# DEVELOPMENT OF ADVANCED COMPOSITE STRUCTURES

# FOR LOCKHEED AIRCRAFT

Warren A. Stauffer and Arthur M. James
Lockheed-California Company

## SUMMARY

Lockheed is currently engaged in three NASA-sponsored Aircraft Energy Efficiency (ACEE) composite structure programs: the L-1011 Advanced Composite Vertical Fin (ACVF), the L-1011 Advanced Composite Aileron (ACA), and a wing study program. These programs are all structured to provide the technology and confidence for a Lockheed committment to the use of advanced composite materials for both primary and secondary structures of future transport aircraft. Material tests conducted on advanced composite material indicate that the 400K (260°F) cure system tested should be limited to structural applications not exposed to high temperature and moisture, while the 450K (350°F) cure system appears adequate for the environment of transport aircraft. The FAA Advisory Circular on certification of composite structure emphasizes the need to identify and account for the effects of the environment (moisture and temperature) on mechanical properties.

## INTRODUCTION

Lockheed is currently engaged in three NASA-sponsored Aircraft Energy Efficiency (ACEE) composite structure programs: the L-1011 Advanced Composite Vertical Fin (ACVF), the L-1011 Advanced Composite Aileron (ACA), and a wing study program. These programs are structured to provide the technology and confidence for a Lockheed committment to the use of advanced composite materials for primary and secondary structures of future transport aircraft.

The ACVF program encompasses the design, fabrication, and certification of an advanced composite vertical fin installed on L-1011 transport aircraft. The primary emphasis of the program is to gain a level of confidence in the structural integrity of advanced composite structures through an expanded series of ground tests. Included are tests of selected subcomponent elements for four years in loading/environment conditions which will simulate twenty years of service, and includes the fabrication of three full-scale components for test and to verify manufacturing cost projections. These tasks were preceded by extensive tests to characterize and determine the effects of airline environments on candidate graphite/epoxy material systems. The latter led to a material change from a 400K (260°F) cure graphite/epoxy system (T300/5209) to a 450K (350°F) cure graphite/epoxy system (T300/5208) to provide greater margins in structure subjected to extreme operational environments.

The current status of these tasks is discussed in this paper. The results of coupon tests for both material systems are presented as well as the ACVF environmental (moisture and temperature) requirements.

The effect of moisture and temperature on the mechanical properties of advanced composite materials as shown in this paper can be quite significant. For this reason, the FAA has recently published an Advisory Circular relating to Certification Guidelines for Civil Composite Aircraft Structure. The requirements set forth in this circular are discussed as they relate to the ACVF.

## SYMBOLS

Values are given in both SI and U.S. Customary Units.

| | |
|---|---|
| $f_{notched}$ | notched laminate strength (gross area), MPa (KSI) |
| $f_{unnotched}$ | unnotched laminate strength, MPa (KSI) |
| RH | relative humidity, percent |
| T | time, hr |
| $T_{AMB}$ | ambient temperature, K ($^{\circ}$F) |
| $T_{MAX}$ | maximum vertical fin box thickness, m (in.) |

## ADVANCED COMPOSITE VERTICAL FIN PROGRAM (ACVF)

### Structural Configuration

The ACVF is defined as the main structural box of the L-1011 vertical fin; it includes the front and rear spars, left and right covers, and all ribs (see Figures 1 & 2). The tip closure rib, hinge and actuator fittings, and the auxiliary spar are metal. The covers feature a single-stage cure assembly of prebled skins and prebled hat stiffeners. At the lower end of each cover, the stiffeners are cut off at a 45$^{\circ}$ angle for the cover to mate with the existing fuselage-to-cover bolted joint. The covers utilize 0$^{\circ}$, ±45$^{\circ}$ T300/5208 graphite/epoxy material with thicknesses and ply orientations selected to match the strength and stiffness requirements of the metal fin. The substructure consists of ten full ribs, one partial rib, and continuous front and rear spars. The spars are molded from T300/5208 graphite/epoxy tape material with integrally molded rib attach angles and stiffeners. Two different rib designs are utilized in the ACVF: solid web ribs integrally stiffened for the upper three ribs and partial rib, and truss ribs with graphite/epoxy molded caps and aluminum diagonals for the remaining seven full-size ribs.

The current indicated production cost of the ACVF (cumulative average for 100 aircraft) is 9% below the estimated cost of a metal fin, determined on the same basis. The indicated weight saving is currently 27.6% (107.3KG (236.6 lbs.)). Composite material utilization is currently predicted to be 77.1% of the redesigned fin box weight.

## Environmental Considerations

Environmental considerations of particular importance to the ACVF are moisture and temperature. Available data on graphite/epoxy composites indicate mechanical property reduction when exposed to high temperature and moisture. The extent of this reduction, particularly on matrix dominated properties, is influenced by the moisture content of the material and the ratio of the operating temperatures to the glass transition temperature of the matrix material. Therefore, it is essential that moisture and temperature conditions be considered in the design of composite structures.

For the ACVF, the moisture absorption for graphite/epoxy is conservatively assumed to correspond to 1.0% weight gain. With 1.5% weight gain assumed for fully saturated graphite/epoxy (T300/5208), 1.0% represents exposure to about 67% relative humidity 100% of the time.

The temperature range applicable to the ACVF is 219K ($-65^{o}$F) to 355K ($+180^{o}$F). The 219K is based on L-1011 specifications, and the 355K was determined from a solar heating analysis conducted on the ACVF. A survey of world commercial airport weather patterns led to the selection of Las Vegas, Nevada as being representative of the most adverse temperature conditions. An ambient temperature of 319K ($114^{o}$F) can occur on the order of once in 10,000 hours as shown in Figure 3.

A review of airline tail logos used on L-1011 aircraft in service indicates that dark colored paints, such as dark blue, are commonly used. The significance of color on skin temperature is indicated in Figure 4. In still air (ambient air temperature of 319K), with the sun position at $45^{o}$ relative to the fin surface, the skin temperature with dark blue paint increases to 365K ($182^{o}$F); with white paint, the skin temperature is 23K ($42^{o}$F) less. The ground time-temperature histories due to solar heating for three structural elements (cover, rib cap, and spar cap) are shown in Figure 5. For these calculations, the horizontal stabilizer was assumed to be unpainted with a solar absorptivity of 0.5 and reflectivity of 0.5. Direct solar heating of the fin and heating due to reflection from the horizontal stabilizer are accounted for in the thermal analysis. Cooling of the sun-facing surface of the fin by external radiation and internal radiation to the shadow exposed surface is also included. The maximum temperature reached for the cover is 363K ($194^{o}$F). Figure 6 gives the temperature time history of the cover during taxi, takeoff, and climb after ground solar heating. It is assumed that during taxi the sun-facing fin surface continued to be exposed to the maximum solar heating intensity, and only during takeoff and climb was this surface assumed to be oriented away from the sun. It can be seen from Figure 6 that the maximum structural temperature for the skin is 355K ($180^{o}$F) at the beginning of takeoff and the maximum structural temperature at the end of the takeoff run is 347K (165°F), which occurs in the crown of the hat stiffener. Since design loads can occur any time during the takeoff run, the temperature of 355K (180°F) has been established as the design maximum temperature for the ACVF.

## Material Characterization

Characterization of the following advanced composite material systems has been completed:

- o Thornel 300/5209
- o Kevlar 49/5209 181 Fabric
- o Thornel 300/5208

The 5209 resin is a 400K (260°F) cure material, and the 5208 resin is a 450K (350°F) cure material. The characterization program was structured to establish basic lamina property behavior as it varies with temperature for both dry and wet conditioning.

The temperature range covered in this program was 219K (-65°F) to 355K (+180°F). This temperature range was assumed to adequately cover that expected in commercial service. Coupons were wet conditioned to ensure a minimum of 1% weight gain. Conditioning was accomplished in a humidity chamber at 95 to 100 percent relative humidity at 339 ±3K (150° ±5°F) for times determined by coupon thickness.

The material characterization results indicate that the effect of moisture and temperature is most pronounced on the 400K (260°F) cure resin material system, particularly on the material properties which are matrix dominated. Figures 7 through 9 show the reduction in compression strength for T300/5209, T300/5208 unidirectional material and Kevlar 49/5209 181 cloth due to the combined effects of moisture and temperature. Whereas elevated temperatures alone do not cause a significant reduction in longitudinal compressive strength, the combination of temperature and moisture does result in drastically reduced compressive strengths for 400K (260°F) cure resin system. This behavior is explained by the fact that moisture conditioning reduces the glass transition temperature of the resin. Subsequently, at elevated temperatures the matrix does not provide sufficient support for the fibers, resulting in low failure stresses. Comparison of the room temperature dry compression data with the 344K (160°F) wet compression data (Figure 7) indicates a 60 percent reduction in compression strength for the T300/5209 material system, while the T300/5208 material system comparison indicates a 13 percent reduction at 355K (180°F). For Kevlar 49, the reduction in compression strength is 62 percent going from room temperature dry to 344K (160°F) wet as shown in Figure 8. It should be noted that the longitudinal modulus is not significantly influenced by either moisture or elevated temperature for the two graphite/epoxy material systems. However, Kevlar 49/5209 181 cloth material system longitudinal moduli are significantly affected as shown in Figure 9. These data indicate that the fiber, in addition to the matrix, is adversely affected by moisture and elevated temperature.

Laminate coupon test results of both T300/5209 and T300/5208 are shown in Figures 10 and 11. The test variables include notched and unnotched coupons, room temperature dry and elevated temperature wet. The coupons are approximately 25.4 x 279.4mm (1 x 11 inches) with fiberglass tabs bonded to the ends to leave a 152.4mm (6 inch) gage length. All coupons were strain gaged. The notched specimens have centrally located 4.76mm (3/16 inch) diameter holes. The ends are clamped with bolted fixtures carefully aligned in the test machine. To prevent buckling, the gage region is clamped between two "I" shaped, Teflon-coated steel plates with cutouts for strain gage and thermocouple wires, see Figure 12. The specimen and fixtures are enclosed in a temperature-controlled box, as shown in Figure 13, and the thermocouples are monitored for coupon temperature. Most of the coupons failed in the gage length. All of the coupons with 50%, $0^{\circ}$ plies, fabricated from T300/5209, failed at the grip, apparently due to misalignment.

The results of the laminate strength tests again indicate a significant drop in compressive strength of the T300/5209 composite material when exposed to elevated temperature and moisture. When compared to room temperature dry properties, the unnotched compressive strength drops 60 percent at 344K ($160^{\circ}$F) with 1% moisture content for laminates with 50%, $0^{\circ}$ plies, and a 45% drop for laminates with 14%, $0^{\circ}$ plies, see Figure 10. For T300/5208 (Figure 11), the unnotched compressive strength drops 10 percent at 355K ($180^{\circ}$F) with 1% moisture for laminates with 50% or 14%, $0^{\circ}$ plies. The compression ultimate of the laminates tested can be characterized by a failure mode involving delamination of the laminate into sublaminates and crippling of the $\pm 45^{\circ}$ sublaminates. The $0^{\circ}$ sublaminates sometimes cripple and delaminate locally and sometimes appear to simply crush and fail without further delamination on a shear plane at about $40^{\circ}$ to the surface. This failure mode is highly sensitive to minor variations in test support such as fixture clamping, minor test eccentricities, and variations in unsupported column length. This sensitivity produces large test scatter, in spite of the special care taken to minimize testing variables. From the nature of the failure mode, one must conclude that the test results are also a strong function of the test specimen detail design, and not necessarily relevant to performance in aircraft structure.

It is quite evident from the results of both the lamina and laminate coupon tests that the application of the 5209 resin system should be limited to structure not subjected to high temperature and moisture.

The effect of notches on the tension and compression strength of T300/5208 laminates is shown on Figure 14. It is quite evident from this figure that a significant reduction in both tension and compressive strength occurs in the presence of a notch. The notch factor, $f_{notched}/f_{unnotched}$ based on gross area for tension is .49 at room temperature and .69 for compression at 219K ($-65^{\circ}$F). The reduction in strength in tension in the presence of a notch far outshadows the reduction in strength in compression in the presence of a notch including moisture and elevated temperature effects.

## Production Readiness Verification Testing

The ACVF program provides for multiple large-scale subcomponents of the structure for evaluation of variability in static strength and for assessment of durability under extended time laboratory tests involving both load and environment simulation. This production readiness verification testing (PRVT) program is supplemental to the ancillary test program. These tests are designed to provide information to answer the following questions:

1. What is the range of production qualities that can be expected for components manufactured under conditions similar to those expected in production, and how realistic and effective are proposed quality standards and quality control procedures?

2. What variability in static strength can be expected for production quality components, and are the design margins sufficient to account for this variability?

3. Will production quality components survive laboratory fatigue tests involving both load and environment simulation of sufficient duration and severity to provide confidence in long-term durability in the service environment?

The questions are not primarily directed towards basic material properties. It is believed that the combination of service experience on secondary structures and coupon tests in the ancillary test program provide confidence in durability of the basic material. The questions are directed instead to the realities of production quality as influenced by cost objectives and by scale-up and complexity effects which will cause structural quality to differ from that represented by idealized small coupons.

Twenty duplicates of each of two key structural elements of the ACVF will be fabricated for test. One element will represent the front spar to fuselage joint area, and the other element will represent the cover to fuselage joint area, see Figure 15. These elements are identical to subcomponent test specimens which will be static tested prior to the fabrication of the PRVT specimens. These specimens will be manufactured as near as practical under conditions anticipated for production hardware and then subjected to NDI procedures. Anomalies from baseline accept/reject criteria will be reviewed and a decision made on whether to accept, accept conditionally, repair, or reject. Ten specimens of each type will then be subjected to static strength tests; and the other ten of each type will be used for durability tests.

At the onset of the durability tests, accept/reject criteria will have been established for many aspects of fabrication variability. Many of these should not affect static strength but are yet unproven from the standpoint of durability for commercial aircraft which have numerous ground-air-ground thermal cycles and the requirement for long life with minimal maintenance. Typical of this category of criteria are those for voids, hole quality, fiber volume ratios, porosity, wrinkles, acceptance of prepreg, etc. If these factors are too closely controlled, they can increase production costs. It too loosely controlled, they could impair durability. Therefore, these criteria must be evaluated carefully. To do this, it is necessary to define the quality that can in fact be achieved during production and second to substantiate that the quality that can meet economic requirements will be able to sustain the expected service loads and environments. Upon completion of the durability portion of the PRVT program, it must have been demonstrated that the potential in-service durability of the ACVF is as good as or better than that of its metal counterpart.

The test fixtures and control system for the durability tests will be designed and fabricated to minimize the need for monitoring and maintenance during the course of the test. In particular, this will include:

- o Computer control of loads and environment
- o Malfunction detection devices for remote monitoring
- o Premium quality load jacks to reduce down time and repairs
- o A central environment mixing chamber with an environment distribution system to individual test specimens

The general approach to test equipment is shown on Figures 16 and 17.

The durability of the ACVF will be demonstrated by testing ten each of the two specimens in an extended time environment/load test. The load and environment spectra will be of the type and general make-up shown on Figure 18. This figure shows anticipated upper and lower bound test temperatures, segments of the flight to be considered and other appropriate conditions to define the flight environmental and load profile. Salient features to be incorporated into the finally selected test/environment spectra will be:

- o The time-at-load for at least the higher loads in the spectra will be near real time. Since the gust loadings for the tail range in duration from five to eight seconds per cycle depending on aircraft speed and altitude, there should be no difficulty in attaining this simulation.
- o Temperature and moisture cycles will be approximately real time in the critical portions where gradient effects are expected to be most critical. In particular, this will include the heat up to 355K (180$^{\circ}$F) and the cool down simulating take-off.
- o Ground storage will be accelerated by adjustment of temperature and humidity so that the anticipated service moisture levels will be approximately maintained.
- o Moisture/temperature control at the central environment chamber will be the same for all specimens. Chamber

design and ducting controls will differ for the cover and spars to convert the baseline environment into that for the cover and spar. For the cover it is anticipated that the heating and cooling will be accomplished from the outside surface with the reverse side permitted to seek a cycle similar to that expected of the ACVF interior. For the spar, an effort will be made to control the temperature/humidity cycle to approximate that of the near surface and interior of the ACVF by appropriate simulation of the surface structure in the design of the chamber. Thus by using the central environment chamber to approximate the external ACVF environment, along with approximations of the manner in which the structure is being heated and cooled, the gradient effects will be approximated during the course of the tests. The actual cycle development will be accomplished by a combination of analyses, small coupon tests, and trial and error simulations during the development and setup of the test facility.

The durability specimens will be tested side-by-side to load/environmental spectra of sufficient severity to assure that twenty years of service usage will be duplicated in four years of laboratory testing. Periodically each specimen will be inspected per field service inspection technique. Any damage will be monitored for growth, or will be repaired using field service repair techniques, depending on extent and location of the damage. At the completion of the long-term durability tests, each specimen will be inspected and extent of damage assessed. Five of each of the specimens will be selected for residual static strength tests. The remaining ten specimens will be held, subject to an option to continue durability testing.

## ADVANCED COMPOSITE AILERON (ACA)

### Structural Configuration

The ACA is defined as the main structural box of the inboard aileron on the Lockheed L-1011 aircraft. The inboard aileron is located on the wing trailing edge between the outboard and inboard trailing edge flaps and is directly behind the engine as shown on Figure 19. The existing metallic design shown in Figure 20 is a single cell box beam with added trailing edge wedge, leading edge shrouds, and end fairings. The box consists of a front spar, a rear spar, and upper and lower skins, joined by 18 ribs.

Certain subassemblies used on the aluminum aileron will also be used for the composite aileron. These include the leading edge shrouds, the end fairings, the trailing edge wedge, the shroud supports, feedback fitting and the hinge/actuator fittings. These subassemblies were not redesigned because analysis indicated it would not be cost effective and no significant weight savings could be achieved.

## Engineering Development

Various combinations of structural configurations and materials have been evaluated for the major subcomponents of the aileron. For each configuration, a variety of materials and materials combination was also studied. A total of thirty-one designs were evaluated for the covers, front spar, rear spar, intermediate ribs, main ribs, and rib backup fittings.

Each design was subjected to a quantitative analysis to determine weight and recurring cost and qualitative analysis of such factors as manufacturing processes, inspectability, etc. Results of the studies for each subcomponent are displayed in a matrix format to allow selection of the most viable concept. The evaluation matrix for the covers is shown in Table 1.

Based on the evaluation of the subcomponents, two ACA assembly concepts were selected for further evaluation. The first, shown in Figure 21, is a honeycomb core sandwich cover design with no intermediate ribs. This concept was selected because it offered the greatest cost saving potential. The second, shown in Figure 22, is a multi-rib design with a graphite/syntactic sheet skin cover. This design offered the greatest weight savings potential. A detailed cost and weight analysis, and qualitative assessment of factors such as producibility, maintainability, inspectability, etc. has lead to selection of the multi-rib concept for the ACA.

# CONCLUDING REMARKS

The potential weight and cost saving which will lead to more energy efficient structure have been shown in several ways. However, the needed confidence, data base, and analysis methods are only now materializing. These data are identifying the characteristics of composite materials which are vital for good design, i.e., the effects of moisture, temperature, notches, etc. on material properties. However, the long-term effects of the environment on the durability of composite aircraft structure still need to be identified. These effects will be determined through a controlled environmental load program such as the Production Readiness Verification Test (PRVT) program.

TABLE 1 - COVER EVALUATION MATRIX

| Concept | Sheet Skin | | | | Sandwich | | | Stiffened Skin | |
|---|---|---|---|---|---|---|---|---|---|
| Materials & Construction | Gr Tape | Gr Tape | Gr Tape Syntactic Core | Gr Tape Syntactic Core | K49 Skins Nomex Core | Gr Skins Nomex Core | Hybrid Gr-K49 Skins Nomex Core | Gr Tape 5 Hats | Gr Tape 6 Hats |
| No. of Intermediate Ribs | 6 | 10 | 5 | 4 | 0 | 0 | 0 | 0 | 1 |
| Weight Ratio (1) to Aluminum | 68.6% | 66.2% | 54.0% | 59.1% | 81.7% | 77.7% | 79.7% | 87.2% | 83.1% |
| Cost Ratio (1) to Aluminum | 92% | 120% | 88% | 91% | 67% | 82% | 72% | 99% | 115% |
| Qualitative (2) Assessment | 3.0 | 2.7 | 2.8 | 3.0 | 1.8 | 1.7 | 1.5 | 1.8 | 1.8 |

(1) Includes covers and intermediate ribs (if required).

(2) Includes producibility, inspectability, maintainability, repairability, etc. rated on a scale from 1 to 3 with 3 being best.

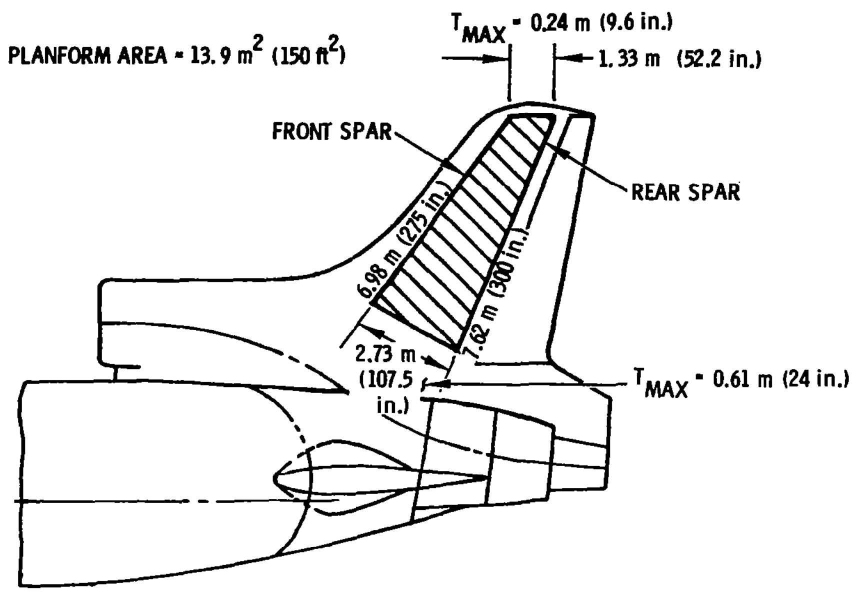


Figure 1.- L-1011 vertical fin box basic data and orientation.

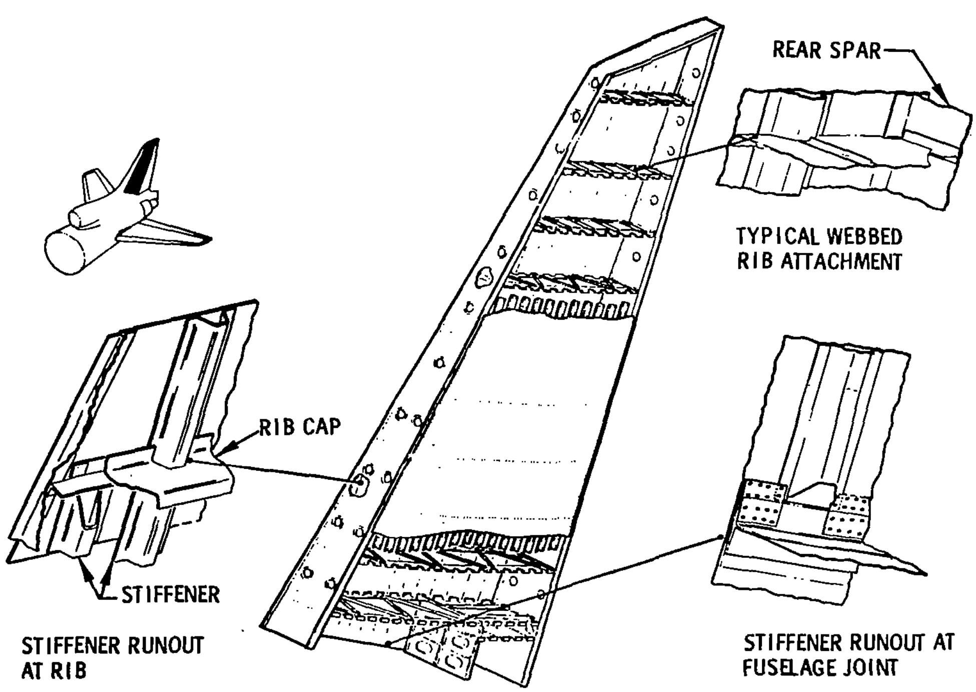


Figure 2.- ACVF configuration.

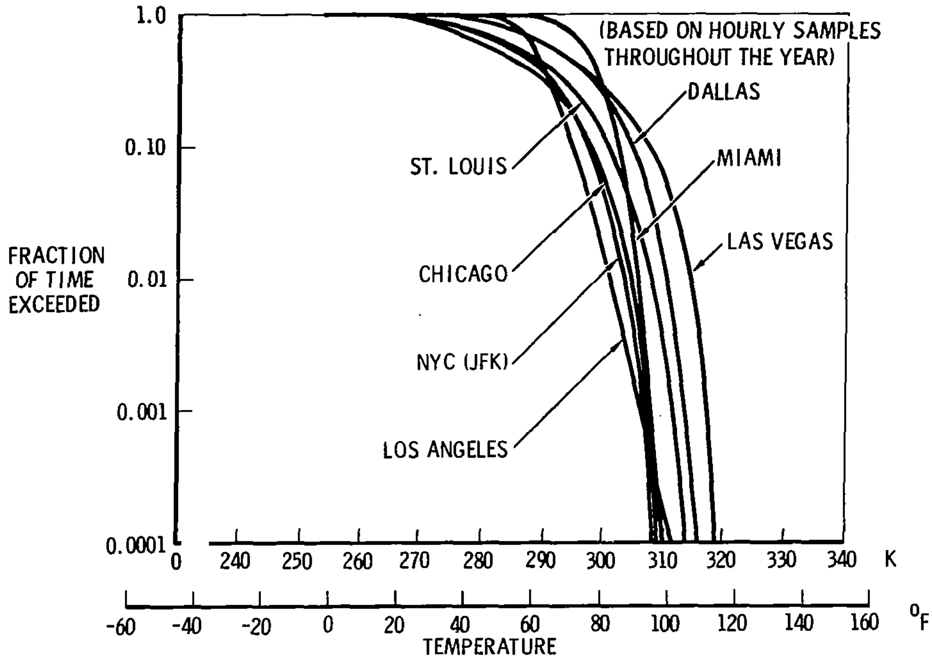


Figure 3.- Fraction of time exceeded — temperature of selected U.S. cities.

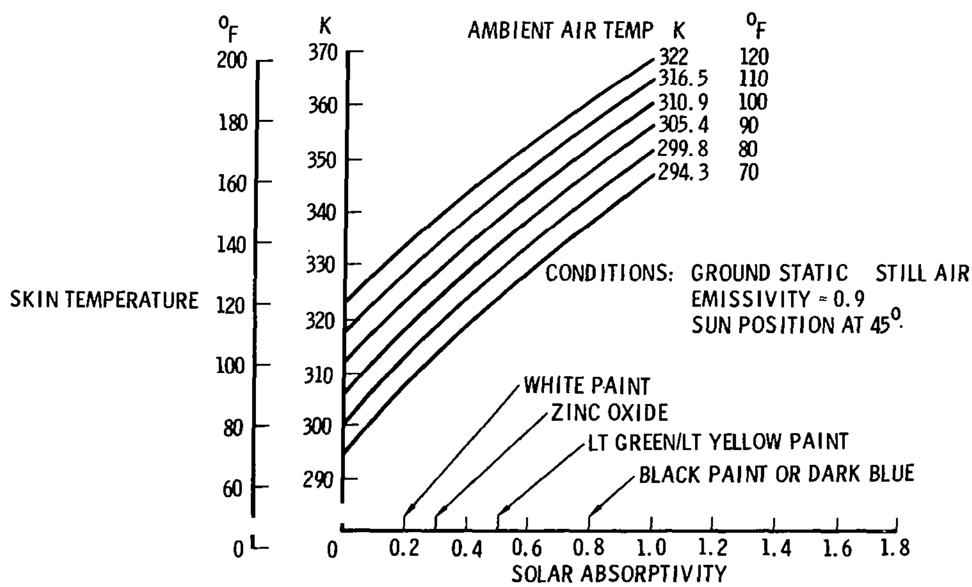


Figure 4.- Steady state — maximum vertical fin skin temperatures.

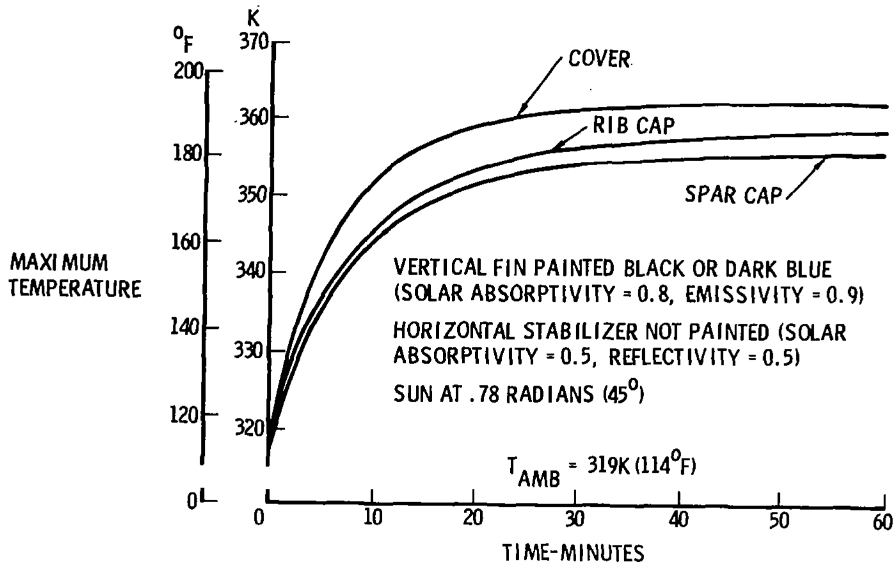


Figure 5.- Time to reach maximum skin temperatures.

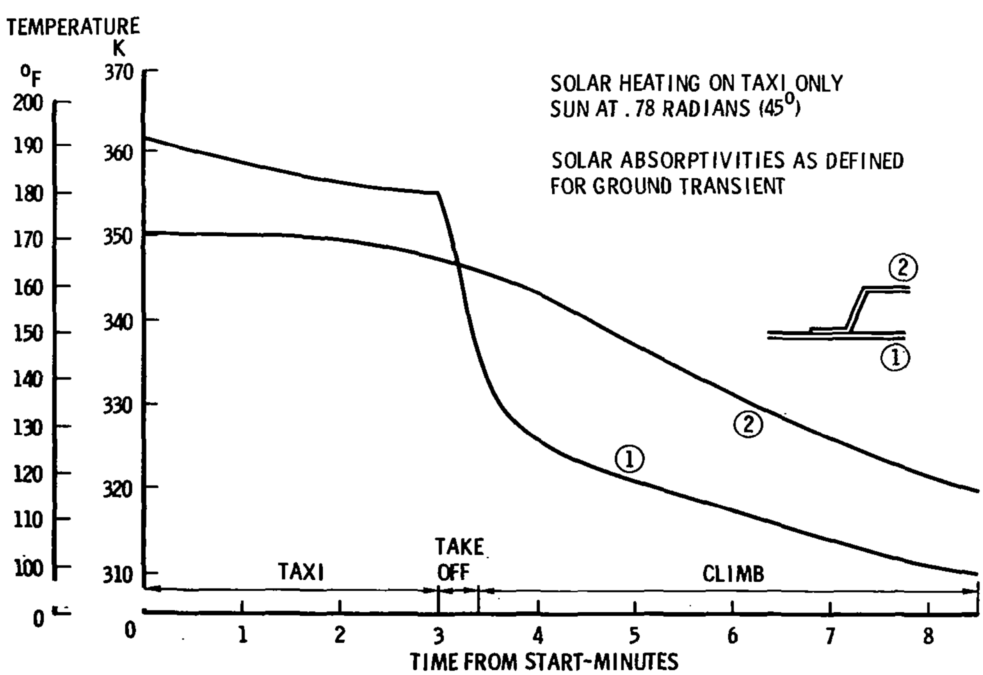


Figure 6.- Temperature-time history — cover.

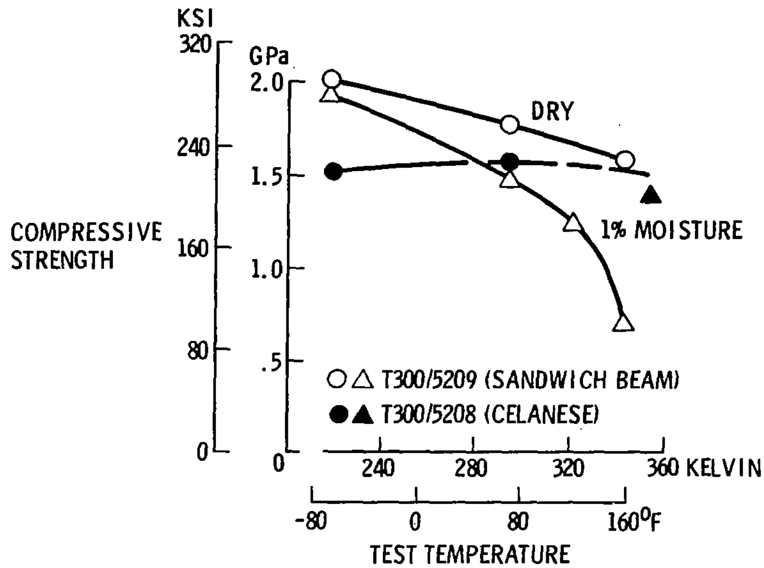


Figure 7.- Effect of moisture and temperature on compressive strength graphite/epoxy (0°).

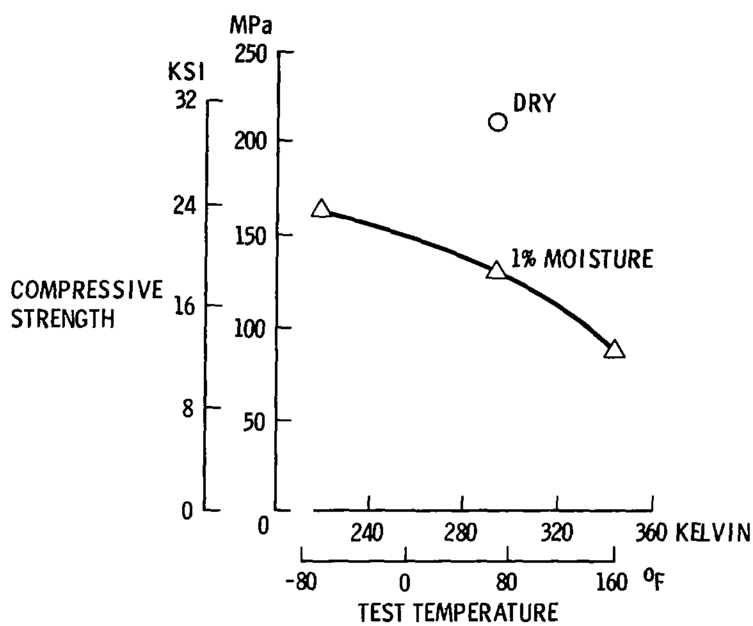


Figure 8.- Effect of moisture and temperature on compressive strength Kevlar 49-181 fabric/5209 (0°). 1.27- by 3.81-cm (0.5- by 1.5-in.) specimen.

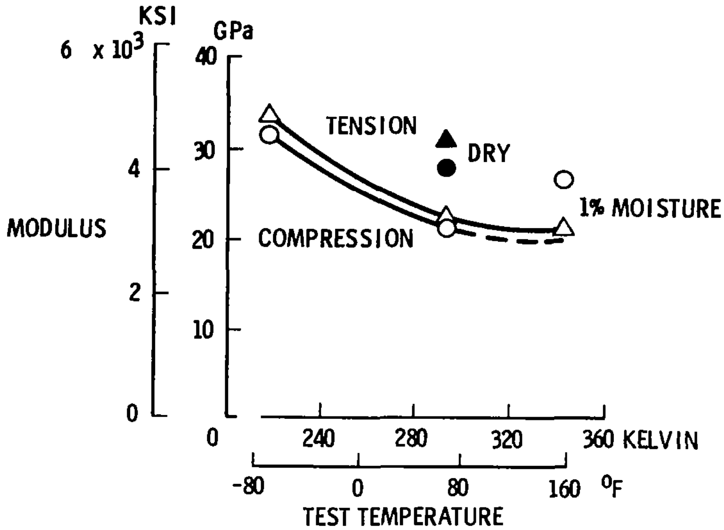


Figure 9.- Effect of moisture and temperature on modulus Kevlar 49-181 fabric/5209 (0°).

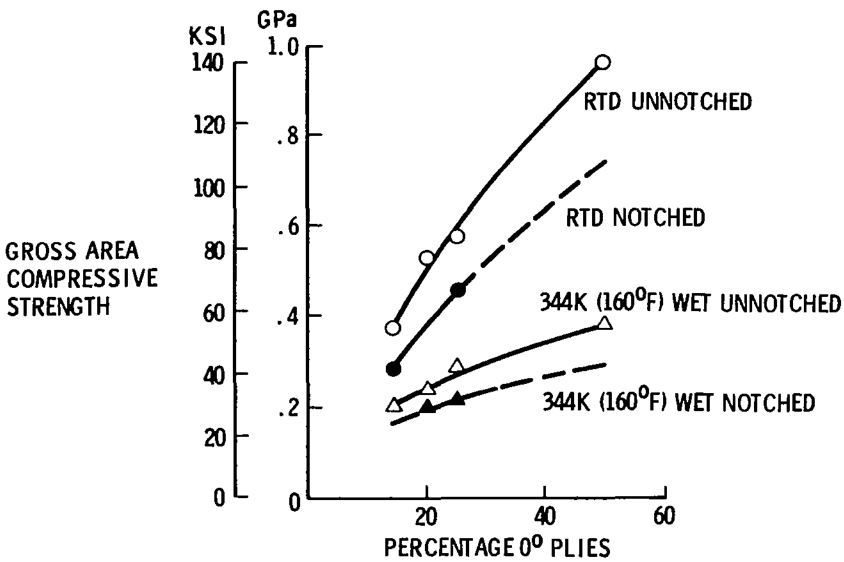


Figure 10.- Laminate compressive strength comparison T300/5209 graphite epoxy laminate ($\pm 45^{\circ}_{i}$, $0^{\circ}_{j}$). Notch: 0.48-cm (3/16-in.) diameter hole; 2.54-cm (1-in.) wide specimen.

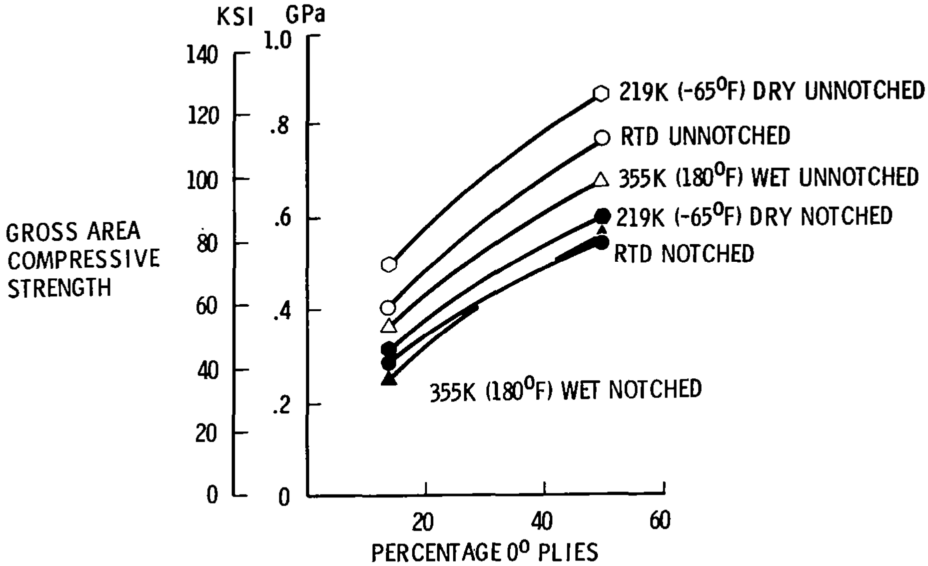


Figure 11.- Laminate compressive strength comparison T300/5208 graphite epoxy laminate ($\pm 45^{o}{}_{i}\ 0^{o}{}_{j}$). Notch: 0.48-cm (3/16-in.) diameter hole; 2.54-cm (1-in.) wide specimen.

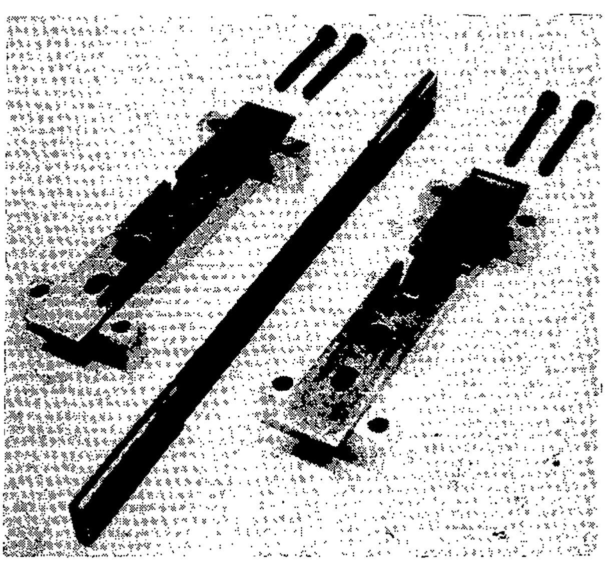

Figure 12.- Full-fixity apparatus-laminate testing.

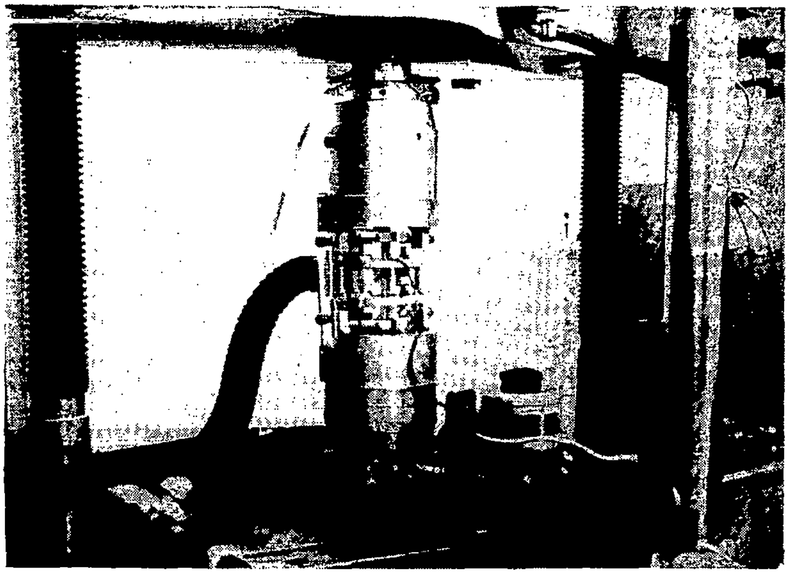

Figure 13.- Laminate compression test apparatus.

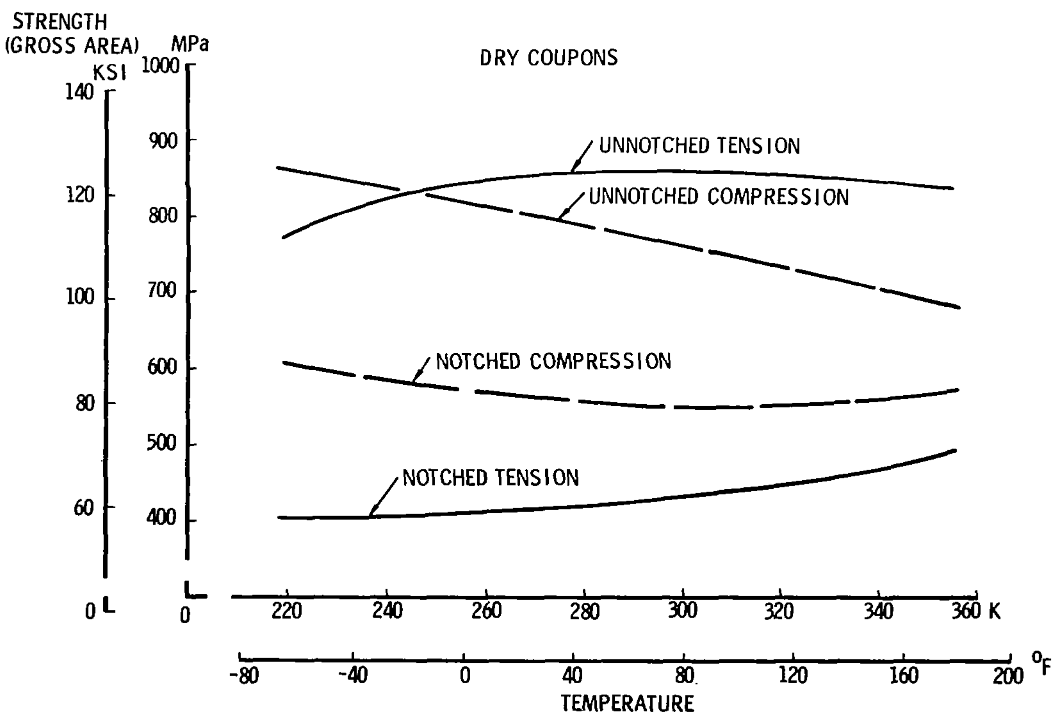


Figure 14.- Advanced composites. T300/5208 graphite/epoxy ($0^{o}_{2}$/$+45^{o}$) 16 ply coupons; 50 percent, $0^{o}$ plies.

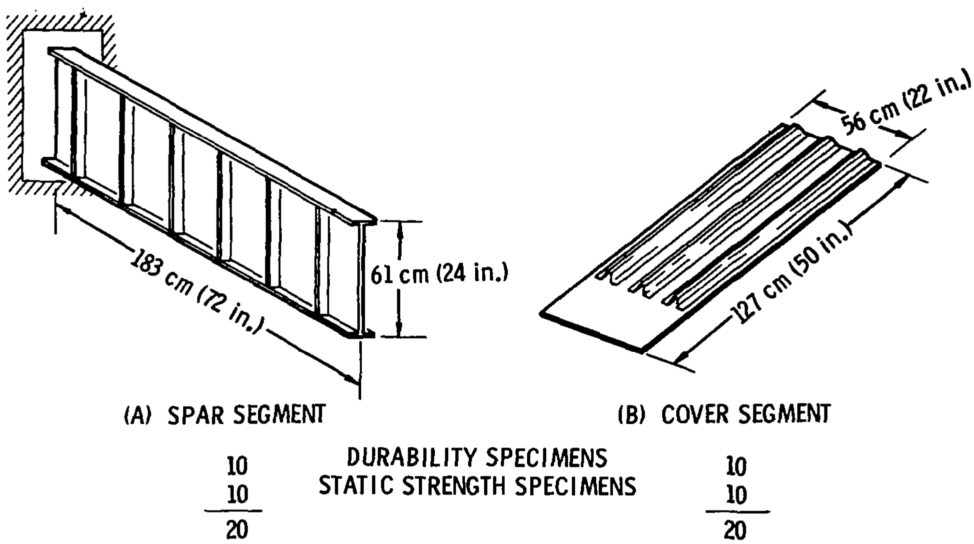


Figure 15.- Production readiness verification tests (PRVT).

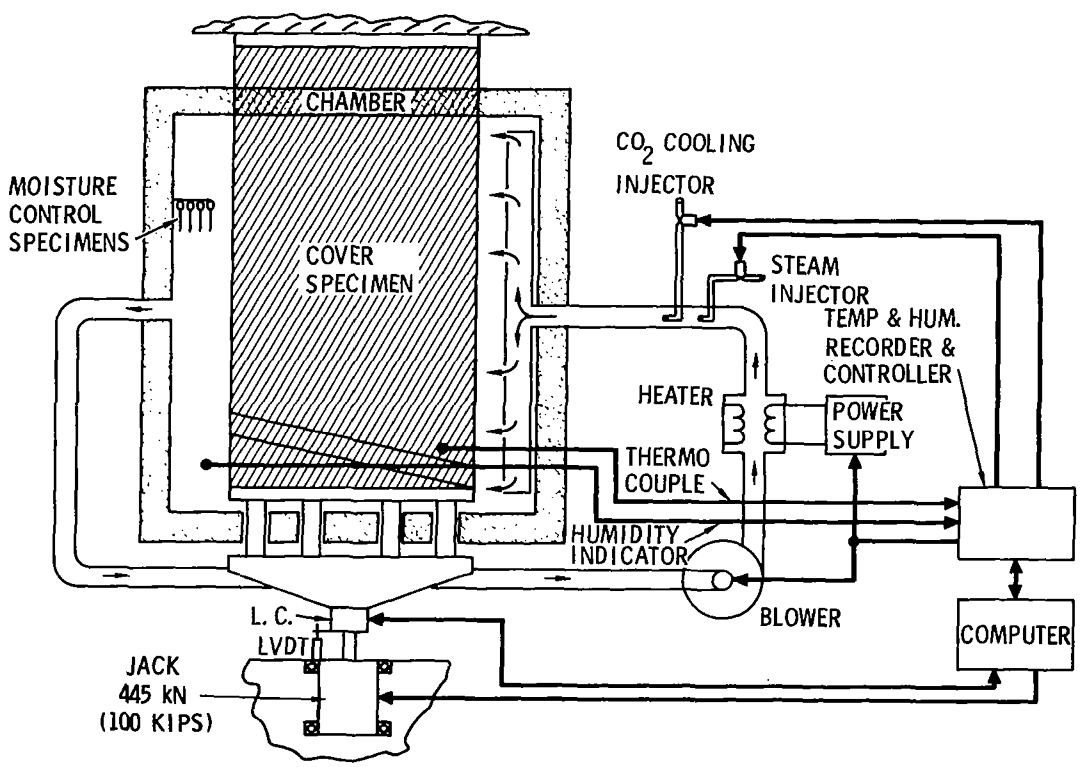


Figure 16.- PRVT schematic of cover test setup.

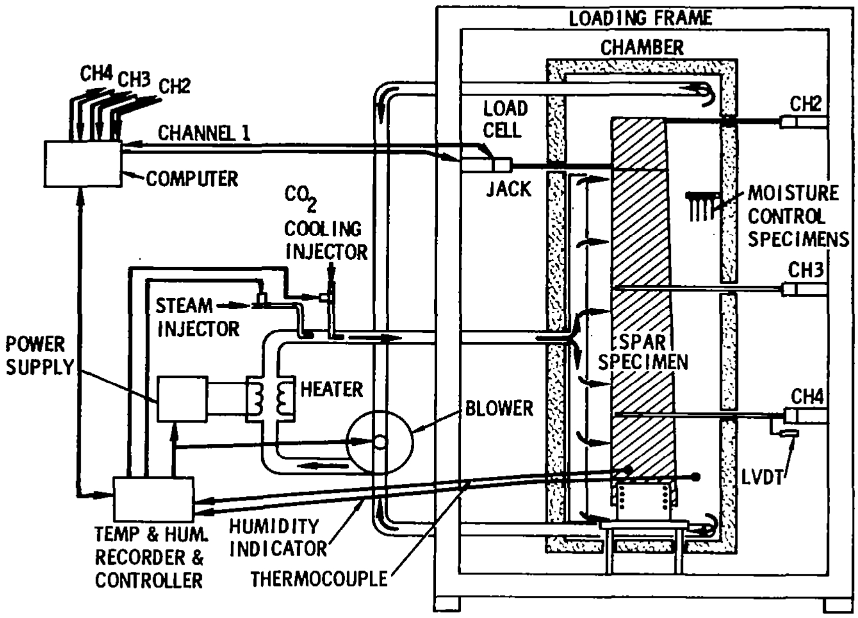


Figure 17.- PRVT schematic of spar test setup.

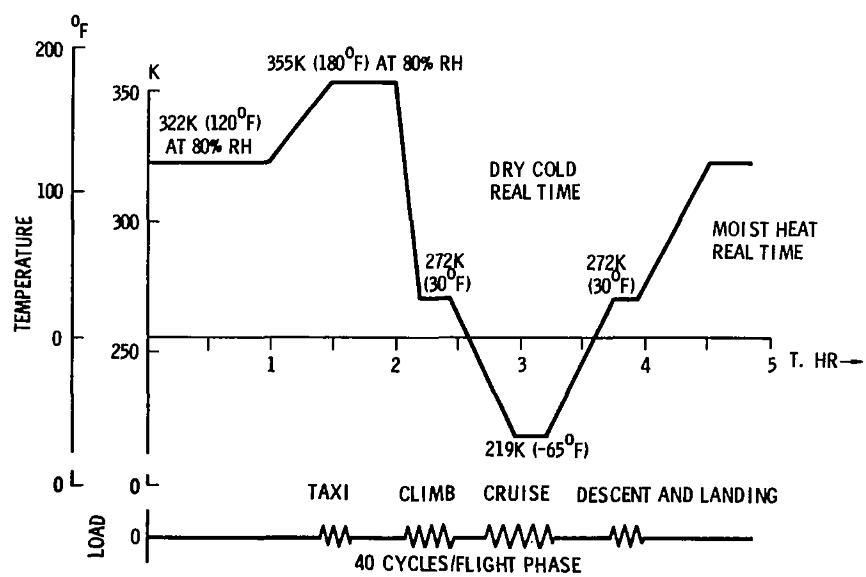


Figure 18.- Preliminary loads/environment simulation basic loading block.

Figure 19.- L-1011 inboard aileron.

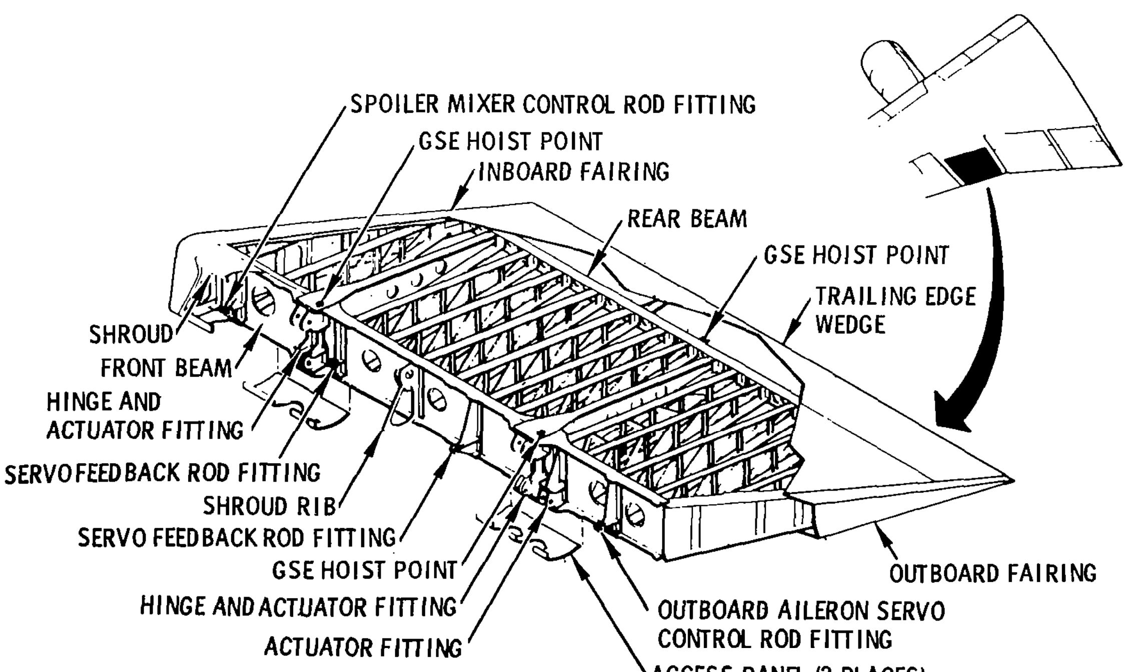


Figure 20.- Inboard aileron.

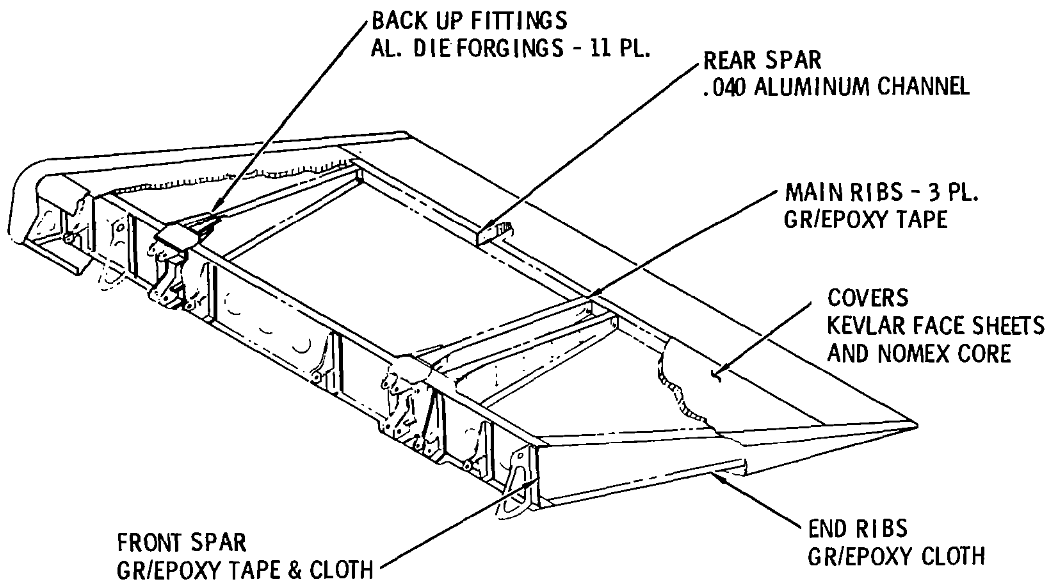


Figure 21.- Advanced composite aileron — concept no. 1.

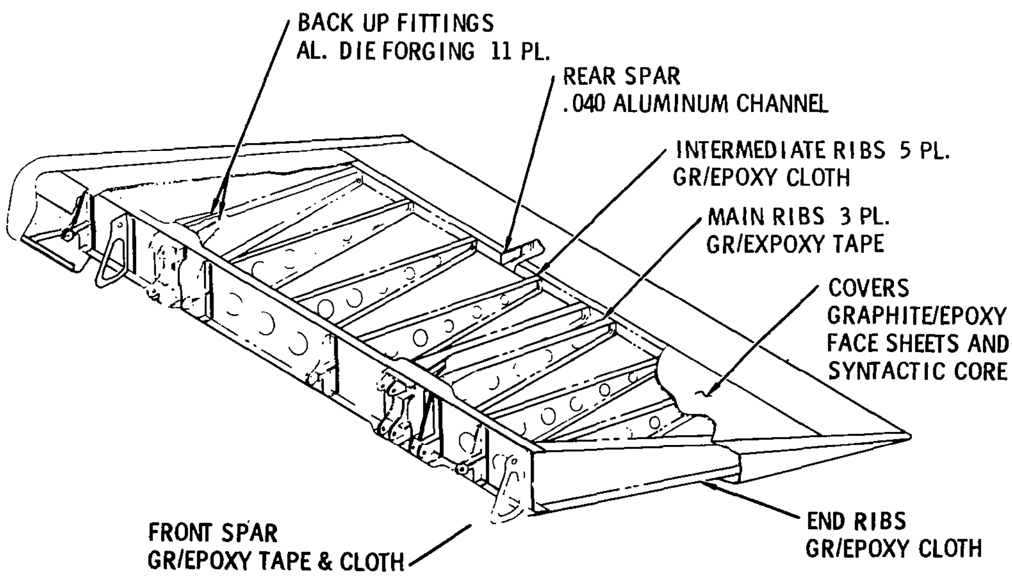


Figure 22.- Advanced composite aileron — concept no. 2.