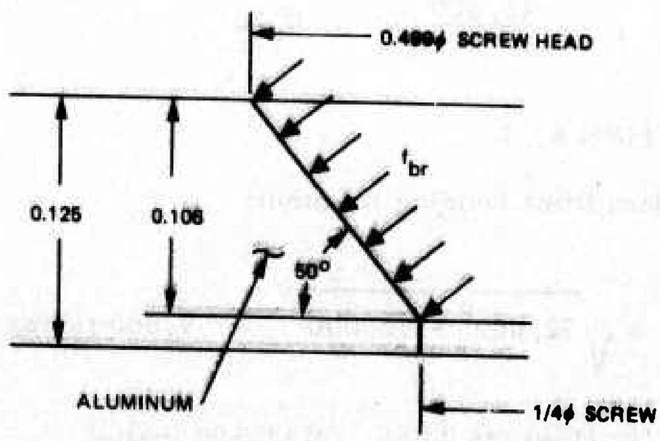
$$P_{S_{ULT}} = \frac{142,500}{1/2\,(12)\,(7.88)} = 3,010 \text{ lbs}$$

Screw shear allowable:

$$P_{S_{ULT}} = 5300 \text{ lbs (Ref: MIL-HDBK-5A)}$$

$$M.S._{ULT} = \frac{5300}{3010} - 1 = 0.76$$

Maximum bearing stress under screw head:



$$f_{BR_U} = \frac{(3010)(\cos^2 50^o)}{(0.106)\left(\frac{0.499 + 0.25}{2}\right)} = 31,400 \text{ psi}$$

Bearing allowable of 6061-TG for e/D = 1.5

$$F_{BR_U} = 67,000 \text{ psi (Ref: MIL-HDBK-5A)}$$

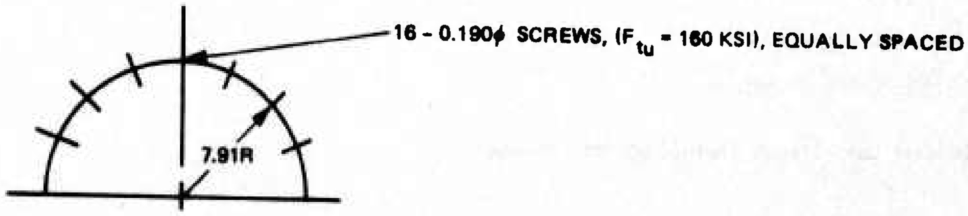
$$M.S._{ULT} = \frac{67,000}{31,400} - 1 = 1.13$$

JOINTS AT STATIONS 84.6 AND 96.8

Maximum limit bending moment is at station 84.6:

$$M_{LIM} = \sqrt{67,000^2 + 56,500^2} = 87,600 \text{ in-lbs}$$

$$M_{ULT} = (1.5)(87,600) = 131,400 \text{ in-lbs}$$



Maximum shear per screw:

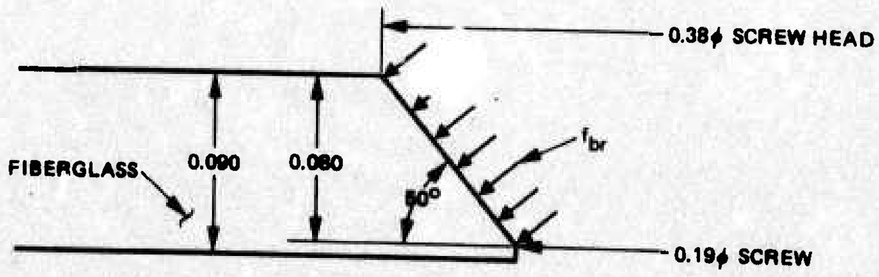
$$P_{S_{ULT}} = \frac{131,400}{1/2\,(16)\,(7.91)} = 2080 \text{ lbs}$$

Screw shear allowable:

$$P_{S_{ULT}} = 3062 \text{ lbs (Ref: MIL-HDBK-5A)}$$

$$M.S._{ULT} = \frac{3062}{2080} - 1 = 0.47$$

Maximum bearing stress on fiberglass under screw head:



$$f_{BR_U} = \frac{(2080)(\cos^2 50^o)}{(0.80)\left(\frac{0.38 + 0.19}{2}\right)} = 37,700 \text{ psi}$$

Bearing allowable of fiberglass, CE9000/7781:

$$F_{BR_U} = 40,000 \text{ psi}$$

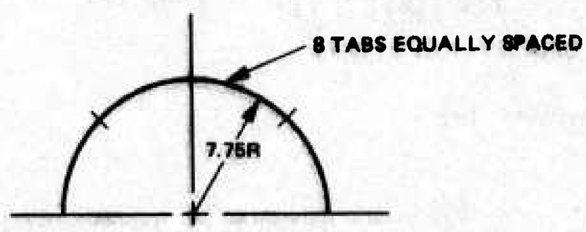
$$M.S._{ULT} = \frac{40,000}{37,700} - 1 = 0.06$$

JOINT AT STATION 98.3

Maximum limit bending moment:

$$M_{LIM} = \sqrt{28,000^2 + 22,200^2} = 35,600 \text{ in-lbs}$$

$$M_{ULT} = (1.5)(35,600) = 53,400 \text{ in-lbs}$$



Maximum shear per tab:

$$P_{S_{ULT}} = \frac{53,400}{(1/2)(8)(7.75)} = 1723 \text{ lbs}$$

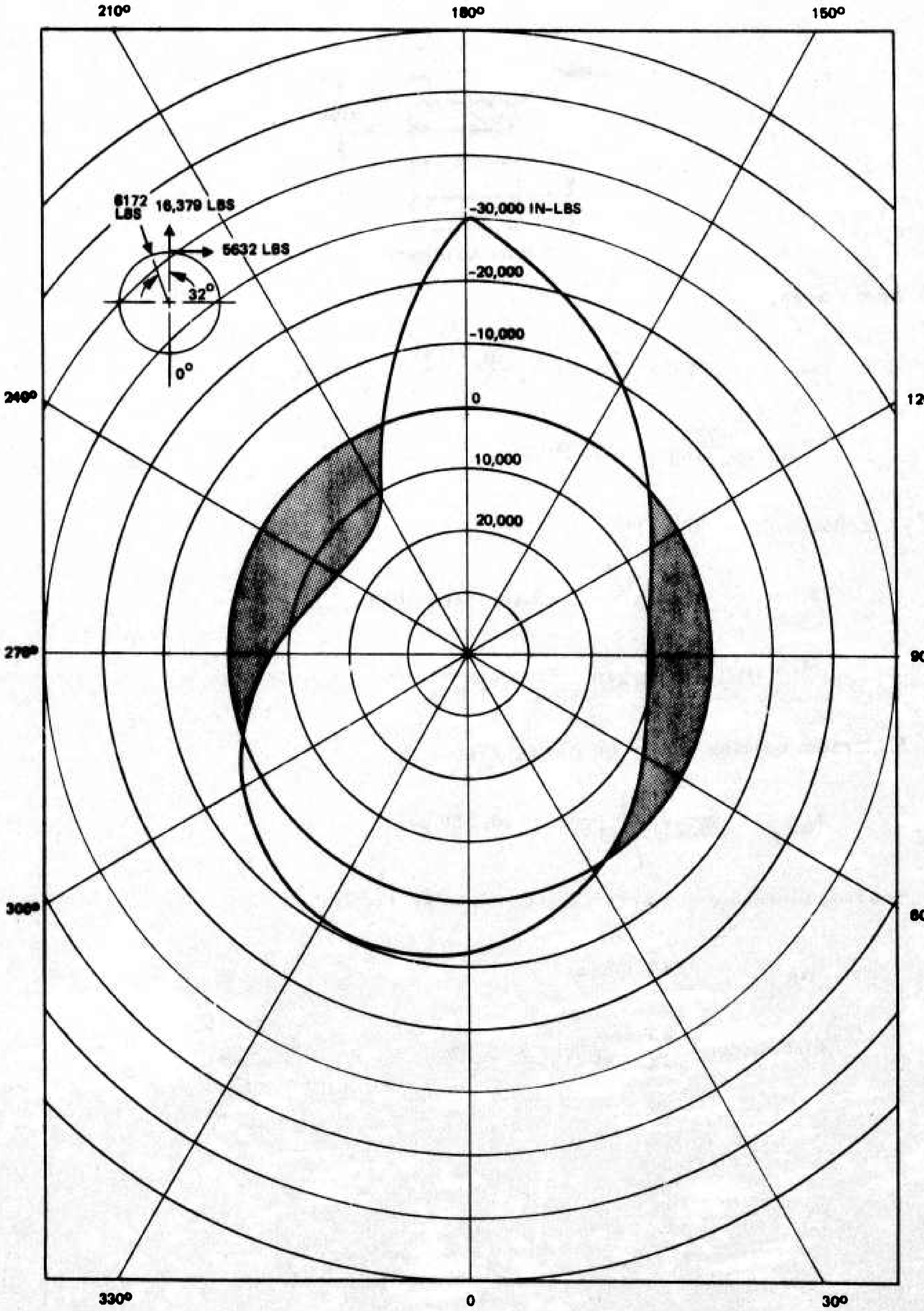
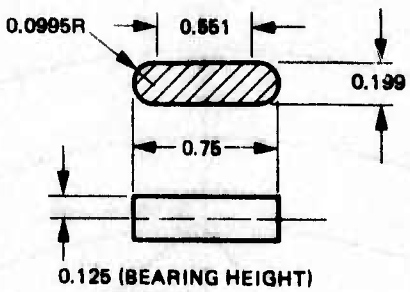


Figure C-7. Aft Ring Bending Moment Diagram



Shear area:

$$A_S = (0.199)(0.551) + \frac{(\pi)(0.199)^2}{4} = 0.1408 \text{ in}^2$$

$$F_S = \frac{1723}{0.1408} = 12,240 \text{ psi}$$

Tab allowable of 6061-T6

$$F_{SU} = 27,000 \text{ psi (Ref: MIL-HDBK-5A)}$$

$$M.S._{ULT} = \frac{27,000}{12,240} - 1 = 1.20$$

Maximum bearing stress on fiberglass:

$$f_{BR_U} = \frac{1723}{(0.75)(0.125)} = 18,380 \text{ psi}$$

Bearing allowable of fiberglass, CE9000/7731

$$f_{BR_U} = 40,000 \text{ psi}$$

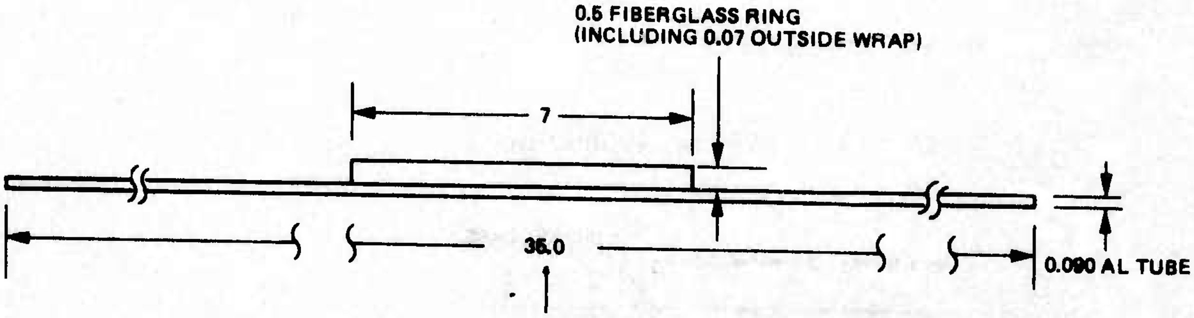
$$M.S._{ULT} = \frac{40,000}{18,380} - 1 = 1.18$$

RING

The critical ring load is at 212^{0}.

$$M_{LIM} = 15,000 \text{ in-lbs}$$

$$M_{ULT} = (1.5)(15,000) = 22,500 \text{ in-lbs}$$



Effective length of aluminum tube

$$= 7 \frac{E_{AL}}{E_{Fiberglass}} = 7 \frac{10 \times 10^{6}}{2 \times 10^{6}} = 0.2 \text{ in.}$$

$$y = \frac{(7)(0.5)(0.34) + (35)(0.050)(0.045)}{(7)(0.5) + (35)(0.090)} = 0.2 \text{ in.}$$

$$I = \frac{(7)(0.5)^{3}}{3} + \frac{(35)(0.09)^{3}}{3} = 0.3 \text{ in}^{4}$$

$$I_{N.A.} = 0.3 - 6.65\,(0.11)^{2} = 0.22 \text{ in}^{4}$$

Maximum flexure stress in fiberglass:

$$f_{bu} = \frac{(22,500)(0.59 - 0.2)}{0.22} = 39,900 \text{ psi}$$

Flexure allowable of fiberglass, CE9000/7781:

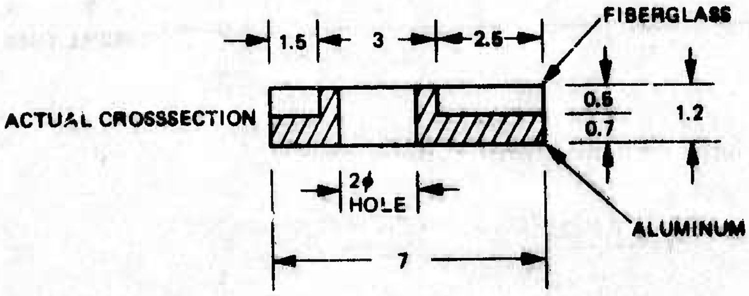
$$F_{bu} = 70,000 \text{ psi}$$

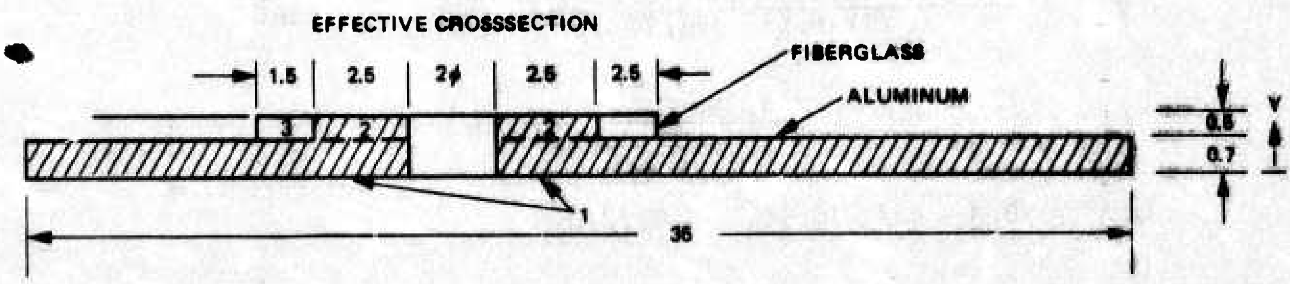
$$M.S._{ULT} = \frac{70,000}{39,900} - 1 = 0.75$$

Check ring stresses at point of load, at 180°:

$$M_{LIM} = 30,000 \text{ in-lbs}$$

$$M_{ULT} = (1.5)(30,000) = 45,000 \text{ in-lbs}$$





Effective length of aluminum = original length $\frac{E_{AL}}{E_{Fiberglass}}$ = 5 (original length)

Item	A	Y	AY	AY^2	I
1	23.1	0.35	8.085	2.83	0.9432
2	2.5	0.95	2.375	2.256	0.0521
3	2.0	0.95	1.9	1.805	1.0417
	27.6		12.360	6.891	1.0370

$$\bar{y} = \frac{12.360}{27.6} = 0.448 \text{ in.}$$

$$I_{N.A.} = 1.037 + 6.891 - (27.6)(0.448)^2 = 2.389 \text{ in}^4$$

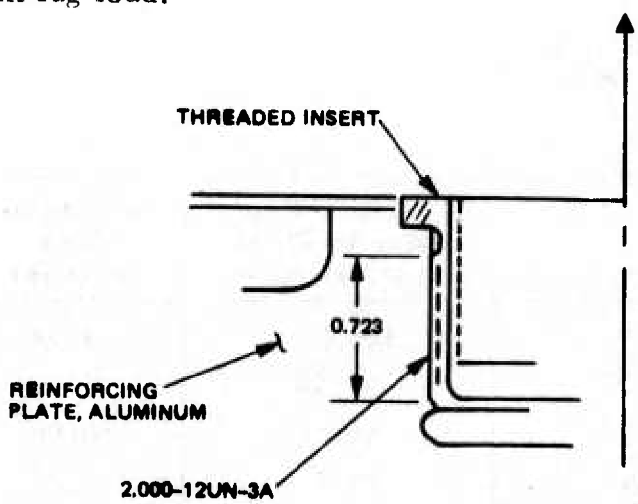
Maximum flexure stress in fiberglass:

$$f_{bu} = \frac{(45,000 (1.2 - 0.448)}{2.385} = 14,160 \text{ psi}$$

Flexure stress not critical.

LUG INTERFACE LOAD-AFT

Maximum lug load:



Ultimate lug load = (1.5) (16.379) = 24,600 lbs.

Shear stress between threaded insert and reinforcing plate; assume first and last threads ineffective, 6 good threads.

$$f_s = \frac{24,600}{(\pi)(1.944)\left(\frac{1}{12}\right)(6)} = 8060 \text{ psi}$$

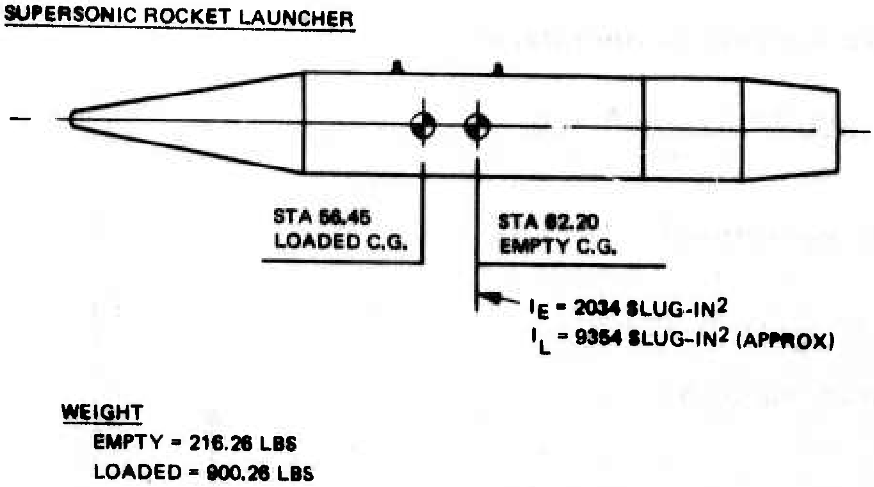
Shear allowable of 6061-T6 plate:

$$F_{su} = 27,000 \text{ psi (Ref: MIL-HDBK-5A)}$$

$$M.S._{ULT} = \frac{27,600}{8060} - 1 = 2.35$$

MASS PROPERTIES

The following tables summarize the mass and initial calculations used to define the launcher.



Item	Name	Distance From Nose to CG of Item (inches)	Weight of Item (pounds)	dw (in-pounds)
1	Aft Fairing	106.35	5.04	536.00
2	Aft Bulkhead	97.50	3.51	342.22
3	Switch	96.88	0.06	5.81
4	Intervalometer	94.51	0.42	39.69
5	Wire Harness	90.69	0.40	36.27
6	Tube Assembly (1P)	90.35	25.60	2312.96
7	Rod (4)	90.79	0.87	78.98
8	Spacer (4)	90.79	0.32	29.05
9	Skin-Base Selt	90.97	4.17	379.34
10	Forward Bulkhead	83.72	4.04	338.22
11	Fiberglass Ring	82.63	3.78	312.34
12	Conduit - Aft	82.50	0.05	4.12
13	Bushing and Conn	79.13	0.18	14.24
14	Fitting - Conn	79.13	0.72	56.97
15	Conduit - Forward	73.50	0.05	3.67

Item	Name	Distance From Nose to CG of Item (inches)	Weight of Item (pounds)	dw (in-pounds)
16	Bushing and Lug	65.63	0.86	54.66
17	Lug Ftg - Aft	65.88	2.60	171.28
18	Glass Ring	66.95	17.62	1179.65
19	Bushing and Lug	51.63	0.86	44.40
20	Lug Ftg - Fwd	51.38	2.60	133.58
21	Glass Ring	51.31	17.62	904.08
22	Tubes:			
	Ctr	42.88	7.14	306.16
	Row 1	46.72	38.78	1811.80
	Row 2	53.25	32.15	1711.98
	Row 3	55.60	29.40	1634.64
23	Foam - Aft	73.81	1.91	140.97
24	Foam - Ctr	54.67	1.28	69.97
25	Foam - Fwd	40.79	0.61	24.88
26	Skin	59.30	13.00	770.90
27	Nose Cap	1.75	0.60	1.05
	Total		216.24	13,451.66

Empty C.G. $= \dfrac{13,451.66}{216.24} = 62.20$

Loaded C.G.:

Rockets = 18 at 38 pounds = 684 pounds with C.G. at Station 54.63
dw = 37,366.92 in-pounds.

$$\sum = \begin{array}{ll} 216.24 & 13,451.66 \\ \underline{684.00} & \underline{37,366.92} \\ 900.24 \text{ pounds} & 50,818.58 \text{ in-pounds} \end{array}$$

Loaded C.G. $= \dfrac{\text{in-pounds}}{\text{pounds}} = 56.45$

Mass moment about empty C.G. (pitch)

$$I = \sum 12 d^2 \left(\frac{W_t}{386}\right) = 2034 \text{ slug-in}^2$$

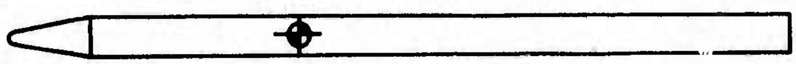
d = Distance from C.G. to C.G. of item

W_t = Weight of item.

Item	Name	d	d^2	$\frac{W_t}{386}$	$d^2 \frac{W_t}{386}$
1	Aft Fairing	44.15	19.50	0.01305	25.42
2	Aft Bulkhead	35.30	1245.00	0.0091	11.32
3	Switch	35.18	1238.00	0.000155	0.20
4	Intervalometer	32.40	1050.00	0.00108	1.14
5	Wire Harness	28.45	821.00	0.00104	0.85
6	Tube Assy (18)	28.15	792.00	0.0664	52.50
7	Rod (4)	28.59	838.00	0.00226	1.89
8	Spacer (4)	28.59	838.00	0.00083	0.69
9	Skin-Base Sect	28.77	855.00	0.0108	9.23
10	Fwd Bulkhead	21.52	463.00	0.0105	4.85
11	Fiberglass Ring	20-43	418.00	0.0098	4.10
12	Conduit-Aft	21.50	462.00	0.00013	0.06
13	Bushing & Conn	18.60	346.00	0.00047	0.16
14	Fitting-Conn	18.40	339.00	0.00187	0.63
15	Conduit-Fwd	13.30	177.00	0.00013	0.02
16	Bushing & Lug	8.46	72.00	0.00223	0.16
17	Lug Ftg-Aft	8.36	70.00	0.00674	0.47
18	Glass Ring	4.75	23.00	0.0457	1.05
19	Bushing & Lug	12.90	166.00	0.00223	0.37
20	Lug Ftg-Fwd	13.10	172.00	0.00674	1.16
21	Glass Ring	10.89	119.00	0.0457	5.44

Item	Name	d	d^2	$\frac{W_t}{386}$	$d^2 \frac{W_t}{386}$
22	Tubes				
	CTR	19.32	374.00	0.0185	7.10
	ROW 1	15.48	240.00	0.1005	24.10
	ROW 2	8.95	80.00	0.0833	6.66
	ROW 3	6.60	44.00	0.0763	3.35
23	Foam-Aft	9.61	92.00	0.00995	0.05
24	Foam-Ctr	7.53	57.00	0.00332	0.05
25	Foam-Fwd	21.41	460.00	0.00158	0.73
26	Skin	2.90	8.00	0.03368	0.27
27	Nose Cap	60.45	3650.00	0.00155	5.52
	Total				169.51

MASS MOMENT OF ROCKETS



C.G. = STA 54.63 OF LCHR

WT = 38 LBS x 18 = 684

$$I_{Rockets} = \frac{18}{12}\left(\frac{38}{32.2}\right)(64)^2 + \left(\frac{684}{32.2}\right)(1.82)^2$$

$$= 7250 + 70$$

$$= 7320 \text{ Slug-in}^2$$

$$I_{Loaded} = 7320 + 2034 = 9354 \text{ Slug-in}^2 \text{ approx.}$$

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AUL (AUL-LSE-70-239)	1
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