Final Report

ULTRA LIGHT HONEYCOMB DEVELOPMENT FOR S-II STAGE IMPROVEMENT

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FINAL REPORT
ULTRA-LIGHT HONEYCOMB DEVELOPMENT
FOR S-II STAGE IMPROVEMENT

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ABSTRACT

The Ultra Light Honeycomb Program has determined that it is feasible to manufacture large, deep core sandwich construction panels. A modified bonding system for titanium was developed by Avco in an independent research program and successfully used in the Ultra Light Honeycomb Program.

Data has been obtained from tests to verify and extend existing data to include deep core sandwich construction. This data is core shear strength, core shear modulus, and proportional limit shear stress. Analysis has shown that for the specimens to be tested in edgewise compression, face wrinkling is the critical mode of failure. An analytical method to determine the ultimate compressive strength for local face wrinkling failure has been determined which closely predicts the failing stress. A weight analysis shows that the approximate weight of the skirt structure of the S-II Stage of the Saturn V Missile, made of deep core aluminum honeycomb with titanium skins, is 3048 lb. with an ultimate load carrying capability of 9275 lb/inch of circumference.

A follow-on program recommends additional feasibility investigation and testing. The investigation and tests will include joint designs to attach the skirt structure to the missile structure and the general instability mode of failure of large curved panels. A full scale cylinder structure will be designed and tested. The end result of this program will be the design of a skirt structure of ultra light honeycomb construction.

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INTRODUCTION

II. INTRODUCTION

A. PURPOSE OF PROGRAM

The purpose of this development program was as follows. One objective was to prove that sandwich panels consisting of titanium skins bonded to aluminum honeycomb core could be manufactured with reliability. Another objective was to determine if the existing theoretical methods of analysis of sandwich design were applicable to thick-core sandwich cylinders of large radii.

The ultimate goal of the tests was to furnish sufficient design data from which the forward skirt structure of the S-II stage of the Saturn V missile, or some future missile structure, could be designed and manufactured with a weight savings over the conventional skin-stringer type structure.

The following objectives of this program are discussed in SUMMARY OF RESULTS AND CONCLUSIONS, page 2.3.0.

1. Determine if presently available manufacturing capabilities are sufficient to efficiently produce deep core sandwich panels that are structurally sound.
2. Determine any materials handling problems that may be of special consequence in the consideration of this type of structure for large shell structures.

3. Determine the limitations of present theory to predict failure modes of deep core honeycomb.
4. Determine if deep core sandwich has deleterious size limitations.
5. Determine necessary quality control procedures and assess consequences of their use on large structures.

B. SCOPE OF WORK

To accomplish the task outlined in the contract, the program was divided into three phases.

Phase I was a test program for which basic data was to be determined concerning elemental fabrication, testing and evaluation of relatively small specimens to substantiate and augment existing design and fabrication data.

Phase II was a continuation of the evaluation of large bonded honeycomb panels. Closeout techniques were developed, handling problems were studied and joint design was evaluated. The characteristics of deep honeycomb panels under static, dynamic and thermal loadings were evaluated.

Phase III of the program consisted of making a 16 mm sound-color film of the manufacture and testing of the specimens during the performance of this program.

The main objective of the tests was to determine the ultimate edgewise compressive load that would produce a local instability mode of failure in the panels. The flatwise compression and tension tests, and the block shear tests, were performed to verify or complement existing manufacturer's data of cores, adhesives and titanium skins. These tests were designed to supply data that was needed in order to calculate the theoretical ultimate strength of the panels. The ultimate stress at failure was to be compared with theoretical predictions of ultimate stress based on the design data obtained from the supplementary tests. From these comparisons, a recommendation was to be made as to which analytical technique showed the best correlation with test results. The scatter of results to be expected upon using this analytical technique was to be determined also.

The results of the test program were to be applied to the design of the forward skirt structure of the S-II stage of the Saturn V missile.

A follow-on program that would provide additional needed data for a more refined design was to be proposed. This follow-on program was to be based on a careful evaluation of the results obtained from this test program.

C. SUMMARY OF RESULTS AND CONCLUSIONS

1. Manufacturing Capabilities

The investigation showed that the problems associated with the fabrication of deep core sandwich construction can be solved by the techniques used in thin or tapered core sandwich structure. The sandwich components must be cut to size, prefitted, cleaned and layed up as an assembly into a bonding tool. This tool must be made to the proper contour, and tolerances must be held so that during the bonding process, crushing of the core does not take place. The curing of the assembly is accomplished in an autoclave in which pressure is applied to the part to hold the assembly in contact with the bonding tool. A controlled temperature cycle accomplishes the cure of the adhesive. The Skirt Structure of the S-II Stage will be manufactured in segments. Any inserts that might be needed will be bonded into place during the primary bonding cycle. The segments will be joined together either by a secondary bond process or by mechanical fasteners.

The only manufacturing problems which were encountered in this program were associated with the titanium skins.

One problem encountered was that of obtaining a good bond between the titanium skins and the aluminum honeycomb core. A technique of cleaning and priming to bond the titanium skins was developed by Avco during an independent development program. This technique provides a bond that is sufficient to develop the required structural properties of the core.

Another problem was the procurement of thin titanium skins in the heat treated and aged condition. The vendor for titanium skins had to "hand make" each skin. This "by hand" process was dependent on operator skill and technique and was very expensive and required a long procurement time. A large structure such as the Saturn V, S-II forward skirt will require large quantities of titanium skin. This "hand made" method of manufacture is obviously not acceptable for the manufacture of large structures. The titanium manufacturers, however, are presently installing new rolling and materials handling equipment which should alleviate this problem.

2. Materials Handling

The handling of materials to manufacture deep core sandwich structure is within the present state-of-the-art. One possible difficulty may arise in the event that the design specifies that there be chemical-milling of titanium skins that are larger than can be accomodated by existing facilities.

The cleaning of large panels will also be dependent upon the size of existing facilities; however, the construction of tanks for the cleaning will not be as critical as that for chemical-milling.

The size of the bonded panels will have to be controlled in the original design so that the sandwich construction segments and the bonding tool or fixtures will fit within the autoclave for curing.

3. Limitation of Present Theory

The test results are tabulated in Appendix A of this report. The edgewise compression tests specimens all failed in the face wrinkling mode. The average failing stress for Phase I specimens was 89,605 pounds per square inch and the average failing stress for Phase II specimens was 113,304 pounds per square inch. A modified bonding system was used on the specimens fabricated for the Phase II tests.

The equation shown below was chosen to be the analytical technique which showed the closest correlation with the test results of specimens which failed by a local face wrinkling mode.

$$\sigma_{cr} = \frac{0.38 \sqrt[3]{E_t E_c G_c}}{1 + 1.39(E_c G_c / E_t F_g)^2 (w/t_f)}$$

where:

σ_{cr} = Critical Buckling Stress, psi

E_t = Tangent Modulus of face material, psi

E_c = Compressive Modulus of core, psi

G_c = Shear Modulus of core, psi

F_g = Flatwise core to skin tension strength or crush strength (whichever is least), psi

w = Waviness, inches

t_f = Face thickness, inches

This equation was taken from Reference 1. The constant was changed from 0.91 to 0.38 which made the test values and the predicted values correlated much better. Also, in Reference 1, the modulus of elasticity of the face sheets appeared in the equation. It was determined that the use of the tangent modulus in its place was more judicious. However, for the range of str stresses encountered in this program, it made no difference whether the modulus or the tangent modulus was used in the equations for predicting the critical stresses.

A statement to define the values to be used for waviness and core shear modulus was lacking in Reference 1. The Phase II test results showed that for a local face wrinkling type failure, the orientation of the core ribbon direction with respect to the loading direction made little difference. The average failing stress of those specimens in which the longitudinal ribbon direction was parallel with the load was 116,198 psi, whereas the average for those specimens with the transverse ribbon direction parallel with the load was 113,182 psi. This represents a difference of less than 3%. Therefore, manufacturer's data on typical core shear modulus for the longitudinal and transverse directions was averaged and the resulting value was used in the theoretical calculations of critical stress. The use of this average value for core shear modulus gave better correlation between test results and theoretical predictions than did the use of different values for the longitudinal and transverse directions or the use of calculated core shear modulus values based on the density of the core.

The face sheets of the specimens were checked, before testing, for flatness by use of a feeler gauge. The deviation from a flat surface was plotted, for both faces in the form of a grid. The maximum deviations across the width of a specimen were averaged for the worst face. This average deviation was used as the waviness factor. Deviations of less than .005 inches were not recorded. Those specimens which had an average deviation of less than .005 inches were assigned a waviness factor of .005.

The waviness of the specimens tested seemed to have little or no effect on the load carrying capability of the specimens. In fact, the specimen that attained the highest stress at failure also had one of the largest waviness factors. Moreover, reducing the waviness factor from .005 to .001 increases the predicted failing stress by assmuch as 15% in some cases. The test results, however, do not show that the critical stress is this dependent upon the waviness.

In the flatwise tensile tests, the majority of the specimens failed in the face-to-core bond. The average tensile strength of the specimens that failed solely in bond was 336 pounds per square inch, with the highest strength being 487 pounds per square inch, and the low, 194 pounds per square inch.

The flatwise compression test results were compared with typical values listed in Reference 3. The following results were obtained:

a. The typical compressive strength of the 2.3 pcf honeycomb, as listed in Reference 3, was 2.4% greater than the average compressive strength of the 2.3 pcf honeycomb specimens that were tested.

b. The typical compressive strength of the 3.1 pcf honeycomb, as listed in Reference 3, was 9.98% greater than the average compressive strength of the 3.1 pcf honeycomb specimens that were tested.

The block shear test results showed the average shear strength of the 3.1 pcf core to be 167 pounds per square inch, while for the 2.3 pcf core, the average shear strength was 101 pounds per square inch. The average test value for core shear modulus was 7.2% greater than the listed value for 3.1 pcf core, in Reference 3, and 1.75% less for the 2.3 pcf core.

The results from the flexure tests indicate that the procedure called out in MIL-STD-401A does not give usable data for thick core honeycomb sandwich construction. The high shear loads produced in flexure tests, combined with the eccentric loading under the load pads, caused by the compressive strain of the upper skin, in thick core sandwich construction cause a premature failure due to core crushing under the load points.

The test results indicate that the materials used may be worked to a high enough stress level that a possible weight savings may be realized.over a conventional sheet-stringer structure.

4. Size Limitations

There were no deleterious size limitations experienced in the use of deep core sandwich construction except that the Flexure Test as outlined in MIL-STD-401A is not recommended as a method to determine the quality of bond for thick core sandwich specimens.

5. Quality Control

The quality control procedures for thick core sandwich construction are similar to those used for thin core sandwich construction. Basically, they are to control materials and thicknesses, dimensions for prefit, process control for cleaning and layup and bonding cycles.

The methods for determining the quality of bond are to bond and test sample coupons along side and through the same cycle as the sandwich segments. Several methods of ultrasonic testing for voids or substandard bond may be used as well as the "coin tap" method.

The contour of the bonded segments should be determined by inspection methods. The amount out of contour, or waviness limit, is determined by analytical methods for a face wrinkling mode of failure.

Core crushing will be determined by radiographic methods. If large areas of core are crushed during the bonding cycle or by handling, the segment must be either repaired by injecting resins to reinforce the buckled core, or the segment will have to be scrapped.

6. Weight

A weight study is shown in Appendix H. The loads in this analysis applied to the thick core sandwich skirt structure have been assumed since the exact loads have not been available. The estimated weight of the complete skirt assembly is 3048# and the assumed ultimate load was 9275# per running inch of circumference. It should be noted that there is not included in the weight analysis, because no information was available, any additional weight due to inserts to support any attachments or equipment to the skirt structure.

TEST PROGRAM

III. TEST PROGRAM

A. EDGEWISE COMPRESSION TESTS

1. Integral Specimens

The modes of failure in edgewise compression are intracellular buckling, face wrinkling, shear instability, general or column instability and compressive yield. A preliminary analysis indicated that none of the specimens tested in this program were either long or large enough to fail due to shear instability or general instability. The specimens were designed to become critically stressed in face wrinkling and intracellular buckling simultaneously. Face wrinkling type of failure did occur in all the specimens that were tested.

There are three separate theoretical expressions considered in this report for predicting the critical face stress for a face wrinkling type of failure. These are:

(a) $$\sigma_{cr.} = \frac{0.38\sqrt[3]{E_t E_c G_c}}{1 + 1.39(E_c G_c/E_t F_g)^2(w/t_f)}$$

(b) $$\sigma_{cr.} = \frac{1.12\sqrt{E_t E_c t_f/h}}{1 + 1.39(E_c G_c/E_t F_g)^2(w/t_f)}$$

(c) $$\sigma_{cr.} = \frac{\frac{.822 E_t t_c}{S^2(1-\mu^2)}(t_f)^3 + .202 E_c S^2}{t_f\left(t_c + .637\frac{E_c}{F_g}w\right)}$$

where: $\sigma_{cr.}$ = Critical Buckling stress, psi

E_t = Tangent Modulus of face material, psi

E_c = Compressive Modulus of core, psi

G_c = Shear Modulus of core, psi

F_g = Flatwise core to skin tension strength or core crush strength (whichever is least), psi

h = Distance, centroid to centroid of skins, inches

w = Waviness, inches

t_f = Face thickness, inches

t_c = Core thickness, inches

S = Cell size (inscribed circle diameter), inches

Equations (a) and (b) are taken from a Douglas Aircraft Company report, Paper 1393, 30 August 1962, but have been modified as a result of a previous test program performed by AVCO/ASD.

Equation (c) is taken from ANC 23, Sandwich Construction for Aircraft, Part II, and is rewritten in an easier to use form. Reference Structural Analysis of Honeycomb Sandwich Construction, AVCO Corporation, Nashville Division, March 1962.

The critical stress predicted by equation (c) correlated quite well with the Phase I test results.[1] However, with a more perfected bonding process, and reduced waviness in the Phase II specimens, equation (c) became unreliable.

In equation (c), the term ($.202E_cS^2$) constitutes only a small percent of the total numerator and hence can be considered virtually a constant. As the waviness approaches zero, the

[1] Tabulated data and correlated data are presented in Appendix A and Appendix B, respectively.

term $(.637\frac{E_c}{F_g}w)$ in the denominator approaches zero, and hence the denominator essentially consists of only two terms, t_f and t_c. In the case of very flat face sheets, the equation can now be approximated by:

$$\sigma_{cr.} = \frac{.822E_t t_c t_f^3}{S^2(1-\mu^2)t_f t_c} \qquad \text{(c-1)}$$

Simplifying, we get:

$$\sigma_{cr.} = \frac{.822E_t t_f^2}{.91S^2} \qquad \text{(c-2)}$$

It can now be seen from equation (c-2) that extremely large stresses are predicted. The tangent modulus does not exist at these stresses and hence the equation becomes useless. Also, note that the equation seems to indicate that the critical stress level is proportional to the square of the face thickness and inversely proportional to the square of the cell size. These proportionalities are disproved by the test results.

Equation (b) seems to consider all of the pertinent parameters, but it considers the critical stress level to be inversely proportional to the square root of the sandwich thickness, h. This relationship is not borne out in the test results. The test results seem to indicate that the 3.5 inch thick coupons attain the highest average stress level, with the 4.0 inch coupons second; and the 3.0 inch coupons attain the lowest average stress level.

Equation (a) predicts more accurately the critical stresses attained in the tests. The comparison of equations (a), (b), and (c) as regards correlation with test data from the phase I tests is shown in Tables 1.1 and 1.2.

Table 1.1

PHASE I TESTS

	Equation (a)	Equation (b)	Equation (c)
Percent of specimens predicted within ± 10% of test results	55.6%	11.1%	38.9%
Percent within ± 20%	72.3%	27.8%	72.3%
Percent within ± 30%	83.4%	38.9%	77.8%

Table 1.1 was compiled by using all of the test results. Of the total number of specimens tested, 4 specimens failed at less than 75% of the average failing stress of the other specimens. There were no specimens that failed in excess of 125% of the average failing stress. With these 4 specimens discarded, the percentages shown in Table 1.2 were obtained.

Table 1.2

PHASE I TESTS

	Equation (a)	Equation (b)	Equation (c)
Percent of specimens predicted within ± 10% of test results	71.5%	7.2%	50%
Percent within ± 20%	93%	21.4%	93%
Percent within ± 30%	100%	36%	100%

Tables 1.1 and 1.2 show that equation (a) fitted the data from Phase I tests more closely than equation (b) or (c). Also, it should be noted that equation (b) consistently predicted values below the test results.

Table 1.3

PHASE II TESTS

	Equation (a)	Equation (b)
Percent of specimens predicted within ± 10% of test results	20.8%	22.9%
Percent within ± 20%	68.8%	47.9%
Percent within ± 30%	77.1%	66.7%

Table 1.3 was compiled by considering all of the test results obtained in Phase II. By discarding those specimens that failed beyond - 30% of the average failing stress of all the specimens, the percentages shown in Table 1.4 were obtained.

Table 1.4

PHASE II TESTS

	Equation (a)	Equation (b)
Percent of specimens predicted within ± 10% of test results	23.8%	23.8%
Percent within ± 20%	78.6%	47.6%
Percent within ± 30%	88.1%	61.9%

Tables 1.3 and 1.4 show that equation (a) fitted the data from Phase II tests more closely than equation (b). A fact that should be noted about equations (a) and (b) is that in no case did either equation predict an ultimate stress level above 122,000 psi. Among those specimens tested in Phase II, 48.0% of them reached an ultimate stress level greater than 122,000 psi.

Conclusions:

Equation (a) was found to have the best correlation with the test data of Phase II and is subsequently preferred for use.

2. Splice-Joint Specimens

The purpose of the splice-joint specimen tests was to determine the joint strength and its effect on the overall compression load-carrying capabilities of the panel. The tests were aimed at determining an optimum type of design that would allow panels to be joined in such a manner to permit the load to be efficiently transferred from one panel to the next.

The splice-joint specimens were made in the following manner. Integral specimens were first cut in half. The two halves were then butted together and splice plates were used to join the two halves by means of a secondary bond process. Two types of adhesives were used: namely, FM-1000, an elevated-temperature-cure adhesive, and Narmco 3135, a room-temperature-setting adhesive.

Three test specimens were to be made by electron-beam welding together the face sheets of integral specimens. Difficulties were encountered in joining the specimens because the weld was contaminated by the bond adhesive of the integral specimens. Only one specimen was made available for testing. Failure of this specimen occurred as face wrinkling and core crushing at the weld. It is believed that the low stress level at failure can be attributed to a reduction in area of the face plates that occurred at one of the welds.

The specimen and loading conditions are shown in Figure 1.1.

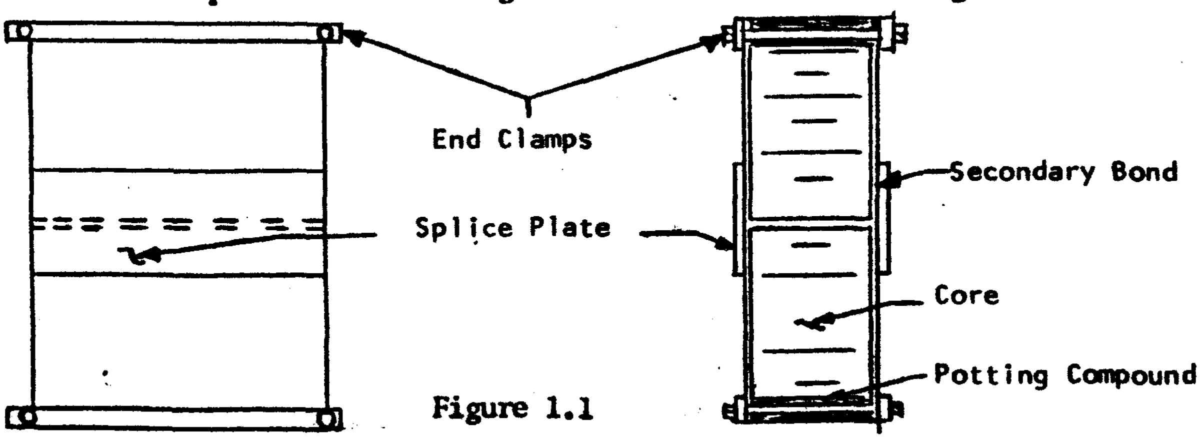


Figure 1.1

A photograph of the typical test set-up is shown in Appendix D. The ends were potted to insure that the load was introduced into the specimen uniformly. The end clamps were provided to insure against a local failure at the ends of the specimen. The Universal testing machine that was used was monitored so that the load rate would be approximately 6,000 pounds per minute. It was assumed that this rate would be slow enough to permit the specimen to relieve possible stress concentrations so that the load could be resisted in the most efficient manner. The load was increased until failure of the specimen occurred. The specimens were designed so that failure due to face wrinkling in the integral specimens, and failure of the splice-joint was expected to occur simultaneously. This design, in effect, would be an optimum-stress type design.

The test data and correlated data are listed in Appendix A and Appendix B, respectively. The majority of the specimens exhibited the expected typical type of failures. Irregularities in two of the specimens were detected prior to testing. Another sample was loaded at a rate approximately three times faster than intended. All three specimens aforementioned failed at stresses which were lower than expected. All three specimens were bonded with the 3135 adhesive. The developed shear stress in the splice adhesives at failure showed little scatter in the specimens which were bonded with the FM-1000 adhesive. The conclusion to be drawn from these tests is that splice joints can be designed so that the full strength of the pieces to be joined may be utilized.

B. FLATWISE TENSILE TESTS

These tests were conducted to determine the flatwise tensile strength of the adhesive system used to bond the face sheets to the core. The tests were performed on specimens that were bonded between heavy metal loading blocks which were pulled apart in a Universal testing machine. A self-aligning loading fixture was employed to insure against eccentric loading. The specimens were cut out of an undamaged area of the edgewise compression specimens after they had been tested. This was done to afford a better cross-reference of data to be used in the theoretical calculations. Flatwise tensile tests were conducted in Phase I only. The values obtained in these tests were used for comparison purposes in determining values for F_g to be used in the Phase I set of theoretical calculations. No flatwise tensile tests were conducted in Phase II because the flatwise tensile strength of the adhesive system was proven in the Phase I tests to be superior to the flatwise compressive strength of the core.

The results of the flatwise tensile tests are listed in Appendix A. The majority of the specimens failed in the bond, as expected. The average tensile strength of the speicmens that failed in bond only was 336 psi, with the highest strength being 487 psi and the low, 194 psi.

C. FLATWISE COMPRESSIVE TESTS

These tests were conducted in order to validate manufacturer's listed values for compressive strengths (Reference 3) for the types of honeycomb used in the edgewise ca pressive tests. The specimens were cut from an undamaged area of those specimens that had been tested in edgewise compression. Load was applied normal to the facings of the specimens through a self-aligning loading block in a Universal testing machine. Flatwise compressive tests were performed in Phase I only. The results of these tests determined the F_g values to be used in one set of theoretical calculations of ultimate stress. Upon comparing the theoretical results obtained from using the test F_g values with those results obtained by using manufacturer's listed typical values, it was determined that the difference did not warrant further flatwise compression testing. In the theoretical calculations for ultimate stress in Phase II, manufacturer's listed typical compressive strengths were used.

The results of the flatwise compressive tests are listed in Appendix A. Honeycomb cores of two different densities were tested, namely, 2.3 pcf and 3.1 pcf. Two specimens were found to give unrepresentative values for compressive strength. The remaining specimens showed the following results:

1. The manufacturer's listed compressive strength of the 2.3 pcf honeycomb was 2.4 percent greater than the average compressive strength of the 2.3 pcf honeycomb specimens that were tested.

2. The manufacturer's listed compressive strength of the 3.1 pcf honeycomb was 9.98 percent greater than the average compressive strength of the 3.1 pcf honeycomb specimens that were tested.

The differences between the compressive strengths listed in Reference 3 and those obtained in the tests had little effect on overall results of the theoretical calculations.

D. BLOCK SHEAR TESTS

Block shear tests were conducted in order that the shear strength, core shear modulus, and proportional limit shear stress of thick honeycomb cores might be determined. The basic purpose of the tests was to determine the correlation of the test results with published data for thin cores.

Steel channels were bonded to the titanium face sheets of the honeycomb specimens. A photograph of the typical test set-up is shown in Appendix D. The ends of the steel channels were loaded in compression through self-aligning blocks. The load was applied uniformly across the width of the specimens and along a line extending through diagonally opposite corners of the specimens. An extensometer was used to measure relative motion between the sandwich faces. Load-deflection curves were drawn for each specimen, from which the core shear modulus, shear strength, and proportional limit stress were evaluated. A typical load-deflection curve is shown in Appendix F.

A table of the results of the block shear tests is shown in Appendix A. The values for core shear modulus of the test specimens showed very good correlation with the values li-ted in Reference 3. For the 3.1 pcf core, the average test value for core shear modulus was 7.2% greater than the manufacturer's listed typical value (Reference 3). For the 2.3 pcf core, the average test value for core shear modulus was 1.75% less than the manufacturer's listed typical value. The average shear strength for the 3.1 pcf core was 167 psi, and for the 2.3 pcf core was 101 psi. The manufacturer's list shows no values for shear strengths for core thicknesses greater than one inch. However, all of the test values fell between manufacturer's listed values for minimum shear strength and typical shear strength. The test results indicate that the shear strength of honeycomb is reduced by a substantial percentage when the thickness of the core is increased from 1/2 inch to 3 inches.

E. FLEXURE TESTS

The flexure tests were conducted to provide a cross-check on the strength properties obtained in the block shear tests. The specimens were designed to fail in core shear and skin compression simultaneously. The specimens were loaded at two quarter-span points through a self-aligning load fixture in a Universal testing machine. Steel bearing plates were prodided at the points of load application in an attempt to reduce local stress concentrations in the test specimens at those points.

The results of the tests are shown in Appendix A. Each specimen failed at the points of load application. The failures were due to local core crushing beneath the edges of the bearing plates. The core shear strength and the compressive strength of the face sheets were not obtained from the flexure test because of the type of failure experienced by each specimen. Because of the extremely localized type of failure, and the low stress levels that were attained at failure in the tests, it does not appear that this type of flexure test, as outlined in MIL-STD-401A, produces data which can be used to determine the design characteristics for thick-core sandwich construction.

F. DYNAMIC TESTS

The purpose of the dynamic tests was to determine the dynamic structural characteristics of thick-core honeycomb sandwich construction. The primary structural characteristic to be determined was the fatigue life of the specimens under dynamic flexure loading. The dynamic test was intended to induce a fatigue failure in shear in the core rather than in bending in the face sheets. Properties such as resonant frequency, amplification factors, and resonant band width were also to be determined during the testing.

A photograph of the typical test set-up is shown in Appendix D. The specimens were fabricated with an aluminum support block at the center and steel weights of approximately 27.2 pounds each, at both ends. The honeycomb core, the titanium face sheets, the center block, and the end weights were fabricated as an integral part. The specimens were mounted at the center support block to an electrodynamic shaker. The specimens acted essentially as two cantilever beams with concentrated loads at their ends. Each specimen was subjected to a sinusoidal resonance search test, during which the resonant frequency, and amplification factor for that specimen were determined. A dwell frequency and input level was selected during the sinusoidal sweep that would produce a critical shear stress in the core of the honeycomb sandwich. Dwell sinusoidal excitation was applied at this frequency and input level until failure of the specimen occurred, or until 10^6

cycles were reached. If no failure occurred at the completion of 10^6 cycles, the vibration input was increased and the specimen was then tested until an additional 10^6 cycles were reached, or until failure occurred.

The results of the dynamic tests are summarized in Appendix A. There were four specimens tested. One specimen entered the first mode of natural resonance too fast to reduce the control input acceleration level and the honeycomb sandwich beam failed in bending. Precautions were taken to prevent this condition from existing in the remaining tests. Two of the specimens failed by shear fatigue in the core, as anticipated. The fourth sample withstood 10^6 cycles at the first sinusoidal dwell, and 10^6 cycles at the same dwell frequency, but with an increased vibration input.

The vibration input levels and the number of cycles to failure that the specimens withstood during these tests indicate that thick-core honeycomb sandwiches have good shear fatigue resistant properties. In no case did the core to face sheet adhesive system show evidence of failure, indicating that the bond is at least as strong as the core in shear fatigue.

G. THERMAL CONDUCTIVITY TESTS

The thermal conductivity tests were performed so that data could be obtained that would enable calculation of the thermal conductance of the honeycomb sandwich panels.

Photographs of the test set-up appear in Appendix D. The tests were carried out in accordance with American Society for Testing Materials Standard (ASTM C236-49T) Thermal Conductance and Transmittance of Built-up Sections by Means of the Guarded Hot Box, as referred by Military Standard MIL-STD-401A, Paragraph 5.2.5. Heat transfer through the sandwich panels was recorded with the hot face temperature stabilized at a maximum of 300°F, and the cool face temperature at 100°F. Observations of heat flow through each specimen were made over a minimum period of eight hours.

The tabulated data from the tests appears in Appendix A. The heat flow and temperature relationships used in the calculation of thermal conductance and thermal conductivity appear in Appendix G.

Due to the lack of available published data on this subject, the data from the thermal conductivity tests could not be checked for correlation with existing data.

RECOMMENDED FOLLOW-ON PROGRAM

It is proposed that the Follow-On Program be broken into three parts.

PART I

Part I will supplement the data obtained in the test program, NAS 8-11807, just completed, by providing additional data on the effects of eccentricities due to joints or panel close-out members on the compression allowables of the sandwich structure.

The task will be accomplished as follows:

At least three joint configurations will be designed. One configuration to be considered will be a design with the inner skin thicker than the outer skin. The length of the specimens must be great enough that the local eccentricities due to the joint design will be distributed through the core into the outer skin. Each design will take advantage of the data obtained from the previous tests of NAS 8-11807 and will be compatible with the required joint to mate the skirt with the adjoining structure. Each joint will have a stress analysis performed to assure that suitable load paths are provided and that the joint will develop the load the panel is required to carry.

The better two (2) designs will be tested to verify the analysis. The selection of a joint design will take into account the aforesaid requirements and will also be cognizant of the limitations from the manufacturing process and material acquisition. The test parts shall be loaded in the tests in such a manner to simulate the loading the skirt will experience. The test support structure will simulate the end conditions and flexibility as nearly as possible. Sufficient instrumentation will be used to assure that above conditions are met during the test.

There shall be three (3) test specimens of each configuration.

Recommendations for possible joint design will be made at the conclusion of this phase of the test program. A meeting will be held with the NASA Contracting Officer and NASA technical personnel to review the test results and recommendations.

PART II

Part II of the Ultra Light Honeycomb Development Program will determine the effects of curvature on the general instability mode of failure of panels of sandwich construction.

The panel sizes will be determined from an analytical study to assure that dimensions will be sufficient to allow a general instability buckle to form. The inside radius of the test panels will be approximately 16.5 feet, and the outside radius may be varied to meet the thickness requirements determined by the analysis. The ends of the test panels will simualte the end conditions and support encountered in the proposed skirt structure, and the sides will be simply supported.

It is recommended that at least three of each configuration be tested and that there be at least three configurations having the core thickness, density and skin thickness as variables.

A study will be made of several of the theoretical techniques of analysis and comparisons will be made with the test results to determine the scatter factor for future design on the skirt structure.

An optimization study will be made of various structural configurations of a cylindrical structure using the design data obtained from the test results. The study will compare structures of unreinforced, thick core, honeycomb sandwich construction with honeycomb sandwich structures reinforced with ring sections to increase the allowable stress due to general instability buckling. The optimization will compare total weight of structure with load carrying capability.

The final report will summarize data obtained during the program and will include recommendations to be used for the design of the Forward Skirt of the S-II stage.

PART III

Part III of the Ultra Light Honeycomb Development Program will consist of the design, manufacture, non-destructive inspection and tests to failure of a full scale cylinder which simulates the design of the forward skirt of the S-II Saturn V missile.

The design of the cylinder will incorporate the data obtained from NAS 8-11807 and from Parts I and II follow-on. The method of attaching to the upper and lower structure shall be the same as that used on the Forward Skirt on the proposed missile. The structure will be designed to the latest load requirements which will be furnished by Marshall Space Flight Center, NASA. The design will incorporate all attachments for equipment and cutouts for access doors which might change the load paths or change the stiffness of the structure.

A design will be made for a test fixture which will support the cylinder and provide the stiffness required to equally distribute the load from the load rams to the cylinder structure. The fixture will be capable of applying a load equal to 1-1/2 times the design ultimate applied load for the cylinder.

The cylinder will be manufactured, using the processes and specifications as set by the design. The compliance to the design will be determined by the quality control department and non-destructive testing methods will be used to determine the quality of the completed article.

The destructive tests will consist of a static load uniformly applied to the ends of the cylinder. The load will be increased in increments to the design limit load. Strain gages will be used to determine the stress levels in the structure. The stress distribution around cutouts and inserts will be determined either by strain gages placed adjacent to these locations or by a photo-elastic coating.

The cylinder will be statically loaded to failure. The load will be increased in increments to the failing load and data will be taken at each increment. This data shall be strain and deflection versus load. The cylinder structure will be torn apart after the failure in the static test and a visual comparison will be made of the quality of bond with the results obtained from the non-destructive quality control test to prove the effectiveness of the quality control methods used.

Recommendations will be made from the data and test results obtained in Part III for the design and manufacture of the Forward Skirt Structure of the S-II of the Saturn V missile.

APPENDICES

APPENDIX A

TABULATED DATA

PHASE I

HONEYCOMB SANDWICH - EDGEWISE COMPRESSION TEST SUMMARY

Specimen Serial Number	DIMENSIONAL						CORE		ULTIMATE LOAD(1)		TYPE OF FAILURE
	Length (Avg) L Inch	Width (Avg) w Inch	Sandwich Thickness (Nominal) T Inch	Skin Thickness t Inch	Stressed Area 2wt Sq. Inch		Description Density x Cell x Foil Thickness P = Perforated lb./ft.³ x inch x 0.001 inch	Ribbon Direction ↓ L → T	Load lbs.	Stress psi	Remarks: All failures caused face wrinkling.
c-1.1	11.931	6.008	3.0	.0126	.1514		3.1x1/8x7P	T	10,600	70013	(2)
c-3.1	11.966	6.004	3.0	.0188	.2258		3.1x3/16x10P	T	21,000	93002	(3)
c-5.1	11.919	6.006	3.0	.0271	.3255		2.3x1/4x10P	T	31,700	97356	(2) (3)
c-501.1	11.828	8.010	4.0	.0131	.2099		3.1x1/8x7P	T	14,800	70510	(2)
c-503.1	11.835	8.012	4.0	.0181	.2900		3.1x3/16x10P	T	23,400	80689	(3)
c-505.1	11.946	8.008	4.0	.0261	.4180		2.3x1/4x10P	T	27,000	64593	(3)
c-507.2	11.934	6.020	3.0	.0129	.1553		3.1x1/8x7P	L	15,440	99420	(3)
c-507.3	11.879	6.016	3.0	.0127	.1528		3.1x1/8x7P	L	14,600	95549	(3)
c-509.2	11.904	6.006	3.0	.0187	.2246		3.1x3/16x10P	L	23,400 27,140(4)	104185 120837	(2)
c-509.3	11.914	6.006	3.0	.0188	.2256		3.1x3/16x10P	L	9,800	43400	(2)
c-511.2	12.002	6.011	3.0	.0264	.3174		2.3x1/4x10P	L	25,800	81285	(3)
c-511.3	11.995	6.019	3.0	.0258	.3106		2.3x1/4x10P	L	26,000	83709	(3)
c-513.2	11.881	8.025	4.0	.0133	.2134		3.1x1/8x7P	L	23,800	111528	(2) (3)
c-513.3	11.939	7.951	4.0	.0129	.2051		3.1x1/8x7P	L	19,200	93613	(2) (3)
c-515.2	11.866	8.027	4.0	.0180	.2890		3.1x3/16x10P	L	27,500	95156	(3)
c-515.3	11.992	8.018	4.0	.0179	.2870		3.1x3/16x10P	L	29,200	101742	(3)
c-517.2	11.936	8.012	4.0	.0259	.4150		2.3x1/4x10P	L	44,600	107470	(3)
c-517.3	11.951	8.014	4.0	.0259	.4151		2.3x1/4x10P	L	36,700	88412	(3)

Skin to Core Bond - Three Component Adhesive System: Bloomingdale Rubber Company FM-47, BR227A and FM-61
Skin = 6 Al-4V Titanium Alloy Core = 5052 Aluminum Alloy

(1) Based on skin thickness
(2) Bond flatwise tension
(3) Core compression
(4) Specimen c-509.2 was eccentrically loaded due to being off center in the Universal Machine 0.25 inch along the three (3) inch thickness direction. By the lever arm method, the failed face was computed to have carried 58% of the total load. 1.768/3.036x100% = 58% where, 3.036 inch = thickness of sandwich -- 1.768 inch = 3.036/2 + 0.25, assuming both faces capable of each carrying a load equal to the failed face, the adjusted ultimate load with no eccentric loading would be 27,140 pound [2(23400 x .58)=27,140] where 23,400 pounds = measured ultimate eccentric load.

PHASE II

HONEYCOMB SANDWICH - EDGEWISE COMPRESSION TEST SUMMARY

Specimen Serial Number	DIMENSIONAL: Length (Avg) L Inch	Width (Avg) w Inch	Sandwich Thickness (Nominal) T Inch	Skin Thickness t(1) Inch	Stressed Area 2wt(1) Sq. Inch	CORE: Description Density x Cell x Foil Thickness P = Perforated lb./ft.³ x inch x 0.001 inch	Ribbon Direction ↑ L → T	ULTIMATE LOAD (2): Load lbs.	Stress psi	TYPE OF FAILURE Remarks: All failures caused face wrinkling.
-23-527	10.716	7.999	4.0	.018	.288	3.1X1/8X7P	L	40,700	141,319	(6) (7)
-23-529	10.642	7.825	4.0	.019	.297	3.1X3/16X10P	L	29,600	99,663	(7)
-23-563	10.855	8.004	4.0	.020	.320	3.1X1/8X7P	T	45,400	141,875	(6)
-23-565	10.630	8.028	4.0	.019	.305	3.1X3/16X10P	T	21,000	68,852	(7)
-21-523	11.835	8.005	4.0	.025	.400	3.1X1/8X7P	L	40,000	100,000	(6)
-21-525	11.800	8.004	4.0	.025	.400	3.1X3/16X10P	L	57,800	144,500	(6) (7)
-21-559	11.869	8.003	4.0	.029	.464	3.1X1/8X7P	T	50,400	108,621	(6) (7)
-21-561	11.769	8.050	4.0	.028	.451	3.1X3/16X10P	T	42,400	94,013	(6) (7)
-19-519	11.700	8.016	4.0	.029	.465	3.1X1/8X7P	L	50,000	107,527	(6)
-19-521	11.760	8.014	4.0	.029	.465	3.1X3/16X10P	L	74,200	159,570	(6) (7)
-19-555	11.800	8.011	4.0	.029	.465	3.1X1/8X7P	T	41,200	88,602	(6) (7)
-19-557	11.704	8.016	4.0	.029	.465	3.1X3/16X10P	T	68,000	146,237	(6) (7)
-17-515	11.855	7.010	3.5	.019	.266	3.1X1/8X7P	L	35,140(5)	143,353	(6) (7)
-17-517	11.689	7.006	3.5	.018	.252	3.1X3/16X10P	L	36,540(5)	145,000	(6) (7)
-17-551	11.885	7.010	3.5	.020	.280	3.1X1/8X7P	T	25,350(5)	90,536	(6) (7)
-17-553	11.841	7.010	3.5	.019	.266	3.1X3/16X10P	T	24,400	91,729	(6) (7)
-15-511	11.842	7.012	3.5	.025	.351	3.1X1/8X7P	L	57,400	163,533	(6) (7)
-15-513	11.715	7.014	3.5	.024	.337	3.1X3/16X10P	L	35,000	[illegible]	(6) (7)
-15-547	11.843	7.013	3.5	.026	.365	3.1X1/8X5P	T	46,000	126,027	(6) (7)
-15-549	11.928	7.013	3.5	.027	.379	3.1X3/16X10P	T	46,400	122,427	(6) (7)
-13-507	11.838	7.006	3.5	.030	.420	3.1X1/8X7P	L	46,000	109,524	(6) (7)
-13-509	11.844	7.013	3.5	.029	.407	3.1X3/16X10P	L	54,820(5)	134,693	(6) (7)
-13-543	11.895	7.002	3.5	.028	.392	3.1X1/8X5P	T	57,100(5)	145,663	(6) (7)
-13-545	11.748	7.014	3.5	.028	.393	3.1X3/16X10P	T	48,200	122,646	(6) (7)
-11-503	11.786	6.012	3.0	.019	.228	3.1X1/8X8P	L	31,800	139,474	(6) (7)
-11-505	11.711	6.011	3.0	.019	.228	3.1X3/16X10P	L	18,000	78,947	(6)
-11-539	11.794	6.005	3.0	.020	.240	3.1X1/8X7P	T	24,100	100,417	(6) (7)
-11-541	11.791	6.007	3.0	.019	.228	3.1X3/16X10P	T	27,800	121,930	(6) (7)
-9-5	11.798	6.012	3.0	.025	.301	3.1X1/8X7P	L	23,600	78,405	(6)
-9-501	11.800	6.005	3.0	.026	.312	3.1X3/16X10P	L	45,200(3)	144,872	(6) (7)
-9-535	11.759	6.002	3.0	.025	.300	3.1X1/8X7P	T	14,000(4)	46,667	(7)
-9-537	11.816	6.006	3.0	.025	.300	3.1X3/16X10P	T	37,400(3)	124,667	(6) (7)
-7-1	11.725	6.003	3.0	.028	.336	3.1X1/8X7P	L	44,800	133,333	(6) (7)
-7-3	11.772	6.002	3.0	.031	.372	3.1X3/16X10P	L	23,400(3)	62,903	(6) (7)
-7-531	11.820	6.001	3.0	.032	.384	3.1X1/8X7P	T	34,800	90,625	(6)
-7-533	11.804	6.003	3.0	.028	.336	3.1X3/16X10P	T	49,800	148,214	(6) (7)

Skin to Core Bond - Three Component Adhesive System: Bloomingdale Rubber Company FM-47, BR227A and FM-61
Skin = 6 AL - 4V Titanium Alloy Core = 5052 Aluminum Alloy

(1) Skin Thickness measured by AVCO prior to specimen fabrication.
(2) Based on skin thickness.
(3) Failure started at bond between potting compound and skin - Restraining clamps were omitted.
(4) Clamp slipped - Failure between potting compound and skin.
(5) Adjusted load values are tabulated. These specimens were eccentrically loaded due to being off center in the Universal Machine 0.25 inch along the specimen thickness direction. For specimen -17-515, the failed face was computed by the lever arm method to have varied 57.1% of the total load; 2.0125/3.525X100% = 57.1%, where 3.525 inch = thickness of sandwich -- 2.0125 = (3.525/2)+0.25. Assuming both faces capable of each carrying a load equal to the failed face, the adjusted ultimate load with no eccentric loading would be 38,140 pounds [2(33,400X.571)=38,140] where 33,400 pounds = measured ultimate eccentric load. Similar calculations were made for the other eccentrically loaded specimens. Observed loads for the specimens were: S/N17-515, 33,400 pounds; S/N17-517, 32,000 pounds; S/N17-551, 22,200 pounds; S/N13-509, 48,000 pounds; and S/N13-543, 50,000 pounds.
(6) Core compression.
(7) Bond flatwise tension.

PHASE II

HONEYCOMB SANDWICH - EDGEWISE COMPRESSION TEST SUMMARY (CONTINUED)

Specimen Serial Number	DIMENSIONAL Length (Avg) L Inch	Width (Avg) w Inch	Sandwich Thickness (Nominal) T Inch	Skin Thickness t(1) Inch	Stressed Area 2wt(1) Sq. Inch	CORE Description Density x Cell x Foil Thickness P = Perforated lb./ft.³ x inch x 0.001 inch	Ribbon Direction ∥ L ⊥ T	ULTIMATE LOAD (2) Load lbs.	Stress psi	TYPE OF FAILURE Remarks: All failures caused face wrinkling.
-15-503	29.875	7.999	4.0	.019	.304	3.1X1/8X7P	L	30,000	98,684	(4) (6)
-15-505	29.671	8.007	4.0	.019	.304	3.1X3/16X10P	L	23,200	76,316	(4)
-15-515	29.921	8.012	4.0	.020	.320	3.1X1/8X7P	T	31,800	99,375	(4)
-15-517	29.828	8.005	4.0	.019	.304	3.1X3/16X10P	T	37,600	123,684	(4) (5)
-17-3	29.890	8.015	4.0	.025	.401	3.1X1/8X7P	L	33,800	84,269	(4) (5) (6)
-17-501	29.765	7.995	4.0	.024	.384	3.1X3/16X10P	L	32,400	84,375	(4) (5)
-17-509	29.859	7.990	4.0	.024	.384	3.1X1/8X7P	T	35,800	93,229	(4) (5) (6)
-17-513	29.859	7.990	4.0	.024	.384	3.1X3/16X10P	T	35,200(3)	91,667	(4) (5)
-19-1	29.859	8.014	4.0	.030	.481	3.1X1/8X7P	L	61,800	128,482	(4) (5)
-19-5	29.796	8.011	4.0	.030	.481	3.1X3/16X10P	L	59,800	124,324	(4) (5)
-19-507	29.828	8.013	4.0	.026	.417	3.1X1/8X7P	T	49,000	117,506	(4) (6)
-19-511	29.890	8.006	4.0	.028	.448	3.1X3/16X10P	T	64,800	144,643	(4) (5)

Skin = 6 AL - 4V Titanium Alloy   Core = 5052 Aluminum Alloy

Skin to Core Bond - Three Component Adhesive System: Bloomingdale Rubber Company FM-47, BR227A and FM-61

(1) Skin Thickness measured by AVCO prior to specimen fabrication.
(2) Based on skin thickness.
(3) Core pre-compressed near end.
(4) Core compression.
(5) Bond flatwise tension.
(6) Core flatwise tension.

HONEYCOMB SANDWICH - SPLICE JOINT COMPRESSION TEST SUMMARY

Specimen Serial Number	SKIN Length (Avg.) L Inch	SKIN Width (Avg.) w Inch	SKIN Thickness (Avg.) t Inch	Sandwich Thickness (Nominal) T Inch	Stressed Area 2wt Sq. Inch	Core Ribbon Direction	SPLICE PLATE Length (Avg.) Inch	SPLICE PLATE Thickness (Avg.) Inch	Splice Adhesive	ULTIMATE LOAD Load Lbs.	ULTIMATE LOAD Stress PSI	TYPE OF FAILURE
J-113-2	10.984	5.445	.0115	3.0	.1197	T	2.0	.020	FM-1000(a)	15,400	123,003	(1) (2)
J-118-1	11.015	5.893	.018	3.0	.1252	L	3.0	.026	FM-1000	25,000	117,758	(1) (2) (3)
J-120-2	10.745	5.867	.019	3.0	.2123	T	3.0	.026	FM-1000	27,500	123,374	(1) (2) (3)
J-125-2	11.809	5.745	.024	3.0	.2229	T	4.0	.026	FM-1000	32,600	118,202	(1) (2) (3) (4)
J-132-2	11.850	5.880	.0275	3.0	.2120	T	5.0	.028	FM-1000	39,300	121,521	(3) (4)**
J-213-3	10.985	5.701	.0105	3.0	.2758	L	2.0	.0195	3135(b)	8,400(c)	70,175	(1) (2)
J-220-1	10.731	5.890	.018	3.0	.2842	L	3.0	.026	3135	11,300(d)	53,302	(1) (2)
J-225-3	11.887	5.799	.0245	3.0	.3234	L	4.0	.026	3135	22,900	80,577	(4)
J-232-3	11.816	5.83	.029	3.0	.3381	L	5.0	.029	3135	24,300(e)	71,872	(4)
J-332-1	11.784	5.89	.02875	3.0	.3367	L	---	----	EBW*	22,500	66,430	***

Skin = 6Al-4V Titanium Alloy   Core = 3.1 x 1/8 x 7P - 5052 Aluminum Alloy
Skin to Core Bond - Three Component Adhesive System: Bloomingdale Rubber Company FM-47, BR227A and FM-61

* Electron Beam Welded by Hamilton-Standard.
** Splice plates wrinkled.
*** Wrinkle at weld.

(1) Face wrinkling.
(2) Core compression.
(3) Core flatwise tension.
(4) Splice plate bond failure.

(a) Bloomingdale FM-1000 Adhesive
(b) Narmco 3135 Adhesive
(c) Specimen subjected to excessive load rate
(d) Flaw in specimen detected before testing
(e) Flaw in specimen detected before testing

Specimen Serial Number	Splice Bond Area (in^2)	Ultimate Load (lbs)	Developed Adhesive Shear Stress (psi)
J-113-2	10.890	15,400	1,414
J-118-1	17.694	25,000	1,413
J-120-2	17.602	27,500	1,562
J-125-2	22.980	32,600	1,419
J-132-2	29.400	39,300	1,337
J-213-3	11.402	8,400	737
J-220-1	17.670	11,300	646
J-225-3	23.116	22,900	991
J-232-3	29.150	24,300	834

FLATWISE TENSION TEST
PHASE I COMPRESSION TEST SPECIMENS

SPECIMEN NUMBER	CORE THK. IN.	AREA $IN.^2$	ULT. LOAD LBS.	CORE TYPE	TENSILE P.S.I.	REMARKS
FT-1A	4.0	4.04	840	*	208	Face Sht. to Adhesive Failure
FT-1B	4.0	4.06	788	*	194	Face Sht. to Adhesive Failure
FT-3A	4.0	4.04	1840	**	456	Face Sht. to Adhesive Failure
FT-3B	4.0	4.09	1670	**	408	Face Sht. to Adhesive Failure
FT-5A	4.0	4.02	2210	***	550	20% Adh. to Face; 80% Core Failure
FT-5B	4.0	4.00	2210	***	552	20% Adh. to Face; 80% Core Failure
FT-C-1.1	3.0	4.06	1600	***	394	Face Sht. to Adhesive Failure
FT-C-3.1	3.0	4.04	970	**	243	Face Sht. to Adhesive Failure
FT-C-5.1	3.0	4.04	1325	*	328	Face Sht. to Adhesive Failure
FT-C-501.1	4.0	4.02	1955	***	487	Face Sht. to Adhesive Failure
FT-C-503.1	4.0	4.05	1710	**	422	Face Sht. to Adhesive Failure
FT-C-505.1	4.0	4.02	1000	*	249	Face Sht. to Adhesive Failure
FT-C-507.2	3.0	4.02	1670	***	416	Face Sht. to Adhesive Failure
FT-C-507.3	3.0	4.04	1240	***	307	Face Sht. to Adhesive Failure
FT-C-509.3	3.0	4.00	1220	**	305	Face Sht. to Adhesive Failure
FT-C-511.2	3.0	4.06	1150	*	283	Face Sht. to Adhesive Failure
FT-C-511.3	3.0	4.04	1265	*	314	Face Sht. to Adhesive Failure
FT-C-513.2	4.0	4.00	2500	***	625	25% Adh. to Face Sht;75% Core Failure
FT-C-513.3	4.0	4.08	2170	***	531	25% Adh. to Face Sht;75% Core Failure
FT-C-515.2	4.0	4.04	1400	**	347	Face Sht. to Adhesive Failure
FT-C-515.3	4.0	4.04	1755	**	435	Face Sht. to Adhesive Failure
FT-C-517.2	4.0	4.06	970	*	239	Face Sht. to Adhesive Failure
FT-C-517.3	4.0	3.97	1405	*	354	Face Sht. to Adhesive Failure

* 2.3 - 1/4 - 10P - 5052
** 3.1 - 3/16 - 10P - 5052
*** 3.1 - 1/8 - 7P - 5052

FLATWISE COMPRESSION TEST
PHASE I COMPRESSION TEST SPECIMENS

SPECIMEN NUMBER	CORE THK. IN.	AREA IN.^{2}	CORE TYPE	ULT. LOAD LBS.	COMPRESSION P.S.I.
FC-1	4.0	4.10	*	546	133
FC-3	4.0	4.10	**	766	187
FC-5	4.0	4.08	***	854	209
FC-C-501.1	4.0	4.06	***	848	209
FC-C-503.1	4.0	4.02	**	980	244
FC-C-505.1	4.0	4.00	*	712	178
FC-C-513.2	4.0	4.02	***	1185	295
FC-C-513.3	4.0	4.05	***	1335	329
FC-C-515.2	4.0	4.08	**	1060	260
FC-C-515.3	4.0	4.00	**	1060	265
FC-C-517.2	4.0	3.95	*	585	148
FC-C-517.3	4.0	4.02	*	720	179
FC-C-1.1	3.0	4.02	***	860	214
FC-C-3.1	3.0	4.03	**	860	213.5
FC-C-5.1	3.0	4.04	*	655	162
FC-507.2	3.0	4.03	***	1180	292
FC-507.3	3.0	4.00	***	1030	258
FC-C-509.3	3.0	4.02	**	1158	288
FC-C-511.2	3.0	4.080	*	624	154
FC-C-511.3	3.0	4.080	*	716	176

* 2.3 - 1/4 - 10P - 5052
** 3.1 - 3/16 - 10P - 5052
*** 3.1 - 1/8 - 7P - 5052

HONEYCOMB SANDWICH - BLOCK SHEAR TEST SUMMARY

Specimen Serial Number	DIMENSIONAL Length (Avg.) L	Width (Avg.) W	Sandwich Thickness (Avg.) T	Skin Thickness (Nominal) t	Stressed Area WL	CORE Description Density-Cell -Foil Thkness P=Perforated	Ribbon Direction	Proportional Limit Load	Proportional Limit Stress	Ultimate Load Load	Ultimate Load Stress	Shear Modulus
	Inch	Inch	Inch	Inch	Sq. Inch	lb./ft.^{3}-x inch x 0.0001 inch		lbs.	psi	lbs.	psi	psi
S-1.1	36.06	6.008	3.044	0.020	216.5	3.1-1/8-7P	L	29,400	136	36,000	166	46,400
S-1.2	36.00	6.000	3.045	0.020	216.0	3.1-1/8-7P	L	30,700	142	36,500	169	46,400
S-3.1	36.00	6.002	3.043	0.020	216.1	3.1-3/16-10P	L	24,000	111	37,300	173	39,400
S-3.2	36.00	5.968	3.047	0.020	214.8	3.1-3/16-10P	L	28,900	135	36,900	172	47,600
S-5.1	36.00	5.959	3.045	0.020	214.5	2.3-1/4-10P	L	15,900	74	21,000	98	29,100
S-5.2	36.00	6.014	3.038	0.020	216.5	2.3-1/4-10P	L	9,700	45	21,400	99	39,700
S-501.1	48.00	8.050	4.049	0.020	386.5	3.1-1/8-7P	L	—	—	64,400	167	45,900
S-503.1	48.18	8.031	4.040	0.020	386.9	3.1-3/16-10P	L	52,600	136	59,500	154	43,200
S-505.1	48.12	8.050	4.053	0.020	387.3	2.3-1/4-10P	L	35,300	91	40,900	106	20,000
S-505.2	48.18	8.047	4.035	0.020	387.7	2.3-1/4-10P	L	—	—	39,600	102	31,100

Skin = 6 Al - 4V Titanium Alloy
Core = 5052 Aluminum Alloy
Bond = FM 61 Adhesive
PR-227A Primer

HONEYCOMB SANDWICH - FLEXURE TEST SUMMARY

SPECIMEN SERIAL NUMBER	CORE TYPE	SPAN LENGTH INCH	SKIN THICKNESS INCH	ULTIMATE LOAD TOTAL LOAD LB.	ULTIMATE LOAD SHEAR STRESS* PSI	ULTIMATE LOAD COMPRESSIVE STRESS** PSI
2-10072-1	3.1-3/16-10P	51.00	.013	8180	102	91,400
2-10072-3	2.3-1/4-10P	52.88	.013	4660	58	60,200
2-10072-5	3.1-1/8-7P	54.00	.025	8000	100	57,400
2-10072-501	3.1-3/16-10P	54.00	.025	9980	125	70,500

Skin = 6Al-4V Titanium Alloy   Core = 5052 Aluminum Alloy

Skin to Core Bond - Three Component Adhesive System:
Bloomingdale Rubber Company FM-47, BR227A, and FM-61

* Computed core shear stress at ultimate load

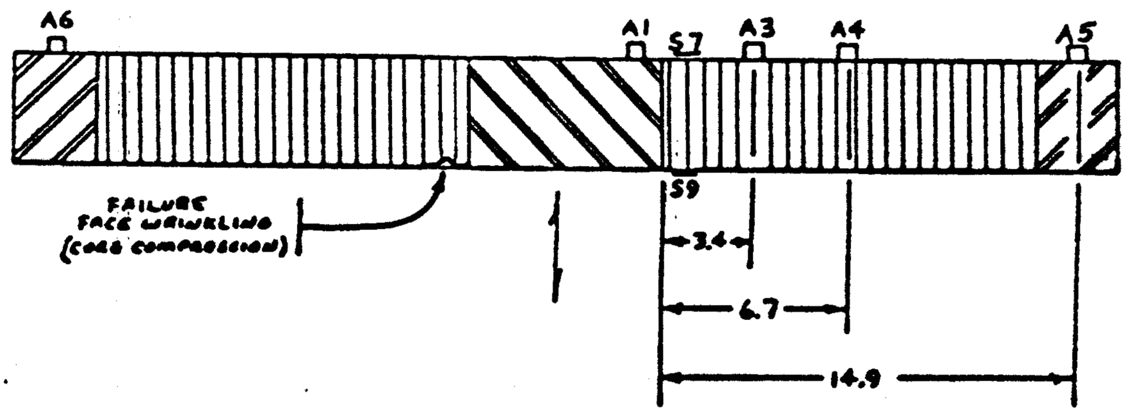
** Computed skin compressive stress at ultimate load

DYNAMIC TEST SPECIMEN CONSTRUCTION

Avco Dwg. Number	Thickness (Inch)	Density (lb./ft.^{3})	Cell Size (Inch)	Thickness (Inch) Perforated	Ribbon Direction Beam. (T or L)
2-10071-1	0.020	2.3	1/4	0.0010 P	L
2-10071-3	0.020	3.1	3/16	0.0010 P	L
2-10071-5	0.020	3.1	1/8	0.0007 P	L
2-10071-505	0.020	3.1	1/8	0.0007 P	T

Note: All specimens were constructed of 6AL-4V Titanium Alloy skin and 5052 Aluminum Alloy core. The skin and core were bonded using Bloomingdale Rubber Company adhesive, Model FM61. All specimens were perforated.

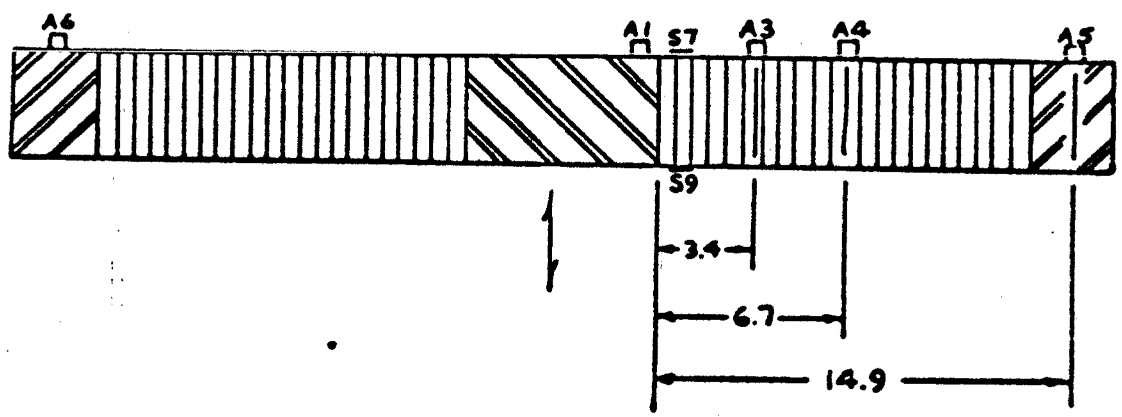
DYNAMIC TEST DATA, S/N 2-10071-505



Frequency (cps)	Acceleration (g peak) Input A_1	Response A_3	A_4	A_5	A_6	Band-width (cps)	Strain (Microinch Per Inch) S_7	S_9
70 sweep	5	15	35	100	95	40	Broken	8,500
800 sweep	1.5	-	50	85	-	25	Broken	<100
1610 sweep	15	250	285	<5	-	-	Broken	<150

Note: 1) Except at 1610 cps, all accelerometers are in phase.

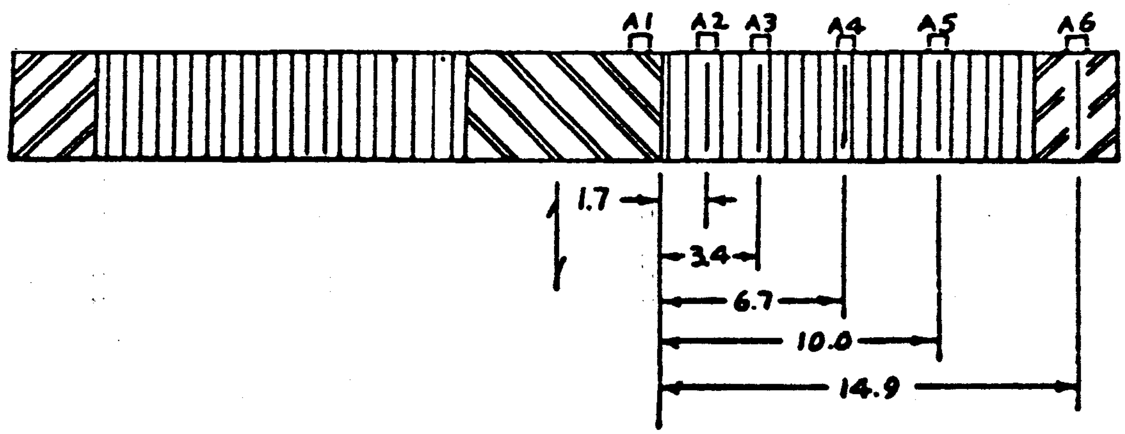
DYNAMIC TEST DATA, S/N 2-10071-1



Frequency (cps)	Acceleration (g peak) Input A_1	Response A_3	A_4	A_5	A_6	Band-width (cps)	Strain (Microinch Per Inch) S_7	S_9
70 sweep	0.7	5	12	24	13	40	2,000	2,700
820 sweep	1.5	28	45	5	2	25	100	100
820 dwell 10^6 cycles	7.2	175	320	26	-	-	250	320
820 dwell 6 minutes (Failure)	50	190	405	75	-	-	625	800

Note: 1) All accelerometers were in phase.

DYNAMIC TEST DATA, S/N 2-10071-5

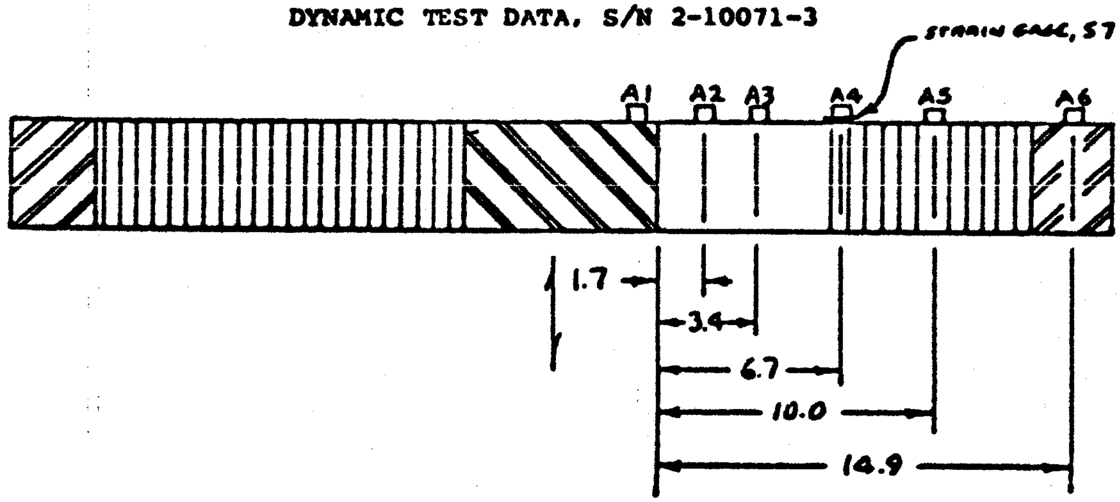


Frequency (cps)	Acceleration (g peak) Input A_1	Response A_2	A_3	A_4	A_5	A_6	Band-width (cps)	Strain (Microinch Per Inch) S_7	S_9
870 sweep	1	13	26	50	55	2.5	25	None	None
870 dwell	6.2	85	180	315	400	13	-	None	None
870 dwell Input Increased Failure after 8.5 Minutes	20	150	440	-	920	26	-	None	None

Note: 1) All accelerometers were in phase.

2) First resonance not recorded. Full sweep not conducted to avoid damage to specimens in first resonance

DYNAMIC TEST DATA, S/N 2-10071-3



Frequency (cps)	Acceleration (g peak) Input A_1	Response A_2	A_3	A_4	A_5	A_6	Band-width (cps)	Strain (Microinch Per Inch) S_7
850 sweep	1	13	26	46	50	1	25	Not plugged in.
850 dwell 10^6 cycles	6.5	110	210	390	420	16.5	-	Not plugged in.
850 dwell 10^6 cycles	20	150	355	620	1000(2)	28	-	1375

No Failure

Note: 1) All accelerometers were in phase.

2) Accelerometer fell off after eight seconds at high level input.

3) First resonance not recorded. Full sweep not conducted to avoid damage to specimen in first resonance.

THERMAL CONDUCTIVITY TEST SUMMARY

Serial Number	Core Density x Cell Size x Foil Thickness (lb/ft^3 x in. x 0.001 in.) (P=Perforated)	Test Duration (Hours)	Thermal Conductivity (k) (1)	Thermal Conductance (c) (2)	Temperatures (F) Hot Face (t_1)	Cold Face (t_2)	Mean $(t_1+t_2)/2$	Heat Input q (3) (Btu/hr.)
7-1	3.1x1/8x7P	8	12.736	3.184	176.5	114.2	145.4	198.5
7-3	3.1x3/16x10P	14	13.560	3.390	170.3	108.6	139.5	209.3
7-5	2.3x1/4x10P	8	11.460	2.865	173.8	108.0	140.9	188.67

Skin to Core Bond - Three Component Adhesive System: Bloomingdale Rubber Company FM-47, BR227A and FM-61

Skin = 6Al-4V Titanium Alloy Core = 5052 Aluminum Alloy

(1) k = Thermal Conductivity, Btu/hr./sq.ft. deg. Fahr./in.

(2) c = Thermal Conductance, Btu/hr./sq.ft. deg. Fahr.

(3) q = Time Rate of heat flow through area A, Btu/hr.

APPENDIX B

CORRELATED DATA

PHASE I

EDGEWISE COMPRESSION

DATA FOR THEORETICAL CALCULATIONS*

SPECIMEN NUMBER	SKIN THICKNESS INCH	SANDWICH THICKNESS INCH	CELL SIZE INCH	E_c PSI	WAVINESS INCH	TEST F_g PSI	MFG. TYPICAL F_g PSI
C-1.1	.0126	3.0	.1250	77000	.010	214	282
C-3.1	.0188	3.0	.1875	77000	.014	213.5	282
C-5.1	.0271	3.0	.2500	55000	.013	162	170
C-501.1	.0131	4.0	.1250	77000	.015	209	282
C-503.1	.0181	4.0	.1875	77000	.024	244	282
C-505.1	.0261	4.0	.2500	55000	.012	178	170
C-507.2	.0129	3.0	.1250	77000	.010	292	282
C-507.3	.0127	3.0	.1250	77000	.009	258	282
C-509.2	.0187	3.0	.1875	77000	.014	**	282
C-509.3	.0188	3.0	.1875	77000	.022	288	282
C-511.2	.0264	3.0	.2500	55000	.0135	154	170
C-511.3	.0258	3.0	.2500	55000	.016	176	170
C-513.2	.0133	4.0	.1250	77000	.011	295	282
C-513.3	.0129	4.0	.1250	77000	.016	329	282
C-515.2	.0180	4.0	.1875	77000	.014	260	282
C-515.3	.0179	4.0	.1875	77000	.014	265	282
C-517.2	.0259	4.0	.2500	55000	.012	148	170
C-517.3	.0259	4.0	.2500	55000	.016	179	170

** No coupon was available for testing.

MATERIALS - Skins - 6Al-4V Titanium - Solution Heat Treated and Aged
Core - 5052 Aluminum Alloy

* G_c = 31,900 psi

PHASE I - EDGEWISE COMPRESSION - TEST VS. THEORETICAL

SPECIMEN NUMBER	ACTUAL ULTIMATE STRESS	THEORETICAL ULTIMATE STRESS* (A)[1]	(B)[1]	(C)[2]	THEORETICAL/TEST PERCENT (A)	(B)	(C)
C-1.1	70,013	82,410	51,389	86,892	117.7	73.4	124.1
C-3.1	93,002	84,156	64,102	74,800	90.5	68.9	80.4
C-5.1	97,388	80,808	73,901	78,626	83.0	75.9	80.7
C-501.1	70,510	69,507	38,274	86,897	98.6	54.3	123.2
C-503.1	80,689	74,696	48,348	64,472	92.6	59.9	79.9
C-505.1	64,593	87,568	68,002	91,334	135.6	105.3	141.4
C-507.2	99,420	99,547	62,811	102,700	100.1	63.2	103.3
C-507.3	95,549	95,788	59,969	99,066	100.3	62.8	103.7
C-509.2	104,185	- - No Test F_g Value - -					
C-509.3	43,400	88,358	67,303	68,919	203.6	155.1	158.8
C-511.2	81,285	75,714	68,342	71,280	93.1	84.1	87.7
C-511.3	83,709	78,023	69,622	66,960	93.2	83.2	80.0
C-513.2	111,528	98,522	54,664	115,415	88.3	49.0	103.5
C-513.3	93,613	93,934	51,329	99,374	100.3	54.8	106.2
C-515.2	95,156	93,830	60,564	84,799	98.6	63.6	89.1
C-515.3	101,742	94,659	60,930	84,571	93.0	59.9	83.1
C-517.2	107.470	76,314	59,088	82,489	71.0	55.0	76.8
C-517.3	88,412	79,183	61,309	78,482	89.6	69.3	88.8

1 Theory from DESIGN AND TESTING OF HONEYCOMB SANDWICH CYLINDERS UNDER AXIAL COMPRESSION, Douglas Aircraft Company, August 1962.

2 Theory from STRUCTURAL ANALYSIS OF HONEYCOMB SANDWICH CONSTRUCTION, AVCO Corporation, Nashville Division, March 1962. (Taken from ANC 23, Sandwich Construction for Aircraft, Part II)

* F_g test values were used in the theoretical calculations.

PHASE I - EDGEWISE COMPRESSION - TEST VS. THEORETICAL

SPECIMEN NUMBER	ACTUAL ULTIMATE STRESS	THEORETICAL ULTIMATE STRESS* (A)[1]	(B)[1]	(C)[2]	THEORETICAL/TEST PERCENT (A)	(B)	(C)
C-1.1	70,013	97,364	60,715	97,025	139.1	86.7	138.6
C-3.1	93,002	98,867	75,308	85,552	106.3	81.0	92.0
C-5.1	97,388	83,688	76,535	80,783	85.9	78.6	82.9
C-501.1	70,510	87,787	48,340	98,879	124.5	68.6	140.2
C-503.1	80,689	83,555	54,082	69,599	103.6	67.0	86.3
C-505.1	64,593	84,932	66,013	89,429	131.5	102.2	138.4
C-507.2	99,420	97,925	61,787	101,410	98.5	62.1	102.0
C-507.3	95,549	100,002	62,607	102,226	104.7	65.5	107.0
C-509.2	104,185	98,743	75,013	84,730	94.8	72.0	81.3
C-509.3	43,400	87,173	66,400	68,114	200.9	153.0	156.9
C-511.2	81,285	81,787	73,824	75,463	100.6	90.8	92.8
C-511.3	83,709	75,865	67,696	65,576	90.6	80.9	78.3
C-513.2	111,528	96,364	53,466	113,769	86.4	47.9	102.0
C-513.3	93,613	85,507	46,724	93,549	91.3	49.9	99.9
C-515.2	95,156	97,846	63,157	87,515	102.8	66.4	92.0
C-515.3	101,742	97,713	62,896	86,624	96.0	61.8	85.1
C-517.2	107,470	84,708	65,585	88,179	78.8	61.0	82.0
C-517.3	88,412	75,986	58,833	76,368	85.9	66.5	86.4

[1] Theory from DESIGN AND TESTING OF HONEYCOMB SANDWICH CYLINDERS UNDER AXIAL COMPRESSION, Douglas Aircraft Company, August 1962.

[2] Theory from STRUCTURAL ANALYSIS OF HONEYCOMB SANDWICH CONSTRUCTION, AVCO Corporation, Nashville Division, March 1962. (Taken from ANC 23, Sandwich Construction for Aircraft, Part II)

* Mfg. F_g typical values were used in the theoretical calculations.

PHASE II - EDGEWISE COMPRESSION DATA FOR THEORETICAL CALCULATIONS

CONSTANTS: E_c = 77,000 psi , Typical F_g = 282 psi
G_c = 31,900 psi*

SPECIMEN NUMBER	SKIN THICKNESS INCH	SANDWICH THICKNESS INCH	CELL SIZE INCH	WAVINESS INCH
-23-527	.018	4.0	.1250	.005
-23-529	.019	4.0	.1875	.005
-23-563	.020	4.0	.1250	.005
-23-565	.019	4.0	.1875	.005
-21-523	.025	4.0	.1250	.005
-21-525	.025	4.0	.1875	.005
-21-559	.029	4.0	.1250	.005
-21-561	.028	4.0	.1875	.005
-19-519	.029	4.0	.1250	.005
-19-521	.029	4.0	.1875	.005
-19-555	.029	4.0	.1250	.005
-19-557	.029	4.0	.1875	.005
-17-515	.019	3.5	.1250	.005
-17-517	.018	3.5	.1875	.005
-17-551	.020	3.5	.1250	.0110
-17-553	.019	3.5	.1875	.010
-15-511	.025	3.5	.1250	.010
-15-513	.024	3.5	.1875	.0108
-15-547	.026	3.5	.1250	.0100
-15-549	.027	3.5	.1875	.005
-13-507	.030	3.5	.1250	.005
-13-509	.029	3.5	.1875	.005
-13-543	.028	3.5	.1250	.005
-13-545	.028	3.5	.1875	.005
-11-503	.019	3.0	.1250	.005
-11-505	.019	3.0	.1875	.005
-11-539	.020	3.0	.1250	.005
-11-541	.019	3.0	.1875	.005
-9-5	.025	3.0	.1250	.005
-9-501	.026	3.0	.1875	.005
-9-535	.025	3.0	.1250	.005
-9-537	.025	3.0	.1875	.005
-7-1	.028	3.0	.1250	.005
-7-3	.031	3.0	.1875	.005
-7-531	.032	3.0	.1250	.005
-7-533	.028	3.0	.1875	.005
-15-503	.019	4.0	.1250	.005
-15-505	.019	4.0	.1875	.005
-15-515	.020	4.0	.1250	.005
-15-517	.019	4.0	.1875	.005

P. E. Pigue	AVCO CORPORATION	PAGE NO. 5 OF 7
PREPARED BY	AEROSPACE STRUCTURES DIVISION	Appendix B
W. A. Griswold	NASHVILLE, TENNESSEE 37202	REPORT NO. R-1050
CHECKED BY		
DATE 18 Oct. 1965	NAS 8-11807	MODEL NO. M.A. 5501

PHASE II - EDGEWISE COMPRESSION DATA FOR THEORETICAL CALCULATIONS

(CONTINUED)

SPECIMEN NUMBER	SKIN THICKNESS INCH	SANDWICH THICKNESS INCH	CELL SIZE INCH	WAVINESS INCH
-17-3	.025	4.0	.1250	.005
-17-501	.024	4.0	.1875	.005
-17-509	.024	4.0	.1250	.005
-17-513	.024	4.0	.1875	.005
-19-1	.030	4.0	.1250	.005
-19-5	.030	4.0	.1875	.005
-19-507	.026	4.0	.1250	.005
-19-511	.028	4.0	.1875	.005

MATERIALS - Skins - 6Al-4V Titanium - Solution Heat Treated and Aged

Core - 5052 Aluminum Alloy

* G_c = Average of manufacturer's typical values for longitudinal ribbon direction and transverse ribbon direction.

PHASE II - EDGEWISE COMPRESSION - TEST VS. THEORETICAL

SPECIMEN NUMBER	ACTUAL ULTIMATE STRESS	THEORETICAL ULTIMATE STRESS* (A)[1]	THEORETICAL ULTIMATE STRESS* (B)[1]	THEORETICAL/TEST PERCENT (A)	THEORETICAL/TEST PERCENT (B)
-23-527	141,319	115,915	74,662	82.0	52.8
-23-529	99,663	116,544	77,115	116.9	77.4
-23-563	141,875	117,117	79,497	82.5	56.0
-23-565	68,852	116,544	77,115	169.3	112.0
-21-523	100,000	119,344	90,516	119.3	90.5
-21-525	144,500	119,344	90,516	82.6	62.6
-21-559	108,621	120,610	98,474	111.0	90.7
-21-561	94,013	120,325	96,545	128.0	102.7
-19-519	107,527	120,610	98.474	112.2	91.6
-19-521	159,570	120,610	98,474	75.6	61.7
-19-555	88,602	120,610	98,474	136.1	111.1
-19-557	146,237	120,610	98,474	82.5	67.3
-17-515	143,383	116,560	82,635	81.3	57.6
-17-517	145,000	115,930	79,996	80.0	55.2
-17-551	90,536	105,331	76,614	116.3	84.6
-17-553	91,729	106,176	75,273	115.7	82.1
-15-511	163,533	110,919	90,201	67.8	55.2
-15-513	105,638	108,991	86,843	103.2	82.2
-15-547	126,027	111,526	92,491	88.5	73.4
-15-549	122,427	120,039	101,447	98.0	82.9
-13-507	109,524	120,895	107,698	110.4	98.3
-13-509	134,693	120,628	105,654	89.6	78.4
-13-543	145,663	120,343	103,571	82.6	71.1
-13-545	122,646	120,325	103,159	98.1	84.1
-11-503	139,474	116,544	88,975	83.6	63.8
-11-505	78,947	116,544	88,975	147.6	112.7
-11-539	100,417	117,117	91,720	116.6	91.3
-11-541	121,930	116,544	88,975	95.6	73.0
-9-5	78,405	119,344	104,411	152.2	133.2
-9-501	144,872	119,694	106,774	82.6	73.7
-9-535	46,667	119,344	104,411	255.7	223.7
-9-537	124,667	119,344	104,411	95.7	83.8
-7-1	133,333	120,325	111,351	90.2	83.5
-7-3	62,903	121,128	117,887	192.6	187.4
-7-531	90,625	121,364	119,985	133.9	132.4
-7-533	148,214	120,325	111,351	81.2	75.1
-15-503	98,684	116,544	77,115	118.1	78.1
-15-505	76,316	116,544	77,115	152.7	101.0
-15-515	99,375	117,117	79,497	117.9	80.0
-15-517	123,684	116,544	77,115	94.2	62.3

PHASE II - EDGEWISE COMPRESSION - TEST VS. THEORETICAL

(CONTINUED)

SPECIMEN NUMBER	ACTUAL ULTIMATE STRESS	THEORETICAL ULTIMATE STRESS* (A)[1]	(B)[1]	THEORETICAL/TEST PERCENT (A)	(B)
-17-3	84,289	119,344	90,516	141.6	107.4
-17-501	84,375	118,967	88,418	141.0	104.8
-17-509	93,229	118,967	88,418	127.6	94.8
-17-513	91,667	118,967	88,418	129.8	96.5
-19-1	128,482	120,877	100,367	94.1	78.1
-19-5	124,324	120,877	100,367	97.2	80.7
-19-507	117,506	119,694	92,568	101.9	78.8
-19-511	144,643	120,325	96,545	83.2	66.7

* No theoretical values were obtained by equation (c).

1 Theory from DESIGN AND TESTING OF HONEYCOMB SANDWICH CYLINDERS UNDER AXIAL COMPRESSION, Douglas Aircraft Company, August 1962.

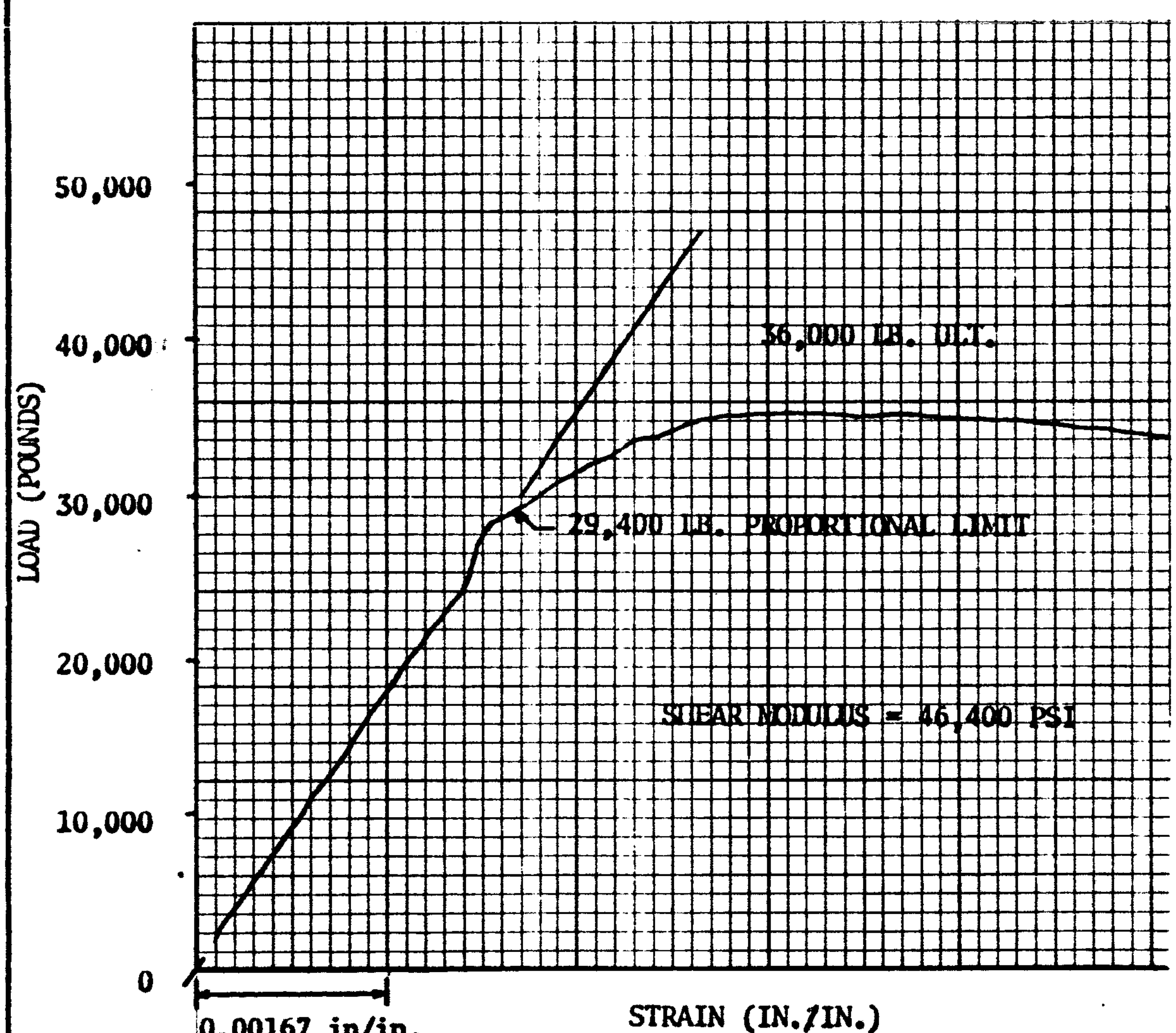
APPENDIX C

LOAD-STRAIN CURVES

BLOCK SHEAR TEST

LOAD VS. SHEAR STRAIN

TYPICAL CURVE

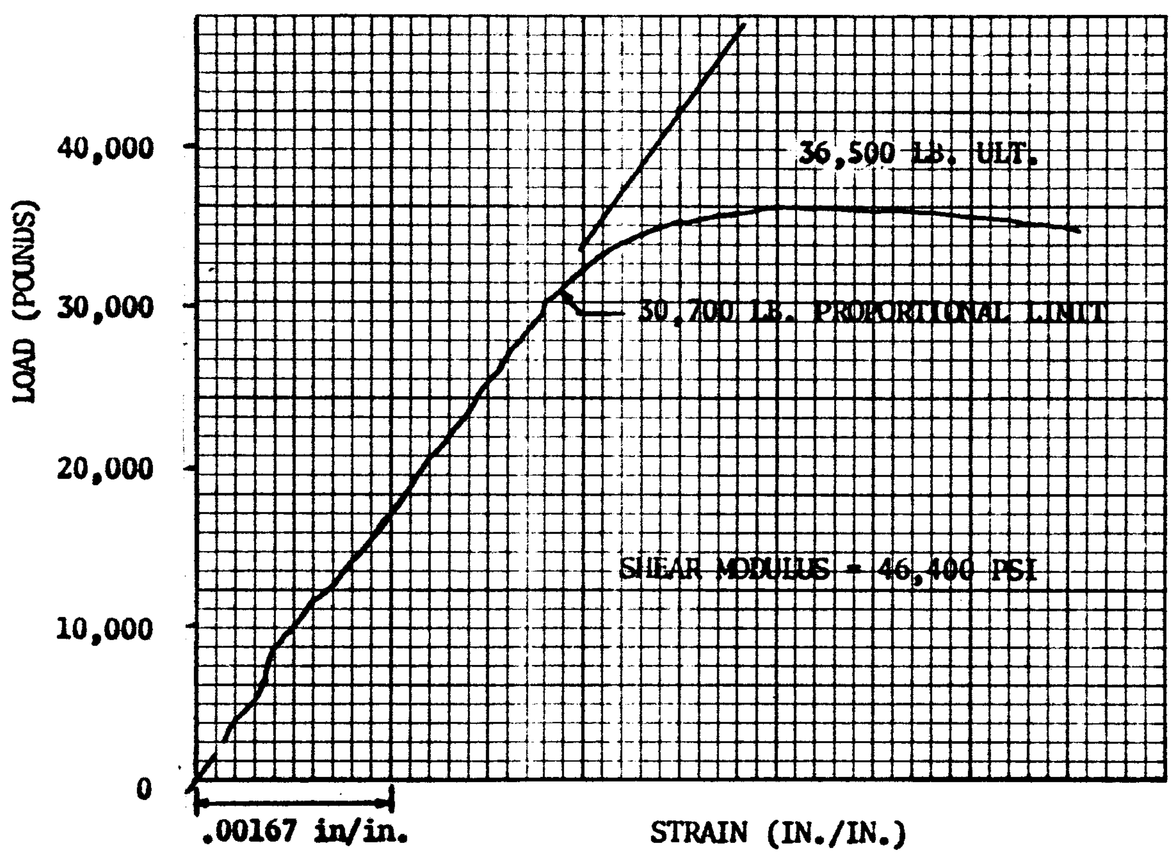


SPECIMEN NO. S-1.1
CORE - 5052 Aluminum Alloy
3.1 pcf - 1/8 Cell Size
SKINS - 6AL-4V Titanium Alloy
0.020 in. Thickness
RIBBON DIRECTION - Longitudinal

BLOCK SHEAR TEST

LOAD VS. SHEAR STRAIN CURVE

TYPICAL CURVE

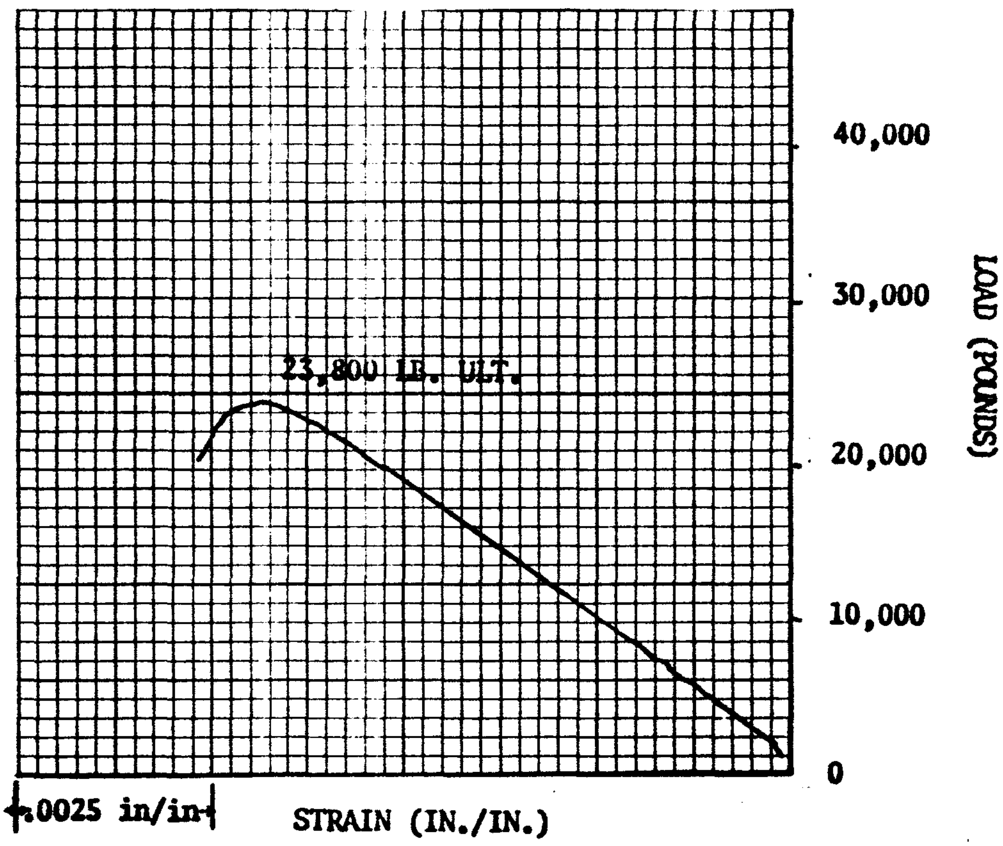


SPECIMEN NO. S-1.2
CORE - 5052 Aluminum Alloy
3.1 pcf - 1/8 Cell Size
SKINS - 6AL-4V Titanium Alloy
0.020 in. Thickness
RIBBON DIRECTION - Longitudinal

EDGEWISE COMPRESSION TEST

LOAD VS. STRAIN

TYPICAL CURVE

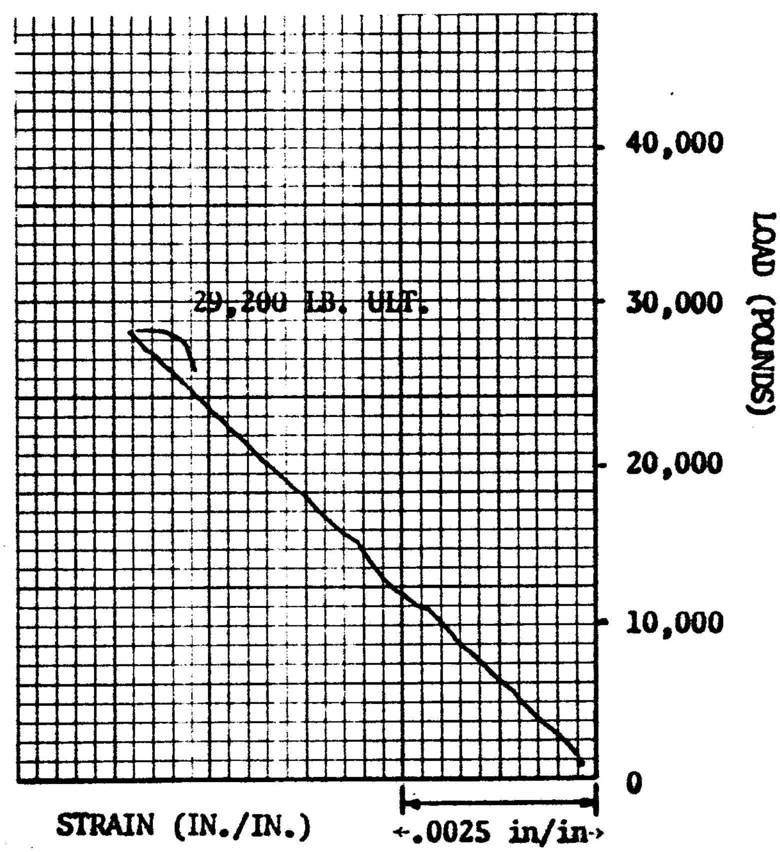


SPECIMEN NO. C-513.2
CORE - 5052 Aluminum Alloy
3.1 pcf - 1/8 Cell Size
SKINS - 6AL-4V Titanium Alloy
0.012 in. Thickness
RIBBON DIRECTION - Longitudinal

EDGEWISE COMPRESSION TEST

LOAD VS. STRAIN

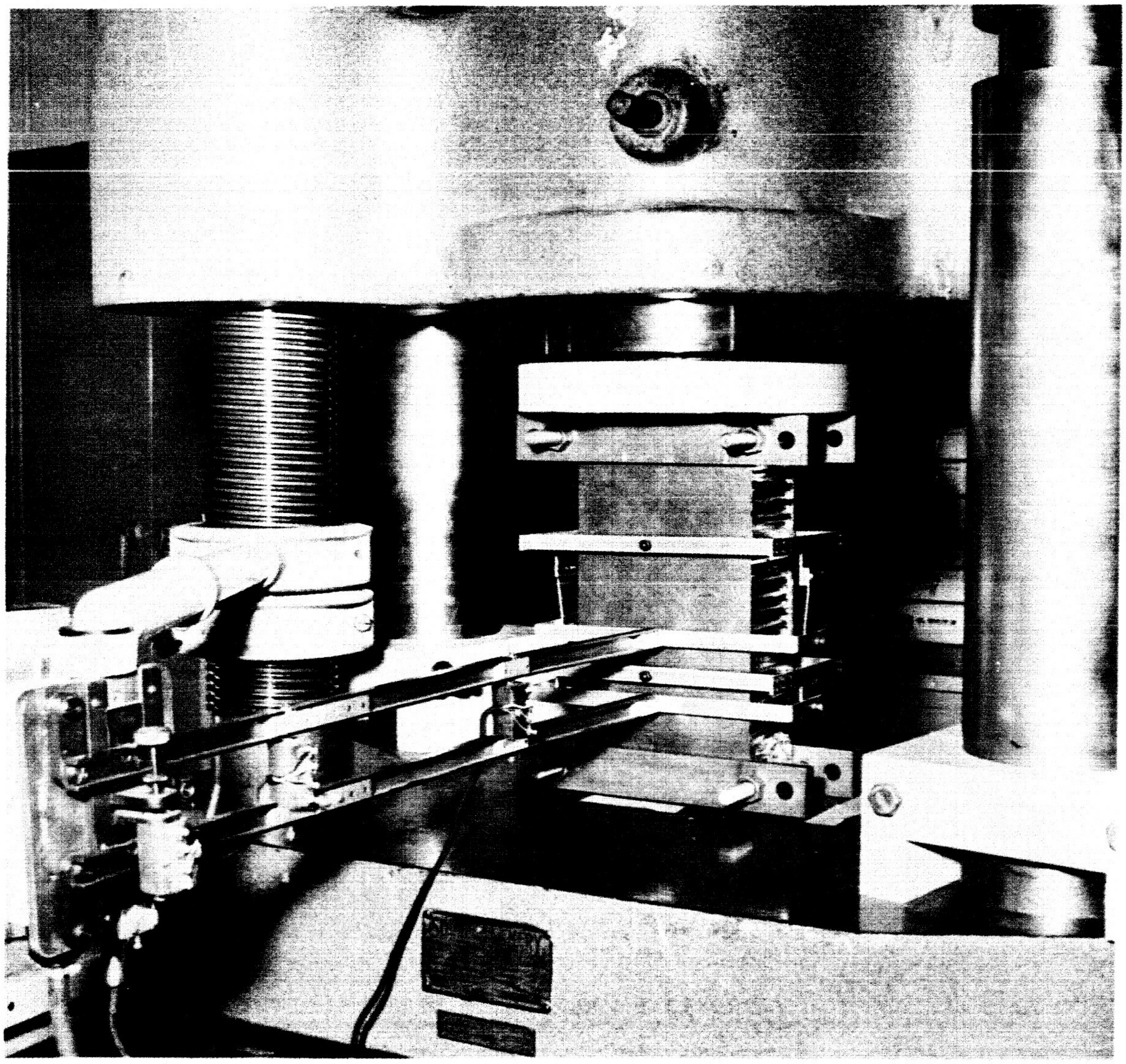
TYPICAL CURVE



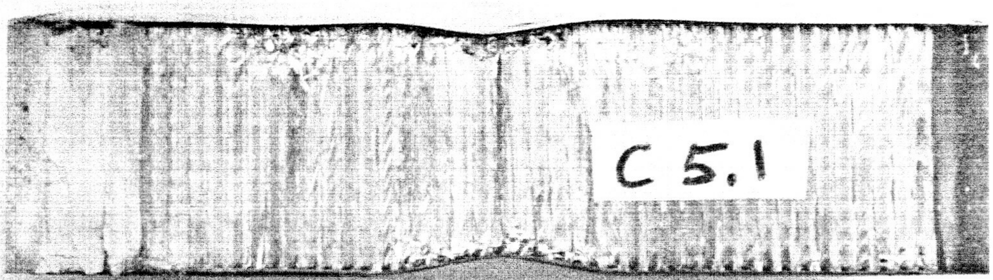
SPECIMEN NO. C-515.3
CORE - 5052 Aluminum Alloy
3.1 pcf - 3/16 Cell Size
SKINS - 6AL-4V Titanium Alloy
0.018 in. Thickness
RIBBON DIRECTION - Longitudinal

APPENDIX D

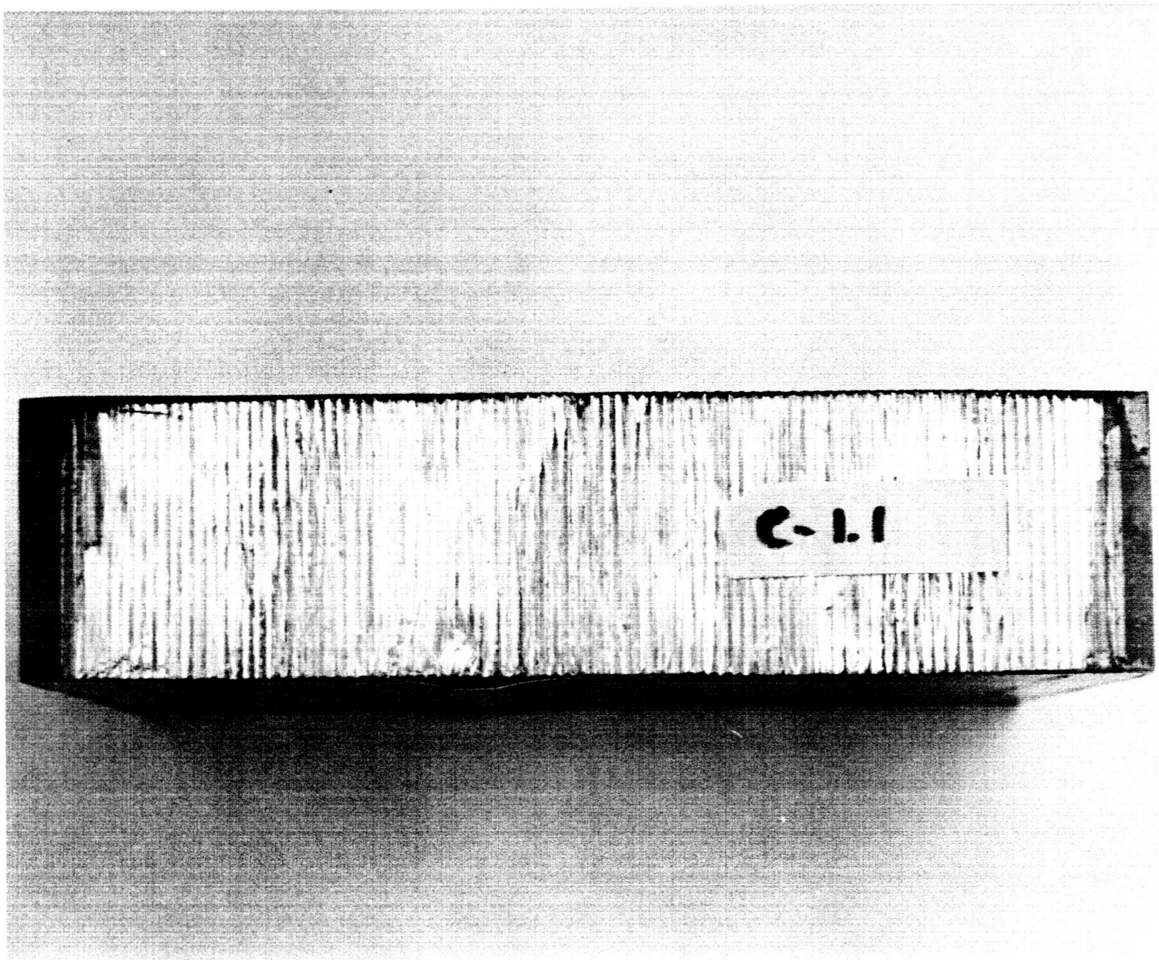
PHOTOGRAPHS



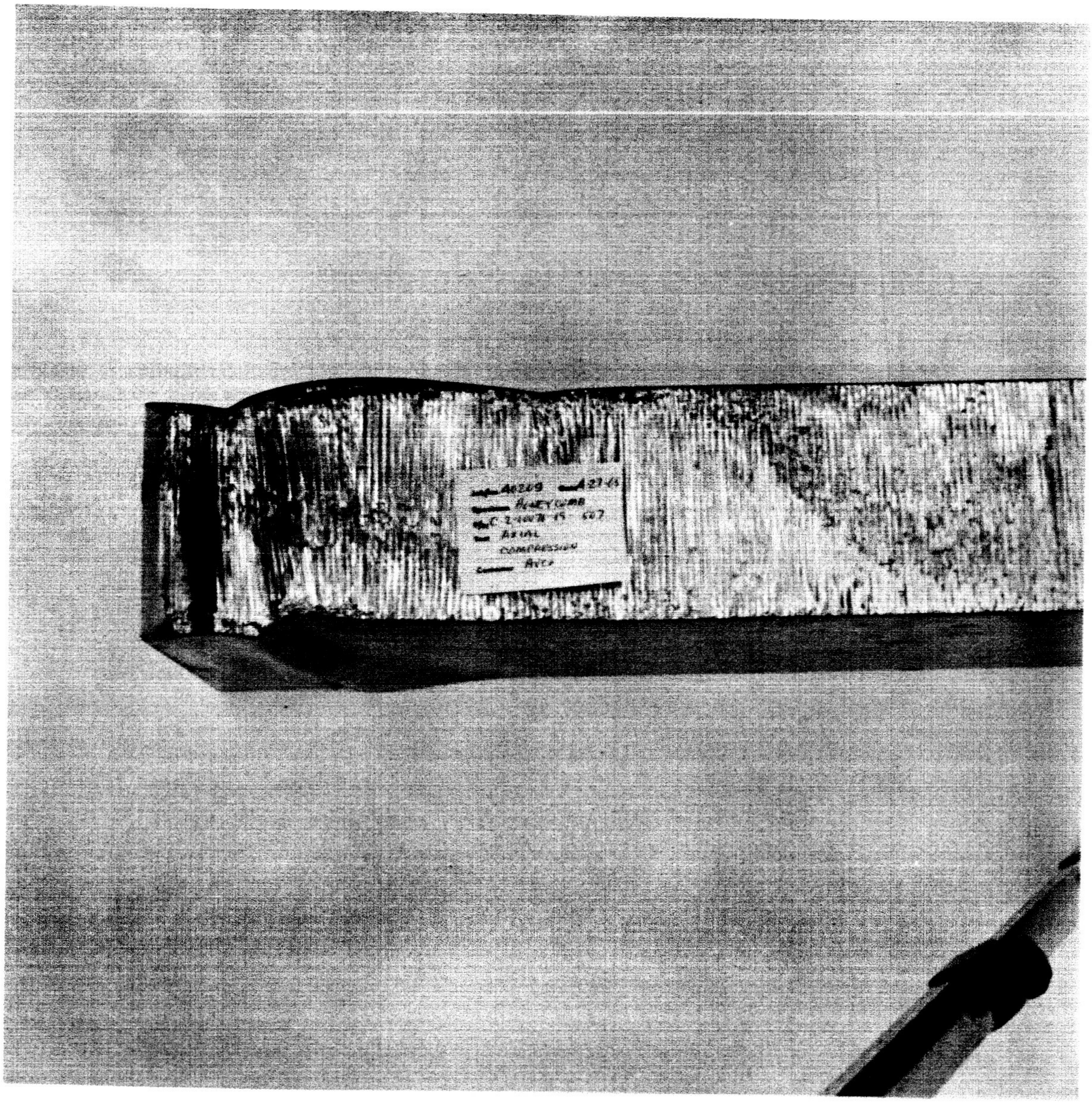
EDGEWISE COMPRESSION TEST
INTEGRAL SPECIMEN
TYPICAL TEST SETUP



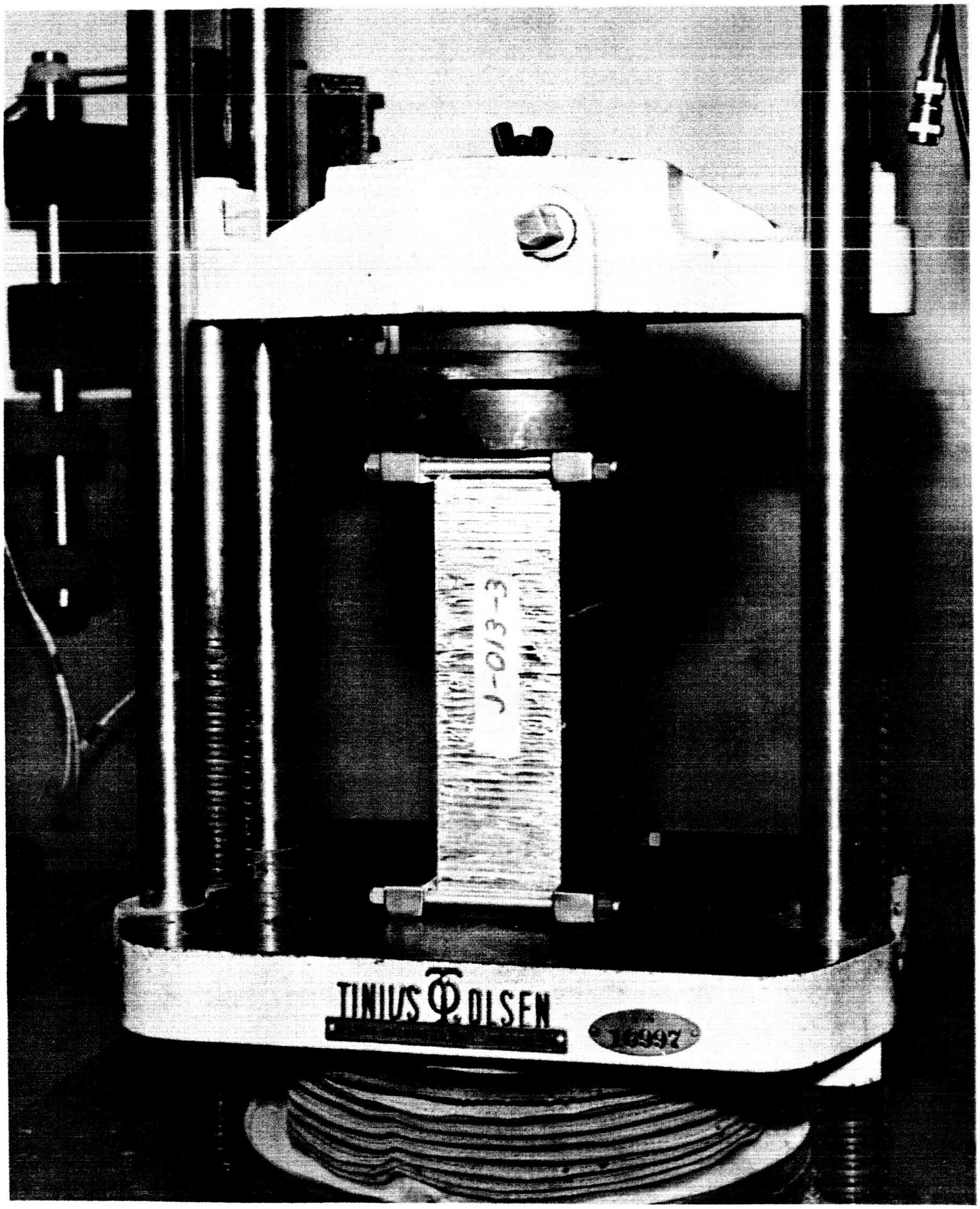
EDGEWISE COMPRESSION TEST
INTEGRAL SPECIMEN
TYPICAL CORE CRUSH FAILURE



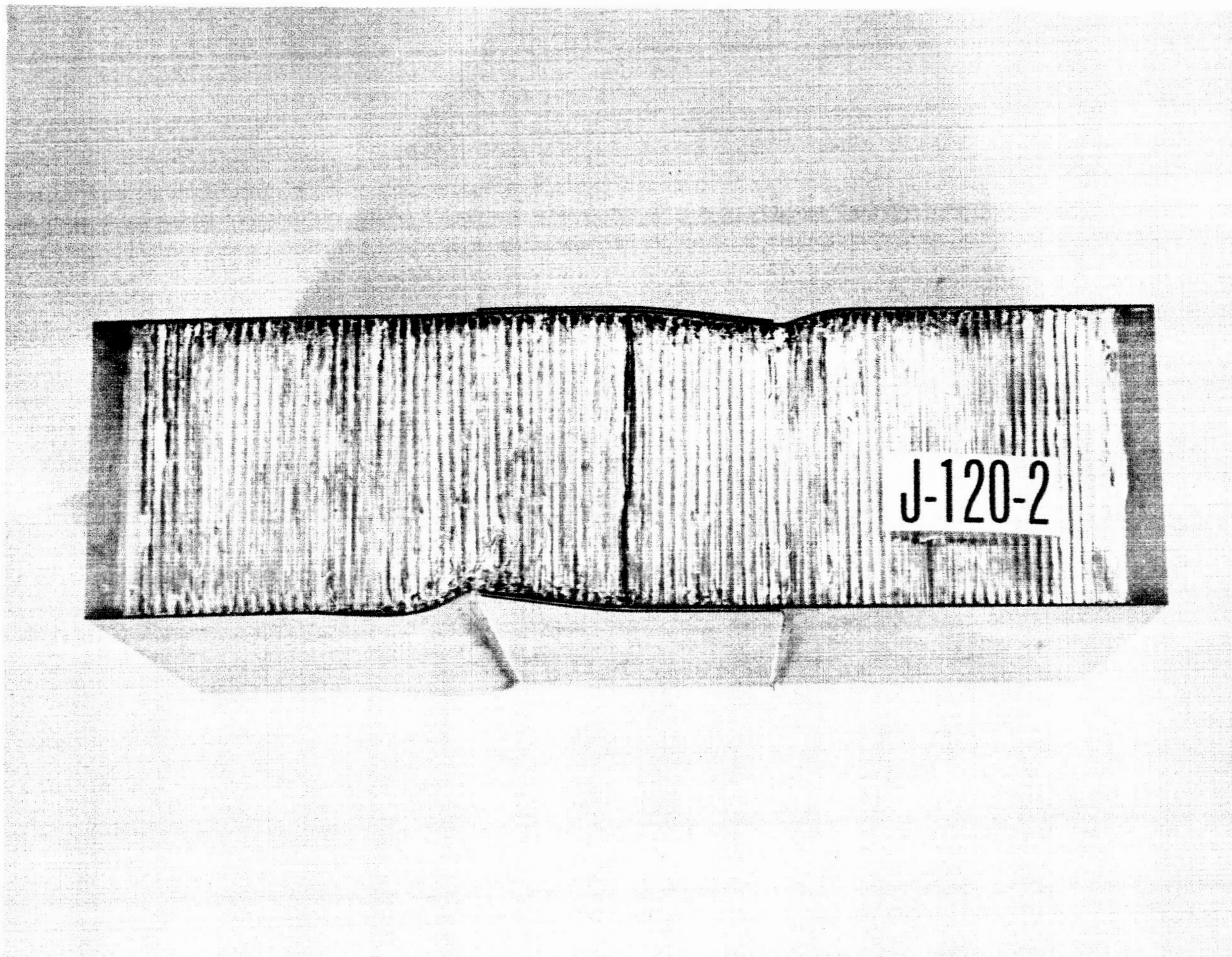
EDGEWISE COMPRESSION TEST
INTEGRAL SPECIMEN
TYPICAL BOND FLATWISE TENSION FAILURE



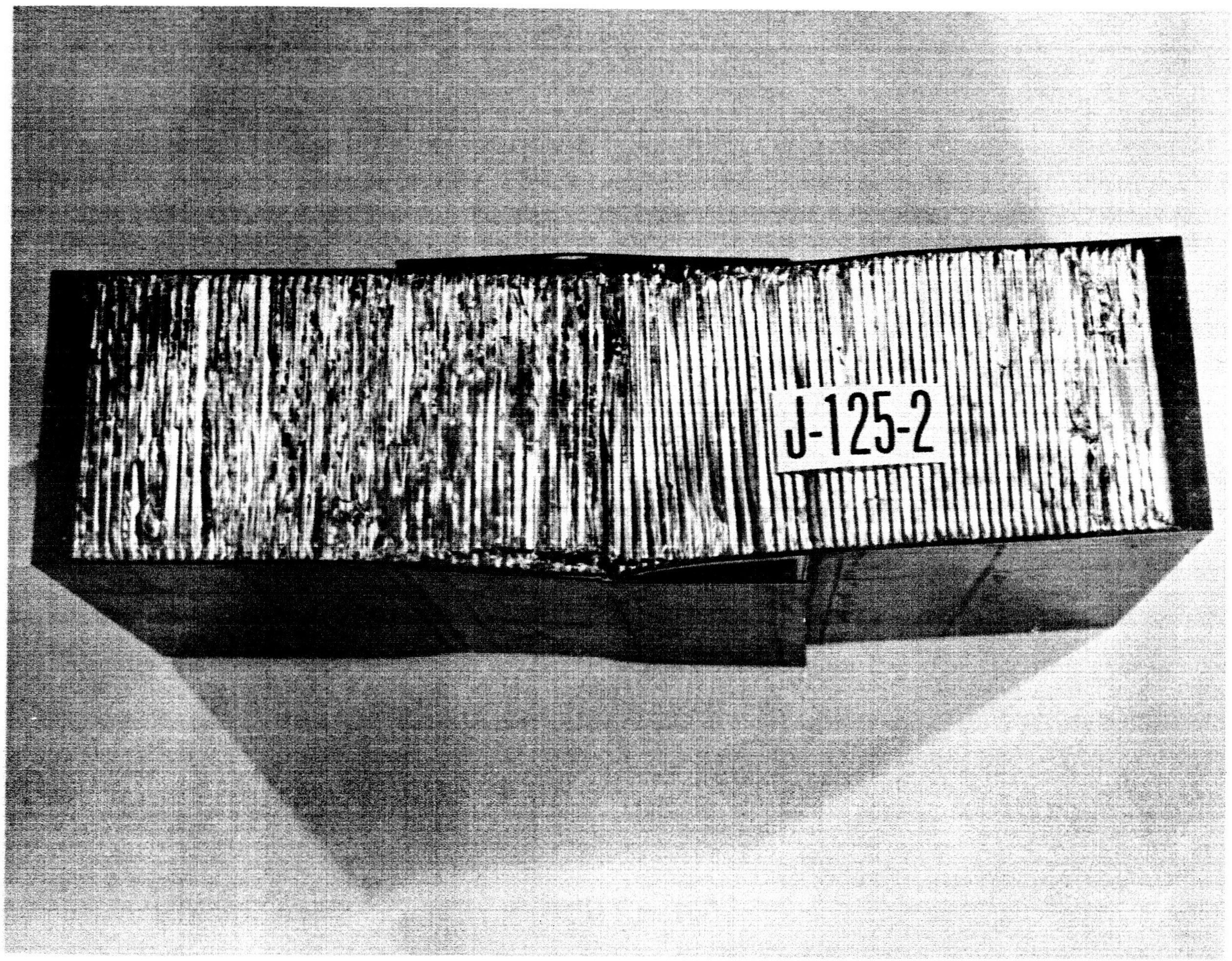
EDGEWISE COMPRESSION TEST
INTEGRAL SPECIMEN
TYPICAL CORE FLATWISE TENSION FAILURE



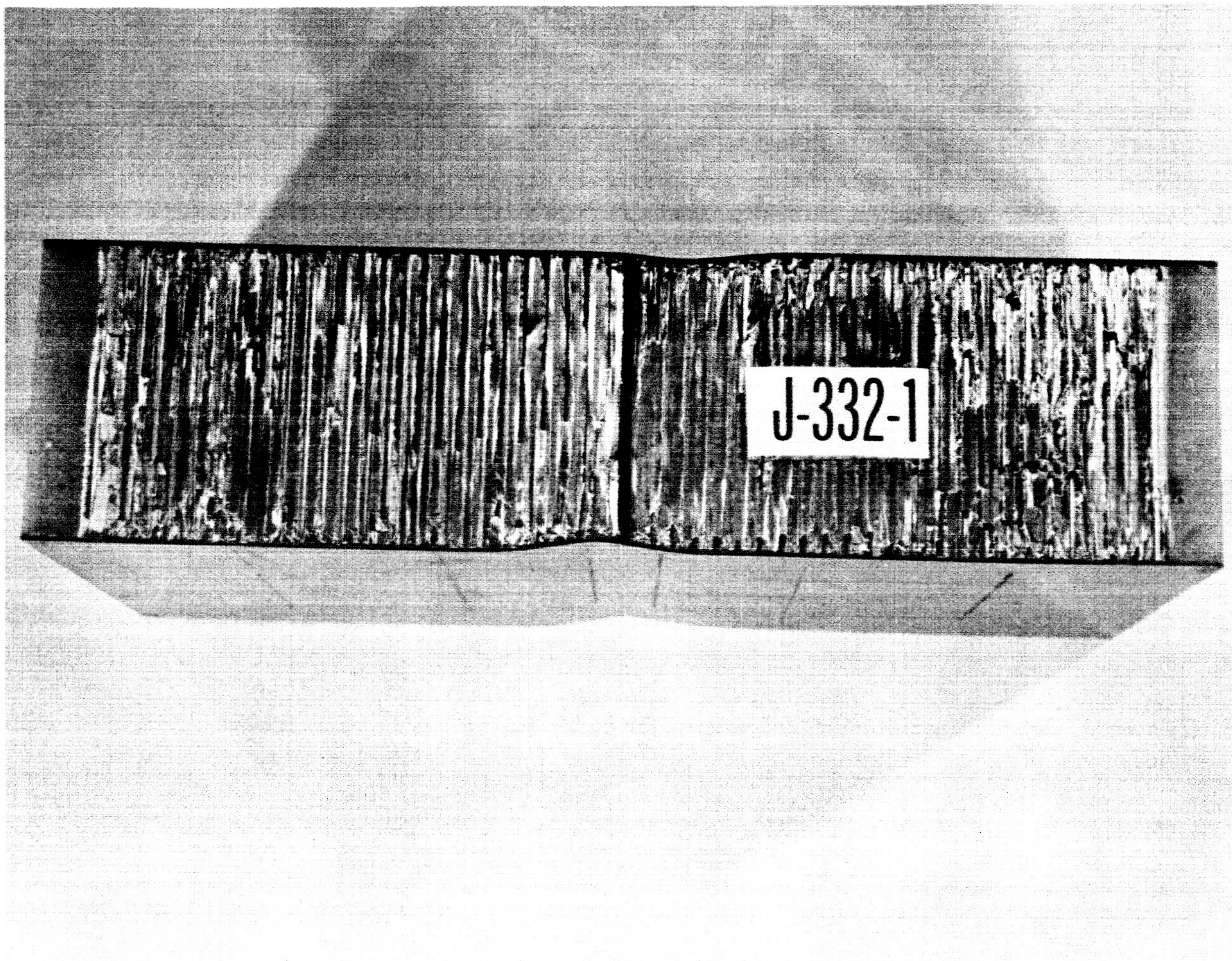
EDGEWISE COMPRESSION TEST
SPLICE JOINT SPECIMEN
TYPICAL TEST SETUP



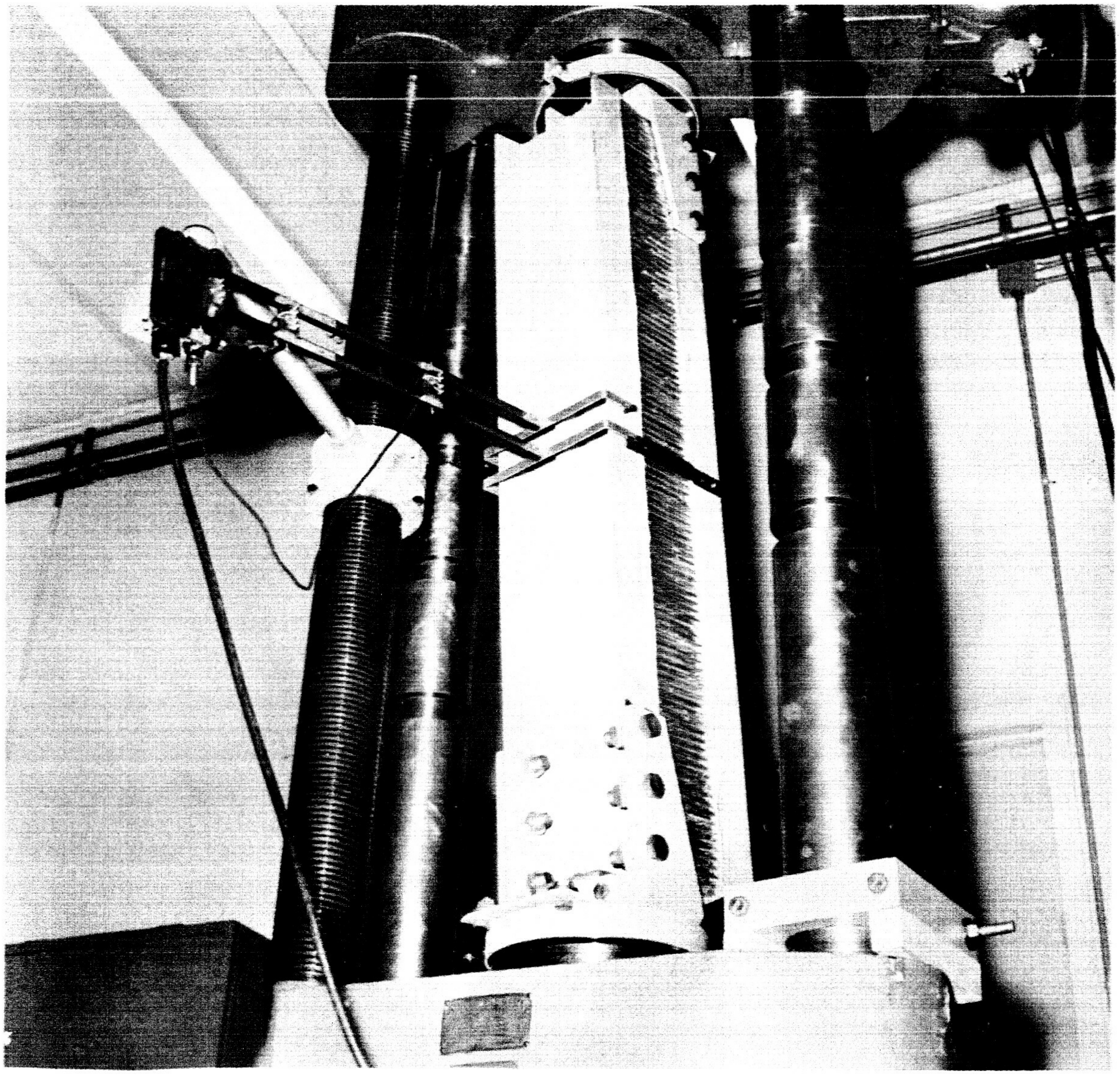
EDGEWISE COMPRESSION TEST
SPLICE JOINT SPECIMEN
TYPICAL CORE CRUSH FAILURE



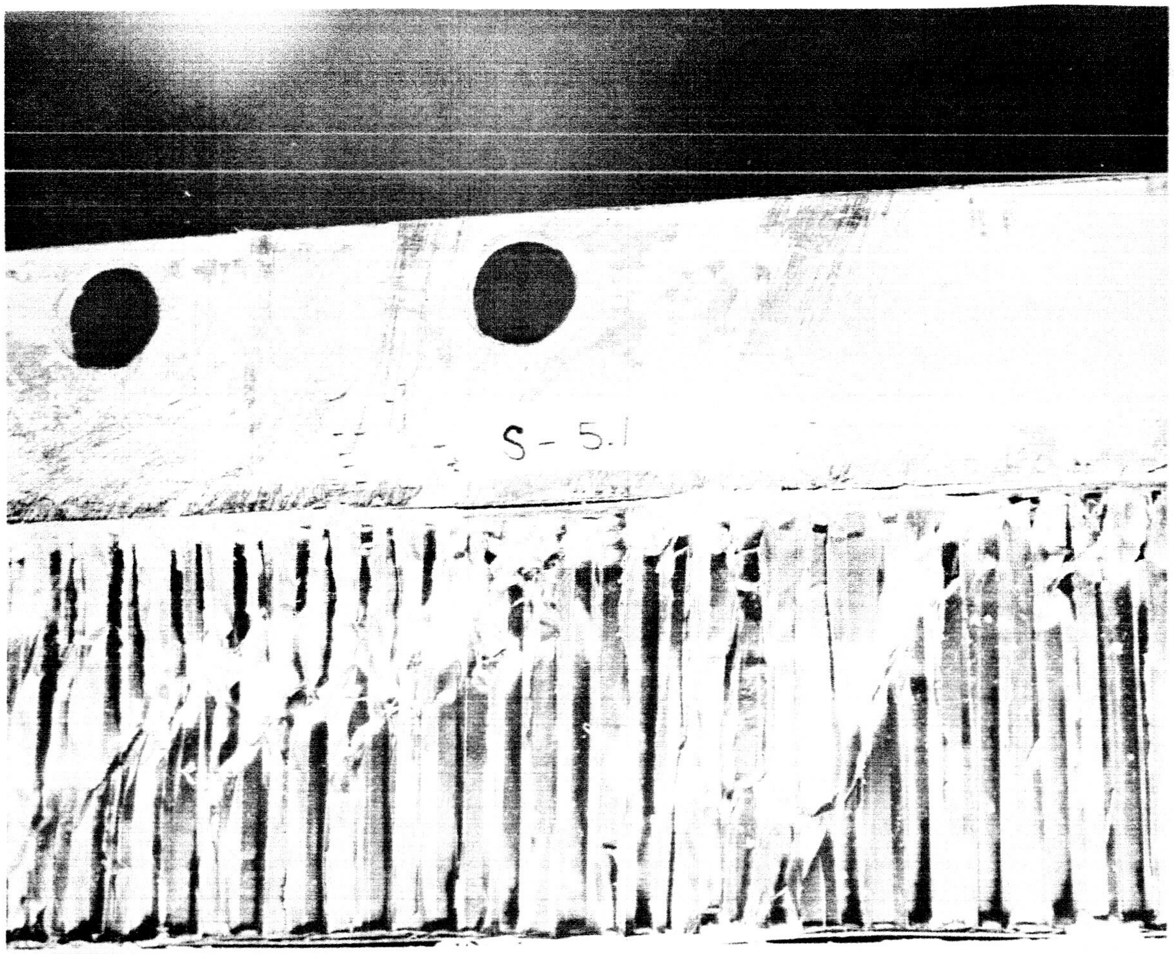
EDGEWISE COMPRESSION TEST
SPLICE JOINT SPECIMEN
TYPICAL CORE FLATWISE TENSION
AND SPLICE PLATE BOND FAILURE



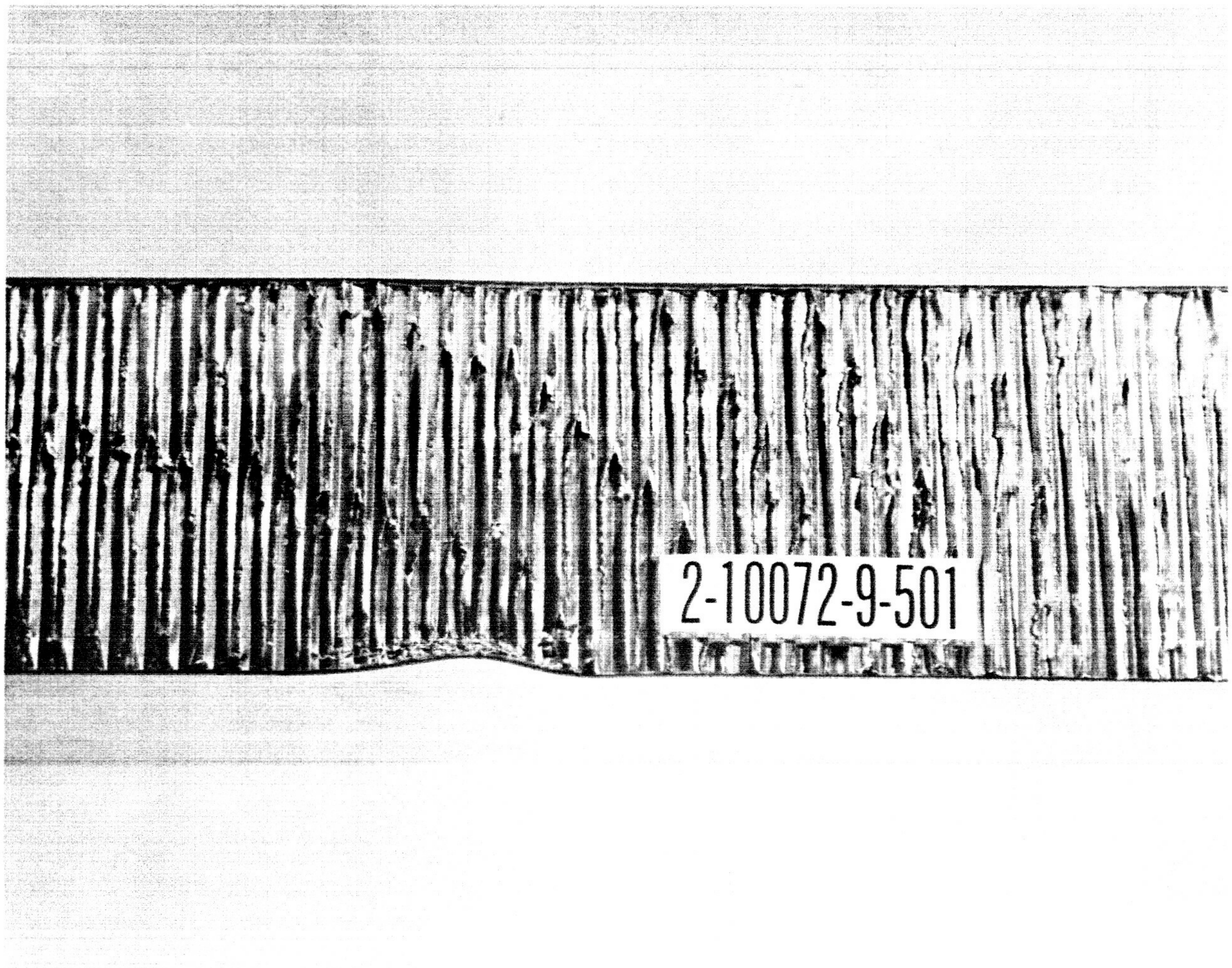
EDGEWISE COMPRESSION TEST
ELECTRON BEAM WELDED SPECIMEN
FACE WRINKLING FAILURE



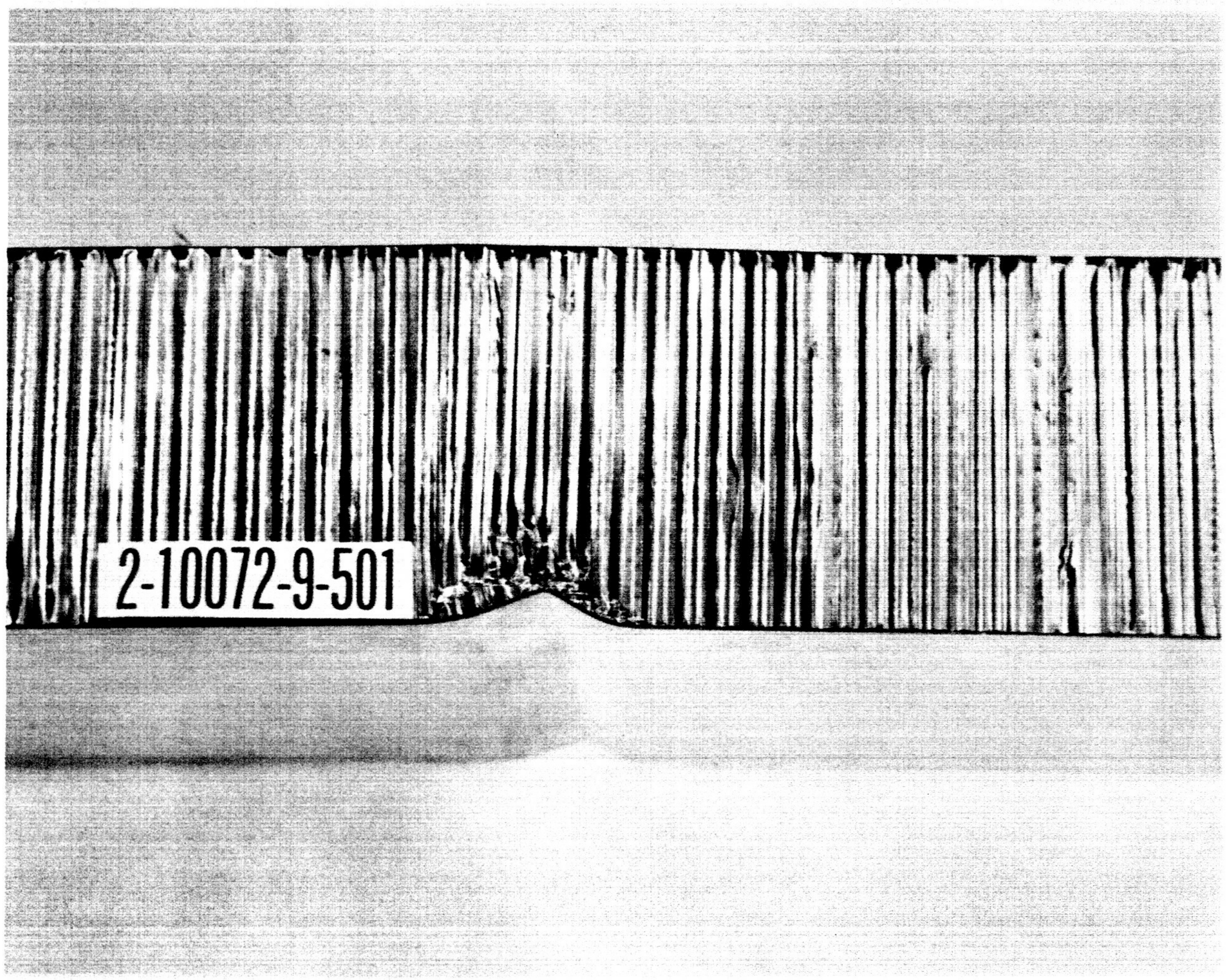
BLOCK SHEAR TEST
TYPICAL TEST SETUP



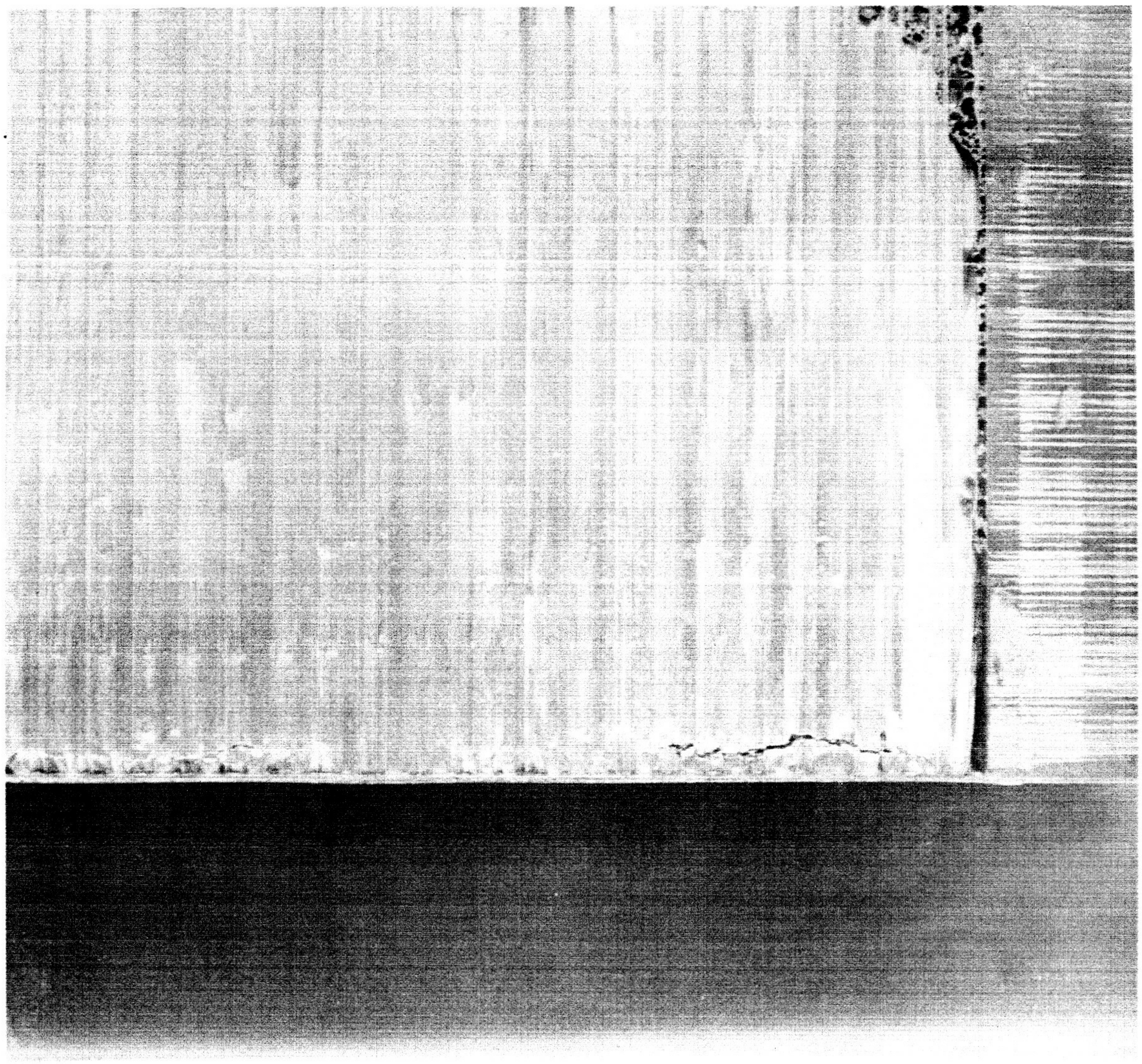
BLOCK SHEAR TEST
TYPICAL CORE SHEAR FAILURE



FLEXURE TEST
TYPICAL FAILURE
(RIGHT HALF OF SPECIMEN)

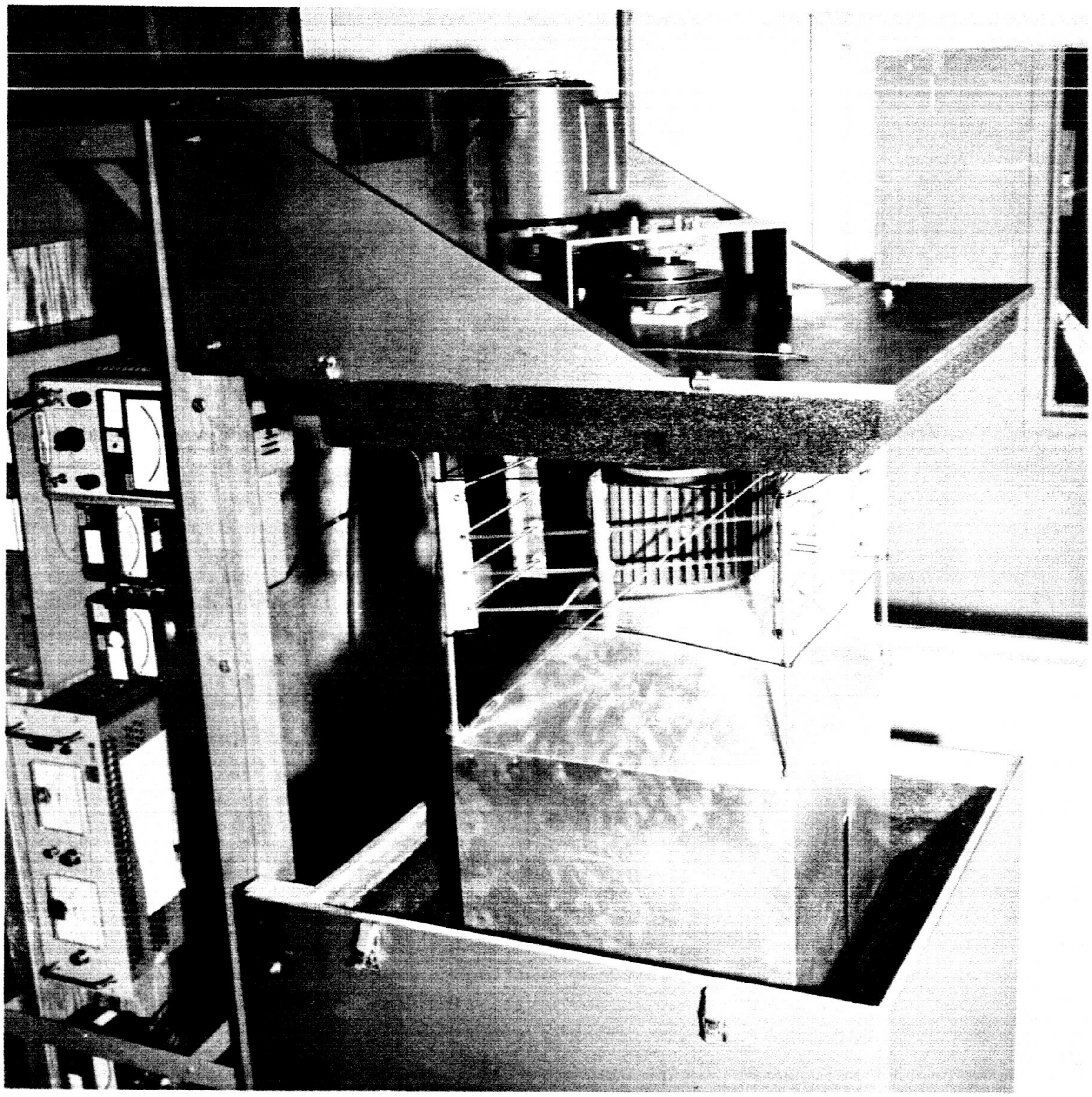


FLEXURE TEST
TYPICAL FAILURE
(LEFT HALF OF SPECIMEN)



DYNAMIC TEST
TYPICAL CORE SHEAR FATIGUE FAILURE

DYNAMIC TEST
FACE BENDING FATIGUE FAILURE



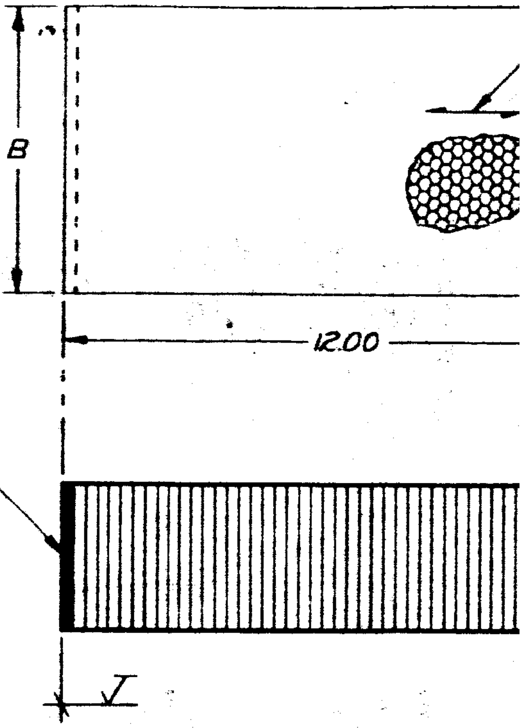
THERMAL CONDUCTIVITY TEST
GUARDED HOT BOX

THERMAL CONDUCTIVITY TEST
INSTALLATION OF SPECIMEN

APPENDIX E

DRAWINGS

A	B	ASSEMBLY
3.000	6.00	-1, -3, -5, -507, -509, -511
4.000	8.00	-501, -503, -505, -513, -515, -517



ROUT OUT CORE AFTER BONDING AND POT WITH A 828. 5

5. COMMERCIAL PRODUCT: SHELL CHEMICAL CO., PITTSBURG, CALIF., CODE IDENT NO. 86961.

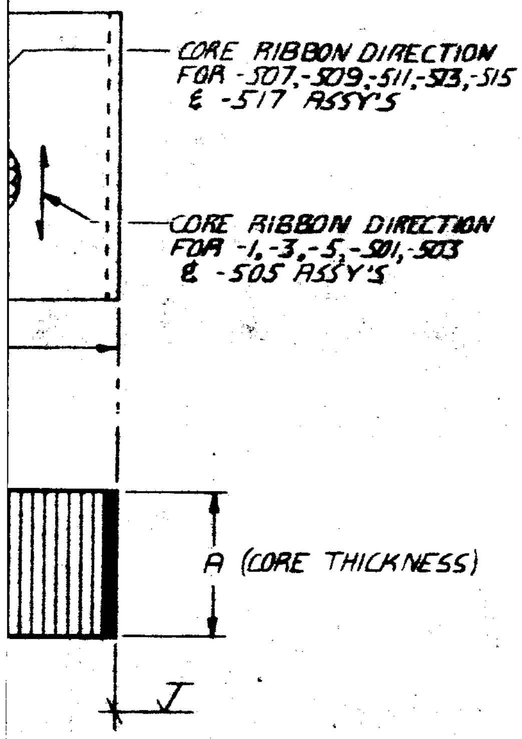
4. COMMERCIAL PRODUCT: BLOOMINGDALE RUBBER CO., HAVRE DE GRACE, MARYLAND, CODE IDENT NO. 07542.

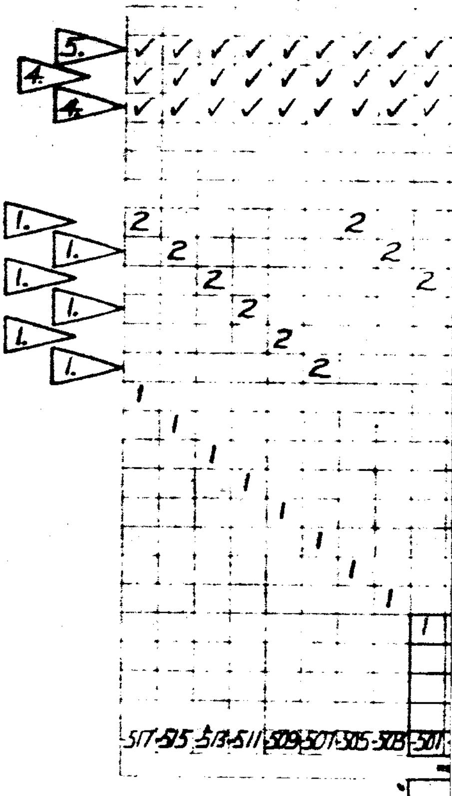
3. MAKE 1 EACH OF -1, -3, -5, -501, -503, -505 ASSEMBLIES AND 2 EACH OF -507, -509, -511, -513, -515, -517 ASSEMBLIES.

2. BOND ALL PANELS PER AVCO/ASD PROC SPEC 11.65.

1. SOLUTION HEAT TREAT AND AGE ALL SKINS: TENSILE 160,000 PSI ULTIMATE, 145,000 PSI YIELD, 3% ELONGATION.

NOTES:


CORE RIBBON DIRECTION
FOR -507,-509,-511,-513,-515
& -517 ASSY'S
CORE RIBBON DIRECTION
FOR -1,-3,-5,-501,-503
& -505 ASSY'S
A (CORE THICKNESS)
J


517-515-513-511-509-507-505-503-501

CHG LTR	S.O. SERIAL NO.	INCORPORATED	DATE	CHECKED	DATE

-5	-3	-1	PART NUMBER	DESCRIPTION	STOCK SIZE	MATERIAL	SPECIFICATION	FIN H.T.	ZN	UNIT WT
✓	✓	✓	A 828	POTTING COMPOUND						
✓	✓	✓	BR-227A	PRIMER						
✓	✓	✓	FM61	ADHESIVE						
			-41	SKIN	.025×8.0×12.0	6AL-4V TI ALY	MIL-T-9046 CLASS 2			
			-39		.018×8.0×12.0					
			-37		.012×8.0×12.0					
2			-35		.025×6.0×12.0					
	2		-33		.018×6.0×12.0					
		2	-31	SKIN	.012×6.0×12.0	6AL-4V TI ALY	MIL-T-9046 CLASS 2			
			-29	CORE	4.000×8.0×12.0	2.3-1/4-10P 5052 AL ALY	MIL-C-7438			
			-27		4.000×8.0×12.0	3.1-3/16-10P 5052 AL ALY				
			-25		4.000×8.0×12.0	3.1-1/8-7P 5052 AL ALY				
			-23		3.000×6.0×12.0	2.3-1/4-10P 5052 AL ALY				
			-21		3.000×6.0×12.0	3.1-3/16-10P 5052 AL ALY				
			-19		3.000×6.0×12.0	3.1-1/8-7P 5052 AL ALY				
			-17		4.000×8.0×12.0	2.3-1/4-10P 5052 AL ALY				
			-15		4.000×8.0×12.0	3.1-3/16-10P 5052 AL ALY				
			-13		4.000×8.0×12.0	3.1-1/8-7P 5052 AL ALY				
1			-11		3.000×6.0×12.0	2.3-1/4-10P 5052 AL ALY				
	1		-9		3.000×6.0×12.0	3.1-3/16-10P 5052 AL ALY				
		1	-7	CORE	3.000×6.0×12.0	3.1-1/8-7P 5052 AL ALY	MIL-C-7438			

QTY REQD/ASSEM

LIST OF MATERIAL

UNLESS OTHERWISE SPECIFIED ALL DIMENSIONS ARE IN INCHES. DIMENSIONS ARE AFTER PLATING. TOLERANCES: ± .005 ANGULAR ± 0°30' ; ± .030 DRILLED HOLES ; ± .010 PER AND 10387

DRN BY	M.W. Lee	7-24-61
GR. ENGR.		
CHECK		
STRESS		
WEIGHTS		
PROCESS		
TOOLING		
PROJECT	[illegible signature]	
APPR. BY		
APPR. BY		

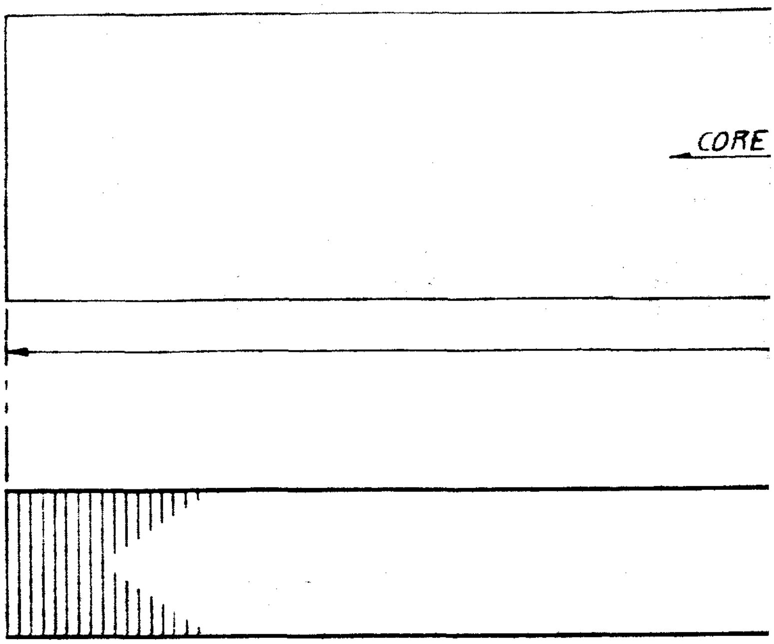
AVCO CORPORATION — AEROSPACE STRUCTURES DIV. NASHVILLE 1, TENNESSEE

TITLE: COUPONS – COMPRESSION TEST
NASA/MSFC NAS8-11807 PHASE I

CODE IDENT.	SIZE	DWG NO.
93221	D	2-10062

SCALE NONE | REL. | SHEET 1 OF 1

2-10062



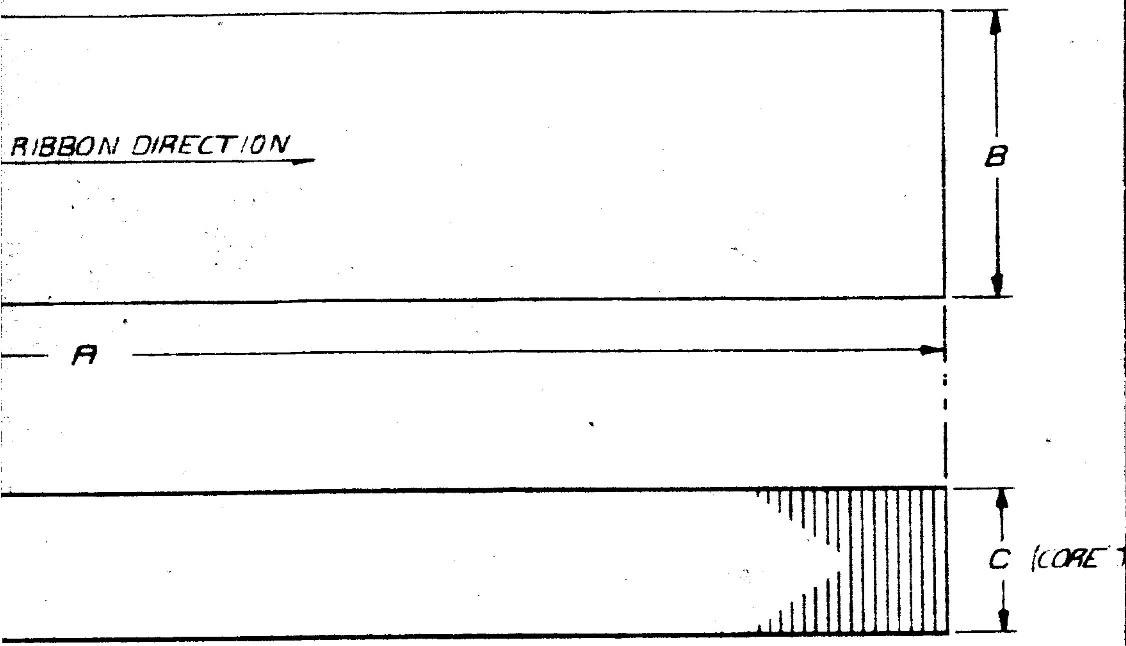
A	B
36.00	6.00
48.00	8.00

3. BOND ALL COUPONS PER AVCO/ASD PROC SPEC 11.65.

2. COMMERCIAL PRODUCT: BLOOMINGDALE RUBBER CO., HAVRE DE GRA
MARYLAND. CODE IDENT NO. 07542.

1. SOLUTION HEAT TREAT AND AGE ALL SKINS;
TENSILE 160,000 PSI ULTIMATE, 145,000 PSI YIELD, 3% ELONGATION.

NOTES:



C	ASSEMBLY
.000	-1, -3, -5
.000	-501, -503, -505

2.
2.
✓ ✓ ✓
✓ ✓ ✓
1.
1.
2 2 2
1
1
1

-505 -503 -501

E,

WT	DASH NO.	NEXT ASSEM	USED ON	SERIAL NO
		APPLICATION		EFFECTIVITY

CHG LTR	E.O. SERIAL NO	INCORPORATED	DATE	CHECKER	DATE

THICKNESS)

				PART NUMBER	DESCRIPTION	STOCK SIZE	MATERIAL	SPECIFICATION	FIN HT	ZN	UNIT WT
✓	✓	✓		FM61	ADHESNE						
✓	✓	✓		BR-227A	PRIMER						
				-21	SKIN	.020-8.0-48.0	6AL-4V TI ALY	MIL-T-9046 CLASS 2			
2	2			-19	SKIN	.020-6.0-36.0	6AL-4V TI ALY	MIL-T-9046 CLASS 2			
				-17	CORE	4.000-8.0-48.0	2.3-1/4-10P 5052 AL ALY	MIL-C-7438			
				-15		4.000-8.0-48.0	3.1-3/16-10P 5052 AL ALY				
				-13		4.000-8.0-48.0	3.1-1/8-1P 5052 AL ALY				
				-11		3.000-6.0-36.0	2.3-1/4-10P 5052 AL ALY				
1				-9		3.000-6.0-36.0	3.1-3/16-10P 5052 AL ALY				
	1			-7	CORE	3.000-6.0-36.0	3.1-1/4-1P 5052 AL ALY	MIL-C-7438			
		2		-505	ASSY						
		1		-503							
		1		-501							
		2		-5							
		2		-3							
		2		-1	ASSY						
-3	-1										
REQD ASSEM											

LIST OF MATERIAL

UNLESS OTHERWISE SPECIFIED
ALL DIMENSIONS ARE IN INCHES
DIMENSIONS ARE AFTER PLATING
TOLERANCES:
ACE ± .060 ANGULAR ± 0°30'
LACES ± .030 DRILLED HOLES
PLACES ± .010 PER AND 10387

DRN. BY	T.V. Lee	7-24-6
GR. ENGR		
CHECK		
STRESS		
WEIGHTS		
PROCESS		
TOOLING		
PROJECT		
APPR. BY		
APPR. BY		

AVCO CORPORATION
AEROSPACE STRUCTURES DIV.
NASHVILLE 1, TENNESSEE
[redacted]

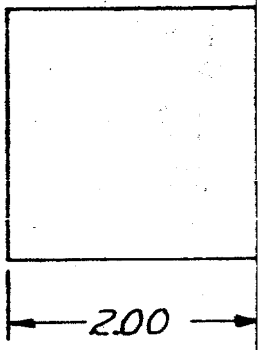
TITLE
COUPONS-
BLOCK SHEAR TEST
NASA MSFC NAS8-11807 PHASE I

CODE IDENT.	SIZE	DWG. NO
93221	D	2-10063

SCALE NONE REL. SHEET 1 OF 1

2-10063

3

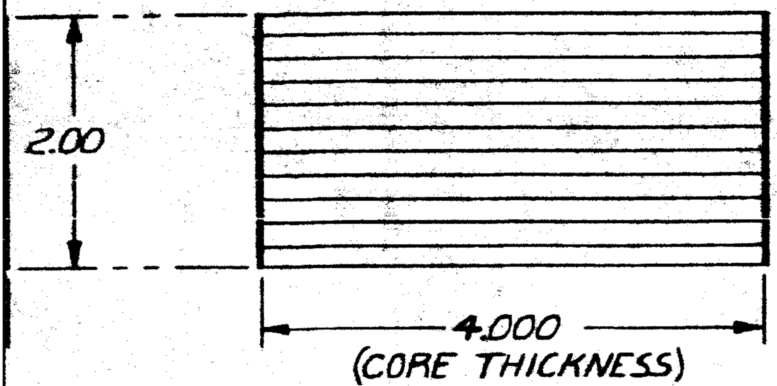


3. BOND ALL COUPONS PER AV

2. COMMERCIAL PRODUCT: BL
GRACE, MARYLAND. CODE

1. SOLUTION HEAT TREAT A
TENSILE 160,000 PSI ULTIMATE

NOTES:



	-5	-3	-1			
2.	✓	✓	✓		FM 61	AL
2.	✓	✓	✓		BR-227A	P
1.	2	2	2		-13	S
	1				-11	C
		1			-9	C
			1		-7	C
				3	-5	A
				3	-3	A
				3	-1	A
	-5	-3	-1		PART NUMBER	
	REQD/ASSEM.					

CO/ASD PROC SPEC 11.65.
OOMINGDALE RUBBER CO., HAVRE DE
IDENT NO. 07542.
AND AGE ALL SKINS;
, 145,000 PSI YIELD, 3 % ELONGATION.

UNLESS OTHERWISE SPECIFIED
ALL DIMENSIONS ARE IN INCHES
DIMENSIONS ARE AFTER PLATING
TOLERANCES:

ONE PLACE ± .060 ANGULAR ± 0°30'
TWO PLACES ± .030 DRILLED HOLES
THREE PLACES ± .010 PER AND 10387

THREADS PER HANDBOOK H-28 AND MIL-STD-9
MACHINE SYMBOLS PER MIL-STD-8
WELD SYMBOLS PER JAN-STD-19
BEND RADII PER AVCO/N 2A20
RIVET SPACING PER AVCO/N 2A21
BREAK SHARP EDGES .005 TO .030R

MARK PER AVCO/N 2A22-

125 SURFACE ROUGHNESS PER MIL-STD-10

DRN. BY
GR. ENGR.
CHECK
STRESS
WEIGHTS
PROCESS
TOOLING
PROJECT
APPR. BY

CHG LTR	E.O. SERIAL NO.	INCORPORATED	DATE	CHECKED	DATE

DESCRIPTION	STOCK SIZE	MATERIAL	SPECIFICATION	FIN H. T.	ZN.	UNIT WT.
…DHESIVE						
…RIMER						
…KIN	.020×2.0×2.0	6AL-4V TI ALY	MIL-T-9046 CLASS Z			
…ORE	4.000×2.0×2.0	3.1-1/8-7P 5052 AL ALY	MIL-C-7438			
…ORE	4.000×2.0×2.0	3.1-3/16-10P 5052 AL ALY	MIL-C-7438			
…ORE	4.000×2.0×2.0	2.3-1/4-10P 5052 AL ALY	MIL-C-7438			
…SSY						
…SSY						
…SSY						

LIST OF MATERIAL

2-10064

W.H. Lee	7-24-64
W. …	1-28-64

Avco CORPORATION

AEROSPACE STRUCTURES DIV.
NASHVILLE 1, TENNESSEE

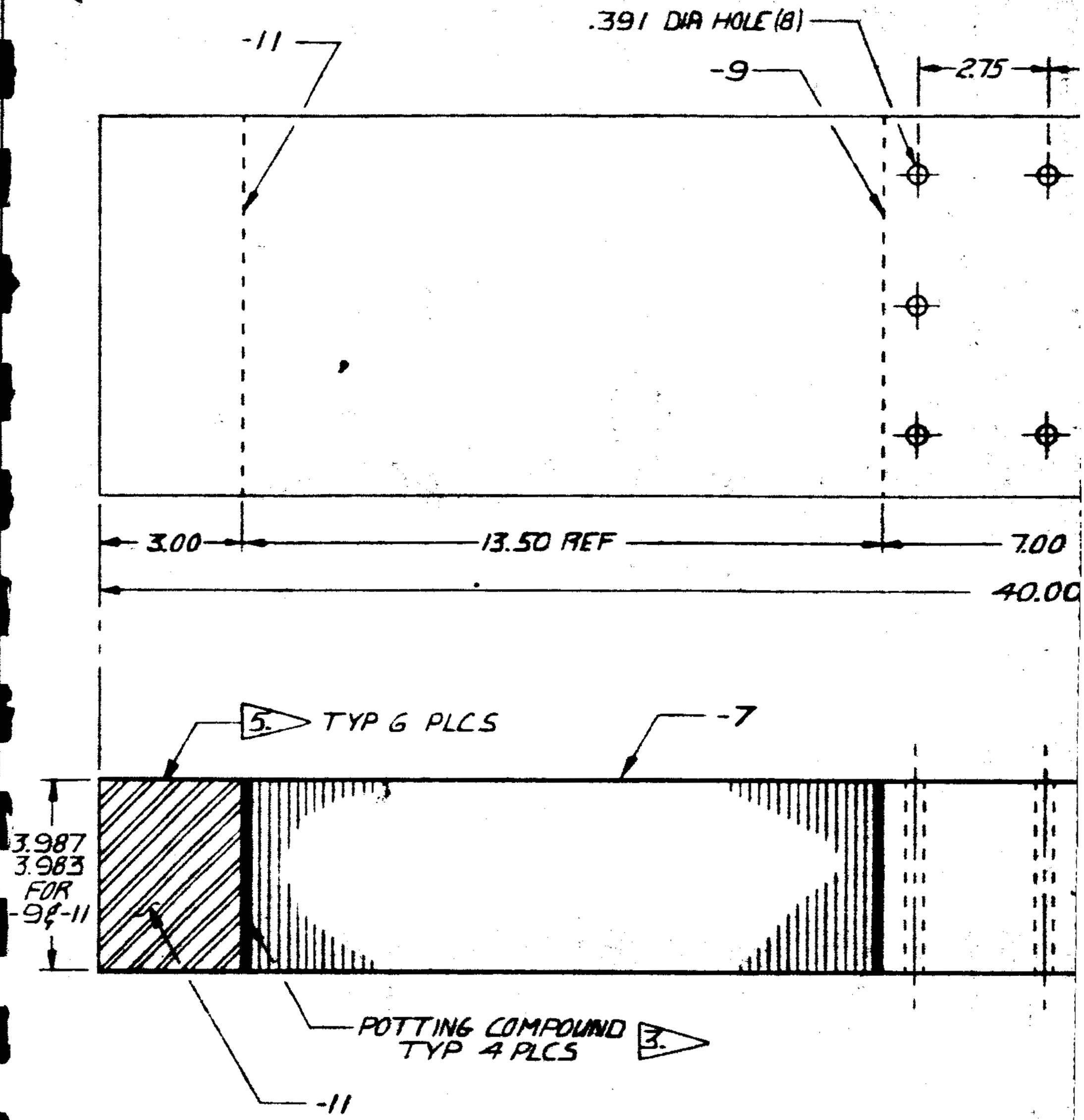
TITLE

COUPONS-
FLATWISE TENSION TEST

NASA MSFC NAS8-11807 PHASE I

CODE IDENT.	SIZE	DWG. NO.
93221	C	2-10064

SCALE 1/1 REL SHEET 1 OF 1



7. NOMINAL THICKNESS OF METAL TO METAL BOND LINE TO BE .009.

6. ALL METAL TO CORE BONDING TO BE ACCOMPLISHED USING FM 61 ADHESIVE AND BR-227A PRIMER 3.

5. ALL METAL TO METAL BONDING ADHESIVE TO BE FM 1000 3.

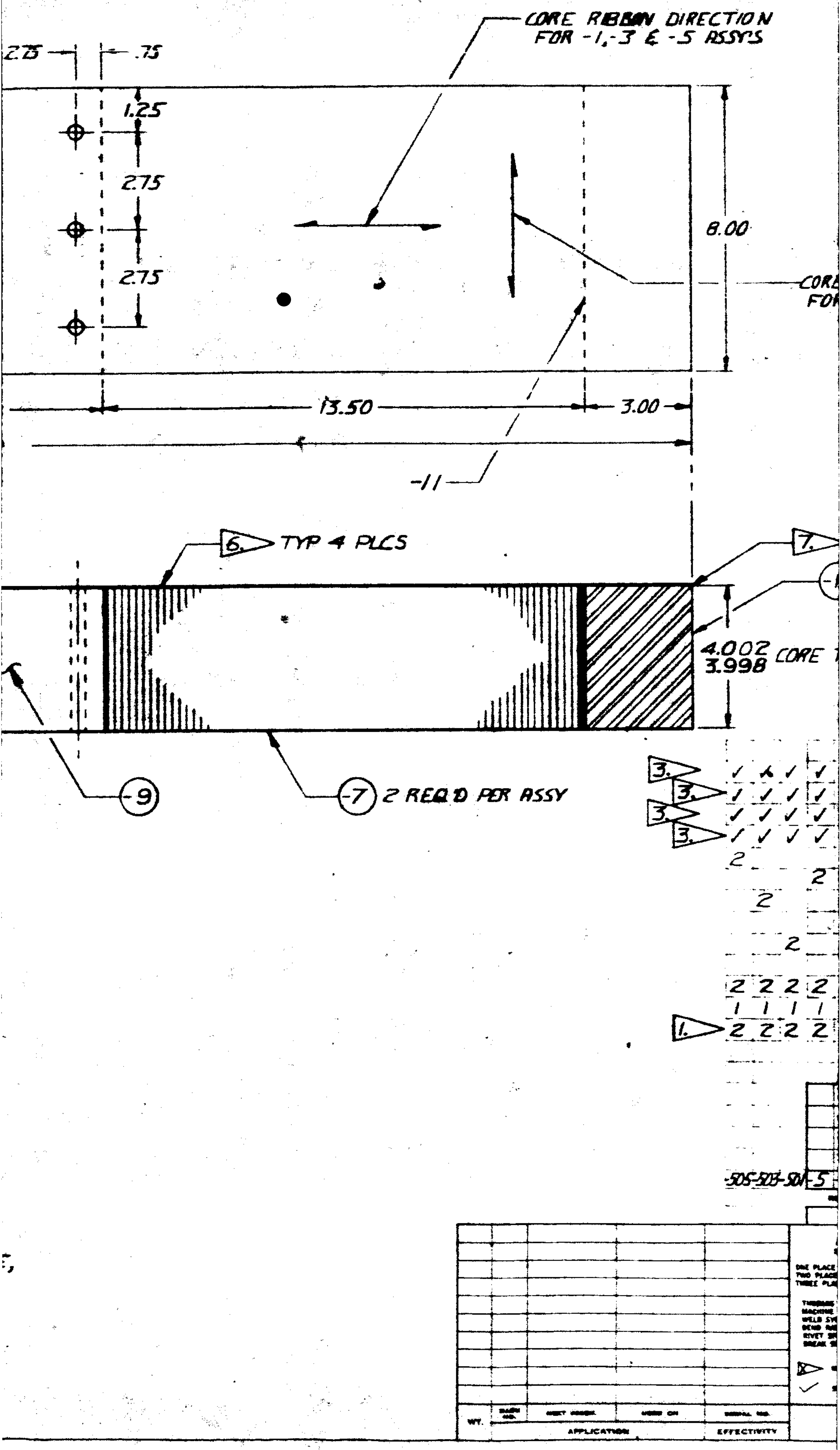
4. NOT USED.

3. COMMERCIAL PRODUCT: BLOOMINGDALE RUBBER CO., HAVRE DE GRAC[E] MARYLAND. CODE IDENT NO. 07542.

2. BOND. ALL PANELS PER AVCO/ASD PROC SPEC 11.65.

1. SOLUTION HEAT TREAT AND AGE ALL SKINS: TENSILE 160,000 PSI ULTIMATE, 145,000 PSI YIELD, 3% ELONGATION.

NOTES:



CORE RIBBON DIRECTION FOR -1, -3 & -5 ASSY'S
2.25
.75
1.25
2.75
2.75
8.00
CORE
FOR
13.50
3.00
-11
6. TYP 4 PLCS
7.
4.002
3.998
CORE
-9
-7 2 REQ'D PER ASSY
3.
1.
-505-503-501 -5
APPLICATION
EFFECTIVITY

CHG LTR	E.O. SERIAL NO.	MODIFICATION	DATE	ENGINEER	DATE

RIBBON DIRECTION
-501, -503 & -505
ASSY'S

(1) 2 REQ'D PER ASSY

THICKNESS

			PART NUMBER	DESCRIPTION	STOCK SIZE	MATERIAL	SPECIFICATION	FIN H.T.	ZN	UNIT WT.
✓	✓		HT 424	POTTING COMPOUND						
✓	✓		BR-227A	PRIMER						
✓	✓		FM 61	ADHESIVE						
✓	✓		FM 1000	ADHESIVE						
			-23	CORE	4.000×8.0×13.5	3.1-1/8-7P 5052 AL ALY	MIL-C-7438			
			-21			3.1-1/8-4P 5052 AL ALY				
			-19			3.1-3/16-2P 5052 AL ALY				
2			-17			3.1-3/16-[illegible]P 5052 AL ALY				
			-15			2.3-1/4-[illegible]P 5052 AL ALY				
	2		-13	CORE	4.000×8.0×13.5	2.3-1/4-[illegible]P 5052 AL ALY	MIL-C-7438			
2	2		-11	WEIGHT	3×5×8.0	STEEL BAR COLD FINISH	AISI 1018			
1	1		-9	SPACER	4½×7.0×8.0	6061-T6511 AL ALY	QQ-A-325			
2	2		-7	SKIN	.020×8.0×40.0	6AL-4V TI ALY	MIL-T-9046 CLASS Z			
		1	-505	ASSY						
		1	-503	ASSY						
		1	-501	ASSY						
		2	-5	ASSY						
		2	-3	ASSY						
		1	-1	ASSY						
3	-1									

...DD/ASSEM

LIST OF MATERIAL

...LESS OTHERWISE SPECIFIED
...LL DIMENSIONS ARE IN INCHES
...MENSIONS ARE AFTER PLATING
TOLERANCES
± .XXX ANGULAR ± 0°30'
± .XX DRILLED HOLES
...S ± .XXX PER ANS 10007
...ER HANDBOOK H-28 AND MIL-STD-9
...MBOLS PER MIL-STD-8
...OLS PER JAN-STD-10
... PER AVCO/N 3830
...INS PER AVCO/N 2421
...IP EDGES .005 TO .030R
...K PER AVCO/N 2422-
...FACE ROUGHNESS PER MIL-STD-10

DRN. BY	M. H. Lee	2-29-[illegible]
GR ENGR		
CHECK		
STRESS		
WEIGHTS		
PROCESS		
TOOLING		
PROJECT		
APPR. BY		
APPR. BY		

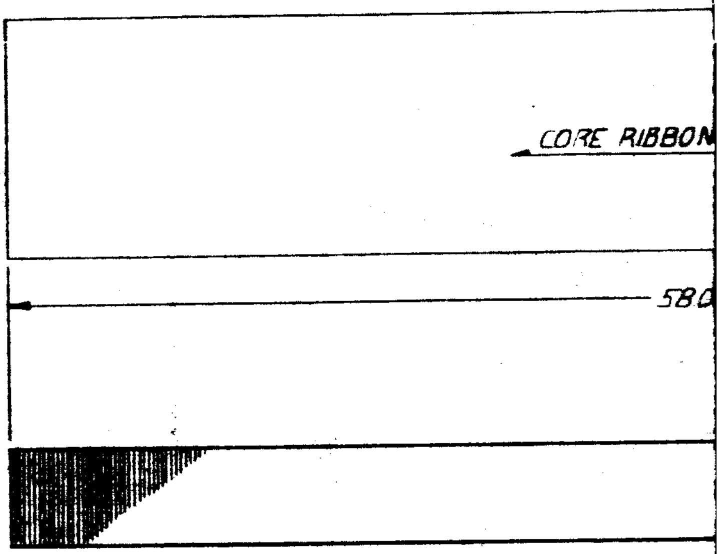
Avco CORPORATION AEROSPACE STRUCTURES DIV NASHVILLE 1, TENNESSEE

TITLE
COUPONS-
DYNAMIC TEST
NASA MSFC NAS8-11807 PHASE II

CODE IDENT.	SIZE	DWG. NO.
93221	D	2-10071

SCALE 1/2 REL SHEET 1 OF 1

2-10071



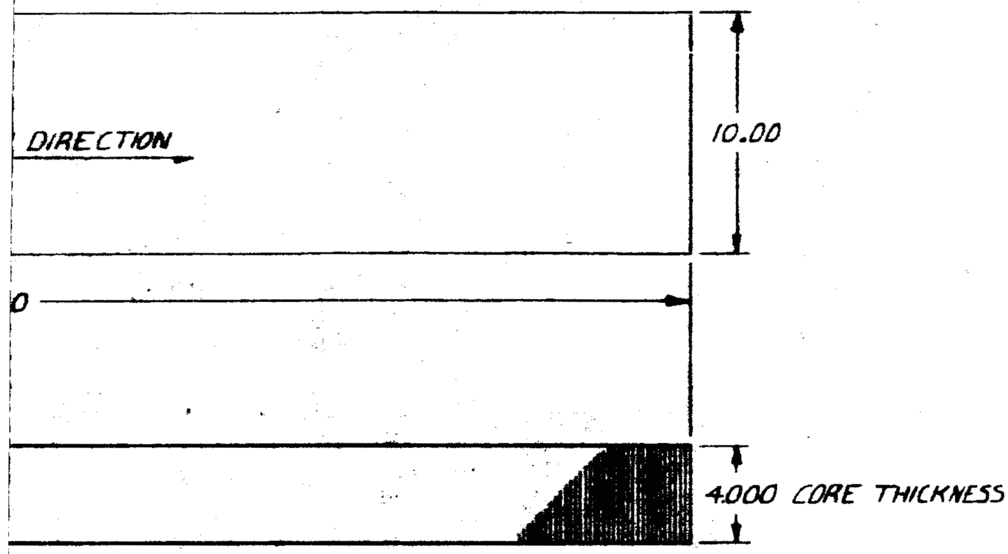
4. MAKE 1 OF EACH ASSY SHOWN.

3. BOND ALL COUPONS PER AVCO/ASD PROC SPEC 11.65.

2. COMMERCIAL PRODUCT: BLOOMINGDALE RUBBER CO., HAVRE DE GRACE, MARYLAND. CODE IDENT NO. 07542.

1. SOLUTION HEAT TREAT AND AGE ALL SKINS; TENSILE 160,000 PSI ULTIMATE, 145,000 PSI YIELD, 3% ELONGATION.

NOTES:



2.

2.

1

1.

1.

2

WT	DASH NO	NEXT ASSEM	USED ON	SERIAL NO
	APPLICATION			EFFECTIVITY

2

-5	-3	-1	PART NUMBER	DESCRIPTION	STOCK SIZE	MATERIAL	SPECIFICATION	FIN HT	ZN	UNIT WT
✓	✓			FM61 ADHESIVE						
✓	✓			BR-227A PRIMER						
	1		-15	CORE	4.000×10.0×5.80	2.3-1/4-10P 5052 AL ALY	MIL-C-7438			
		1	-13	CORE	4.000×10.0×5.80	3.1-3/16-10P 5052 AL ALY	MIL-C-7438			
1			-11	CORE	4.000×10.0×5.80	3.1-1/4-7P 5052 AL ALY	MIL-C-7438			
2			-9	SKIN	.025×10.0×5.80	6AL-4V TI ALY	MIL-T-9046 CLASS 2			
	2	2	-7	SKIN	.013×10.0×5.80	6AL-4V TI ALY	MIL-T-9046 CLASS 2			

... OD ASSEM

LIST OF MATERIAL

UNLESS OTHERWISE SPECIFIED
ALL DIMENSIONS ARE IN INCHES
DIMENSIONS ARE AFTER PLATING
TOLERANCES
± .010 ANGULAR ± 0°30'
± .030 DRILLED HOLES
ES ± .010 PER AND 10387

PER HANDBOOK H-28 AND MIL-STD-9
SYMBOLS PER MIL-STD-8
BOLS PER JAN-STD-19
PER AVCO/N 2A20
ONS PER AVCO/N 2A21
MP ... TO ...

... PER AVCO N 2A20

... FACE ROUGHNESS PER MIL-STD-10

DRN. BY	W. D. Lee	2-29-64
GR. ENGR.		
CHECK		
STRESS		
WEIGHTS		
PROCESS		
TOOLING		
PROJECT		
APPR. BY		
APPR. BY	3	

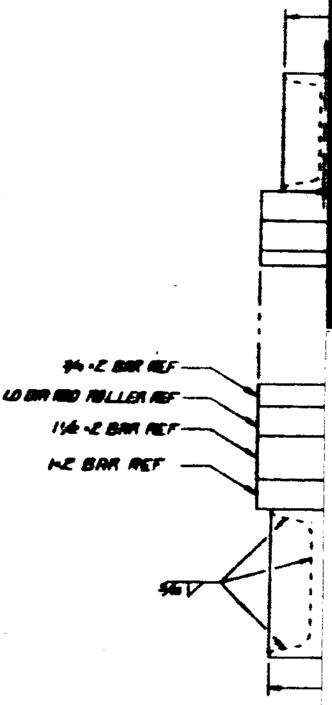
Avco CORPORATION
AEROSPACE STRUCTURES DIV
NASHVILLE 1, TENNESSEE

TITLE
COUPONS-FLEXURAL TEST
NASA MSFC NAS8-11807 PHASE II

CODE IDENT.	SIZE	DWG. NO.
93221	D	2-10072

SCALE 1/4 REL SHEET 1 OF 1

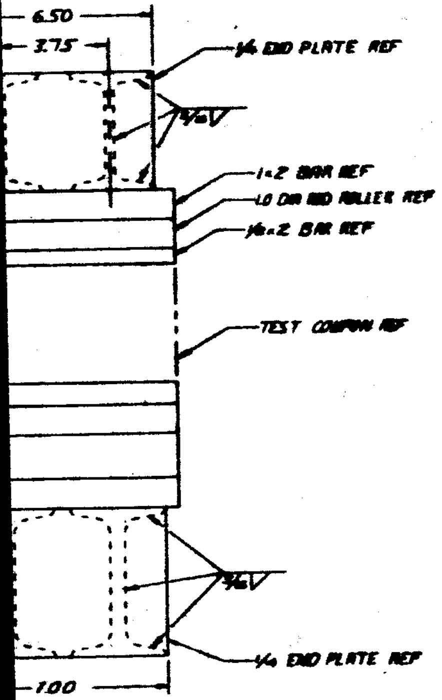
2-10072



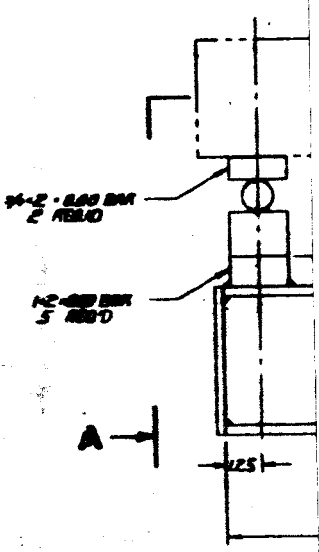
H

G

1

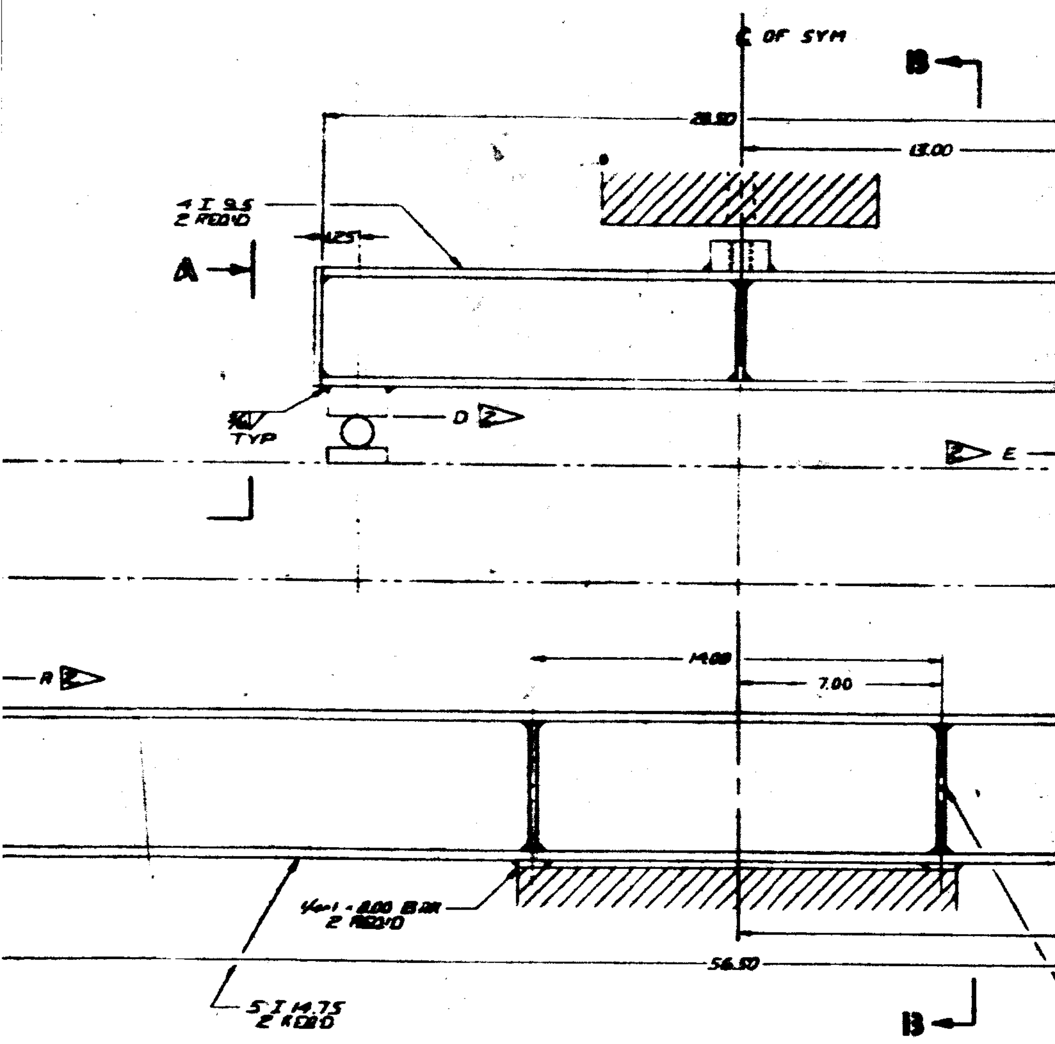

6.50
3.75
1/4 END PLATE REF
1×2 BAR REF
1.0 DIA ROD ROLLER REF
1/2×2 BAR REF
TEST COUPON REF
1/4 END PLATE REF
7.00

A—A


2 REQ'D
5 REQ'D
A
.125

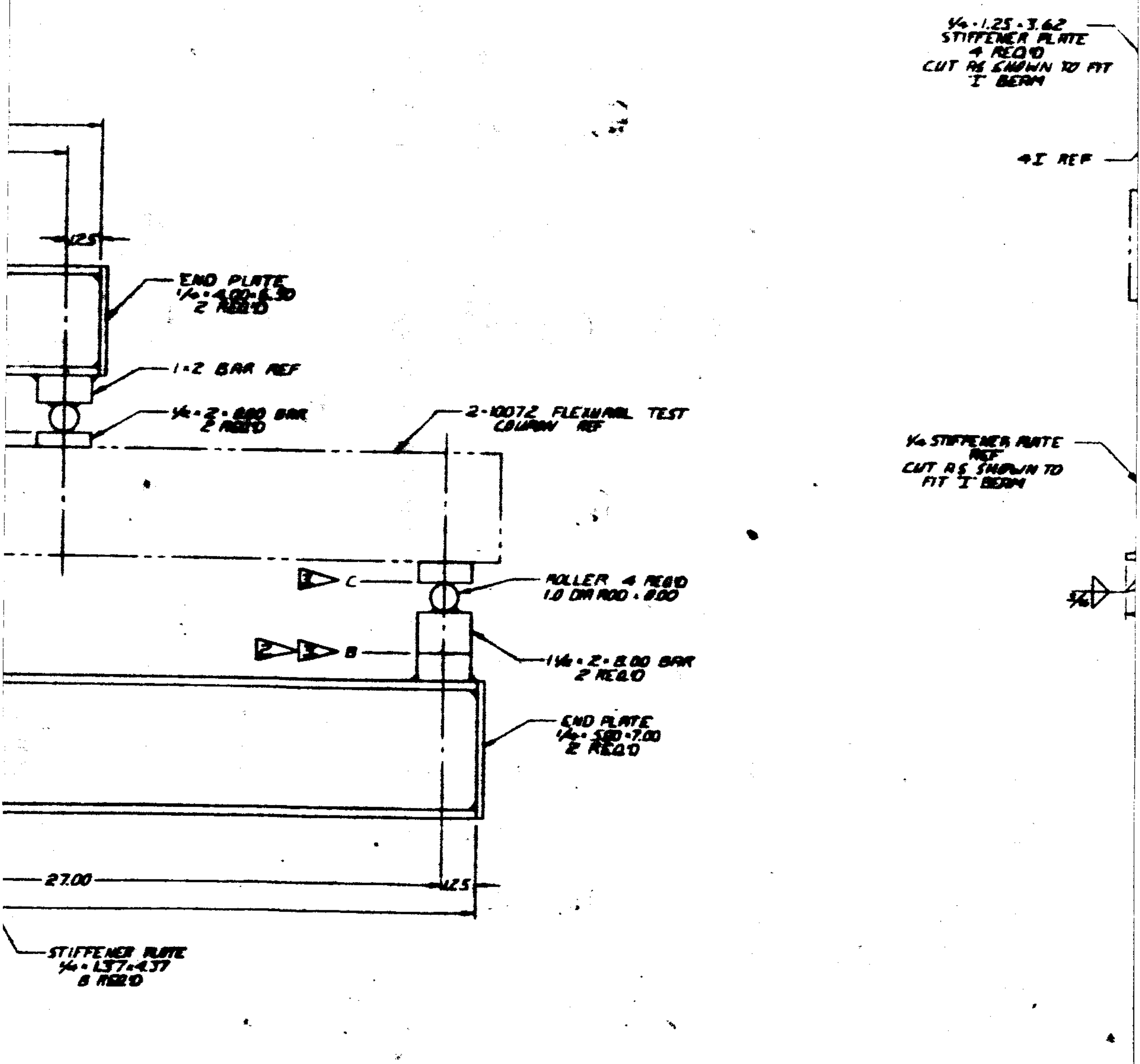
E

2



℄ OF SYM
B
28.50
13.00
4 I 9.5
2 REQ'D
1.25
A
TYP
D
E
R
14.00
7.00
1/4 x 1 - 8.00 BAR
2 REQ'D
56.50
5 I 14.75
2 REQ'D
B

E D
3



NOTES:

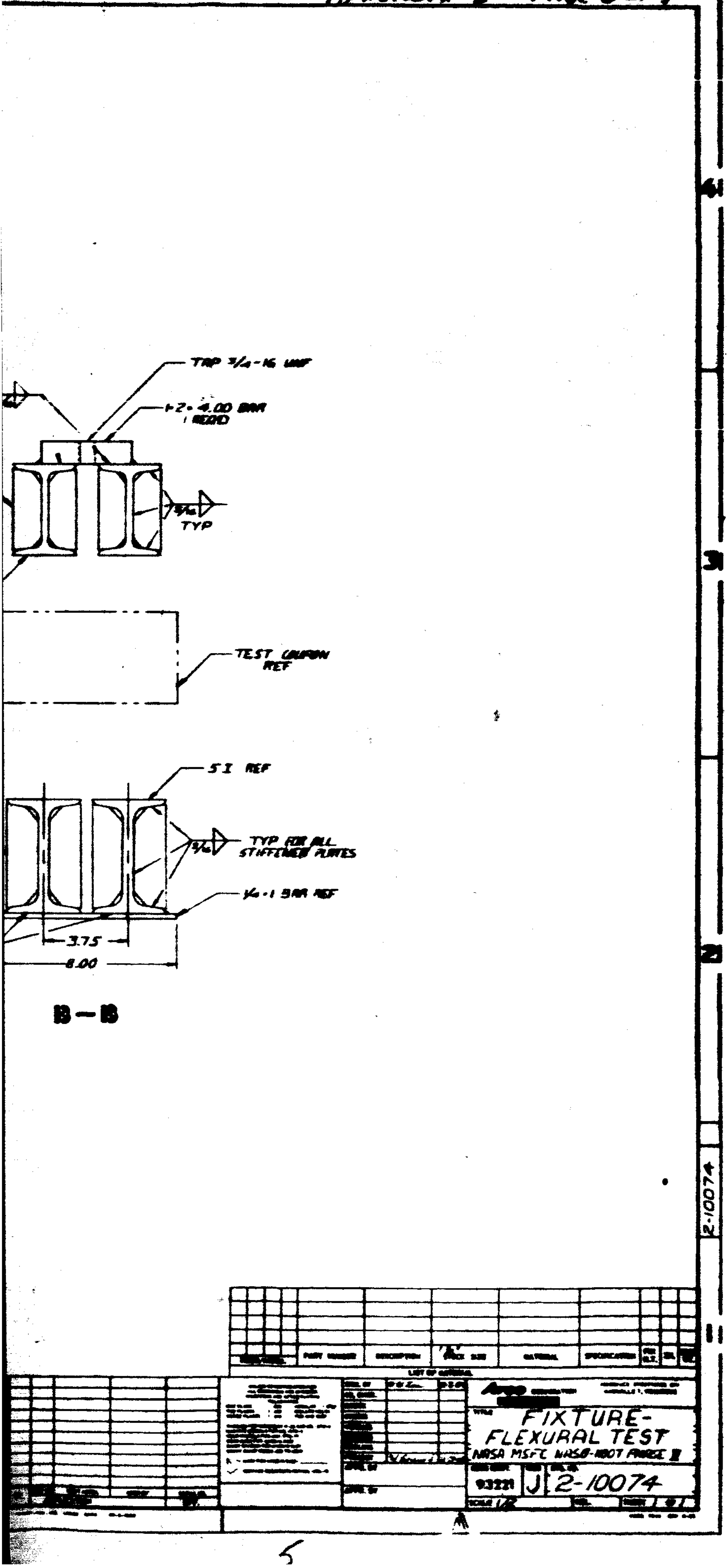
1. ALL MATERIAL TO BE A7 STRUCTURAL STEEL.

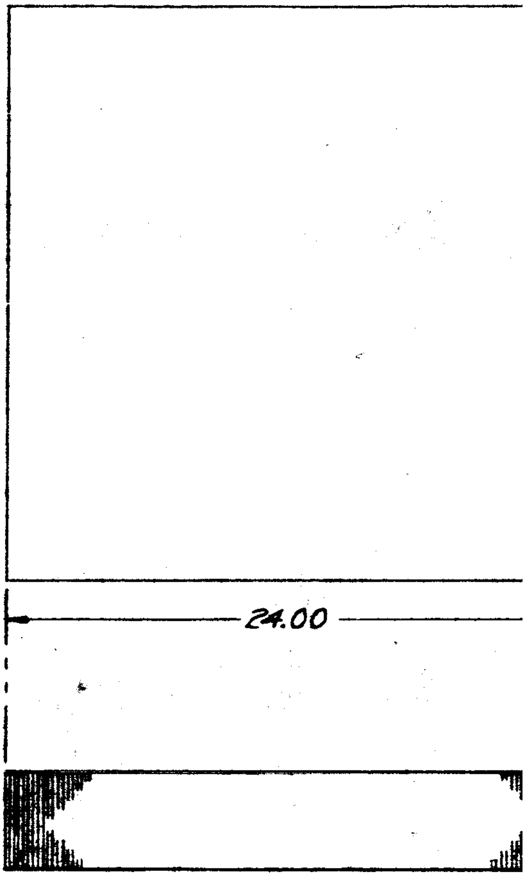
2. AFTER ASSEMBLY OF LOWER BEAM, SURFACE "B" MUST BE PARALLEL TO SURFACE "E" WITHIN ±.005. LIKEWISE, SURFACE "D" AND SURFACE "E" OF UPPER BEAM MUST BE PARALLEL WITHIN ±.005.

3. AFTER WELDING 1.0 DIA AND 10 1 1/4 × 2 BAR, SURFACE "B" MUST BE PARALLEL TO SURFACE "C" WITHIN ±.005.

C

4


TAP 3/4-16 UNF
1·2·4.00 BAR 1 REQD
3/16 TYP
TEST COUPON REF
5 I REF
3/16 TYP FOR ALL STIFFENER PLATES
1/4·1 BAR REF
3.75
8.00
B—B
LIST OF MATERIAL
FIXTURE-
FLEXURAL TEST
NASA MSFC NAS8-1807 PHASE II
93221
J
2-10074
SCALE 1/2

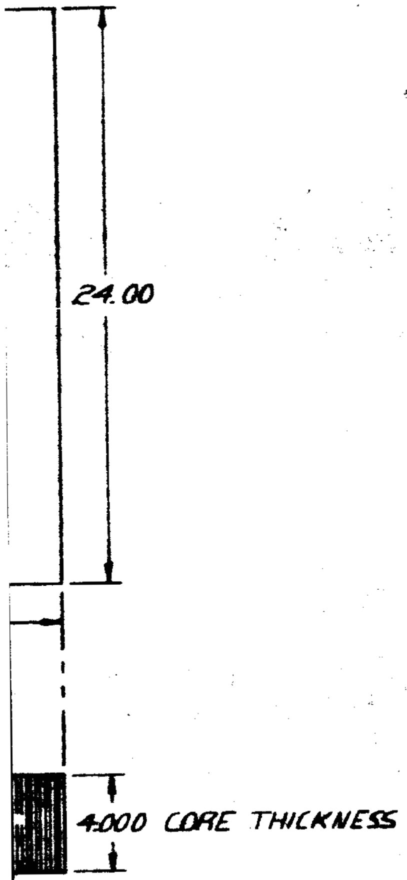


NOTES:

1. SOLUTION HEAT TREAT AND AGE ALL SKINS: TENSILE 160,000 PSI ULTIMATE, 145,000 PSI YIELD; 3% ELONGATION.

2. COMMERCIAL PRODUCT: BLOOMINGDALE RUBBER CO., HAVRE DE GRACE, MARYLAND, CODE IDENT NO. 07542.

3. BOND ALL COUPONS PER AVCO/ASD PROC SPEC 11.65.



2. ✓ ✓
2. ✓ ✓

1
1

1. 2 2

-5 -3
REQD

ONE PLACE
TWO PLACES
THREE PLACES

WT.	DASH NO.	NEXT ASSEM	USED ON	SERIAL NO.
	APPLICATION		EFFECTIVITY	

✓		BR-227A	PRIMER						
✓		FM61	ADHESIVE						
		-13	CORE	[illegible]-24.0-24.0	2.3-1/4-[illegible] 5052 AL ALY	MIL-C-7438			
		-11	CORE	[illegible]-24.0-24.0	3.1-3/16-[illegible] 5052 AL ALY	MIL-C-7438			
1		-9	CORE	[illegible]-24.0-24.0	3.1-1/8-[illegible] 5052 AL ALY	MIL-C-7438			
2		-7	SKIN	.020-24.0-24.0	6AL-4V TI ALY	MIL-T-[illegible] CLASS 2			
	1	-5	ASSY						
	1	-3	ASSY						
	2	-1	ASSY						
-1		PART NUMBER	DESCRIPTION	STOCK SIZE	MATERIAL	SPECIFICATION	FIN H T	ZN	UNIT WT.

LIST OF MATERIAL

DRN. BY	[illegible]	12-23-[illegible]
GR. ENGR.		
CHECK		
STRESS		
WEIGHTS		
PROCESS		
TOOLING		
PROJECT		
APPR. BY		
APPR. BY		

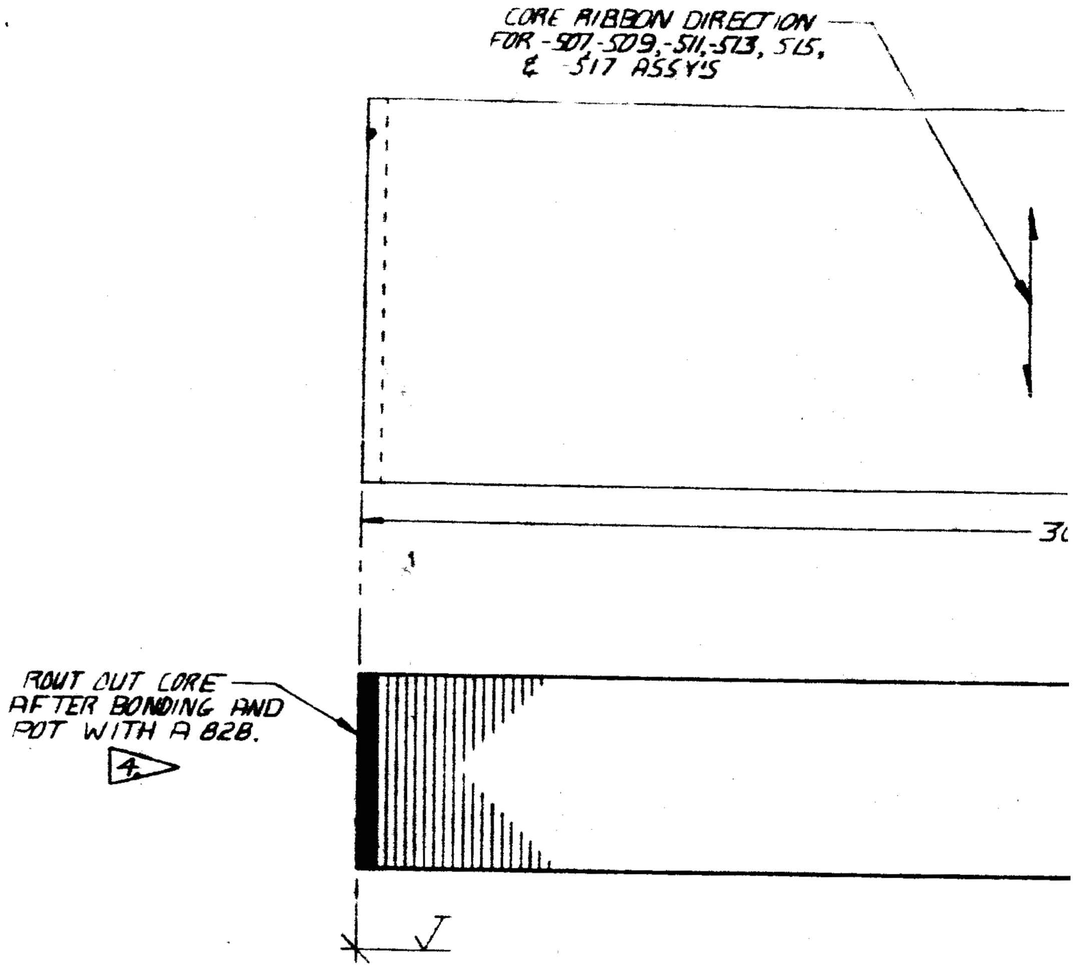
Avco CORPORATION
AEROSPACE STRUCTURES DIV.
NASHVILLE 1 TENNESSEE

TITLE
COUPONS-
THERMAL TEST
NASA MSFC NAS8-11807 PHASE II

CODE IDENT.	SIZE	DWG. NO.
93221	D	2-10075

SCALE 1/4 REL SHEET 1 OF 1

2-10075



5. MAKE 1 OF EACH ASSY SHOWN.

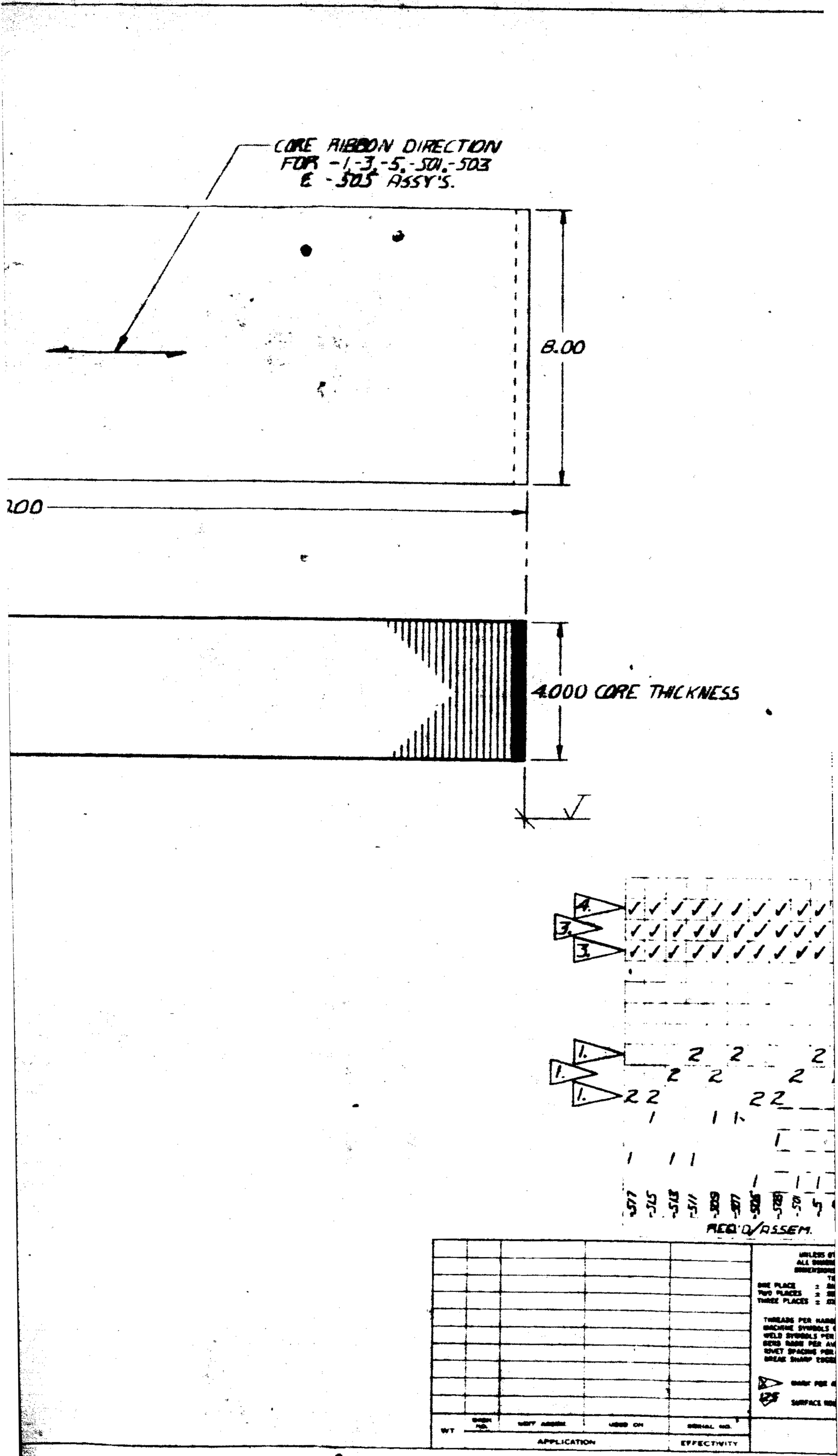
4. COMMERCIAL PRODUCT: SHELL CHEMICAL CO., PITTSBURGH, CALIF., CODE IDENT NO. 86961.

3. COMMERCIAL PRODUCT: BLOOMINGDALE RUBBER CO., HAVRE DE GRACE, MARYLAND, CODE IDENT NO. 07542.

2. BOND ALL COUPONS PER AVCO/ASD PROC SPEC 11.65.

1. SOLUTION HEAT TREAT AND AGE ALL SKINS: TENSILE 160,000 PSI ULTIMATE, 145,000 PSI YIELD; 3% ELONGATION.

NOTES:


CORE RIBBON DIRECTION
FOR -1,-3,-5,-501,-503
& -505 ASSY'S.
8.00
200
4.000 CORE THICKNESS
-517
-515
-513
-511
-509
-507
-505
-503
-501
-5
REQ'D/ASSEM.
APPLICATION
EFFECTIVITY

✓✓ A 828 POTTING COMPOUND
✓✓ BR-227A PRIMER
✓✓ FM 61 ADHESIVE

		PART NUMBER	DESCRIPTION	STOCK SIZE	MATERIAL	SPECIFICATION	FIN H. T	ZN	UNIT WT.
2		-19	SKIN	.032×8.0×30.0	6AL-4V TI ALY	MIL-T-9046 CLASS 2			
2		-17	SKIN	.025×8.0×30.0	6AL-4V TI ALY	MIL-T-9046 CLASS 2			
		-15	SKIN	.020×8.0×30.0	6AL-4V TI ALY	MIL-T-9046 CLASS 2			
		-13	CORE	4.000×8.0×30.0	3.1-1/8-7P 5052 AL ALY	MIL-C-7438			
1	1	-11			3.1-1/8-7P 5052 AL ALY				
		-9			3.1-3/16-2P 5052 AL ALY				
		-7	CORE	4.000×8.0×30.0	3.1-3/16-2P 5052 AL ALY	MIL-C-7438			

LIST OF MATERIAL

DRN. BY	T.Y. Lee	12-29-69
GR. ENGR.		
CHECK		
STRESS		
WEIGHTS		
PROCESS		
TOOLING		
PROJECT		
APPR. BY		
APPR. BY		

Avco CORPORATION — AEROSPACE STRUCTURES DIV, NASHVILLE 1, TENNESSEE

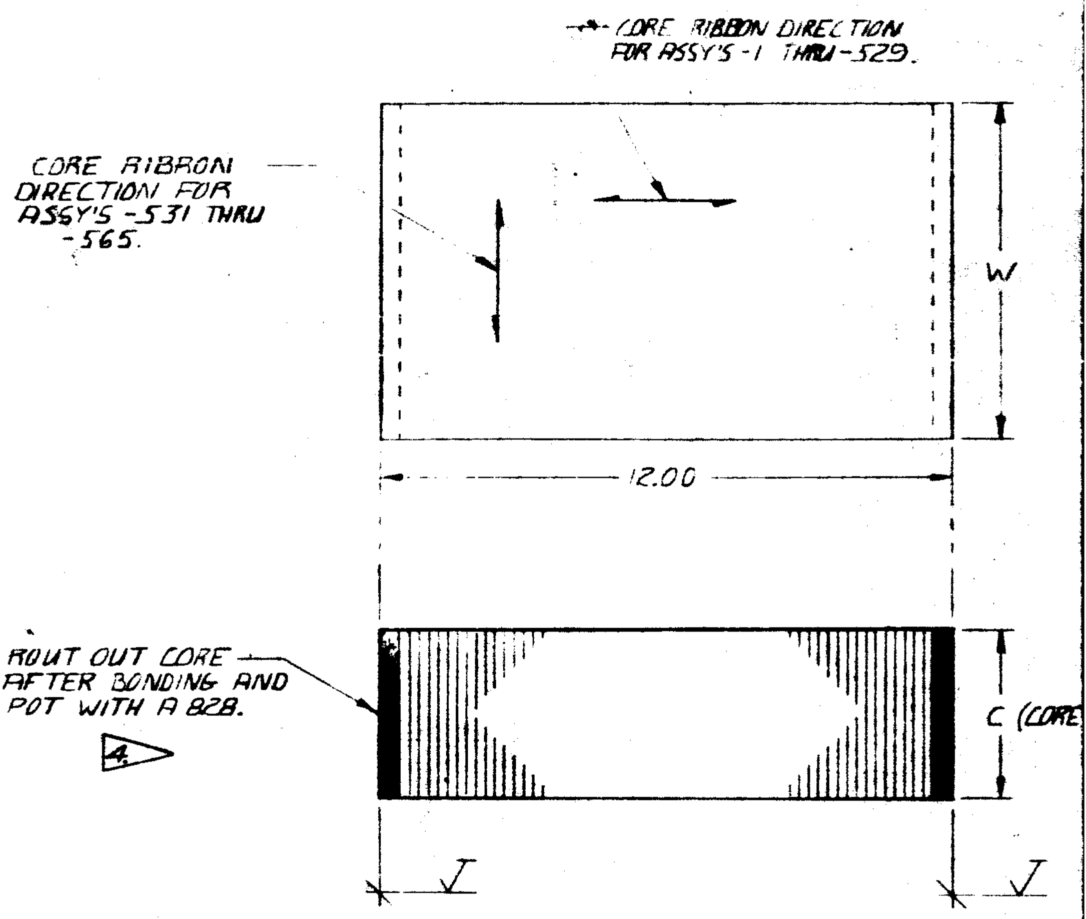
TITLE: COUPONS- AXIAL COMPRESSION TEST NASA MSFC NAS8-11807 PHASE II

CODE IDENT.	SIZE	DWG. NO.
93221	D	2-10076

SCALE 1/2 | REL | SHEET 1 OF 1

2-10076

3



5. MAKE 1 OF EACH ASSY SHOWN.

4 COMMERCIAL PRODUCT: SHELL CHEMICAL CO., PITTSBURGH, CALIF., CODE IDENT NO. 86961.

3 COMMERCIAL PRODUCT: BLOOMINGDALE RUBBER CO., HAVRE DE GRACE, MARYLAND. CODE IDENT NO. 07542.

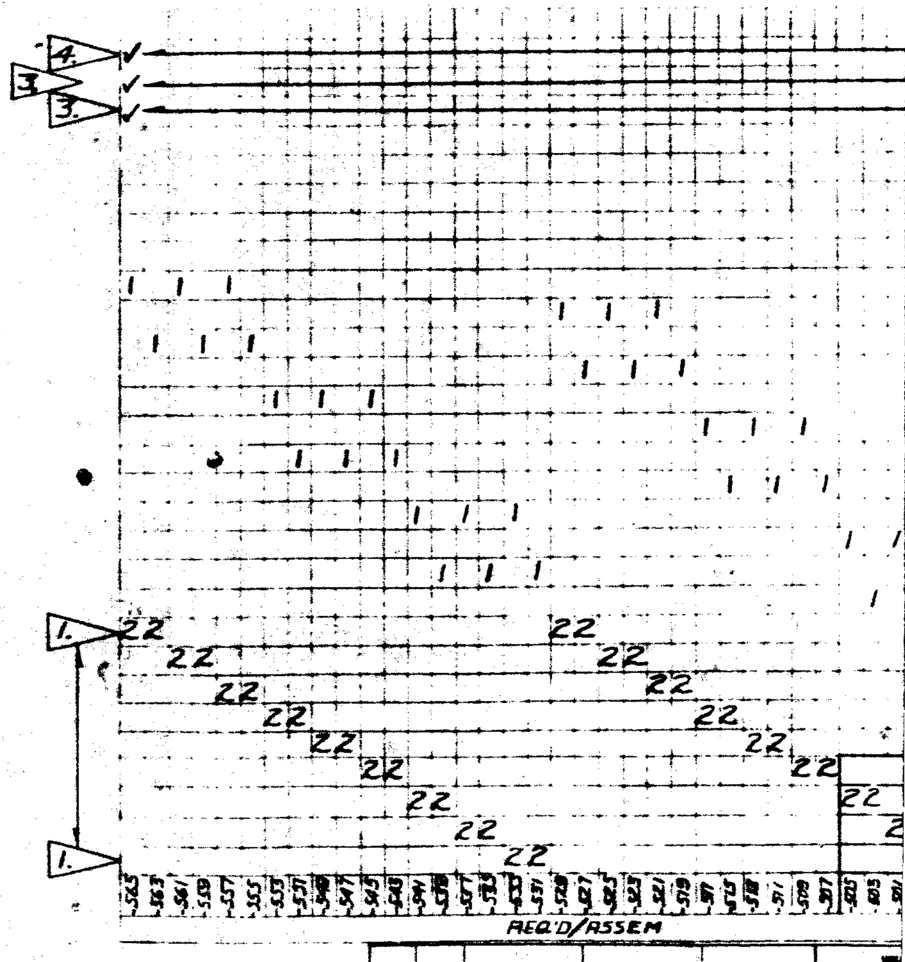
2. BOND ALL COUPONS PER AVCO/ASD PROC SPEC 11.65.

1 SOLUTION HEAT TREAT AND AGE ALL SKINS: TENSILE 160,000 PSI ULTIMATE, 145,000 PSI YIELD; 3% ELONGATION.

NOTES:

W	C	ASSY
6.00	3.000	-1,-3,-5,-501,-503,-505, -531,-533,-535, 537,-539,-541
7.00	3.500	-507,-509,-511,-513,-515,-517, -543,-545,-547,-549,-551,-553
8.00	4.000	-519,-521,-523,-525,-527,-529, -555,-557,-559,-561,-563,-565

THICKNESS)



WT	SHEET NO.	NEXT ASSEM	USED ON	SERIAL NO.	
	APPLICATION			EFFECTIVITY	

2

CHG LTR	E.O. SERIAL NO.	INCORPORATED	DATE	CHECKED	DATE

✓ A8Z8 POTTING COMPOUND
✓ BR-227A PRIMER
✓ FM61 ADHESIVE

PART NUMBER	DESCRIPTION	STOCK SIZE	MATERIAL	SPECIFICATION
-47	CORE	4.000×8.0×12.0	3.1-3/16-10P 5052 AL ALY	MIL-C-7438
-45		4.000×8.0×12.0	[illegible]	
-43		4.000×8.0×12.0	[illegible]	
-41		4.000×8.0×12.0	[illegible]	
-39		3.500×7.0×12.0	[illegible]	
-37		3.500×7.0×12.0	[illegible]	
-35		3.500×7.0×12.0	[illegible]	
-33		3.500×7.0×12.0	[illegible]	
-31		3.000×6.0×12.0	[illegible]	
-29		3.000×6.0×12.0	[illegible]	
-27		3.000×6.0×12.0	[illegible]	
-25	CORE	3.000×6.0×12.0	5052 AL ALY	MIL-C-7438
-23	SKIN	.020×8.0×12.0	6AL-4V TI ALY	MIL-T-9046 CLASS 2
-21		.025×8.0×12.0		
-19		.032×8.0×12.0		
-17		.020×7.0×12.0		
-15		.025×7.0×12.0		
-13		.032×7.0×12.0		
-11		.020×6.0×12.0		
-9		.025×6.0×12.0		
-7	SKIN	.032×6.0×12.0	6AL-4V TI ALY	MIL-T-9046 CLASS 2

LIST OF MATERIAL

DRN. BY W. H. Lee 2-28-64

Avco CORPORATION
AEROSPACE STRUCTURES DIV.
NASHVILLE 1, TENNESSEE

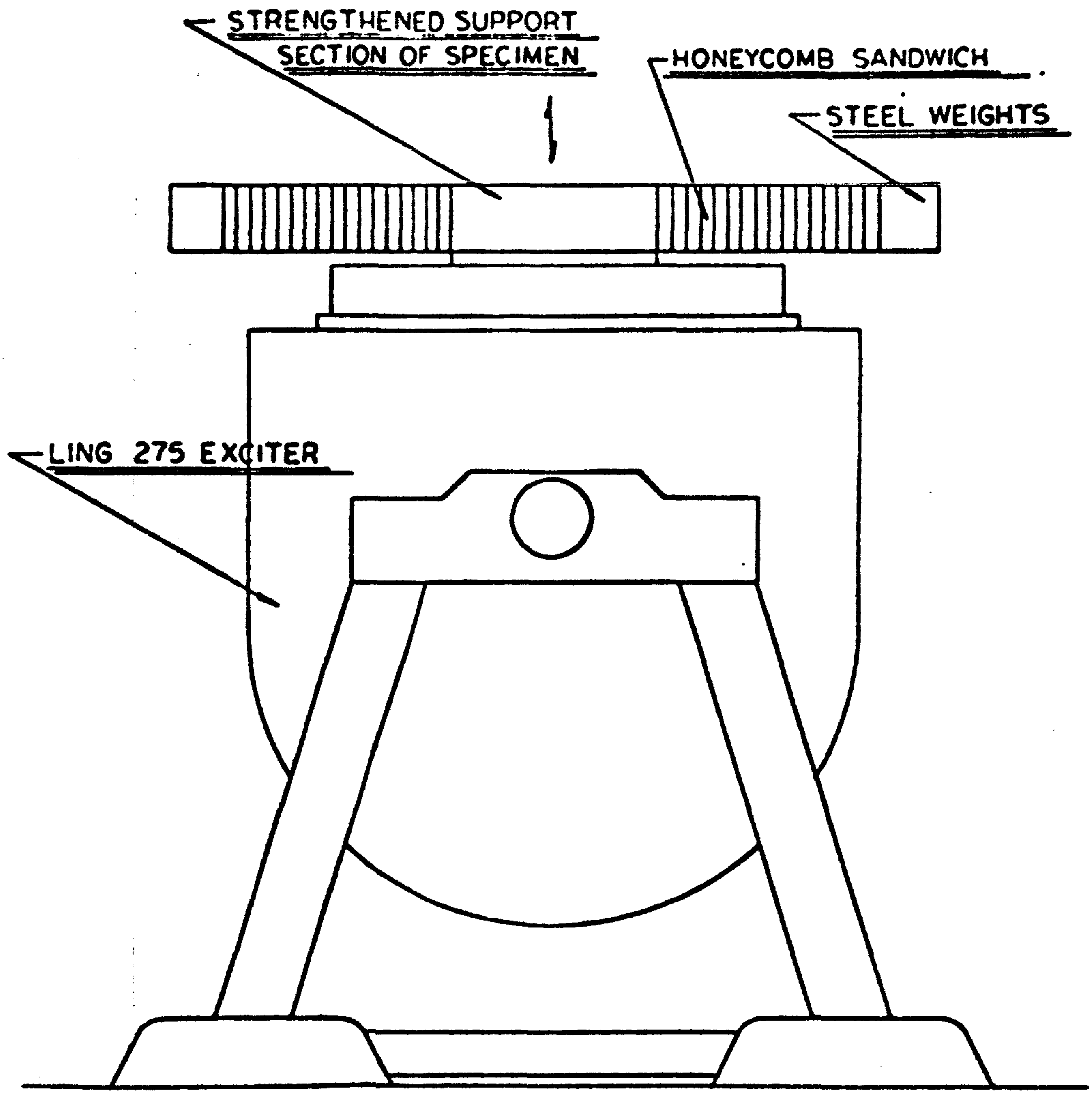
TITLE
COUPONS - TEST,
EDGEWISE COMPRESSION
NASA MSFC NAS8-11807 PHASE II

CODE IDENT. 93221 SIZE D DWG. NO. 2-10077

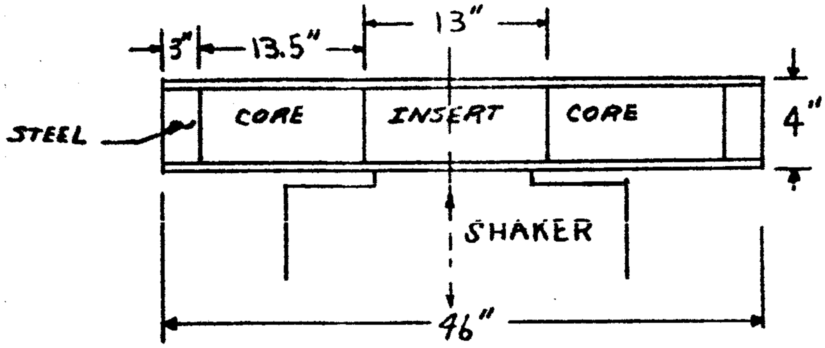
SCALE 1/2 SHEET 1 OF 1

APPENDIX F

LOAD CALCULATION FOR DYNAMIC TEST



DYNAMIC TEST SETUP



Weight of Steel Mass = 3 x 4 x 8 x .283 = 27.2#

The skin will be stressed to 130,000#/in^2 or 85% of 153,000#/in^2 (yield)

$$f = \frac{Mc}{I}$$

$$M = PL = \frac{fI}{c}$$

$$w = 27.2\#$$

$$P = gW \qquad g = \text{Number of Gravities}$$

$$P = \frac{fI}{Lc}$$

$$g = \frac{fI}{CLW} \qquad L = 13.5 + 1.5 = 15''$$

$$I = 2 \times .012 \times 8 \times 2^2 = .768 \text{ in}^4$$

$$c = 2.0''$$

$$g = \frac{130{,}000 \times .768}{2 \times 15 \times 27.2} = \underline{\underline{122.2}}$$

$$a = 122.2 \times 32.2 = 3940 \text{ ft/sec}^2 = 47{,}200 \text{ in/sec}^2$$

$$r = \Delta = \frac{P\ell^3}{3EI} = \frac{122.2 \times 27.2(13.5 + 3/2)^3}{3 \times 16 \times 10^6 \times .768} = \underline{\underline{.305 \text{ in.}}}$$

$$w^2 r = w^2(.305) = 47{,}200 \qquad w^2 = 155{,}000$$

$$w = \underline{\underline{394}} \ \frac{\text{RADIANS}}{\text{SEC.}} = 62.6 \ \frac{\text{CYCLES}}{\text{SEC.}}$$

$$fs = \frac{P}{A} = \frac{27.2 \times 122.2}{8.0 \times 4.0} = 104 \ \#/\text{in}^2$$

APPENDIX G

HEAT FLOW & TEMPERATURE RELATIONSHIPS

APPENDIX G

HEAT FLOW AND TEMPERATURE RELATIONSHIPS

The following relationships were used in calculating the values reported in the Thermal Conductivity Test Summary:

1) $$k = \frac{q}{A(t_1 - t_2)/L}$$

2) $$c = \frac{q}{A(t_1 - t_2)}$$

3) $$U = \frac{q}{A(t_h - t_c)}$$

4) $$W + M \text{ (thermopile reading)} = AU(t_h - t_c)$$

k = Thermal conductivity, Btu/hr./sq.ft. deg. Fahr./in.

c = Thermal conductance, Btu/hr./sq.ft. deg. Fahr.

q = Time rate of heat flow through area A, Btu per hour.

A = Area normal to heat flow, sq.ft.

L = Length of path of heat flow (thickness of specimen), in.

t_h = Temperature of hot air, deg. Fahr.

t_1 = Temperature of the hot surface, deg. Fahr.

t_2 = Temperature of the cold surface, deg. Fahr.

t_c = Temperature of cold air, deg. Fahr.

W = Time rate of electric power input to metering box.

M = Heat flow meter coefficient.

The power input to the metering box was corrected by application of the heat flow meter coefficient, M, to the thermopile reading. To obtain this coefficient, the guard box air temperature was adjusted for steady temperatures with the guard box air first slightly warmer, and then slightly cooler than the metering box air. Two equations in the form of 4) above were then written, and assuming U and M to be constants, the equations were solved for M.

APPENDIX H

WEIGHT STUDY CALCULATION

APPENDIX H

JOINT, CLOSE-OUT WEIGHT STUDY

The purpose of this report is to determine that percentage weight of a honeycomb panel which is attributable to the close-out members. The honeycomb panel is part of a cylinder and carries compressive axial loads as shown in Figure 1.

The compressive axial loads obtained are from the test data gathered at AVCO/ASD.

Two types of close-outs are shown in this report. Type 1 is shown in Figure 2, and is necessary to transfer axial loads from lower skin to upper skin and vice versa. Type 2 is shown in Figure 3 and is necessary for transferring loads around the circumference through the welds.

Type 2 close-outs consist of strips equal to skin thickness and .5" width, and are bonded along the length of the panel to the upper and lower skins along both edges. During assembly, the panels will be welded along the length of the cylinder.

The cylinder in the example used has 16.5' inside radius, and is 11' long; therefore, honeycomb panels of approximately 6' along the circumference by 11' long will be built.

For weight calculation, a honeycomb panel of 6' x 11' is used and assumed to be flat. Percentage weights of the close-outs to the honeycomb panel are determined and a summary of estimated forward skirt is made on page 8.

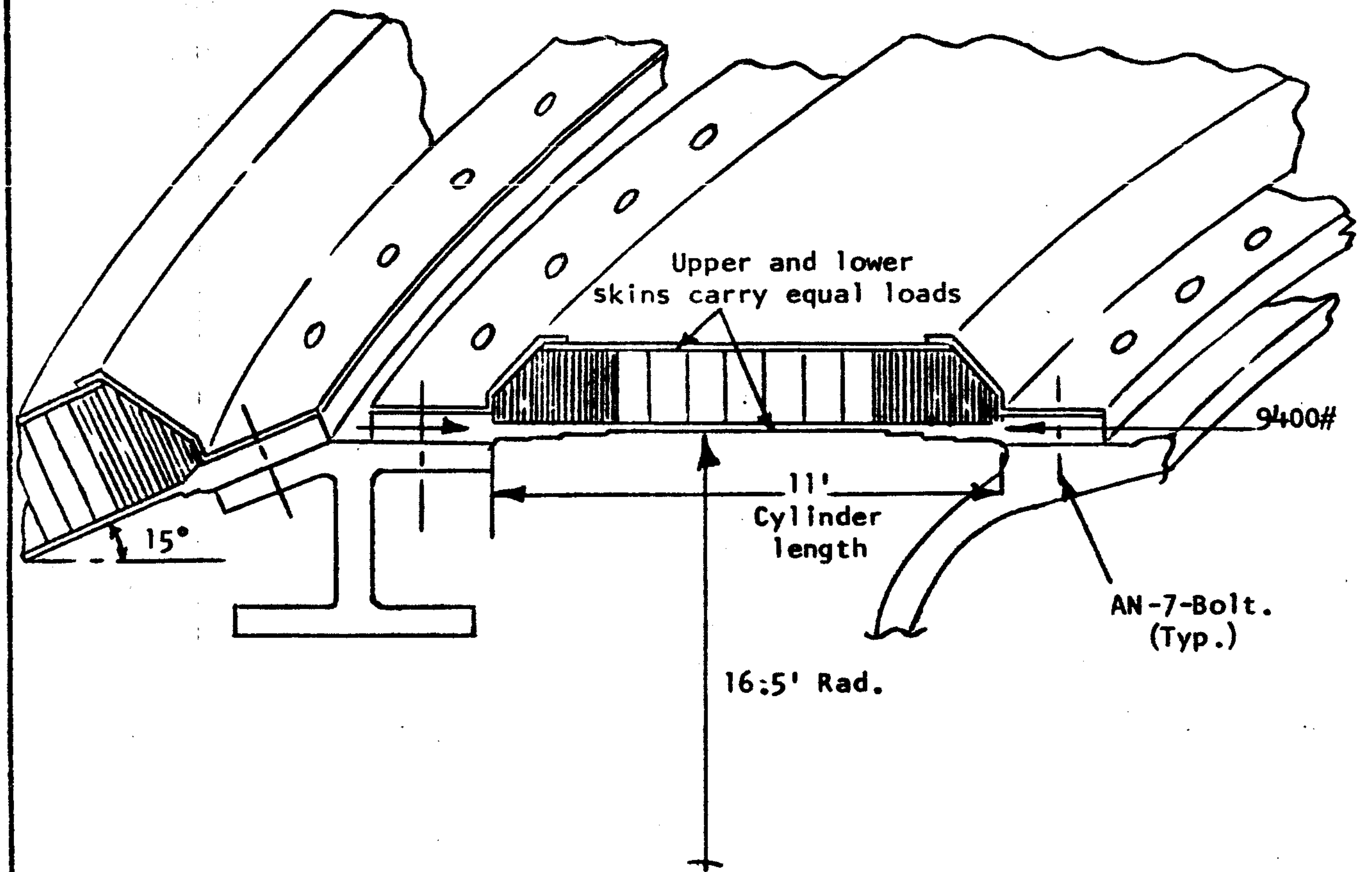
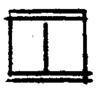


FIGURE 1.

8.1 - 1/8 - 20P (5052) Core



3.1 - 1/8 - 7P (5052) Core

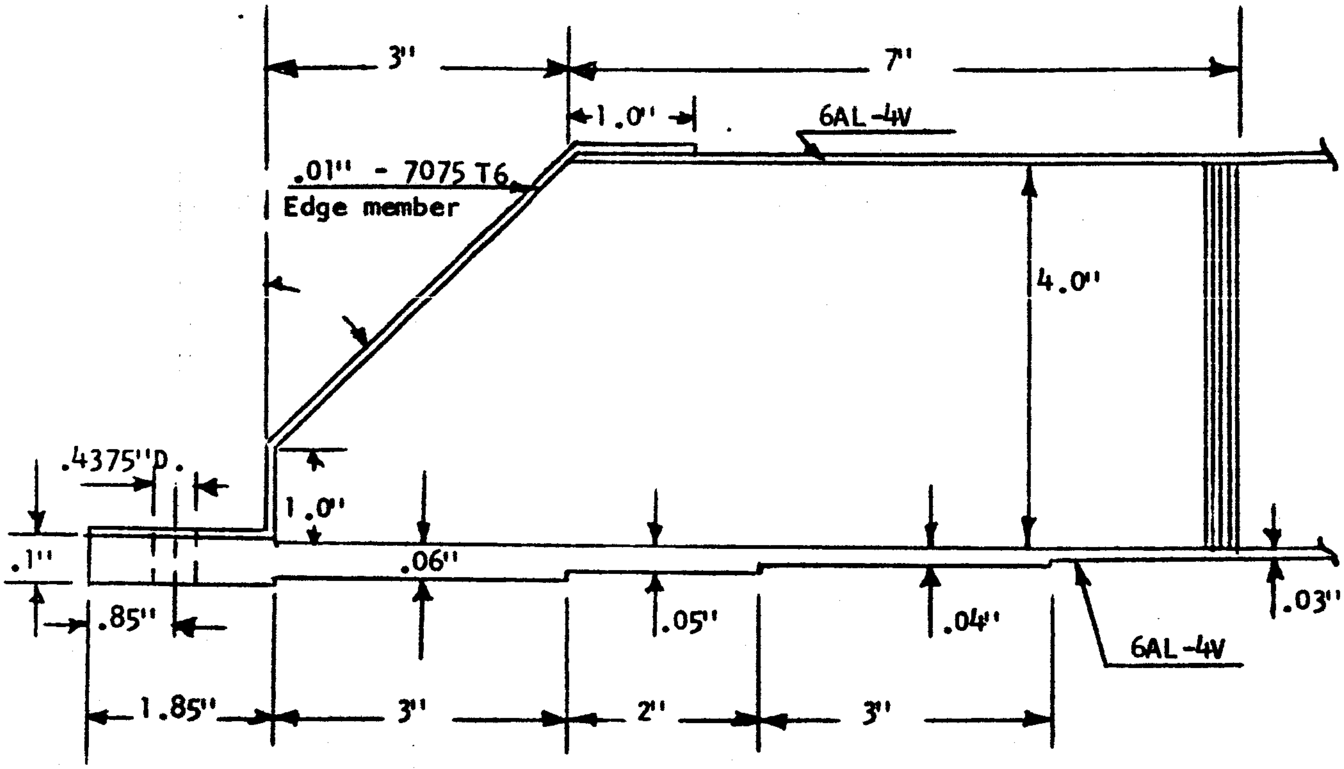


FIGURE 2.

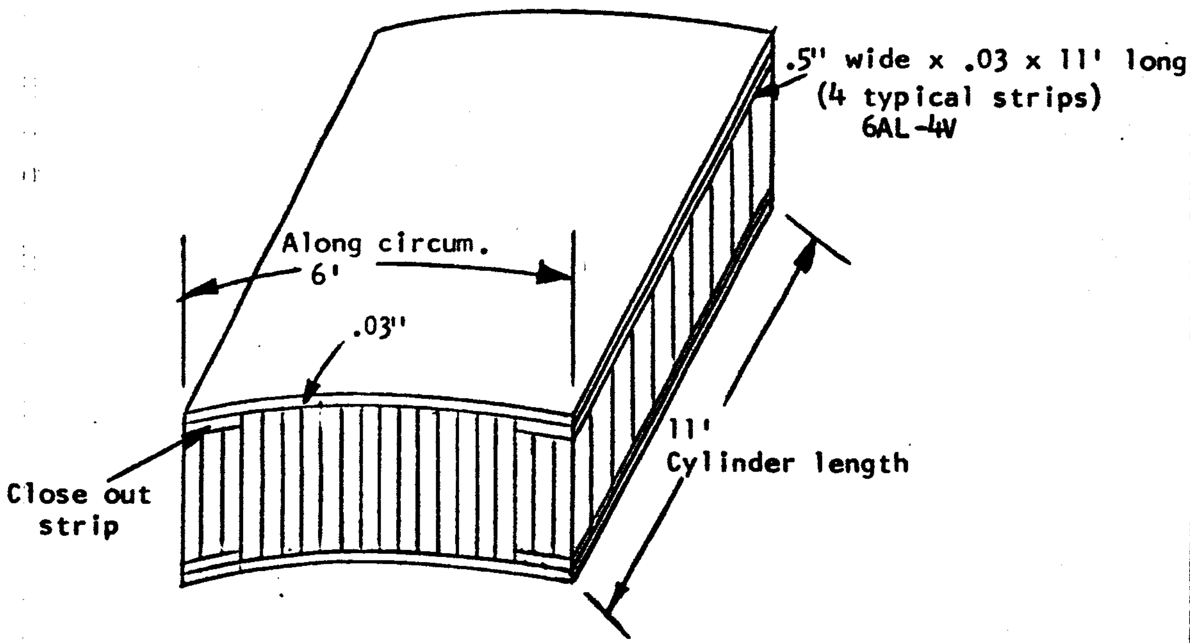


FIGURE 3.

From Tests-----------(Ref. 1)

Ultimate Compressive Axial Load = 74200#

Panel Width = 8"

Skin Thickness = .03"

Material - 6AL-4V - Solution treated and aged (room temperature)

Load/Inch Width = 74200/8 = 9275#

There are 2 skins.

Load/Skin = 9275/2 = 4638#

For design purposes of close-outs, take 4700# per skin.

6AL-4V Properties ---------(Ref. 2)

Ult. Tensile Stress = 170,000 psi
Yield Tensile Stress = 150,000 psi
Solution treated and aged.

Ult. Bearing Stress = F_{bru} - e/D = 2 = 244,000 psi

Yield Bearing Stress = F_{bry} - e/D = 2 = 198,000 psi Annealed.

Shear Stress = 76,000 psi

The loads as shown in Figure 4 are balanced by hoop tension loads at "A" and "B" and a hoop compression load at "C".

The moment induced in the core due to the transfer of 4700# axial load from lower skin to upper skin = 4700 (4 + .03 + .015) = 19,012#.in.

This moment is reacted over 8" length of 8.1#/cu.ft. core.

$\therefore$ The Force = 19,012/8 = 2376#

$\therefore$ Shear in the Core = 2376/4 = 594#/in.

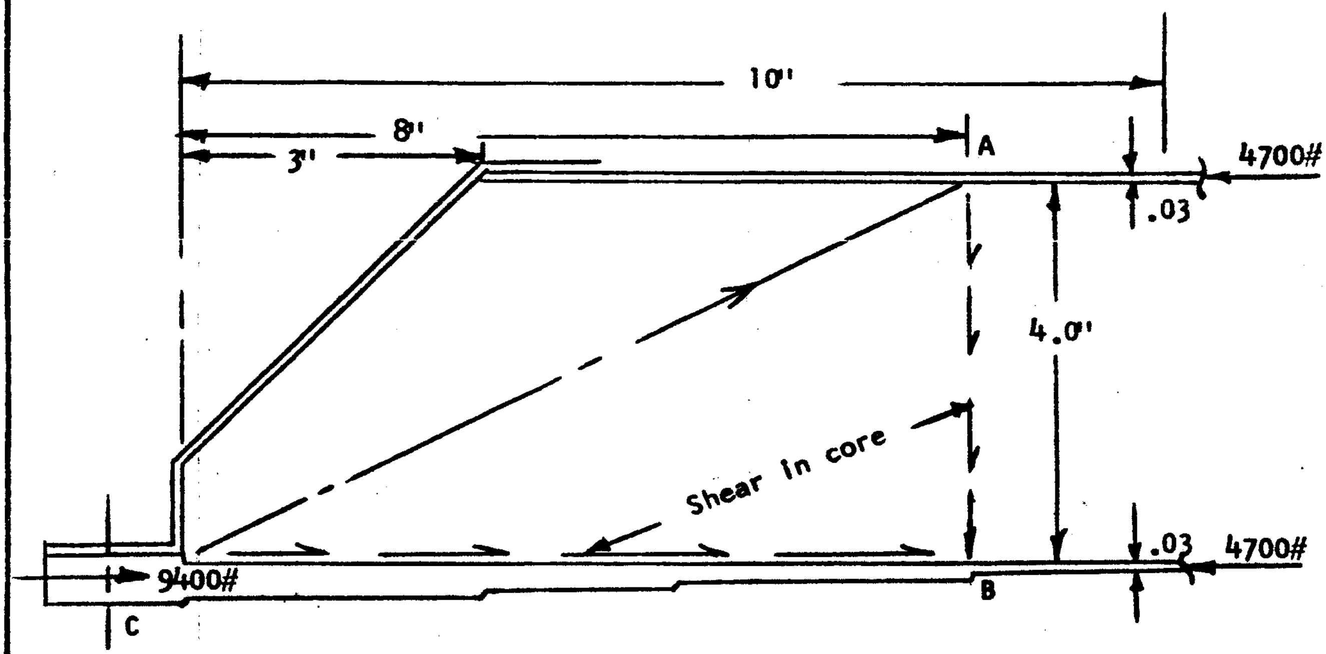


FIGURE 4.

The core is capable of carrying 670#/in-------------(Ref. 1)

.˙. M.S. = 670/594 -1 = .12 ---------------- .12

8.1#/cu.ft. density core is required 2.0" beyond point A (Figure 4) for transferring horizontal load in the core to the upper skin.

7/16" dia. - AN-7 bolts are used for transferring 9400# in single shear per inch around the circumference.

Shear allowable for the bolt = 10,500# -------(Ref. 2)

Checking plate at attachment point.

Bearing area = dia. x thickness.

$= .4375 \times .1 = .04375 \text{ in}^2$

Allowable bearing load = F_{bru} x area

= 24,400 x .04375

= 10,675#

.˙. Bearing is not critical ------------------------------O.K.

This is a compression load. Therefore shear tear-out is not critical. However, in case of load reversal,

Shear tear-out area = 2(.85).1

= .17 in^2

Allowable shear load = 76,000 x .17

= 12,920#

.˙.Shear tear-out is not critical-----------------------O.K.

Lower Skin.

9400# are carried at .06" thickness.

.˙. f_{tu} = 9400/.06 = 156,700 psi

.˙. M.S. = 170,000/156,700 -1 = .08-----------------.08

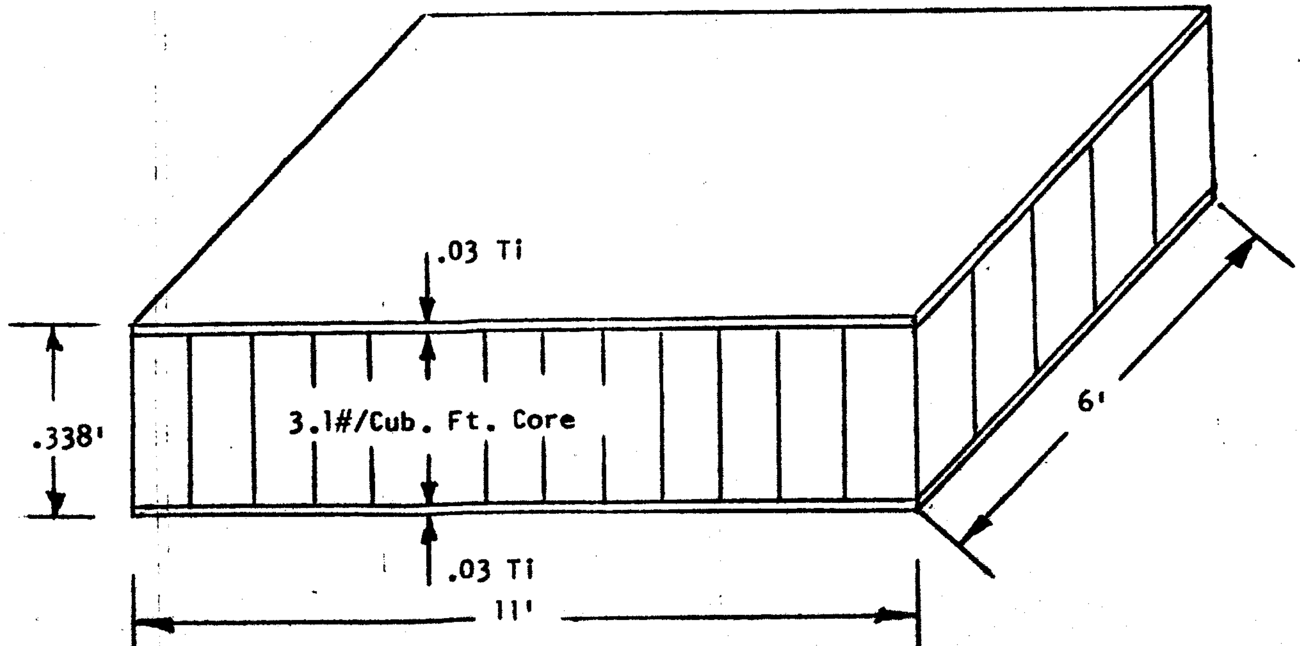


FIGURE 5.

Panel weight for a panel 6' x 11' (assume the panel to be flat for weight study.)

Core Volume = 11 x 6 x .33 = 22.0 cu.ft.

∴ Core weight = 3.1 x 22.0 = 68.2#

Skin volume = [(11 x 12)(6 x 12).03]2 = 570.24 cu.in.

∴ Skin Wt. = .16 x 570.24 = 91.24#

∴ Total panel weight = 159.44#

Weight of 6' x 11' panel with close-outs as shown in Figure 1.

<u>CLOSE-OUT</u> ------(Fig. 2)

Core Volume = .833(6).33 - (.25 x .25/2)6

= 1.501 cu.ft.

∴ Core Weight = 8.1(1.501) = 12.16#

Upper Skin Vol. = 7(72).03 = 15.12 cu.in.

∴ Upper Skin Wt. = .16(15.12) = 2.42#

Lower Skin Vol. = [(.03 x 2)+(.04 x 3)+(.05 x 2)+(.06 x 3)+.1 x .185)]72

= 46.44 in^3

∴ Lower Skin Wt. = .16(46.44) = 7.43#

Vol. of Aluminum Edge Member = (1 x .01+4.3 x .01+.185 x .01)72

= 5.15 in^3

∴ Wt. of Aluminum Edge Member = .1 x 5.15

= .515 .52#

Subtract weight of 7/16" dia. holes

Weight = Number of holes x vol. x density

= 66 x .0152 x .16

= .16#

∴ Net close-out wt. = 12.16 + 2.42 + 7.43 + .52 - .16 = 22.37#

There are 2 close-outs ∴ Total Wt. = 2 x 22.35 = 44.74#

CLOSE-OUT -------- (Fig. 3)

4 edge strips for upper and lower skins --- .5" wide x .03" x 132" long

Volume = 4(.5 x .03 x 132) = 7.92 in^3

∴ Weight = .16(7.92) = 1.28#

∴ Total Wt. of all Close-Outs = 4.74 + 1.28 = 46.02#

Weight of Honeycomb Panel 6' x 9' with 3.1#/ft^3 density core.

Core Volume = 6 x 9 x .33 = 18.0 ft^3

∴ Core Weight = 3.1 x 18.0 = 55.6#

Skin Volume = [(9 x 12)(6 x 12).03]2 = 466 in^3

∴ Skin Weight = .16 x 466 = 74.7#

∴ Total Weight = 130.3#

∴ Total Wt. of Panel Including Close-Outs = 130.3 + 46.02 = 176.32#

∴ Weight Attributable to Close-Outs = 176.32 - 159.44 = 16.88#

∴ Per Cent of Close-Out Weight in a Complete Panel as shown in Fig. 1 =

16.88/176.32 = 9.5% ------------- 9.5%

Circumference = 33 x π = 103.8'   103.8/6 = 17.3 panels

Weight of Complete Skirt Assy = 176.32 x 17.3 = 3048#

REFERENCES:

1. Hexcell Products - Mechanical Properties of Honeycomb Material-TSB120;2/20/64.

2. MIL-HDBK-5 - August 1962.