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RECENT NASA PROGRESS IN COMPOSITES

By R. R. Heldenfels

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ABSTRACT

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RECENT NASA PROGRESS IN COMPOSITES

By R. R. Heldenfels

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INTRODUCTION

Recent progress made by NASA in research and technology for composite structures of aerospace vehicles will be reviewed in this paper. The material presented will cover accomplishments since RECAST and will address the topics listed in figure 1. The major recommendations of RECAST will be reviewed and the NASA composites program carried out in response to RECAST will be summarized. Specific applications to space vehicles, aircraft engines, and aircraft structures will be discussed in detail. Finally, the progress of the flight service evaluation program will be reviewed; the objective of this program is to increase confidence by accumulating flight service experience on composite aircraft structural components.

NASA PROGRAM

Early in 1972 over 100 representatives of industry, universities, and government agencies met to develop an Air Force/NASA Long Range Planning Study for Composites. This activity was called RECAST.

RECAST identified two major barriers to increased utilization of composites: lack of confidence and high costs (fig. 2). To build confidence in composites, far reaching programs were recommended in which the production/service performance of composites under realistic flight service environment could be experienced. Recommendations were also made to encourage applications of composites to new systems being developed for future use. A third recommendation to improve confidence of users was to conduct indepth studies to provide needed technology development and an increased data base for composite materials and structures. This was to be accomplished through additional work in laboratory simulation testing and in development of design criteria/philosophy.

The next most visible barrier was cost. While this is a very volume-sensitive item, the needed reductions were expected to come from improved structural concepts, improved design procedures, and from materials and fabrication innovations such as the use of hybrids of low and high cost filaments, quick curing processes and adhesives, simple tooling, automation, and better design for manufacturing.

The NASA program in response to RECAST has emphasized confidence building based on a sound program of technology development, indepth studies of applications to new vehicle systems, and by selected flight service programs on military and commercial aircraft. Cost reduction has been included as a secondary emphasis in design studies and in component developments that feature materials, structural arrangements, and manufacturing methods that are cost-competitive with similar metal components.

Figure 3 presents an overview of the NASA composites program. The program consists of technology activity, design and cost studies, and hardware applications. In the technology area, the research ranges over materials, fatigue, analysis, design, manufacturing, and component testing and evaluation. Design and cost studies (refs. 1 and 2) are undertaken to identify high payoff areas for future composite applications and help to provide a basis for selecting applications programs. In the applications area, the focus is on space shuttle, aircraft engines, commercial and military aircraft, and helicopters. These applications will be described in more detail later.

In the fiscal year following RECAST (FY 73) the total resources applied to the NASA composite program amounted to approximately \$9,000,000, which included 80 man-years of NASA manpower. The program has grown such that both annual funding and manpower will be doubled in the coming fiscal year (FY 76). Approximately 80% of the resources will be devoted to airframe and space structures, the remainder to engines. The basic technology area is supported by 60% of the resources and 45% of the funds support the NASA staff and laboratories devoted to composites technology.

Much of the new technology developed under NASA's program, although of a general nature, used a specific vehicle application for a focus. Consequently, progress in the technology development is discussed under the appropriate vehicle applications.

SPACE VEHICLE APPLICATIONS

Space Systems

Composites are expected to play a major role in the development of future space systems. The potential weight savings coupled with high specific stiffness makes composites an ideal candidate for many weight and/or stiffness critical space vehicles.

A number of composite applications have been made or are under study for a variety of space vehicles. Figure 4 illustrates two typical applications to NASA spacecraft wherein small amounts of composites are used where they perform their functions best. In such cases, cost has not been a significant factor. Spacecraft designs inherently place high dollar values on weight savings so that composite space applications are usually cost effective.

The Radio Astronomy Spacecraft, Explorer 49 (RAE-B), launched in June, 1973, utilizes boron/epoxy for both its stabilization and control nozzle booms (ref. 3). The inertia stabilization booms, located at the top of the spacecraft (see fig. 4), were fabricated of 11-ply unidirectional boron/epoxy construction, were 116 inches long, 0.74 inch diameter, and weighed 1.1 pounds each. The attitude control booms, mounted on the side of the spacecraft, were fabricated of 6 plies of boron/epoxy on 1/2-inch diameter tubes, were 25.2 inches long, and weighed 0.17 pounds each.

Applications for composite struts were also found for both Pioneers 10 and 11. As shown in figure 4, boron-epoxy tubular struts (covered with glass scrim cloth) were used to support the antenna dish of Pioneer 10. Additional applications for boron-epoxy were found on struts supporting the conical feed horn visible at the top of the photograph and a 4-foot long magnetometer boom. The spacecrafts were launched in March 1972 and April 1973 and have both executed planet Jupiter fly-bys. By 1987 a composite structure will have escaped from our solar system.

Thermal Expansion Control

The relatively low coefficient of thermal expansion of some composites provides the capability of developing thermally stable structures. Such structures have significant implications for spectacular increases in communications capability via large nondistorting reflectors and antennas and ultra high-resolution capabilities for earth/space resource monitoring.

An example of the application of composites to such spacecraft is shown in figure 5.

A half-scale graphite/epoxy metering structure (GEMS) is shown in the figure. The structure is about 5-1/2 feet in diameter, 10 feet long, and weighs 128 pounds. It was constructed by General Dynamics as part of a technology development for a future program being considered by NASA - the Large Space Telescope (LST) (ref. 4). Such a telescope should give U. S. astronomers an order of magnitude increase in resolution over the best earth-based telescopes.

The function of the graphite structure is to hold the two mirrors of the telescope a fixed (metered) distance apart. Requirements are for exceedingly small thermal distortions with an effective coefficient of expansion of $.35 \times 10^{-6}$ in/in/^{o}F desired. As suggested in the bar graph (fig. 5), graphite/epoxy offers the potential of an order of magnitude decrease in coefficient of expansion over the best known metal expansion-controlling material, Invar.

The Marshall Space Flight Center (MSFC) is supporting technology studies of two such structures - the General Dynamics shell shown and a Boeing truss concept (ref. 5). The shell structure was recently tested in a large thermal vacuum chamber with simulated sun loadings. The test was marginally satisfactory with some difficulties experienced in obtaining the precision expansion measurements required and in simulating appropriate LST thermal gradients during a typical viewing cycle. Additional mirror contamination tests have indicated that outgassing of candidate materials is not a serious problem.

These graphite/epoxy components are early pathfinders in a new class of space structures capable of extraordinary capabilities for passive thermal expansion control.

Space Shuttle

The importance and value of weight saving in the Space Shuttle has encouraged much development and applications activity. Figure 6 displays some of the developmental components built and tested during the design study phase. The fuselage frame and shear web were built by the Boeing Aerospace Company (refs. 6 and 7). The frame is a 1/3-scale model approximately 60 inches wide at the base and represents one-half of a major fuselage frame near the aft end of the payload bay. Webs for the frame are titanium I-shaped sections and the caps are reinforced with unidirectional plies of boron/epoxy. The frame supported 108 percent of design ultimate load at failure and, compared to an aluminum frame, a weight saving of 25 percent is estimated.

The shear web represents a section from a center-loaded thrust beam in an early orbiter design, is 36 inches high by 47 inches long and has a titanium-clad $\pm 45^{\circ}$ boron/epoxy web plate with vertical boron/epoxy reinforced aluminum stiffeners. The web supported 96 percent of design ultimate load at failure and the weight saving is estimated to be 24 percent.

The 24-inch by 47-inch fuselage panel has a .036 titanium face sheet stabilized by eight boron/aluminum hat section stiffeners which are spot-welded to the face sheet. This panel was designed, fabricated, and tested by Convair Aerospace and a 33 percent weight saving, compared to an all-titanium panel, was projected (ref. 8).

A 1/3-scale section of an aft thrust structure is shown in the lower center photograph of figure 6. The configuration was obtained from an early booster design and utilizes boron/epoxy tubular members with titanium joint clusters. The truss was designed and fabricated by Grumman Aerospace Corporation (refs. 9 and 10). Tests were performed at the Langley Research Center (LaRC) and the truss supported 99 percent of design ultimate load at failure. Compared to an all-titanium structure, a weight saving of 30 percent is projected.

The center section, including hinges and latches, of a main landing gear door for an early orbiter configuration was designed, fabricated, and tested by McDonnell-Douglas Astronautics (ref. 11). The door construction consists of full-depth aluminum honeycomb sandwich with 6-ply (0, ± 45, 90) graphite/epoxy face sheets. Static and cyclic loads were imposed on the door which failed at 1.6 times the design ultimate load. Compared to an all-titanium door, a weight saving of 65 percent is estimated.

The composite applications now going into the construction of the first Shuttle Orbiter are shown in figure 7. These structures are currently baseline design for five components; the aft thrust structure, mid-fuselage truss tubes, orbital maneuvering pods (OMS), payload bay doors, and purge and vent lines. The largest weight savings, 1070 pounds, is obtained with the payload bay doors which utilize graphite/epoxy face sheet/nomex honeycomb core sandwich stabilized with graphite/epoxy hat-section frames. The aft thrust structure which utilizes boron/epoxy reinforced titanium truss members accounts for the next largest weight savings, 900 pounds. Mid-fuselage frame truss members are boron/aluminum tubes with titanium end fittings and are the only baseline application of metal-matrix composites. Weight savings for the boron/aluminum tubular members is 180 pounds per vehicle. An OMS pod shell of graphite/epoxy and Kevlar 49/epoxy purge and vent lines are being designed. Preliminary estimates of the weight savings for each item are 300 and 200 pounds, respectively.

Composites have also found application to Shuttle pressure vessels. In addition to the advantages of reduced weight in comparison to metal and cost effectiveness, the overwrapped pressure vessels have a significant safety advantage. Failure of metal tanks under high pressure generally results in high velocity shrapnel that is dangerous to personnel and also hazardous to adjacent systems. The overwrapped tanks have a much less destructive failure mode (ref. 12). The liner fails, the contained fluids and gases leak through the overwrap, but the liner pieces are contained within the fiber envelope.

The weights, savings, and tank particulars with respect to the application to the Shuttle Orbiter are displayed in figure 8, along with a photo of an experimental vessel. This vessel was made by wrapping Kevlar 49 fiber, impreganted with an epoxy matrix polymer, on an aluminum liner approximately 0.10-inch thick. The tank was made by Structural Composites Industries. However, both Brunswich Corporation and McDonnell-Douglas have made similar applications to Shuttle. Various liners have been used in the development of these tanks. Titanium, stainless steel, and Inconel have been studied, in addition to aluminum, to determine optimum design configurations. Titanium is usable for storing helium and nitrogen, but is, of course, hazardous for storing oxygen. For oxygen, Inconel or stainless steel should be used.. The Kevlar 49 fiber used for the overwrap has a modulus of 20,000,000 psi and a strength of approximately 350,000 psi. In addition, it is lighter than glass, having a density of 0.053 lb/in^3. The density of S-glass fiber is 0.092 lb/in^3.

Technology Spinoff

One application of NASA pressure vessel technology to civil sector needs is shown in figure 9. Composite tanks for fireman's breathing apparatus are a spinoff from the LeRC developed technology. This technology was utilized by the Crew Systems Division of Johnson Space Center (JSC) to find ways of reducing the weight of the breathing systems for astronauts. It was, of course, recognized that the same purpose could be accomplished in the case of the fireman's breathing apparatus. Structural Composites Industries and

Martin-Marietta Corporation built the tanks. Testing is being conducted by Lewis Research Center (LeRC) to determine whether or not environmental factors can play a role in the life of the tanks. Prototype tanks have been placed in service in the New York, Houston, and Los Angeles fire departments for test purposes.

In another spinoff, the Boeing Company has contracted with Structural Composites Industries to build composite overwrapped pressure vessels for containing the high pressure gas required to operate the passenger escape slide systems in the Boeing 747 SP aircraft.

AIRCRAFT ENGINE APPLICATIONS

The unique properties of composite materials makes them prime candidates for application to the low-temperature sections of turbofan engines. Use of composites in fan blades, fan frames, nacelles, and early stage compressor blades can result in reductions in engine weight, specific fuel consumption, and direct operating costs leading to an increased return on investment.

Fan Blades

The application of composites for the large fan blades of high by-pass ratio engines (ref. 13) is depicted in figure 10. The largest of the three blades is approximately 32 inches long. Blades of this type can effect weight savings of the order of 30 percent compared to metal blades. In addition, lighter blades permit further savings to accrue because the disk can be made lighter, and then the bearing supports and frame in turn can be made lighter.

Fibers that are being considered for use in this application include graphite, boron, Kevlar 49, and glass. Graphite and boron are advantageous because they have both a high strength-to-density ratio and a high stiffness-to-density ratio. In general, strength in the sense of resisting stresses due to centrifugal forces, bending, and forced vibration is not the major problem. Of more importance is the stiffness, in order to provide high natural frequencies, freedom from flutter, and elimination of the part-span shroud.

The figure shows, in each case, a metallic leading edge sheath made from thin sheet metal, such as a stainless steel alloy. This sheath is adhesively bonded to the blade and provides dust, sand, and rain erosion protection. The sheath also protects against foreign object damage from hail, cowl lip ice, gravel from the runway, and other debris that is often drawn into the engine.

The blade shown on the right is a spar-shell concept built by Hamilton-Standard. The middle blade is a JT9D configuration made by Pratt & Whitney, and the blade on the left is a TF-39 fan blade made by General Electric. The latter two blades are of solid laminate construction.

Figure 11 illustrates a graphite/polyimide composite fan blade that was fabricated using the NASA PMR (Polymerization of Monomer Reactants) process which eliminates the need for the separate polymerization step, a significant cost savings (refs. 14 and 15). In the PMR process, fiber is drawn through the solution of monomer reactants, eliminating the need for using highly toxic solvents and providing greater safety to fabricators of polyimide composites.

Improved Impact Resistance

Polymer matrix composite materials have not, as yet, been introduced into the fan rotor assembly of commercial turbofan engine. The primary reason for this is that polymer matrix composites are quite brittle and have been unable to withstand impact from large objects, such as birds. A comprehensive research program is underway at the LeRC to develop material and manufacturing procedures leading to improved impact resistance (ref. 16).

The goal of the program is to produce blades capable of withstanding ingestion of birds ranging from 3-oz starlings to sea gulls and geese that can weight 2 to 4 pounds and more. Our objective is to design and develop composite fan blades that can meet the FAA criteria when these kinds of objects are ingested. In general, the criteria require that brids of the starling category be ingested with no blade damage; 1- to 2-pound birds must be ingested with the capability of subsequently operating the engine at the 75 percent power level; larger bird ingestion requires only that the engine be capable of being shut down without jeopardizing the aircraft (by unbalance, for example).

Considerable progress has been made toward meeting these objectives. Blades have now been developed to the point where the starling size is not a problem. In addition, the ingestion of birds larger than 2 pounds does not appear to be a problem, since it should be possible to shut the engine down without trouble (unbalance forces are less than with metal blades). The problem is therefore narrowed down to the acceptance by the engine of birds in the 1- to 2-pound class while maintaining 75 percent power. This means that small portions of the blade tips can be lost, but that major damage to the blade is not permissible.

This progress was made possible by a comprehensive NASA technology program to improve the impact resistance of composites. Some of the pertinent results are shown in figures 12 to 14.

To improve the impact resistance of graphite composites, a concept of hybridization was developed which involved mixing fibers other than graphite with the graphite plies. The success of this technique is illustrated in figure 12. Here a graphite/glass hybrid composite blade withstood the impact of a 24-ounce slice of a 2-1/2-pound bird with only tip damage, while a conventional graphite/epoxy blade was completely destroyed by the impact of a slice only 1/2 the weight (12 oz.).

To develop metal composites that meet FAA foreign-object-damage requirements, studies are underway to increase the impact resistance of boron/aluminum composites. The new approach being considered is to use large diameter fibers in more ductile matrices together with lower temperature processing (ref. 17).

The improved impact strengths obtained with B/Al composite impact specimens by using these approaches are illustrated in figure 13. The use of a more ductile matrix, with no change in fiber diameter, led to nearly a fourfold increase in impact strength. Note the cleavage type of failure exhibited by the specimen made with the brittle matrix and the ductile failure of the specimen made with the ductile Aluminum matrix. An even more marked improvement in impact strength was obtained by using larger diameter fibers in the ductile matrix. This specimen had an impact strength of 68 ft-lb and did not fracture upon impact. This is much greater impact strength than titanium alloys now used in fan blades (15 ft-lb).

These approaches are being used in a joint NASA/USAF program to fabricate boron/aluminum compressor blades for the J-79 engine. Improved FOD resistance is obtained by using larger diameter fibers in more ductile or in hybrid matrices, together with improved (lower temperature) processing and design. The use of large diameter fibers results in increased interfiber separation which permits the strain capability of the matrix to be more effectively utilized (i.e., greater strain can occur upon impact). Figure 14 illustrates the impact resistance of two J-79 B/Al compressor blades. Both blades shown in the chart were impacted in a whirling arm test with birds weighing 3 ounces. The failed blade did not utilize the advanced techniques described, whereas the advanced blade on the right showed no visible change after impact. The ultimate objective of this jointly sponsored program is to demonstrate the viability of B/Al J-79 compressor blades in flight tests using an F4 airplane.

Short-Haul Engines

The QCSEE (Quiet Clean Short-Haul Experimental Engine) program has progressed to the point where the engine design phase is nearing completion. In about a year the fabrication phase for the first engine, which is the UTW (Under the Wing) engine, is scheduled for completion. This engine employs several advanced technology features including composites (refs. 18 and 19). Figure 15 shows a cross section of the UTW engine and nacelle with the advanced composite technology components identified. The UTW engine employs the F101 engine core, which fits well into the QCSEE engine designs. A major technical advance in the UTW engine is the composite nacelle, consisting of the inlet duct, the fan duct outer wall, the acoustic splitter and the core cowl. The acoustic treatment is integral with the structure and serves the dual functions of acoustic suppression and structural load carrying capability, effecting a light-weight, low-cost design. The digital control system and accessories are at the top of the engine, making the engine accessories connections to the aircraft more convenient.

The composite fan frame is the primary structural element in the engine; supporting the fan, nacelle components, and front portion of the core.

Also, the engine thrust loads are carried through it to the aircraft. The frame is made of graphite/epoxy, saving frame weight and reducing fabrication costs. Composite materials are also used in the fan blades for the same reasons. The engine has a variable area fan nozzle, consisting of four hinged flaps moved by hydraulic actuators. The reduction gears and fan variable-pitch mechanism are located inside the fan spinner.

The General Electric Company is the prime contractor. The UTW engine design is complete; fabrication of the engine will be completed in 1976. Following engine and nacelle tests at General Electric, it will be delivered to NASA late in 1977 for tests at LeRC. These ground tests will consist of performance and acoustic evaluations; foreign object damage and composite durability will not be included. No flight tests are planned.

Technology Spinoff

Another application for composite fan blades is proposed for large wind energy generators (refs. 20 and 21). A 100 KW wind energy generator is presently under construction at the NASA Plum Brook Station near Sandusky, Ohio, and is shown schematically in figure 16. The 100 KW capacity is attained at a wind velocity of 18 miles per hour. It is a constant speed machine with variable pitch blades to maintain a speed of 40 rpm. A wind direction sensor supplies a signal that initiates rotation of the generator housing to keep the blades faced into the wind. At wind speeds above 18 mph, the pitch is changed to keep the power level constant and still maintain constant speed. Initially, metal blades will be used and are constructed using a spar-shell concept. Composite blades such as the glass/epoxy blade shown at the bottom of the figure are presently being designed and will be fabricated and retrofitted at a later date. It is anticipated that costs will be reduced by going to composite blades and that the blades will be lighter.

AIRCRAFT STRUCTURAL APPLICATIONS

The primary emphasis in the NASA composite program is directed toward structural applications for commercial aircraft. Indepth design studies and comprehensive technology programs were carried out to guide the selection of flight service programs, to improve design procedures, and to increase confidence of potential users. A few highlights of the technology program will be reviewed before describing the flight service programs.

Technology Program

Design Studies. - In order to illustrate the character of the design studies, two studies, with substantially different components and requirements will be discussed. The first study was conducted by Lockheed-California for LaRC and considered a composite replacement structure for the inboard aileron on the L-1011 aircraft (see fig. 17) (refs. 22, 23, and 24). As shown in the figure, the 4-foot by 8-foot metal aileron is a multirib structure of riveted aluminum construction. The replacement composite design weighs 101 lbs. and is 28 percent lighter than the metal design. The design proposed uses both Kevlar 49/epoxy and graphite/epoxy materials in a sandwich type construction for maximum cost effectiveness. This approach has led to a 61 percent reduction in the number of parts; projections for a 200 item production run indicate that 20 percent reductions in cost are achievable.

Subcomponents of the aileron including a full-scale section of the front spar, a rib, and a key attachment fitting were designed and tested successfully to demonstrate the integrity of the proposed design. This type of aileron poses some challenging aspects of composite design in that it is exposed to high acoustic loadings and elevated temperatures as well as impacts from foreign objects strewn by jet engine exhaust.

The second study deals with applications of composites to the helicopter. Since the advent of advanced composites, most helicopter R&D effort has focused on rotor blade design to the point where major airframe fabricators are now firmly committed to the production of all-composite blades. Only recently has R&D attention turned to the composite design of fuselage structure.

A NASA/Army/Sikorsky study of the CH-53D cargo helicopter (ref. 25) has concluded that, with existing composite materials and fabrication techniques, composite fuselage structure can be expected to cost 5 to 10 percent more than conventional light alloy construction, weigh 15 to 20 percent less, and reduce total system operating costs by about 5 percent. It was also concluded that a significant effort in composite fuselage manufacturing technology over the next few years would reduce initial composite fuselage costs to the point where they would be 5 to 10 percent cheaper than aluminum structure. The material cost savings could be attained by maximizing the use of broad goods and reducing the use of the more expensive graphite tape to a minimum.

The reduction in fabrication and assembly costs are achievable through the development of a reliable cocured skin/stringer/frame concept involving Kevlar 49 skins and foam-supported graphite stringers and frames as shown in figure 18. Panels, employing this concept, have been fabricated. Static tests on skin/stringer/frame segments have been encouraging. Post buckling panel fatigue tests are about to get underway. The cost of manufacturing these panels is significantly lower than the costs of fabricating composite stringer and frame details and subsequently bonding them to the skins.

The cost reductions in systems installation and outfitting are achieved by the use of segmented construction rather than by the fabrication of large fuselage halves or full circumferential shell segments. The use of moderate sized segments reduces lay-up time, permits full use of existing autoclave facilities, eliminates major scale-up problems, and provides ready access for systems installation prior to final assembly.

Panel Studies. - A considerable effort is also being expended to develop designs that permit maximum exploitation of the potential of composite materials. As an example, NASA is conducting in-house programs to develop the fundamental structural data necessary for design confidence. In-depth design studies such as shown on figure 19 are being conducted at LaRC on compression and shear panels representative of primary aircraft wing structure or spacecraft shell structures (ref. 26). Typical design study results for compression panels are shown in the structural efficiency chart on the right of the figure. The vertical axis is proportional to the weight of a panel design; the horizontal axis is the intensity of compression loading normalized by the panel length. Data points shown for aluminum panels represent the results of nearly 2,000 NACA panel tests performed in the late 1940's. Most of the minimum weight aluminum designs shown are bounded by the dashed curve predictions for efficiently designed hat-section stiffened panels.

Highly efficient designs have been analytically developed for hat-stiffened or corrugated graphite panels. The solid curves shown indicate that weight reductions of over 50 percent of the weight of metal panels may be achievable. The design concept employed achieves high efficiency by using heterogeneous structure. As indicated in the corrugated cross-section sketch, plies with fibers in the direction of loading are separated by thin low stiffness webs to maximize the bending stiffness of the panel to resist buckling. Similar minimum weight designs have been developed for graphite shear panels.

Laboratory tests have been conducted on approximately 30 compression panels and two large shear panels as illustrated in the figure. Test results indicate weight savings of 32-42 percent are experimentally achievable in test articles which do not have stringent quality control. Test programs are continuing with higher quality panels and with other stiffening configurations. The interaction of theoretical predictions corrected with selected experimental data should lead to reliable design procedures for advanced minimum weight configurations.

Long-Term Environmental Effects. - Long-term environmental exposure effects on the properties of composite materials are of major concern to potential users. Both laboratory simulation and onsite exposure followed by testing of exposed specimens are presently underway and are illustrated in figures 20 and 21 for subsonic aircraft and in figure 22 for supersonic aircraft. Figure 20 presents interim results of the effects of long-term exposure in an outdoor, groundbased environment on the strength of graphite/epoxy material test specimens. The circle symbols on the figure represent the averages of results of short beam shear tests conducted on replicate specimens fabricated from graphite/epoxy 5 years ago and subsequently placed in an outdoor exposure rack at LaRC from which they have been periodically removed for testing. Strength is shown as strength retention ratio which is the ratio of the strength after exposure to the baseline average strength of the material before exposure began. It can be seen that after 57 months (41,600 hours) of outdoor exposure with no stress other than the residual fabrication stresses, strength retention remains within the initial 15 percent scatter band based on shear strength. As shown by the square symbols a separate investigation with tensile specimens, showed that the introduction of sustained stress had no significant effects in the strength ratio after 6 months of exposure.

To maintain a matrix-controlled strength, the tensile specimens are designed at $\pm 45^{\circ}$ laminates of graphite/epoxy. A level of sustained stress equal to 25 percent of tensile ultimate was applied throughout the exposure; and after removal from the exposure fixture, the specimen was pulled to tensile failure. The results are plotted in figure 20. These tests as well as many others on boron/epoxy, Kevlar 49/epoxy, and graphite/polyimide are continuing with planned durations of up to 10 years in order to build confidence in the durability of these composite materials.

Materials used in NASA flight service programs are being exposed to the local environmental conditions where the flight service aircraft are home based. The site locations were dictated by the world wide distribution of the B-737 aircraft equipped with composite spoilers plus the LaRC site and the one at the United Airlines Maintenance Facility in San Francisco.

The upper left photograph in figure 21 shows the rack for exposing unstressed specimens; interlaminar shear (top row), flexure (second and third rows) and compression (fourth row). Seven materials are being exposed at various exposure sites (ref. 27). The materials include

1. Narmco (Thornel 300/5209) ┐
2. Union Carbide (Thornel 300/2544) ├─ Spoiler Program, Phase I
3. Hercules (Type AS/3501) ┘
4. Kevlar-49/epoxy (F-155) ┐
5. Kevlar-49/epoxy (F-161) ┘─ L-1011 Fairing Panels
6. Graphite/Polysulfone - Spoiler Program, Phase II
7. Narmco (Thornel 300/5208) - DC-10 Rudder Program

The exposure sites are airline terminals at Wellington, New Zealand; São Paulo, Brazil; Frankfurt, West Germany; Honolulu; San Diego; and at the LaRC site. At specified intervals, specimens are removed from the racks and shipped to LaRC for testing. Except for the São Paulo site, test results have been obtained for the first five listed materials after 1 year of exposure. These test results showed little or no degradation in mechanical properties. The planned testing intervals are 1, 3, 5, 7, and 10 years.

The lower right photograph shows the rack for exposing both stressed and unstressed specimens of the graphite epoxy (Narmco - Thornel 300/5208) being used in the DC-10 rudder program. The specimens are eight-ply laminates $(0/45/90/-45)_S$ fabricated into 6-inch dog bones. The stressed specimens are loaded at 40 percent of ultimate strength in the individual spring fixtures. The rack is located atop a United Airlines Maintenance Facility building in San Francisco. Specimen exposures at the two sites began in October 1974, and, no test results have been obtained to date. The planned testing intervals are 1, 3, 5, 7, and 10 years.

A program to establish the time-temperature-stress capabilities of composites for advanced supersonic applications is being conducted under NASA contract with General Dynamics-Convair. This work is outlined in figure 22. The objective of the contract is to establish these capabilities through comprehensive environmental tests and analyses of five advanced filamentary composite material systems: 4 mil boron/P105A polyimide, HTS graphite/710 polyimide, and 5.6 mil boron/6061 aluminum.

The elements of the testing environments include outdoor and laboratory exposures at ambient and elevated temperatures for times to 50,000 hours, ambient and reduced pressure static exposures, and multiparameter exposures. Baseline and residual property test temperatures range from -65° to 800°F. The specimens include tensile strips, compression strips and beams, and shear strips and beams. Materials are tested in unidirectional and crossply layups. The test parameters include baseline properties (tensile, compressive, shear, fatigue, creep, Poisson's ratio), thermal and ambient static aging, supersonic flight simulation with both accelerated and real time loading, and residual properties after exposure.

The complex flight simulation equipment shown in the photograph, in which both accelerated and real time tests are conducted, applies random load spectra on a flight-by-flight basis and programed temperature histories with independent load and temperature levels for each of the materials systems under test. Up to 100 specimens can be tested simultaneously. The static exposure and accelerated flight simulation data are used in analyses based on modified wearout concepts to predict materials behavior after long flight simulation exposures. If the 50,000-hour exposure data correlate with these predictions, a significant advance will have been made towards efficient design of advanced composite components for long time, elevated temperature aircraft service.

Currently, baseline and short-term simulation data are on hand for all five materials. Long-term testing on boron/polyimide was canceled because of excessive variability of matrix-controlled properties and rapid degradation of the polyimide during short time exposures. Ten thousand hours static exposure data will be on hand for the epoxy and aluminum matrix materials in July 1975.

High Temperature Panel Technology. - As part of the Supersonic Cruise Aircraft Research Program (SCAR), LaRC is studying advanced fabrication methods for both high temperature composites and metals. The YF-12 wing panel indicated in the upper right-hand view of figure 23 has been selected as the focus for the research. Five types of panels noted in the upper left are included in the program (ref. 28). A view of a titanium skin-stringer panel fabricated in this program is shown in the lower left and the type of test program conducted on the panels is shown in the lower right.

The original YF-12 panel is integrally stiffened and was machined from a 1-inch thick titanium plate. The finished panel weighs 8.5 lb. The panel dimensions are 16 in. by 28 in. and the primary loading is shear.

The three types of composite panels currently included in the SCAR YF-12 panel program include graphite/polyimide, boron/aluminum and Borsic/aluminum. These panels are illustrated in figure 24 and are to be flown on the YF-12 aircraft in the same location as the titanium panels. Representative ground test specimens will be subjected to temperatures between 400°F to 1000°F for

10,000 hours and cycled from -65°F to 600°F for 1000 3-hour cycles. The graphite/polyimide and the Borsic/aluminum panel are being fabricated at LaRC and the boron/aluminum panel is being fabricated by McDonnell Douglas Astronautics Company - East (MDAC-E).

The LaRC graphite/polyimide panel detailed in the upper left portion of figure 24 will consist of graphite/polyimide skins bonded to a glass polyimide honeycomb core using a LaRC developed polyimide adhesive known as LaRC-13/AI. Titanium interleaves are cocured in the outer skin to carry the bearing loads around the fasteners used to attach the panel to the aircraft.

The original graphite/polyimide system selected was P-13N polyimide and HTS graphite filaments. This selection was made primarily on the potential 600°F capability of the system. However, after extensive evaluation, this system was found to possess inadequate and erratic peel and interlaminar shear properties. Therefore, the P-13N system was replaced by a relatively new polyimide system developed by NASA LeRC designated PMR-15 (refs. 14 and 15). The PMR-15 system has the potential of adequate performance at 550°F and is easier to process than the P-13N system. The weight of the graphite/polyimide panel is estimated to be approximately 4 lbs or about 55 percent lighter than the original YF-12 titanium panel.

The B/Al panel shown in the upper right of figure 24 consists of titanium-clad boron/aluminum skins brazed to titanium honeycomb-core which in turn is brazed to a titanium edge member. The concept of using titanium-clad boron/aluminum results from studies conducted at LaRC and promises to alleviate brazing problems encountered with an earlier design. Fabrication studies by MDAC-E have also resulted in the development of a new braze alloy designated 713-C which promises to simplify the problem of brazing B/Al composites. A design verification panel is expected to be tested in July and if successful, flight testing should commence in September or October 1975. This panel will weigh approximately 6 lb or approximately 30 percent less than the original titanium panel.

The LaRC Borsic/aluminum panel shown in the lower left corner of the figure consists of Borsic/aluminum skins brazed to a titanium honeycomb core and titanium frame structure. Borsic/aluminum rather than boron/aluminum was selected for the skin material because Borsic/aluminum is less susceptable to degradation during brazing. This panel is estimated to weigh approximately 5.7 lb or approximately 32 percent less than the original titanium panel. A process verification panel shown at the lower right has been fabricated and inspected by radiography and C-scan ultrasonics. The inspection indicates that satisfactory fabrication parameters have been developed. A design verification panel is expected to be tested in July and it is hoped that flight and ground testing will be initiated in September 1975.

Flight testing time above Mach 2.6 is so short that this program cannot be considered a flight service project. The principal environmental data comes from the ground tests conducted at the NASA Flight Research Center and LaRC.

FLIGHT SERVICE PROGRAMS

The NASA flight service programs are conducted to obtain practical experience in design, manufacturing, and operational considerations for a variety of aircraft components (refs. 28 and 29). The primary focus for the programs have been commercial airliners. The reasoning behind this approach is multifaceted. First, if significant payoffs in composite applications are to be realized, both civil aircraft manufacturers and operators must have confidence in the safety, reliability, and cost effectiveness of a composite aircraft fleet. Second, civil aircraft have high utilization rates necessary for rapid accumulation of flight time on the test components. In addition, airlines must be convinced that there will be no penalties associated with maintenance and repair of composite structures in a normal service environment.

In order to build confidence in composites to a point where both manufacturers and operators are willing to make long-term production commitments using composites, NASA has initiated a systematic program to introduce composites into airline fleet service. As will be seen in subsequent figures, initial applications were proposed as selective reinforcement of metallic components on military aircraft. Small secondary airline components were added to the program; larger composite reinforced metallic components were placed in service; larger secondary components are being fabricated; and large, primary structural components are in the design or planning stages.

This approach has permitted available knowledge to be applied with minimum risk; has stimulated the development of new technology for later application; and has resulted in considerable flight service experience at comparatively small cost. In the process, both manufacturers and operators are becoming more trustful of composites as a prime structural material.

Typical Flight Service Program

The phases that constitute a typical flight service evaluation program and the time span are indicated in figure 25. The advanced development, detail design, fabrication and ground tests phases generally require on the order of 3 years to complete. The environmental tests phase begins during the fabrication phase and continues throughout the flight service phase. The flight service phase is 5 years or more. Note that the minimum time required to accomplish this type of program is 8 years.

Typical elements that comprise each of the phases are noted in the figure. Some variation in these elements may occur depending upon the flight service program and the contractor performing the project. The program elements noted in the advanced development phase are concept selection, tool try and subcomponent tests. Under detail design are structural design, tooling development, and component tests. In the fabrication phase are tooling, ground test components, and flight service components. The ground tests phase contains static and fatigue tests, vibration tests, and FAA certification. In the environmental tests phase are included outdoor and controlled laboratory exposure tests, thermal cycling, and exposure at airline terminals and at LaRC. The last phase, flight service, consists of periodic residual and ultrasonic inspections, removal of selected components for testing, and annual reporting.

Flight Service Summary

The scope of the current flight service program is summarized in figure 26. The aircraft and component are arranged chronologically by date of entry into flight service. Details of each are discussed subsequently. The first four types of components have already entered service, beginning with the CH-54B helicopter tail cone in March 1972, now into its fourth year of service. The last three types of components are scheduled to enter flight service at various times during the next 3-1/2 years. All together, 156 components will be in service on 48 individual aircraft operated by the U.S. Army, the U.S. Air Force, and 12-14 commercial airlines. Five different advanced composite material systems have been used in the fabrication of these components.

A wide spectrum of flight service experience is being accumulated as indicated by the cumulative flight hours tabulated for the individual high time aircraft and for the total number of components. Current accumulations as of June 1, 1975, are over 400,000 flight hours and projections to December 1981, are for more than 2-1/2 million flight hours with several of the high time aircraft components approaching 10 years of service. The totals show that 9 years after RECAST we will have accumulated an average of 16,000 hours on each of 156 components of secondary and primary structure. Such a widespread accumulation of service experience with composite components, inspected, monitored, and maintained jointly by the participating airlines and the aircraft manufacturers should provide a substantial base for confidence in long-term durability for future applications.

CH-54B Helicopter Tail Cone. - The Sikorsky Aircraft Division designed and fabricated a boron/epoxy reinforced aluminum tail cone for the CH-54B helicopter under contract to LaRC, and the Langley Directorate of the Army Aviation Materiel Research and Development Laboratories (refs. 30 and 31). The CH-54B helicopter shown in figure 27 is the U.S. Army's Flying Crane and is utilized in a variety of heavy-lift loading, unloading and transport situations. An undesirable dynamic resonance condition was experienced under certain combinations of payload weight and sling cable length. A production fix has been applied that increased the tail cone weight substantially.

The composite reinforcement design provided the equivalent increase in stiffness by adhesively bonding unidirectional strips of boron/epoxy to the webs of the stringers on the top and bottom skins of the tail cone. The interior of the tail cone is shown on the left side of the figure and the portion of the aircraft of interest is shown as the shaded area in the right-hand photo. Twelve of these boron/epoxy strips are distributed over the 17-foot long, 3.5- by 3-foot cross section tail cone. The composite represents only 8 percent of the 397 pound total weight, yet achieves a 14 percent weight saving.

In the development program, tests of structural elements and panels were performed to verify the design performance requirements for strength, stiffness, and fatigue life. One full-scale component was fabricated and installed on the production line by Sikorsky, subjected to a ground vibration test and then put through a flight test program to demonstrate its adequacy for normal, unrestricted flight service.

This CH-54B helicopter was delivered to the U. S. Army in March 1972, and has been in regular service continuously with the 355th Aviation Company at Fort Eustis, Virginia. Under the terms of the NASA contract, Sikorsky has provided regular inspection services of the composite portion of the structure. Performance continues to be highly satisfactory, although due to the realities of fuel conservation measures, the Army is only flying this helicopter about 6 hours per month at present, about 30 percent of previous normal usage.

L-1011 Fairing Panels. - The Lockheed-California Company is under contract to LaRC to obtain longtime flight service experience with 18 Kevlar-49/epoxy composite fairing panels which have been installed on TWA, Eastern, and Air Canada L-1011 aircraft for a 5-year flight service evaluation (ref. 32). Locations of the panels and a preinstallation photograph are shown in figures 28 and 29. Three different configurations were selected for evaluation; (1) wing-to-body fairing, (2) center engine fairing, and (3) wing-to-body fillet. Right and left fairings for each of the three configurations were installed on each of the three participating aircraft.

The wing-to-body fairing is 60 by 67 inches and has a slight curvature. This sandwich panel has a Nomex honeycomb core with Kevlar-49/epoxy face sheets. This panel weighs 16 pounds and is 25 percent lighter than the replaced fiberglass panel. Each panel has an aluminum flame spray coat on the external surface to discharge electrical potential from the fairing surface to the adjacent structure. Internal and external pressurization tests were conducted on a full-scale panel for FAA certification. The panel failed at a pressure that was 125 percent of design ultimate.

The second largest fairing panel (fig. 29) is a triangular configuration approximately 82 inches long and 30 inches wide. This panel is similar in construction to the wing-to-body fairing except a higher temperature epoxy is required because of its proximity to the center engine. This panel weighs 5 pounds and represents a 30 percent weight saving compared to its fiberglass counterpart. No testing was required since the loadings are less than the loads on the wing-to-body panel tested.

The smallest fairing is about 5.5 by 32.5 inches and weighs about 2 pounds. This represents a 32 percent weight saving. This fillet panel is of solid Kevlar-49/epoxy laminate construction, 0.090-inch thick, tapering to 0.030-inch thick edge closeouts. No testing was required for this panel other than the design allowable data requirements for all panel configurations.

Flight service evaluation of the fairings was initiated in January 1973. The fairings have been inspected twice and they are performing in a manner similar to the standard fiberglass fairings (ref. 33). The TWA fairings have been out of service for the last year because of fire damage to the TWA aircraft. The fairings have been reinstalled on a new TWA aircraft scheduled for delivery in August 1975.

All fairings for this program were fabricated by the Heath-Tecna Corporation, Kent, Washington.

B-737 Spoilers. - The largest element in the program in terms of numbers of components in service is the flight service of B-737 spoilers illustrated in figure 30. The spoilers are indicated as the dark portion of the trailing edge on each wing in the photo. The Boeing Commercial Airplane Company has fabricated 139 graphite/epoxy spoilers such as shown in the right photo for the 737 transport aircraft under contract to LaRC (ref. 34). Three spoilers were static tested to failure for FAA certification, 25 will be tested at LaRC to study as-fabricated variations in strength, 3 are being held as spares and 108 spoilers have been installed by 6 commercial airlines, 4 spoilers to an aircraft, for worldwide flight service. The 6 airlines are Piedmont and PSA in the continental United States, Aloha in Hawaii, New Zealand National, Lufthansa in Germany and VASP in Brazil. The graphite/epoxy spoilers were designed to be interchangeable replacements for the standard aluminum production spoilers. They are 52 inches long by 22 inches wide and weigh 13 pounds. Three different graphite/epoxy composite materials have been used in fabricating the cover skins which are adhesively bonded to the production aluminum honeycomb core and aluminum leading edge spar and hinge fittings. The finished spoiler is 35 percent composite by weight and is 17 percent lighter than the production aluminum spoiler.

The first 4 spoilers installed were placed in flight service in July 1973. The last of the 108 flight installations was placed in service in August of 1974. All are expected to remain in service for at least 5 years with the exception of certain randomly selected spoilers which have been designated for removal at annual intervals for complete inspection and static test to verify service life experience. Spoilers receive normal airline inspections at scheduled intervals and in addition are inspected visually once a year on the aircraft by Boeing. The first of the annual removals has been completed and the tests and inspections indicated a very satisfactory service. No evidence of moisture or corrosion in the aluminum honeycomb core was found. Service experience to date by all airlines has been satisfactory.

The fabrication experience in producing the graphite/epoxy spoilers for flight service on 737 transport aircraft is shown on figure 31. The band represents a certain amount of engineering judgment on the number of direct fabrication manhours required to produce a flightworthy spoiler as a function of the number of spoilers produced. Detailed manhours expended by individual task and individual part number were not tracked generally; however, the total manhours were accumulated at several intervals in the production run. The band represents a 74 percent "learning curve" - that is, if the fifth spoiler, for instance, requires 100 manhours to fabricate, the tenth spoiler (double the unit number) will require 74 percent or 74 manhours to fabricate. Fabrication time for these spoilers includes both tasks which are unique to composites and tasks which are common to production metal honeycomb core bonded structure.

The graphite/epoxy skins were laminated by laying 3-inch wide tape of graphite/epoxy prepreg on a steel tool with a semiautomatic tape machine. Six plies, multidirectionally oriented, were placed over a sufficient area to produce four replicate skins. The graphite/epoxy layup was cured in an autoclave, trimmed to final size, and assembled with the metal details by adhesive bonding. The favorable learning curve achieved for the graphite/epoxy spoilers may be attributed to several factors. The change from production metal spoilers to composite spoilers necessitated that the production team become familiar with the new materials and manufacturing tasks. Increased experience resulted in increased efficiency. Initial prepreg materials were of marginal quality and required considerable handwork in layup with subsequent reductions as the quality of prepreg improved. The fabrication experience with the graphite/epoxy spoilers has shown that even with the favorable learning curve applied to over 135 spoilers, the number of man hours required to produce a composite spoiler is still greater than that required to produce the metal spoiler with a production volume approaching 4000 units.

As a follow-on to the original spoiler program, The Boeing Commercial Airplane Company is fabricating 13 advanced design, all-composite spoilers for the 737 transport aircraft under contract to LaRC. This program is illustrated in figure 32. One spoiler has been static tested to failure as a part of the substantiation for FAA certification. Twelve spoilers will be fabricated and shipped to each of the six airlines participating in the graphite/epoxy spoiler flight service program. Each airline will install two of these advanced, all composite spoilers in positions on 737 aircraft that are vacated as certain selected graphite/epoxy spoilers are removed for evaluation.

The all-composite spoiler is the same size as the production aluminum and the graphite/epoxy spoilers, 52 inches long by 22 inches wide. The graphite/epoxy skins are replaced by graphite/polysulfone skins which are produced by a high temperature thermoplastic process with a potential for lower cost fabrication. The metal details in the substructure are entirely replaced with composite parts: glass/epoxy honeycomb and molded graphite/epoxy hinge fittings and leading-edge spar. Weight increase in the glass core is more than offset by weight reductions in the molded fittings. Thus, the advanced concept spoiler is 100 percent composite with a 20 percent weight saving compared to the production aluminum spoiler.

Although the graphite/epoxy-skinned spoilers have been fabricated with the best known methods of corrosion protection between the graphite and the aluminum substructure, the long-term endurance of this protection will remain a question for many years. The all-composite spoiler removed this issue by eliminating all the metal details. Flight service is expected to begin in August 1975 side by side with the graphite/epoxy spoilers and production aluminum spoilers.

C-130 Center Wing Box. - The Lockheed-Georgia Company has fabricated three boron/epoxy reinforced center wing boxes for the C-130 cargo aircraft under contract to LaRC (ref. 35, 36 and 37). One wing box is being ground tested at Lockheed and two wing boxes have been installed in two new C-130H aircraft for flight service evaluation in the USAF. The boron/epoxy reinforced wing box shown in figure 33 as the shaded area of the photo on the right and shown prior to installation at the left is 36.7 feet long, 6.7 feet wide and 2.8 feet deep and is a direct substitution for the standard all-aluminum wing box. The boron/epoxy reinforced wing box weighs 4,440 pounds which is 500 pounds less than the aluminum wing box. An 8 percent unidirectional boron/epoxy reinforcement of wing planks and stringers results in a 10 percent weight saving for the boron/epoxy reinforced aluminum wing box. Flight service with the USAF was initiated in October 1974 at the 314th Tactical Airlift Wing (TAW) of the Little Rock, AFB. Flight service is expected to continue for 3 years with wing box inspections scheduled at 6-month intervals. The first visual and ultrasonic inspections for the two flight articles have been completed and the wing boxes are performing satisfactorily.

Several subcomponents tests were conducted by Lockheed-Georgia prior to fabrication of full-scale boron/epoxy reinforced wing boxes and are illustrated in figure 34. Tests were conducted to investigate load transfer from an all-metal wing joint to a boron/epoxy reinforced wing section. This test component was 12 inches wide and 40 inches long and included typical wing joints at each end. After eight lifetimes of fatigue loading the component had a residual strength that was 135 percent of design ultimate. Another subcomponent test was conducted to measure the compression buckling strength of a typical boron/epoxy reinforced wing panel. This panel was 20 inches wide and 75 inches long. After proper end design for testing, two panels were tested and they failed at 100 and 110 percent of design ultimate, respectively. The final subcomponent test was a tension fatigue test on a typical wing panel with an access door. This panel was 40 inches wide and 140 inches long. Two panels were tested; the first panel had a residual strength of 109 percent of design ultimate after 6 lifetimes of fatigue loading and the second panel had a residual strength of 92 percent of design ultimate after 8 lifetimes of fatigue loading.

A full-scale boron/epoxy reinforced wing box is being tested by Lockheed-Georgia. Static limit load and ground vibration tests have been successfully completed. The vibration test was conducted on the first C-130H aircraft with a boron/epoxy reinforced wing box. A 4 lifetime or 40,000 simulated flight-hour fatigue test is scheduled for completion in June 1975. Periodic visual and ultrasonic inspections have been conducted and no fatigue cracks or disbonds have been found. The residual strength test to failure is scheduled for July 1975.

<u>DC-10 Aft Pylon Skin</u>. - McDonnell Douglas, under LaRC contract, has designed, manufactured, obtained FAA certification and arranged for flight service evaluation of three boron/aluminum aft pylon skins on DC-10 aircraft (ref. 38). The aft pylon is located within the dashed ellipse on figure 35 and is the light colored portion at the bottom of the subassembly pictured on the left. Each boron/aluminum skin replaces a production titanium skin and results in a weight saving of 26 percent. The aft pylon skin is approximately

67 inches in length and 8 inches wide and contains 11 plies (0, ±45, 90° fiber orientation) of boron/aluminum. The skin is attached to the substructure with mechanical fasteners which pass through a series of holes machined along the perimeter of the skin. During flight, the skin is exposed to acoustic loads and air flow which reach levels of 115 db and Mach 1.2, respectively. Peak temperature of 200°F occurs at the end of the ascent phase of flight. Cruise temperature for the skin is 130°F. Flight service by United Airlines will begin in August 1975 and is planned for a 5-year period. The three aircraft will be periodically inspected by both United and Douglas personnel. Results of the inspections will be reported annually.

DC-10 Upper Aft Rudder. - The Douglas Aircraft Company is under contract to LaRC for the design, manufacture and flight service evaluation of graphite/epoxy upper aft rudders for DC-10 transports (ref. 28). As may be seen in the photograph in figure 36, the DC-10 has a 4-segment rudder system. The upper aft rudder has approximate dimensions of 13.5-feet (spanwise) and 3-feet (chordwise) and it has an area of 32 sq. ft. The graphite/epoxy (Narmco-Thornel 300/5208) rudder is a rib-stiffened-skin design in which the structural box consisting of skins, ribs, and spars is manufactured as a cocured assembly. The rudder leading and trailing edges and the tip assembly are glass/epoxy; but the hinges and actuator fittings are production aluminum. The composite rudder will weigh about 57 pounds with 77 percent being composite materials and it will be about 37 percent lighter than the production aluminum rudder.

Present plans are for Douglas to manufacture 11 rudders, one for ground test and 10 for flight service. Douglas has flight service agreements for the 10 rudders, one rudder per aircraft, with the listed airlines: Air Siam (2), Air New Zealand (3), Swiss Air (3), Western (1), and Trans International (1). Flight service evaluation of the composite rudders will begin in June 1976.

The graphite/epoxy structural box for the DC-10 upper aft rudder is manufactured with form mold die tooling, more commonly identified as the "trapped rubber" process. The manufacturing sequence is shown in figure 37 and begins with the layup and densification at 250°F of the right and left side skin panels (upper left photo) and preforming the spars and ribs. The upper right photo shows the preformed front spar being installed in the tooling. The front spar web has lightening holes that permit the spar to fit over the internal metal mandrels of the tool. The mandrels are centered in the cavity formed by adjacent ribs and the skins and each mandrel is surrounded by blocks of silicone rubber that completely fill each cavity. Expansion of the silicone rubber during the cure cycle produces the required molding pressure. The lower right photo shows the assemblage of spars, ribs and silicone rubber tooling just prior to the installation of the skin panels. After the skin panels are installed, the heavy steel side plates of the tool are attached and the tool is rolled into the curing oven. The 350°F curing temperature is provided by the oven and electrical heating elements contained within each internal mandrel. The center photo shows the completed graphite/epoxy structural box.

L-1011 Vertical Fin. - The most recent flight service program initiated by NASA and the first for a primary structural component for a commercial aircraft is shown in figure 38. The program provides for the design, development, test, and inservice evaluation of the structural box of the Lockheed L-1011 vertical fin which is located by the shaded area of the tail in the photo. The component has a span of about 25 feet and a root chord of about 9 feet. The principal structural material will be graphite/epoxy with Kevlar 49 used in selected areas. The proposed design makes extensive use of composites. Over 80 percent by weight of the redesigned box will be made from composite material and a weight saving of about 25 percent of the current metal box design is anticipated.

Two types of construction are under consideration - a sandwich skin design and a hat-stiffened skin design. The detail shown is a preliminary concept for the hat-stiffened design and shows a typical rib-spar-skin intersection. A single concept will be selected for detail design at the end of a 5-month preliminary design effort.

Two components of the selected design will be fabricated for the inservice evaluation portion of the program. The components will be installed on new aircraft during production. The contractor has entered into discussions with several airlines for flight service of the composite ailerons, but specific arrangements with the airlines have not been completed. The current program and aircraft production schedules show the first component entering service in January 1979, and the second about 5 months later. The components will remain in flight service for a minimum of 5 years.

Aircraft in Flight Service Program

The specific aircraft flying the components already in service are indicated in figure 39. In the upper left is shown the L-1011 fairing panel and the three airlines performing the flight service program on this component, namely, Eastern, Air Canada, and TWA. The corresponding tail numbers are also given. Immediately below is shown the C-130 center wing box and the two USAF airplanes involved are identified. In the lower left is shown the B-737 spoiler and the six participating airlines: Aloha, Lufthansa, New Zealand National, Piedmont, PSA and VASP (Brazil). The tail numbers associated with each airline are given also.

Component Flight Hours

Figure 40 shows how the high time component of each type is expected to accumulate time. The chart indicates the hours of flight service that will be attained by the high time components up through 1981 for each of the flight service components noted on the right hand side of the chart. The number of flight service components in each program is indicated in parentheses. Several of the programs were under development at the time of RECAST and began flight service shortly thereafter. The CH-54B tail cone is currently achieving approximately 84 hours of flight service per year. The high time

L-1011 fairing panel experiences approximately 2600 hours of flight service per year, the B-737 spoiler 2350, the DC-10 aft pylon skin and DC-10 upper aft rudder will achieve an estimated 2920 hours of flight service per year, the C-130 center wing box achieves 720, and the L-1011 vertical fin is estimated at 3000.

The higher rate of utilization of commercial aircraft is evidenced by the projections which show that by 1981, the L-1011 fairing panels, the B-737 spoilers, the DC-10 aft pylon skin and DC-10 upper aft rudder will have attained approximately 20,000 hours of flight service. The L-1011 vertical fin will be approaching 10,000 hours of flight service and the C-130 wing box approximately 5,000 hours and the CH-54B tail cone approximately 1,000 hours of flight service.

Figure 41 indicates the cumulative flight service hours for all the NASA components through 1981. The circled numbers refer to the indicated components and their location on the curve indicates the time the component entered flight service. At the present time, approximately 400,000 cumulative flight service hours have been achieved for all NASA components. One million hours will be achieved early in 1977 and 2.5 million hours by the end of 1981. Most of this experience will be provided by one program - flying over 100 spoilers worldwide on B-737 aircraft.

CONCLUDING REMARKS

In conclusion, NASA has increased its confidence building since RECAST by a technology program in space vehicles, aircraft engines, and aircraft structures applications and flight service. This program is providing extensive data on the serviceability of composite materials with the expectation of much more data in the years ahead.

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33. Wooley, J. H.: Flight Service Evaluation of PRD-49/Epoxy Composite Panels in Wide-Bodied Commercial Transport Aircraft. First Annual Flight Service Report, Lockheed-California Co., NASA CR-132647, July 1974.

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FIGURES

OUTLINE

- MAJOR THRUSTS OF RECAST
- NASA COMPOSITES PROGRAM
- SPACE VEHICLE APPLICATIONS
- AIRCRAFT ENGINE APPLICATIONS
- AIRCRAFT STRUCTURAL APPLICATIONS
- PROGRESS SUMMARY

FIGURE 1. – NASA PROGRESS SINCE RECAST

CONFIDENCE VIA

- FLIGHT SERVICE EVALUATION OF NUMEROUS COMPONENTS
- LABORATORY SIMULATIONS
- DESIGN PHILOSOPHY AND CRITERIA

COST REDUCTION VIA

- INNOVATIVE MATERIALS USAGE AND FABRICATION METHODS

FIGURE 2. – MAJOR THRUSTS OF RECAST

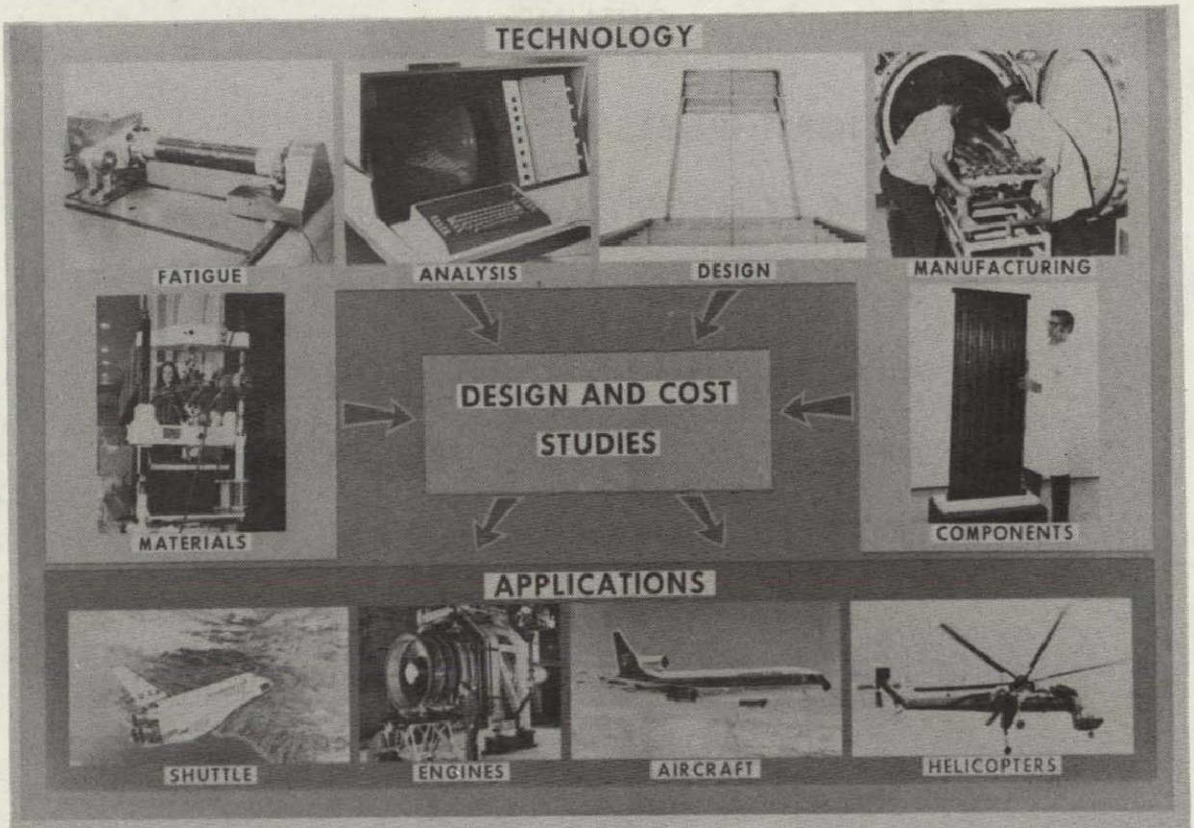


FIGURE 3. – NASA COMPOSITES PROGRAM

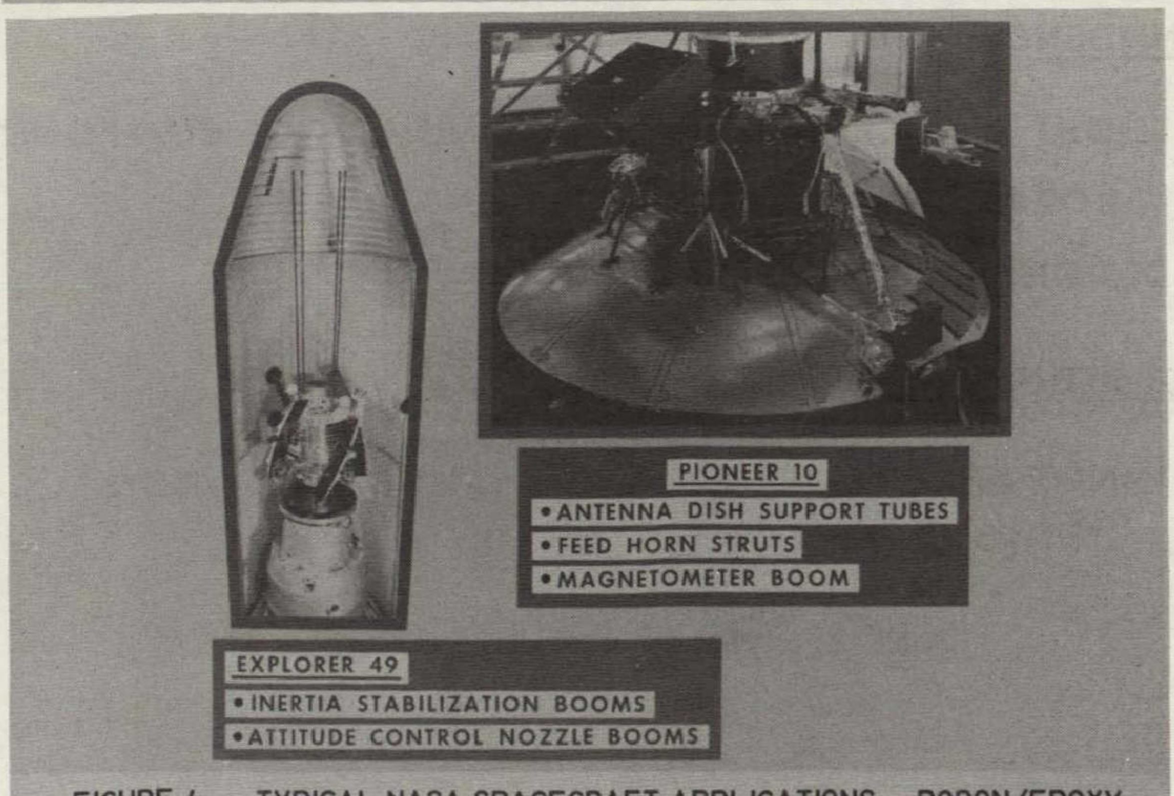


FIGURE 4. – TYPICAL NASA SPACECRAFT APPLICATIONS – BORON/EPOXY

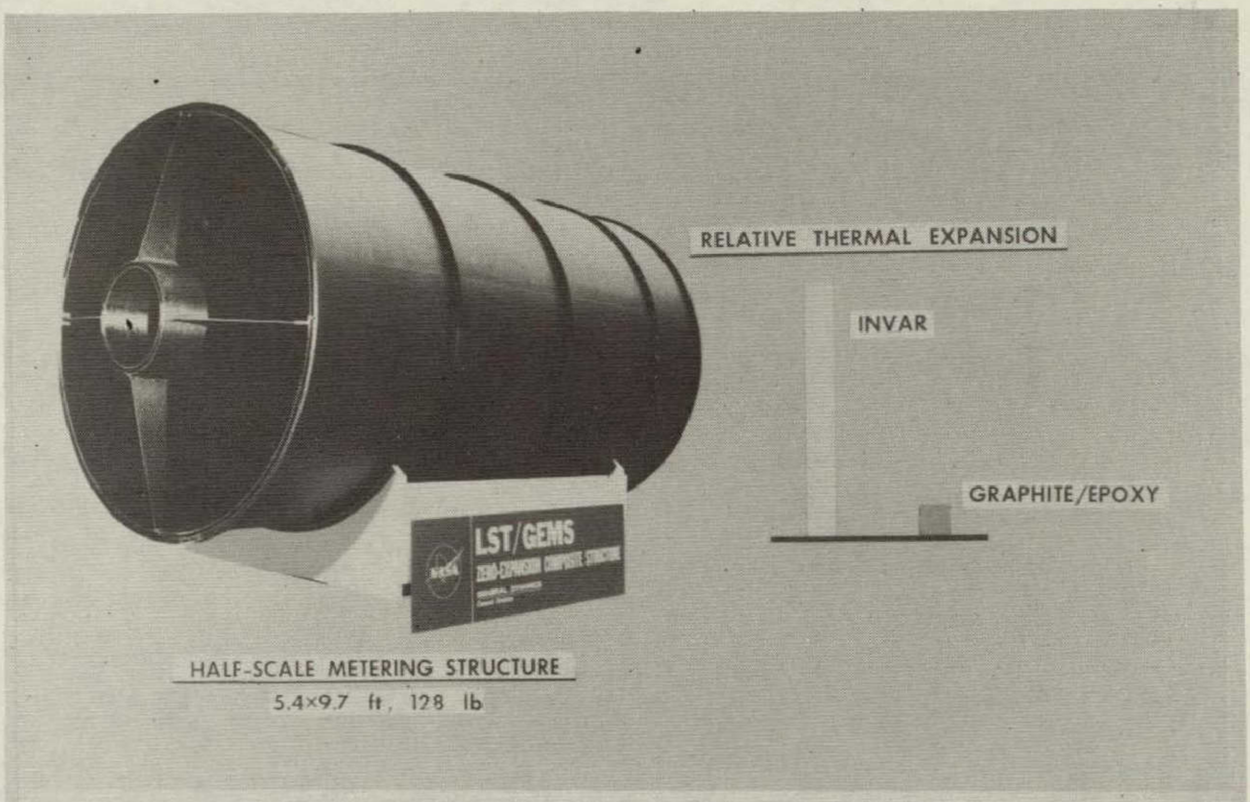


FIGURE 5. – SPACE TELESCOPE TECHNOLOGY

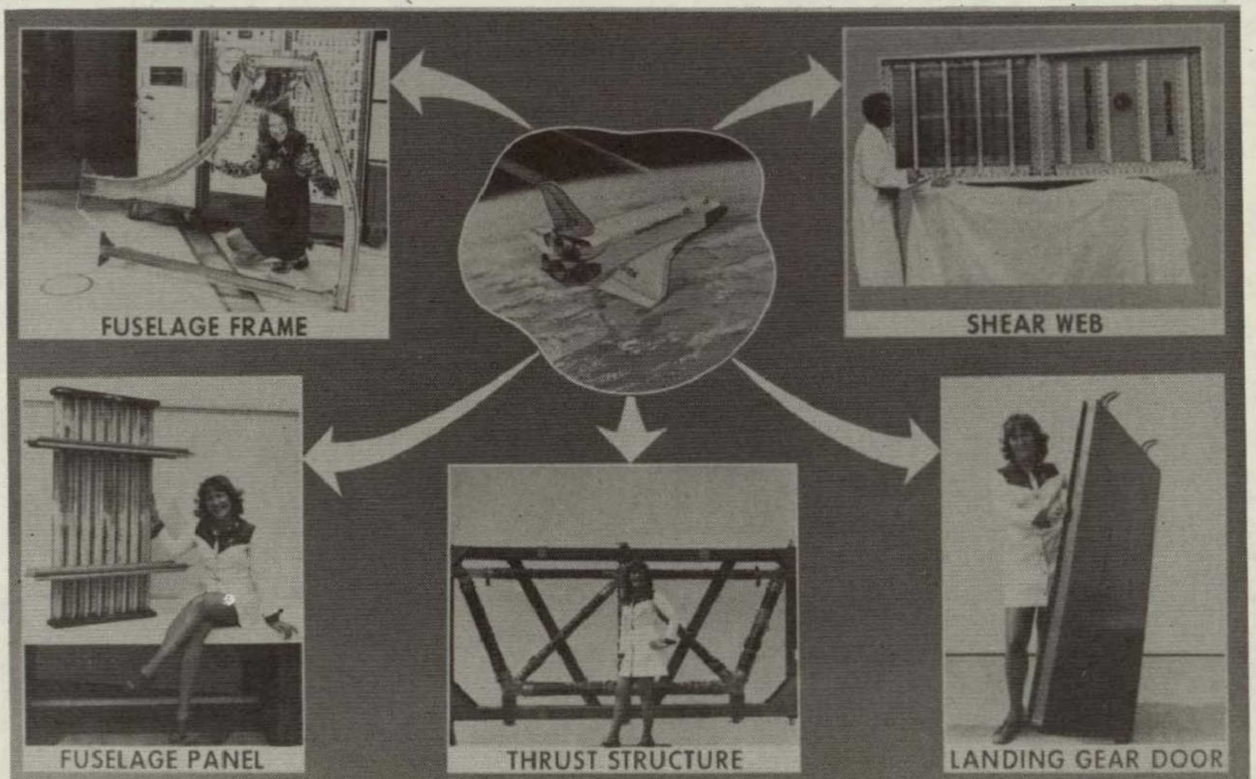


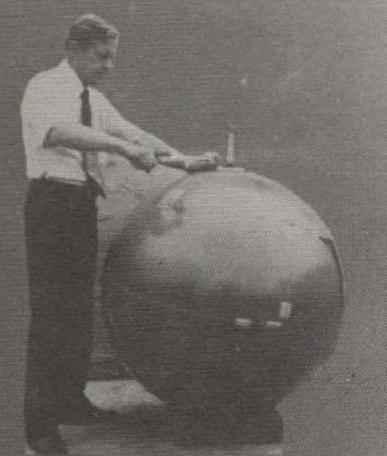
FIGURE 6. – SPACE SHUTTLE COMPONENT DEVELOPMENT

FIGURE 7. – SPACE SHUTTLE ORBITER COMPOSITE APPLICATIONS

SHUTTLE ORBITER APPLICATIONS

SYSTEM	NO.	DIAMETER IN.	WEIGHT PER TANK	WEIGHT SAVED
OMS/HE	2	40	295	21%
MPS/HE	4	25	75	24%
RCS/HE	6	21	25	17%
ECLSS/O_2	1	25	77	43%
ECLSS/N_2	4	25	75	24%
TOTAL SAVINGS			435 LB	23%

EXPERIMENTAL 38 INCH TANK



KEVLAR/EPOXY OVERWRAP
ALUMINUM LINER

FIGURE 8. – COMPOSITE OVERWRAP PRESSURE VESSELS

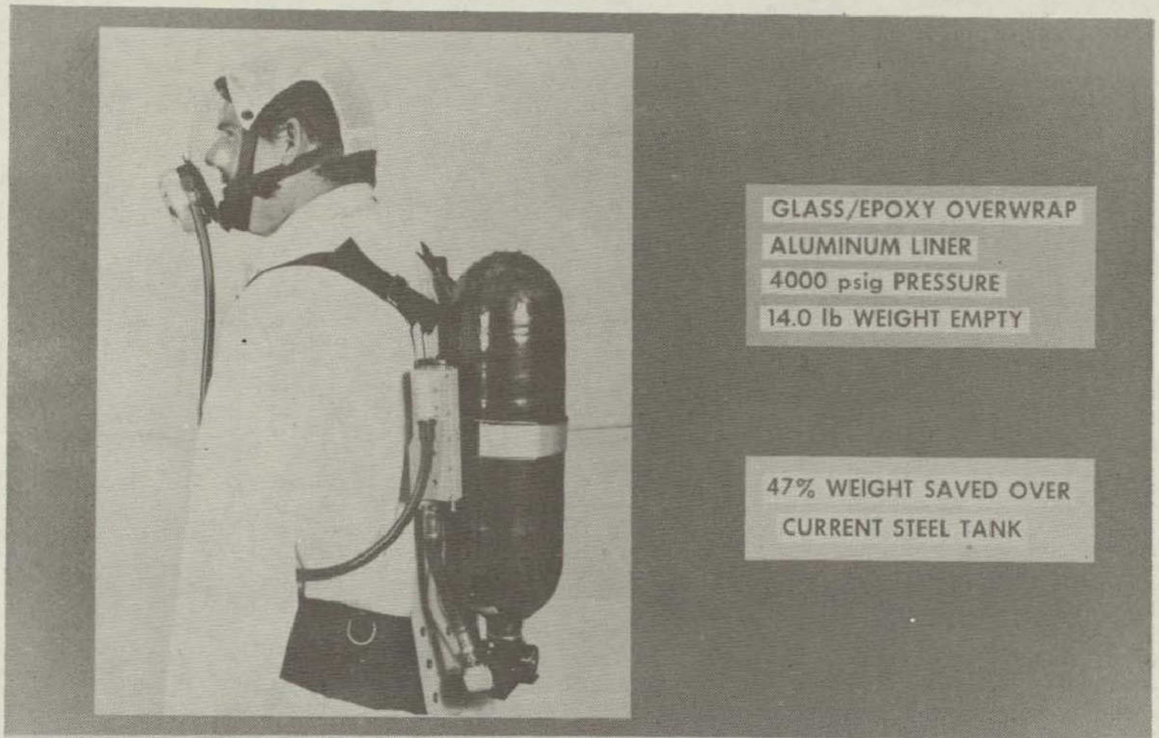


FIGURE 9. – COMPOSITE TANK FOR FIREMAN'S AIR BREATHING APPARATUS

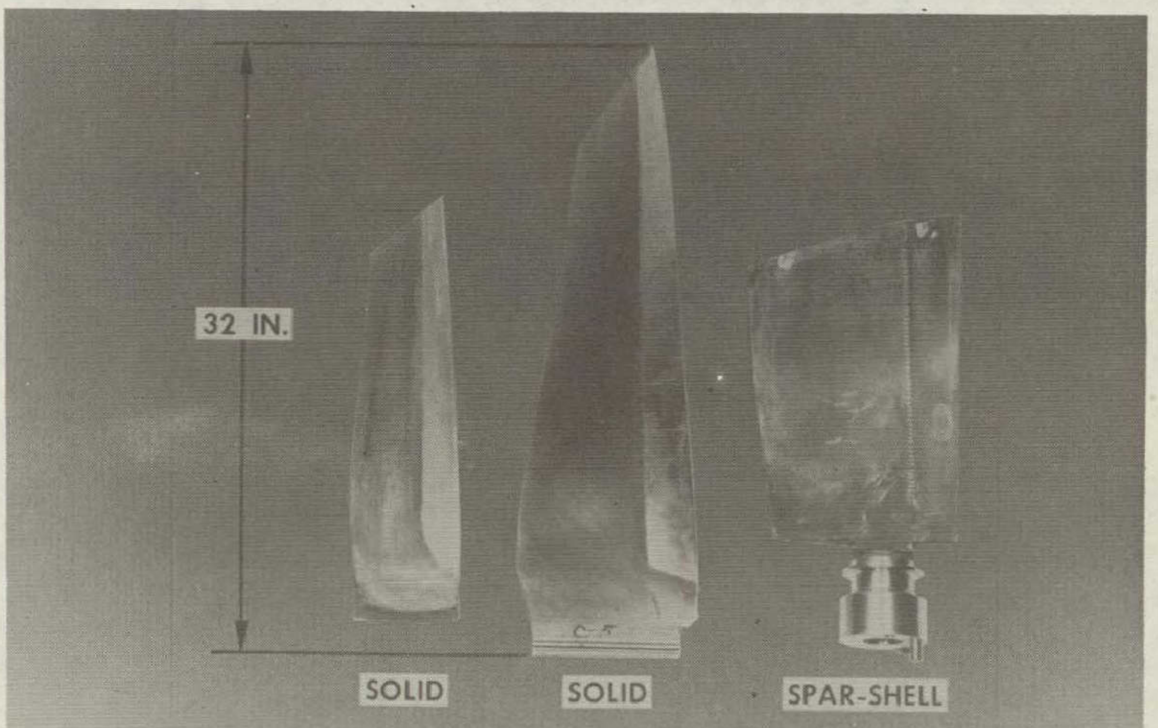


FIGURE 10. – EXPERIMENTAL COMPOSITE FAN BLADES

FIGURE 11. – GRAPHITE/PMR POLYIMIDE FAN BLADE

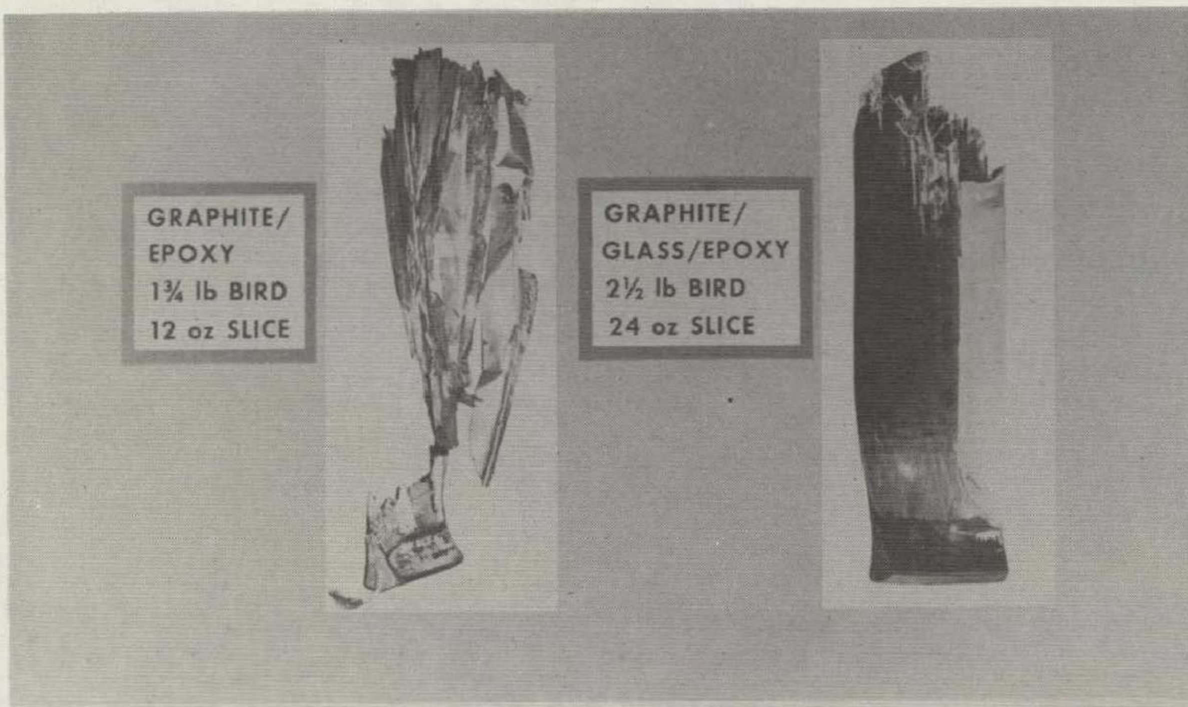


FIGURE 12. – NASA IMPACT IMPROVEMEMT PROGRAM FOR TF39
COMPOSITE FAN BLADE

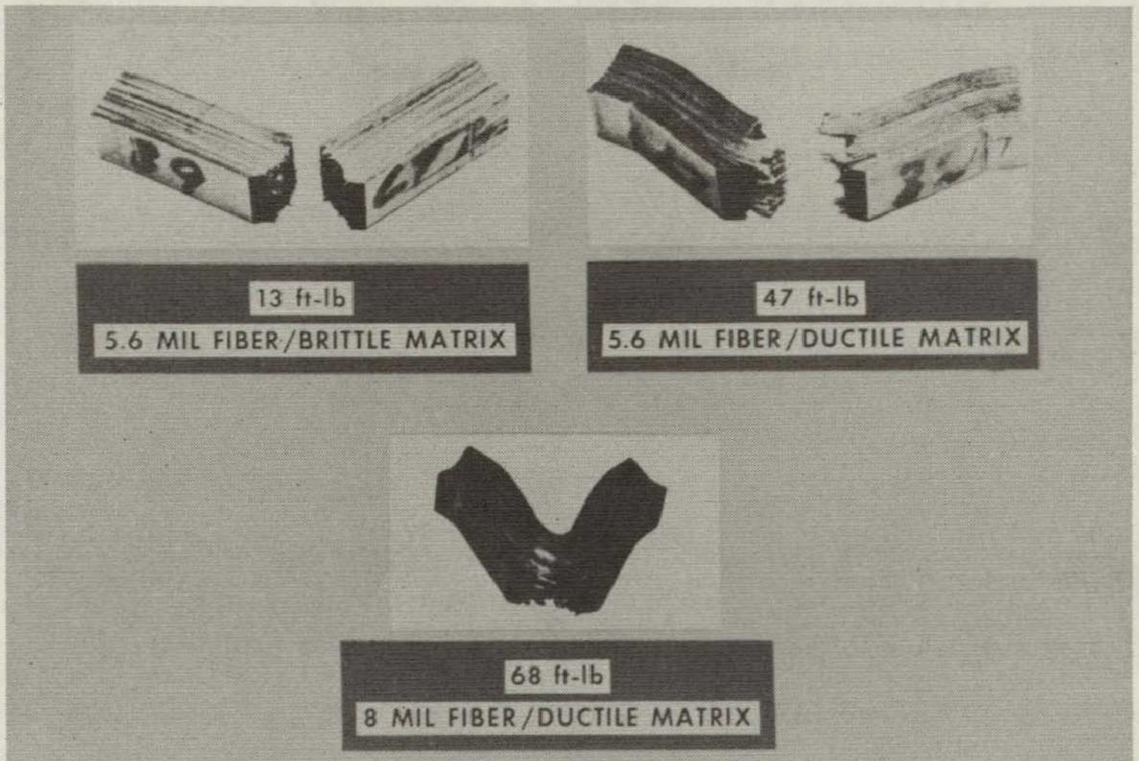


FIGURE 13. – IMPROVED B/Al IMPACT RESISTANCE 50, VOLUME % FIBER

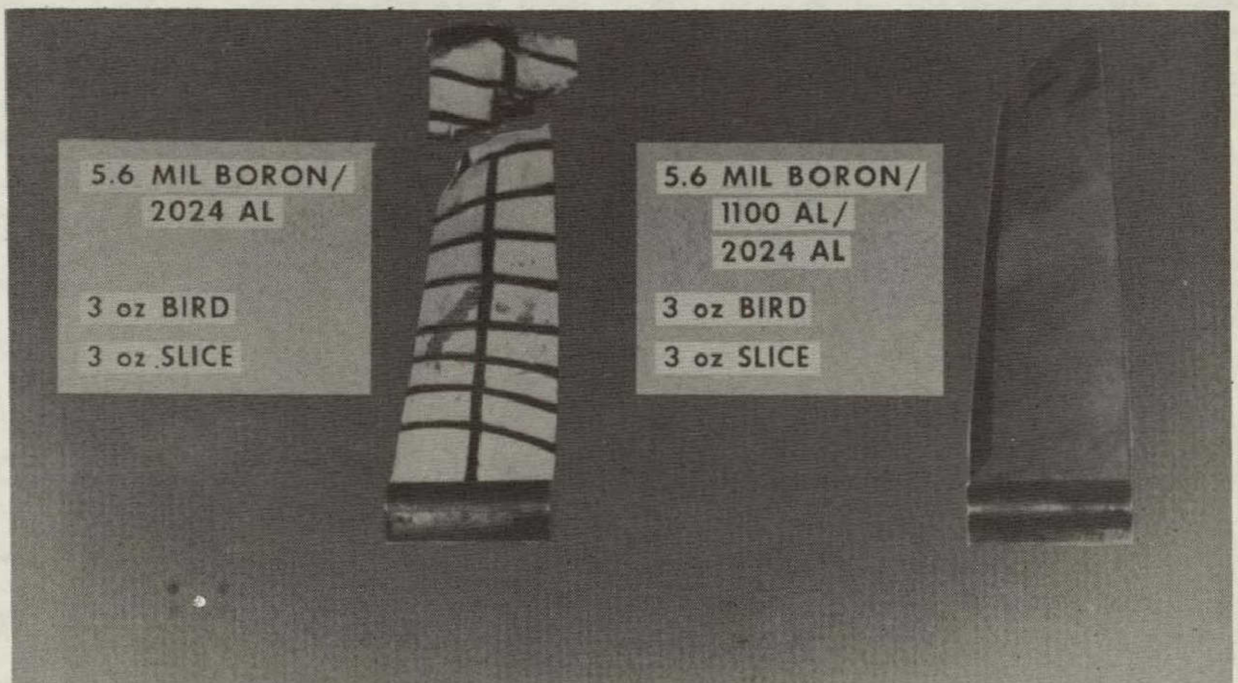


FIGURE 14. – NASA/USAF IMPROVED IMPACT RESISTANCE
J79 B/Al COMPRESSOR BLADE

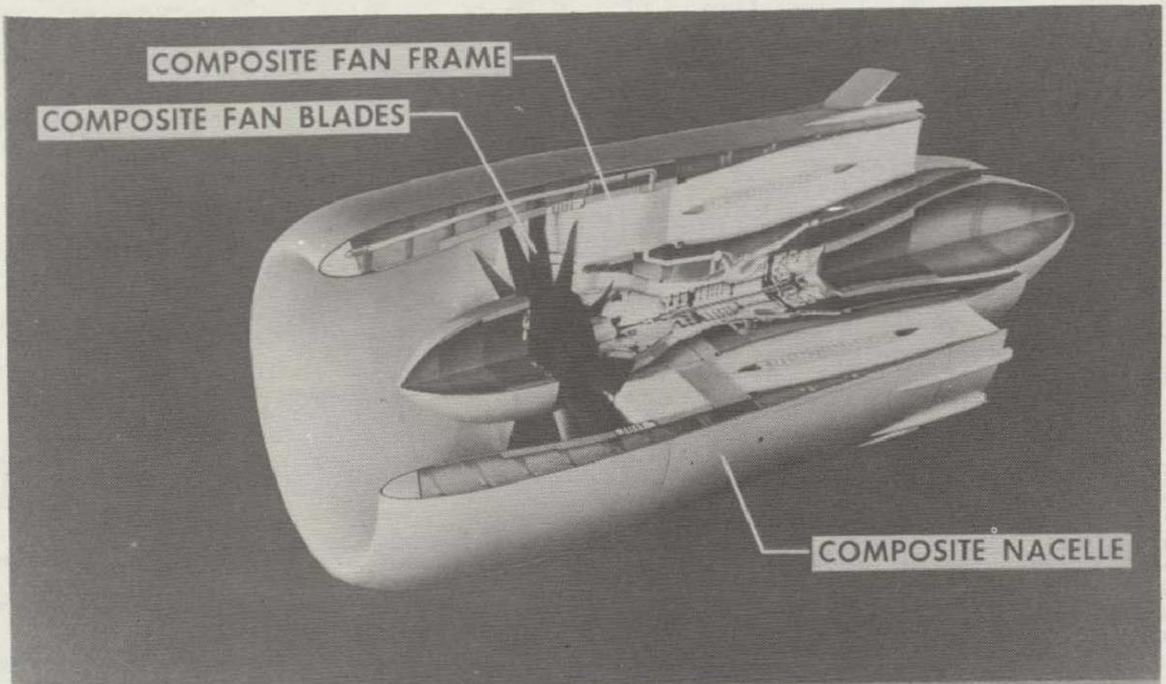


FIGURE 15. – QCSEE UTW ENGINE

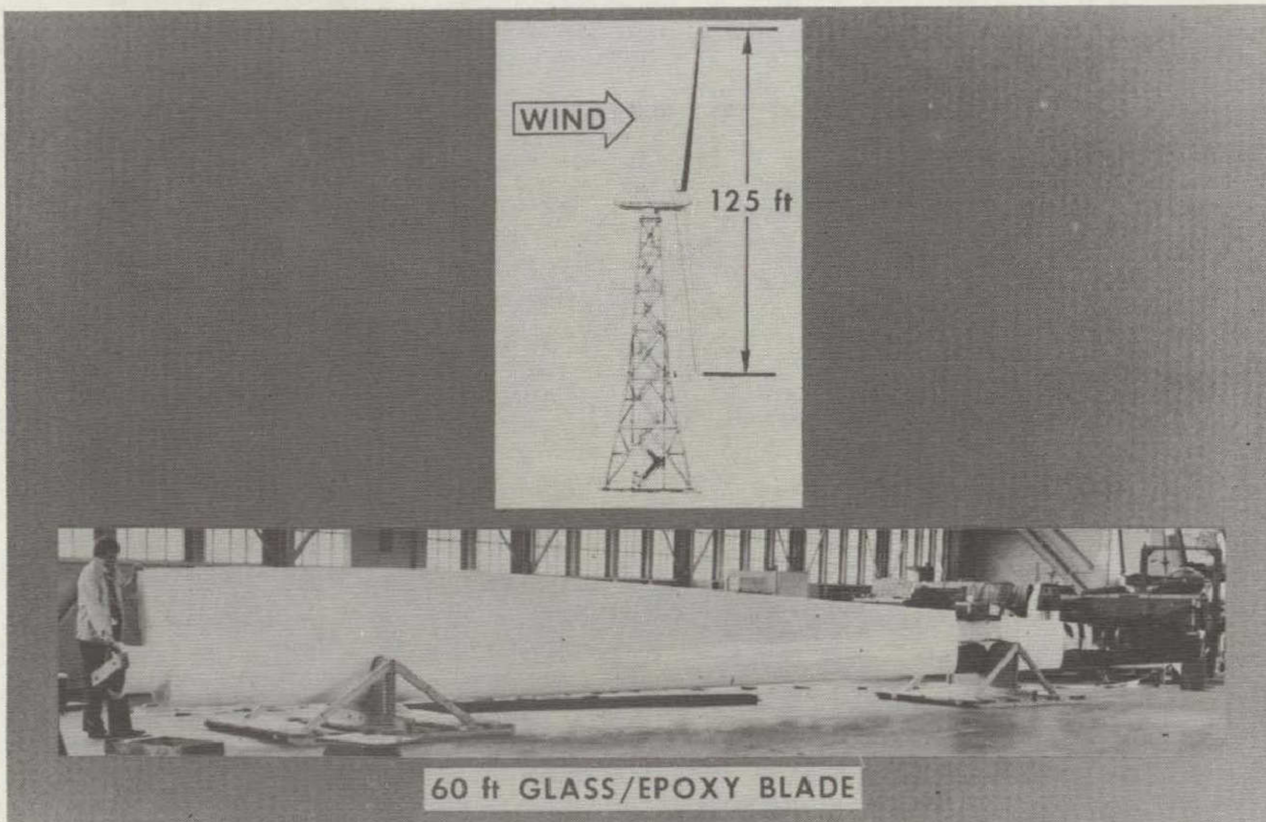


FIGURE 16. – 100 KILOWATT WIND ENERGY GENERATOR

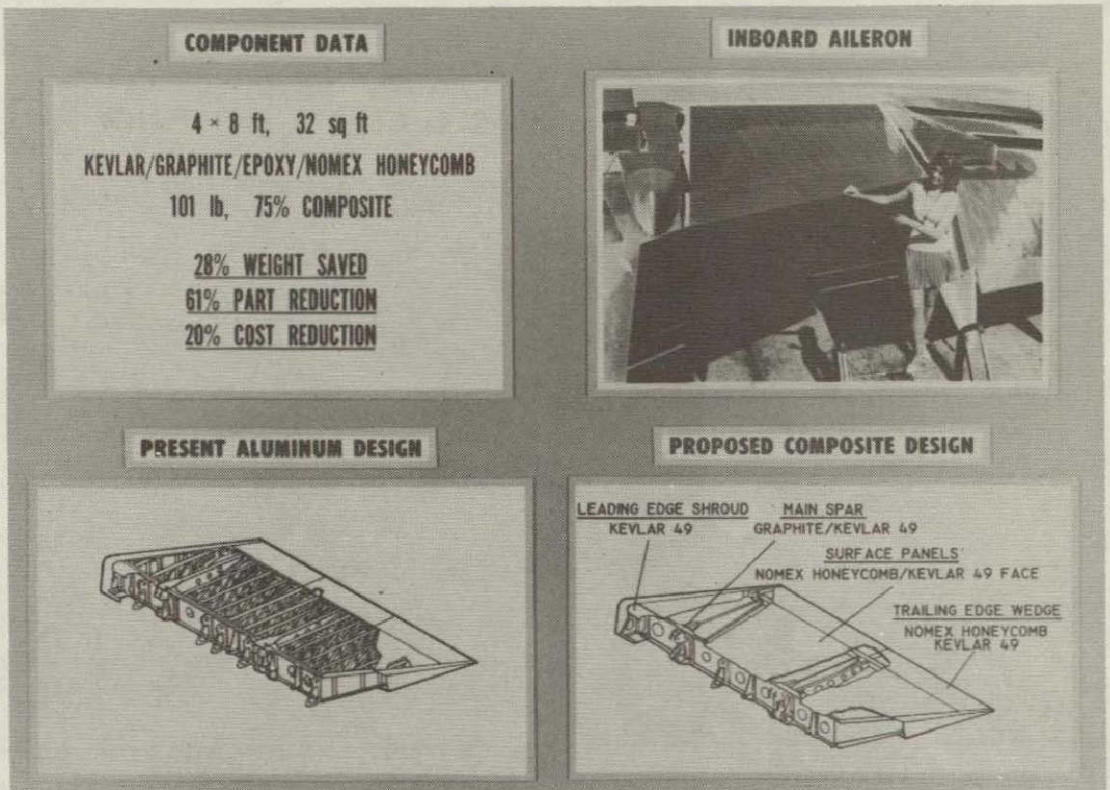


FIGURE 17. – L-1011 COMPOSITE AILERON DESIGN

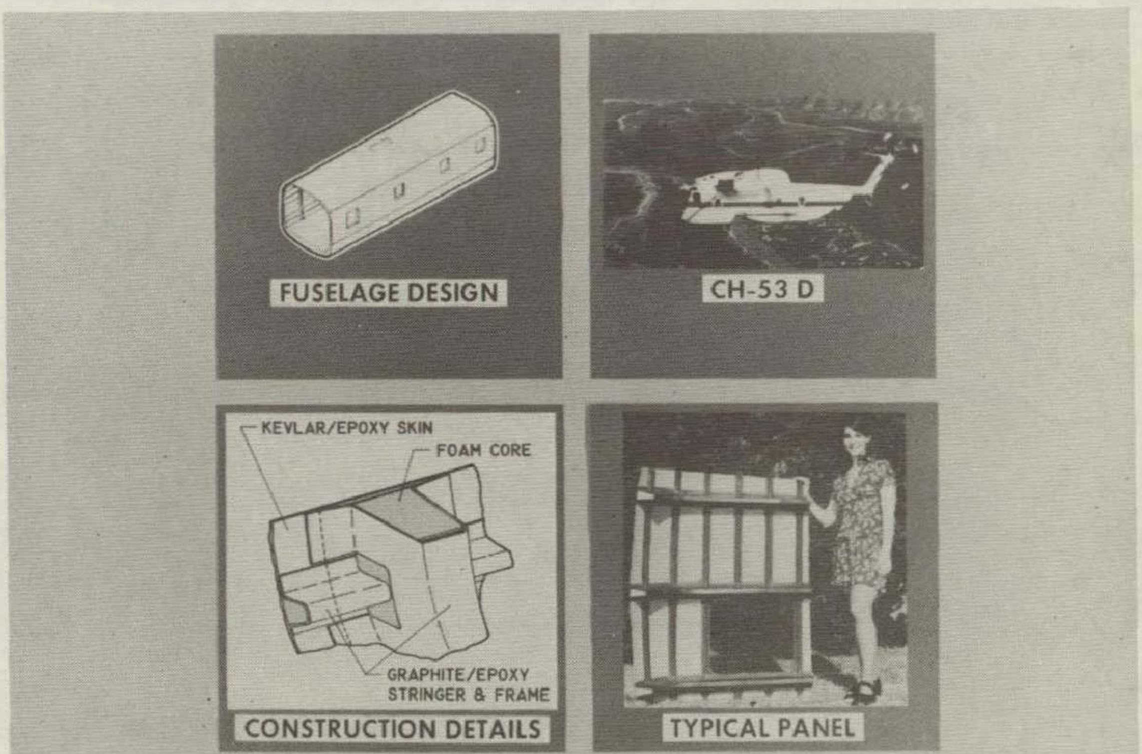


FIGURE 18. – COMPOSITE STRUCTURE FOR HELICOPTERS

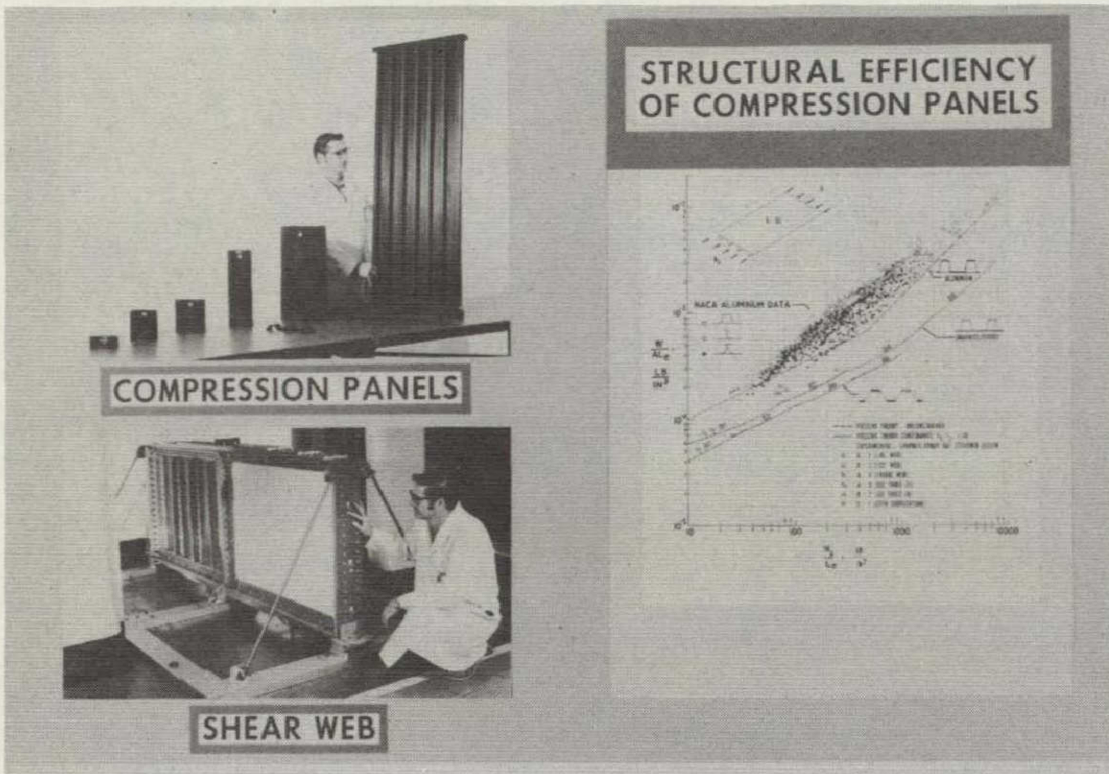


FIGURE 19. – MINIMUM WEIGHT GRAPHITE EPOXY PANELS

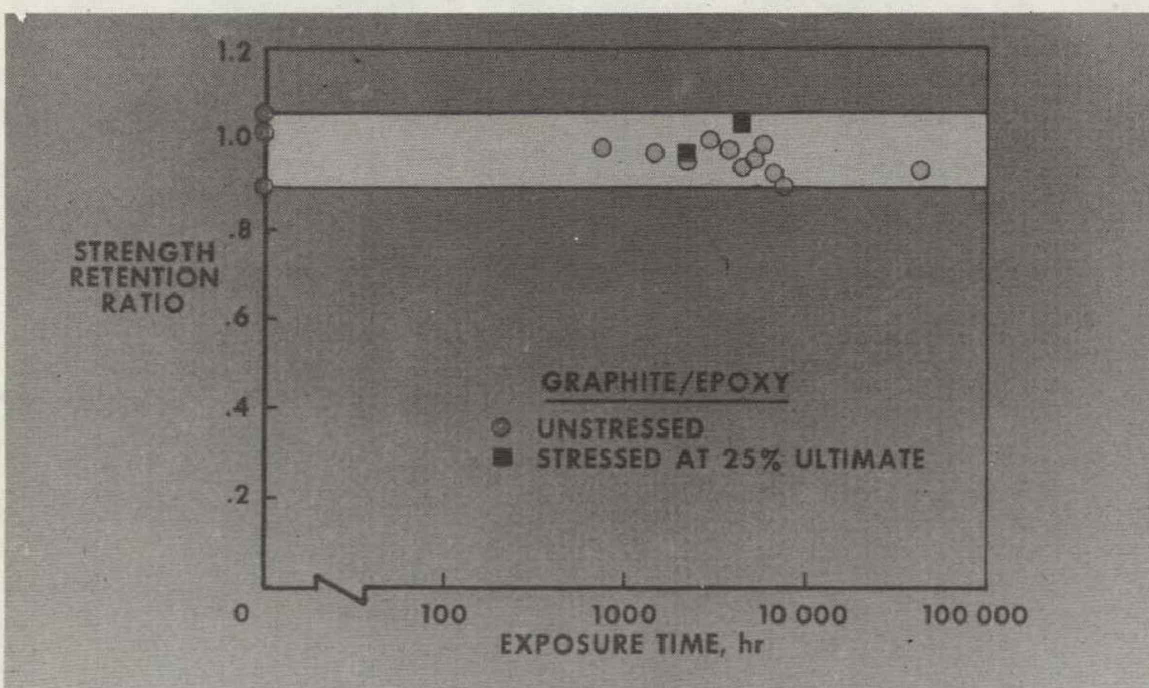


FIGURE 20. – MATRIX CONTROLLED COMPOSITE STRENGTH AFTER OUTDOOR EXPOSURE

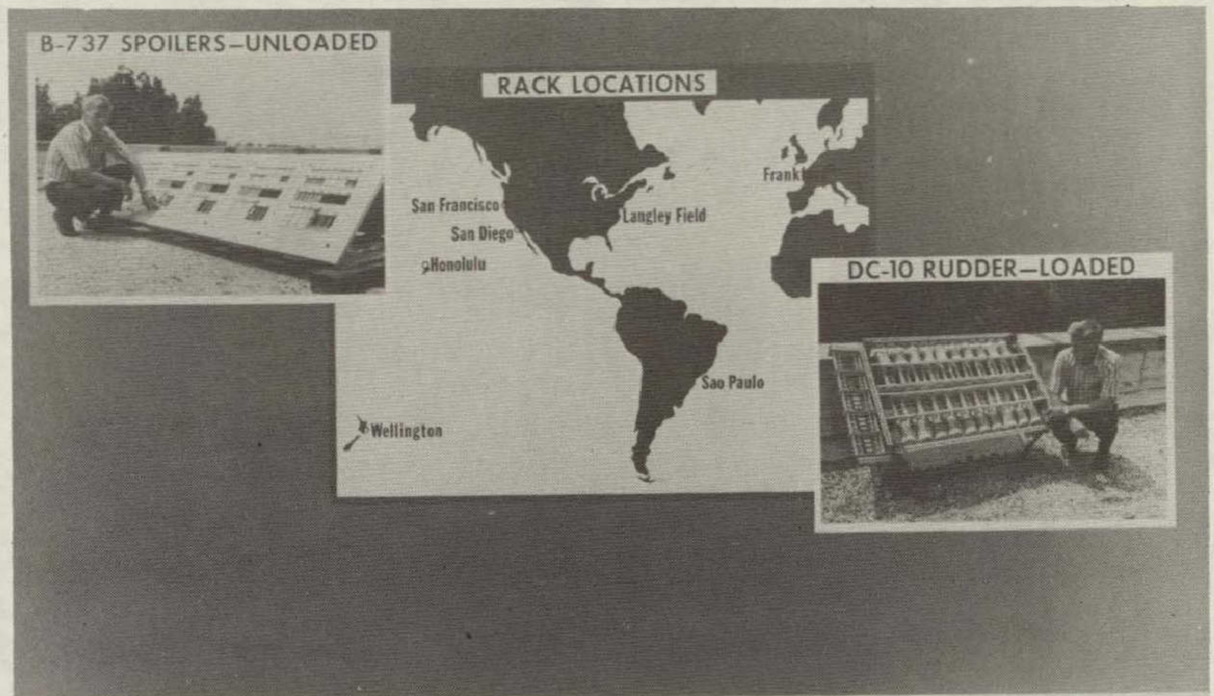


FIGURE 21. — ENVIRONMENTAL EXPOSURE OF COMPOSITE MATERIALS USED IN FLIGHT SERVICE PROGRAMS

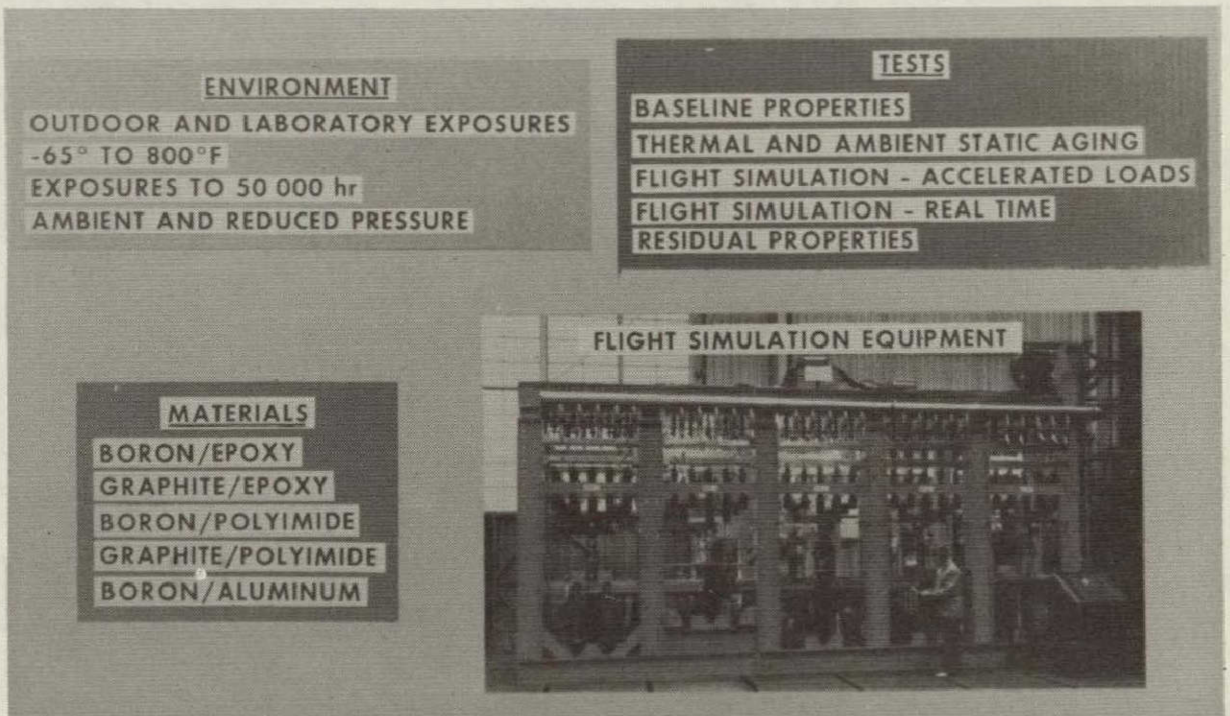


FIGURE 22. — TIME—TEMPERATURE—STRESS CAPABILITIES OF COMPOSITES

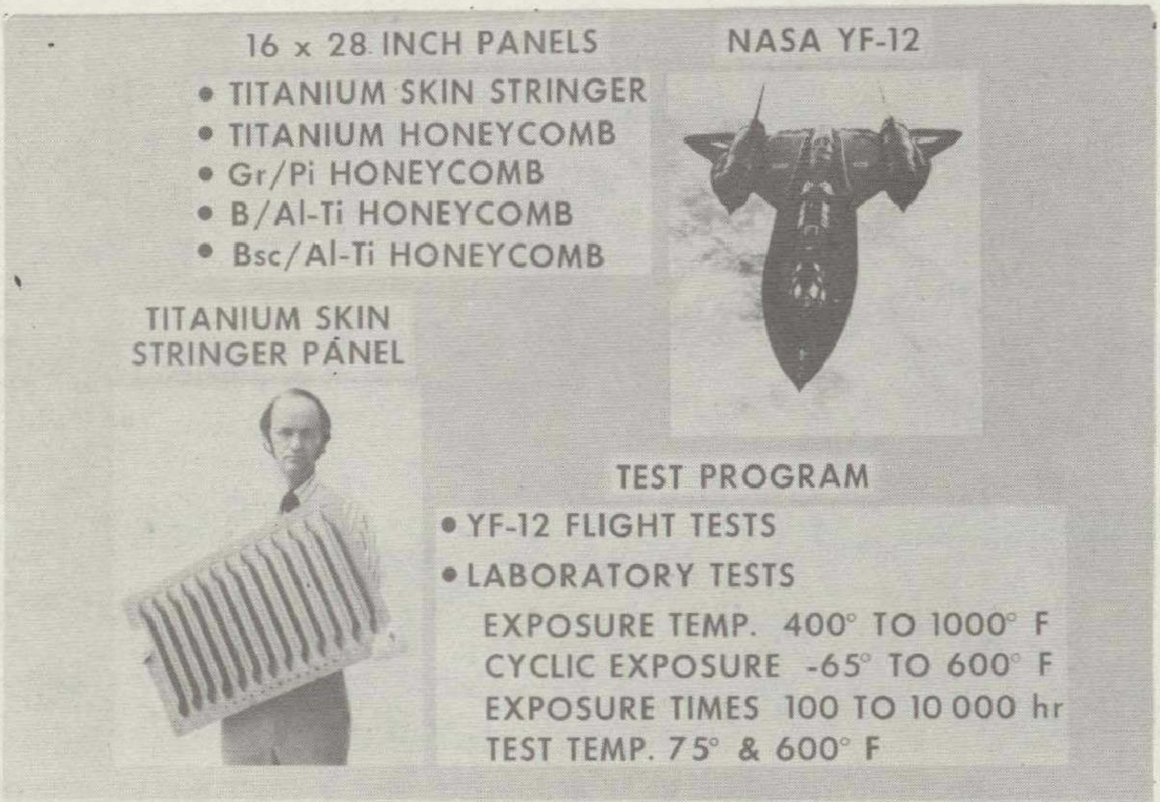


FIGURE 23. – WING PANELS FOR SUPERSONIC AIRCRAFT

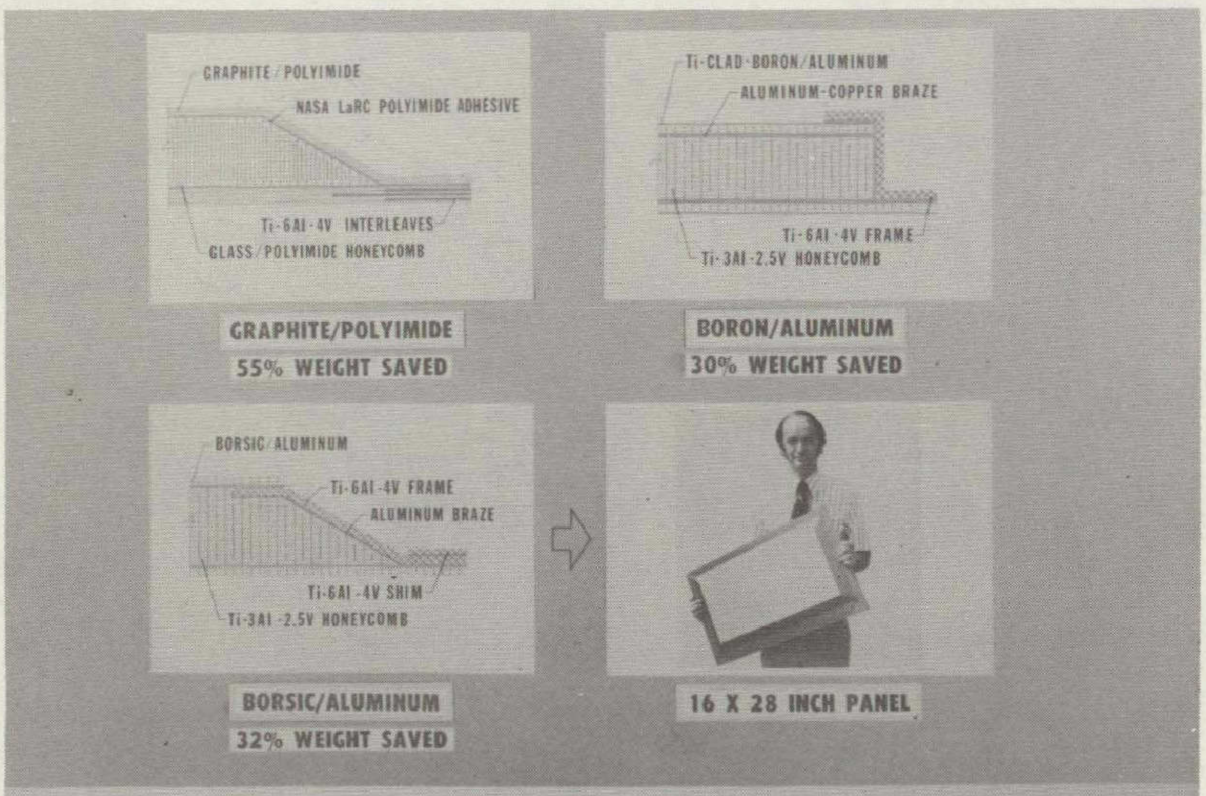


FIGURE 24. – COMPOSITE PANELS FOR SUPERSONIC AIRCRAFT

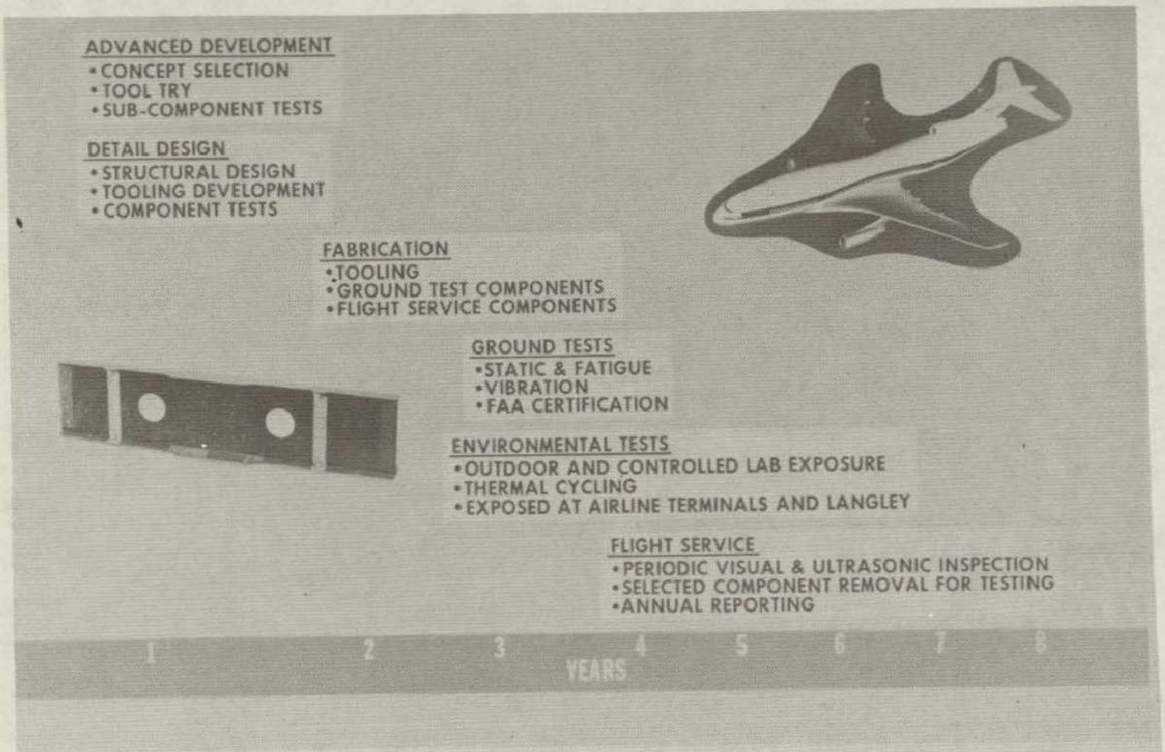


FIGURE 25. — TYPICAL FLIGHT SERVICE EVALUATION PROJECT

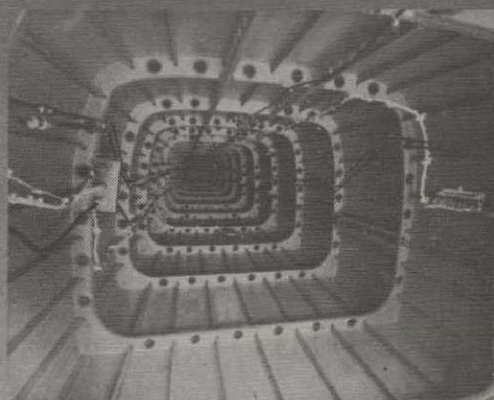
A/C AND COMP. % AND PRINCIPAL COMPOSITE	START OF FLIGHT SERVICE	NUMBER PARTICIPATING		CUMULATIVE FLIGHT HOURS			
				JUNE 1, 1975		DECEMBER 31, 1981	
		A/C	COMP.	HIGH TIME A/C	TOTAL COMP.	HIGH TIME A/C	TOTAL COMP.
CH-54B TAIL CONE 8% BORON/EPOXY (REINF.)	MARCH 1972	1	1	527	527	1000	1000
L-1011 FAIRING PANELS 100% KEVLAR 49/EPOXY	JANUARY 1973	3	18	6293	80 916	23 410	353 800
B-737 SPOILERS 35% GRAPHITE/EPOXY	JULY 1973	27	120	4118	358 000	19 734	1 938 000
C-130 CENTER WING BOX 8% BORON/EPOXY (REINF.)	OCTOBER 1974	2	2	436	781	5176	10 260
DC-10 AFT PYLON SKIN 100% BORON/ALUMINUM	AUGUST 1975	3	3			19 750	59 250
DC-10 UPPER AFT RUDDER 77% GRAPHITE/EPOXY	JUNE 1976	10	10			16 300	155 000
L-1011 VERTICAL FIN 83% GRAPHITE/EPOXY	JANUARY 1979	2	2			8750	17 000
TOTALS		48	156	11 374	440 224	94 120	2 534 310

FIGURE 26. — NASA FLIGHT SERVICE SUMMARY

FLIGHT SERVICE DATA

AIRCRAFT	COMPONENT	USER
1	1	U.S. ARMY 355 th AVIATION CO. FORT EUSTIS, VA.

INITIATED MARCH 1972



COMPONENT DATA

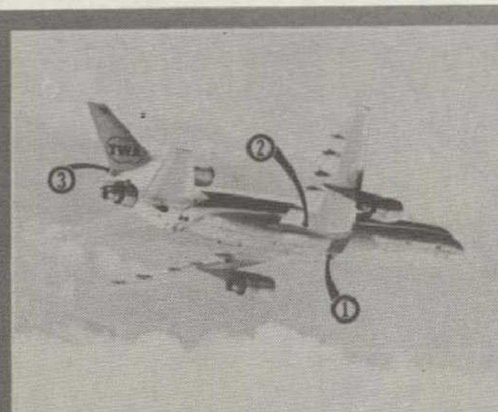
3.5 x 3 x 17 ft, 119 sq ft
BORON/EPOXY REINFORCED ALUMINUM
397 lb 8% COMPOSITE
14% WEIGHT SAVED
1 PER AIRCRAFT

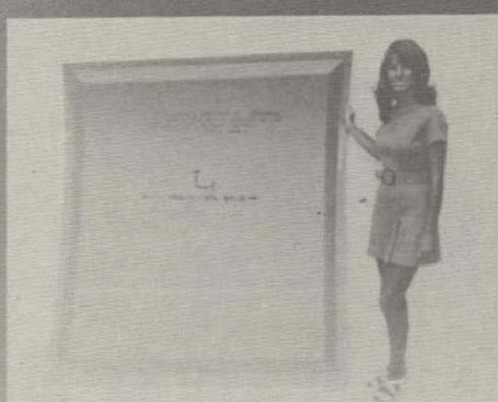
FIGURE 27. – CH–54B TAIL CONE

FLIGHT SERVICE DATA

AIRCRAFT	COMPONENTS	AIRLINE
1	6	TWA
1	6	EASTERN
1	6	AIR CANADA

INITIATED JANUARY 1973





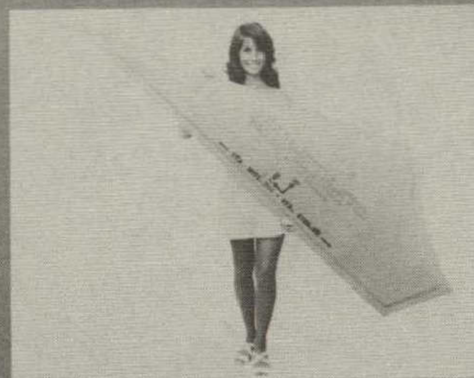
① WING TO BODY FAIRING

60×67 INCHES, 28 SQ FT
KEVLAR/NOMEX HONEYCOMB/EPOXY
16 LB 100% COMPOSITE
25% WEIGHT SAVED
2 PER AIRCRAFT

FIGURE 28. – L–1011 FAIRING PANELS, I

(3) CENTER ENGINE FAIRING

30×82 INCHES, 7 SQ FT
KEVLAR/NOMEX HONEYCOMB/EPOXY
5 LB 100% COMPOSITE
30% WEIGHT SAVED
2 PER AIRCRAFT



(2) WING TO BODY FILLET

5.5×32.5 INCHES, 2 SQ FT
KEVLAR/EPOXY
2 LB 100% COMPOSITE
32% WEIGHT SAVED
2 PER AIRCRAFT

FIGURE 29. – L-1011 FAIRING PANELS, II

FLIGHT SERVICE DATA

AIRCRAFT	COMPONENT	AIRLINE
4	16	ALOHA
6	24	LUFTHANSA
4	16	NEW ZEALAND NATIONAL
8	32	PIEDMONT
1	4	PSA
4	16	VASP

INITIATED JULY 1973

COMPONENT DATA

22 × 52in., 7.5 sq ft
GRAPHITE/EPOXY/ALUMINUM HONEYCOMB, ALUMINUM HINGES & SPAR
13 lb 35% COMPOSITE
17% WEIGHT SAVED
4 PER AIRCRAFT

FIGURE 30. – B-737 SPOILERS

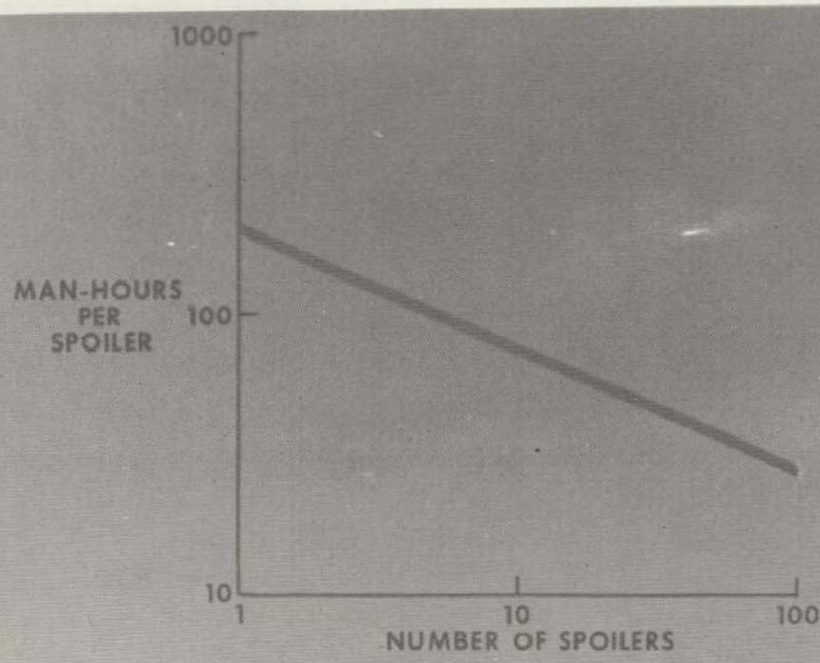


FIGURE 31. – FABRICATION MAN-HOURS FOR B-737 GRAPHITE/EPOXY SPOILERS

FLIGHT SERVICE DATA		
AIRCRAFT	COMPONENT	AIRLINE
2	2	ALOHA
2	2	LUFTHANSA
2	2	NEW ZEALAND NATIONAL
2	2	PIEDMONT
1	2	PSA
2	2	VASP

TO BE INITIATED AUGUST 1975

COMPONENT DATA
22 × 52 in., 7.5 sq ft
GRAPHITE/POLYSULFONE/GLASS HONEYCOMB, GRAPHITE/EPOXY HINGES & SPAR
12.5 lb 100% COMPOSITE
20% WEIGHT SAVED
1 PER AIRCRAFT

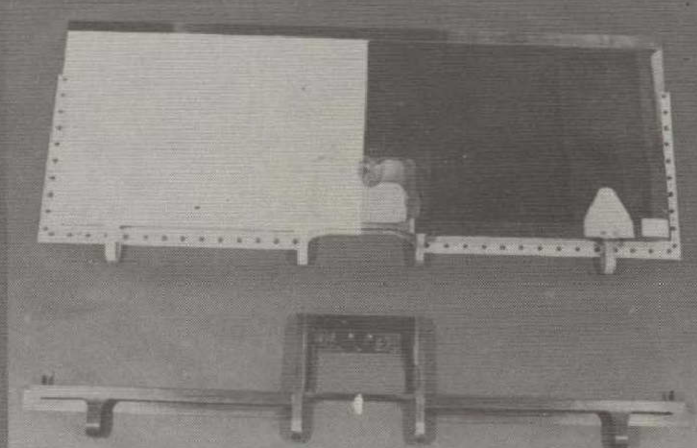


FIGURE 32. – B-737 ADVANCED SPOILERS

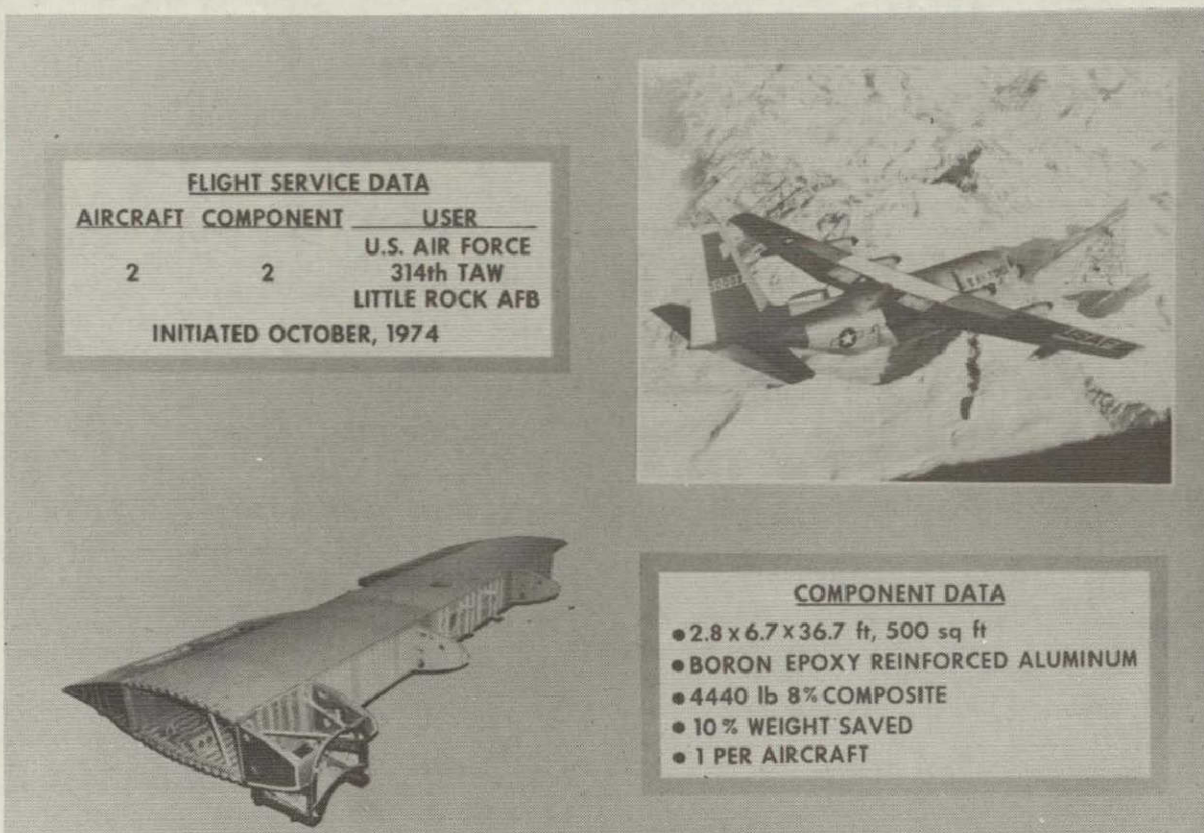


FIGURE 33. – C–130 CENTER WING BOX

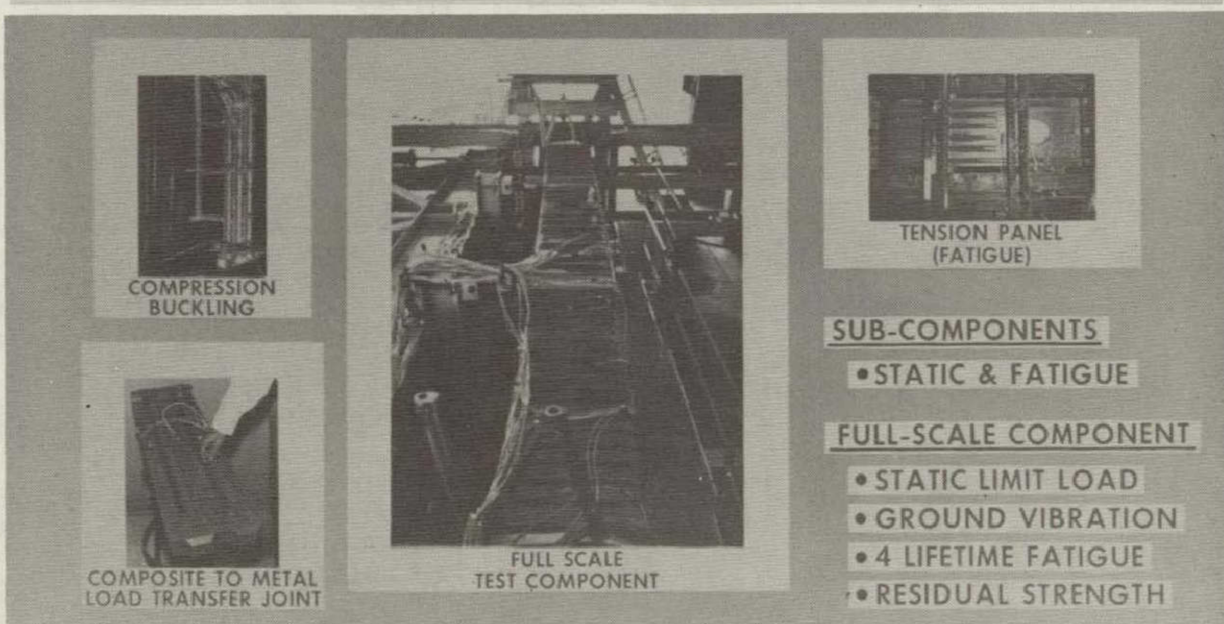
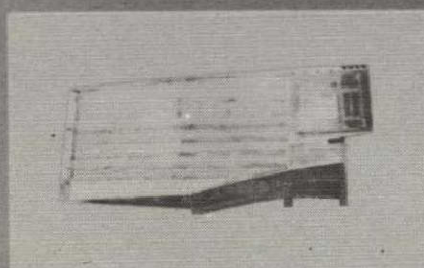


FIGURE 34. – C–130 COMPOSITE REINFORCED CENTER WING BOX TEST PROGRAM

FLIGHT SERVICE DATA

AIRCRAFT	COMPONENTS	AIRLINE
3	3	UNITED

TO BE INITIATED AUGUST, 1975



COMPONENT DATA

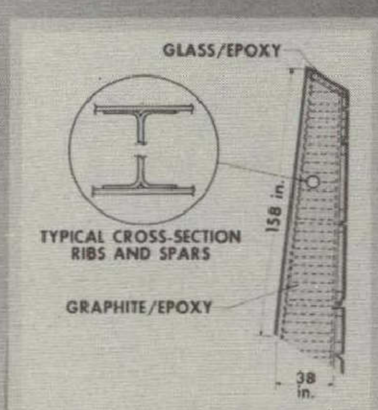
- 67 x 8 in., 3.2 sq ft
- BORON/ALUMINUM
- 3.45 lb 100% COMPOSITE
- 26% WEIGHT SAVED
- 1 PER AIRCRAFT

FIGURE 35. – DC–10 AFT PYLON SKIN

FLIGHT SERVICE DATA

AIRCRAFT	COMPONENT	AIRLINE
2	2	AIR SIAM
3	3	NEW ZEALAND
3	3	SWISS AIR
1	1	TRANS INTERNATIONAL
1	1	WESTERN

TO BE INITIATED JUNE 1976



COMPONENT DATA

- 3 x 13.5 ft, 32 sq ft
- GRAPHITE/EPOXY/GLASS
- 57 lb 77% COMPOSITE
- 37% WEIGHT SAVED
- 1 PER AIRCRAFT

FIGURE 36. – DC–10 UPPER AFT RUDDER

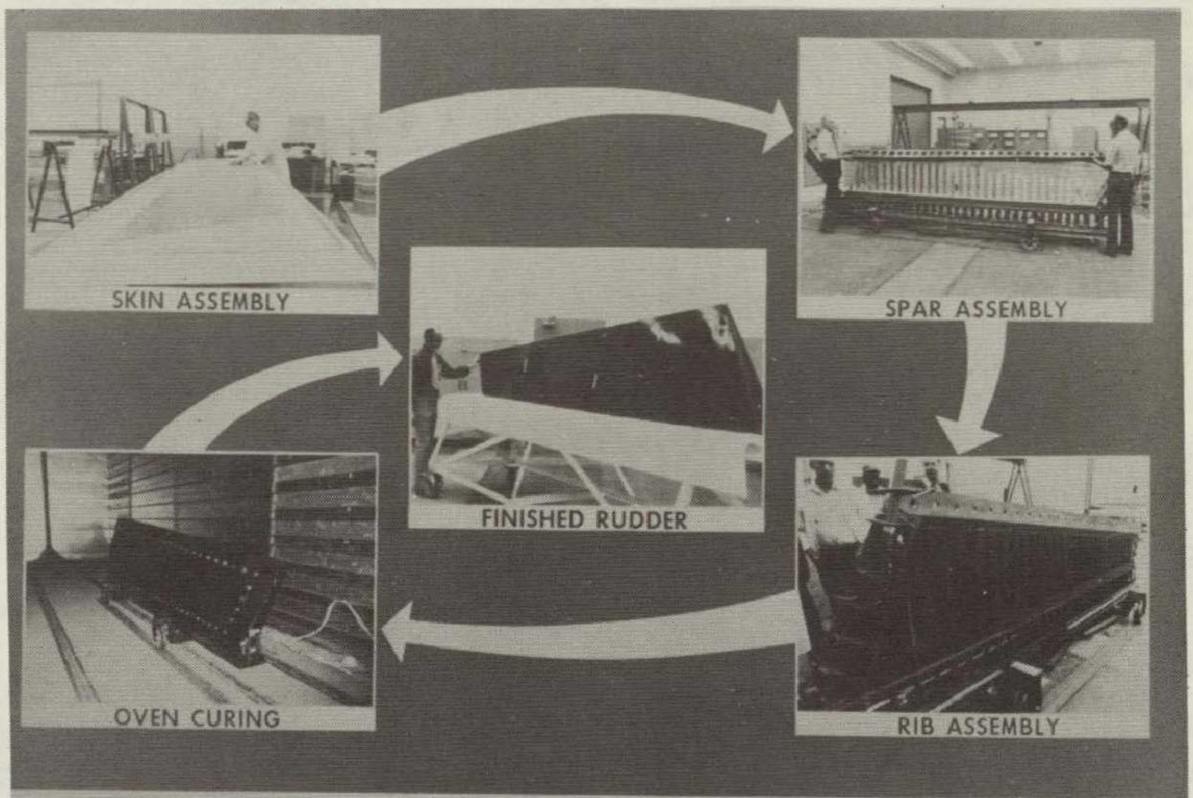
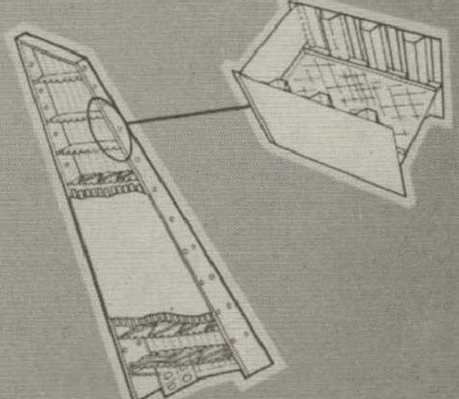


FIGURE 37. – DC–10 UPPER AFT RUDDER MANUFACTURING SEQUENCE

FLIGHT SERVICE DATA

AIRCRAFT	COMPONENTS	AIRLINE
1	1	
1	1	

TO BE INITIATED JANUARY 1979



COMPONENT DATA

9x25 ft, 150 sq ft
GRAPHITE/KEVLAR/EPOXY
640 lb 83% COMPOSITE
25% WEIGHT SAVED
1 PER AIRCRAFT

FIGURE 38. – L–1011 VERTICAL FIN

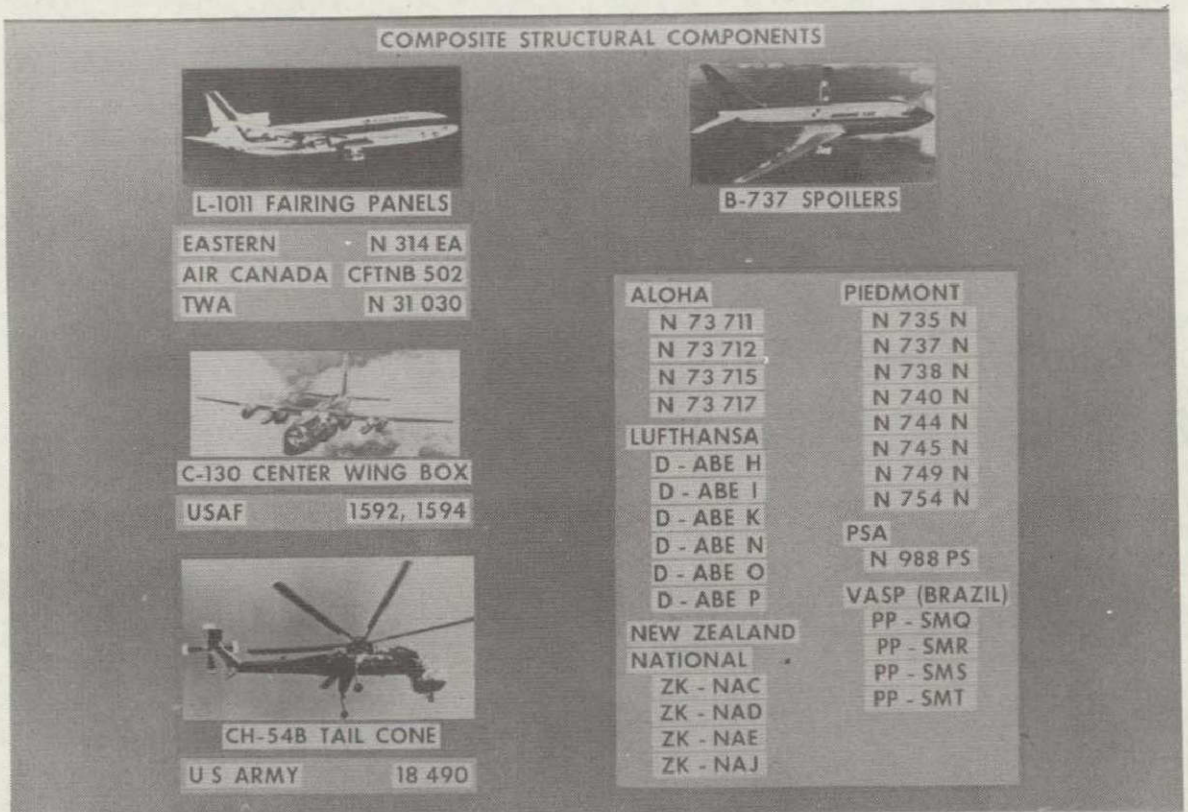


FIGURE 39. – AIRCRAFT IN NASA FLIGHT SERVICE PROGRAM

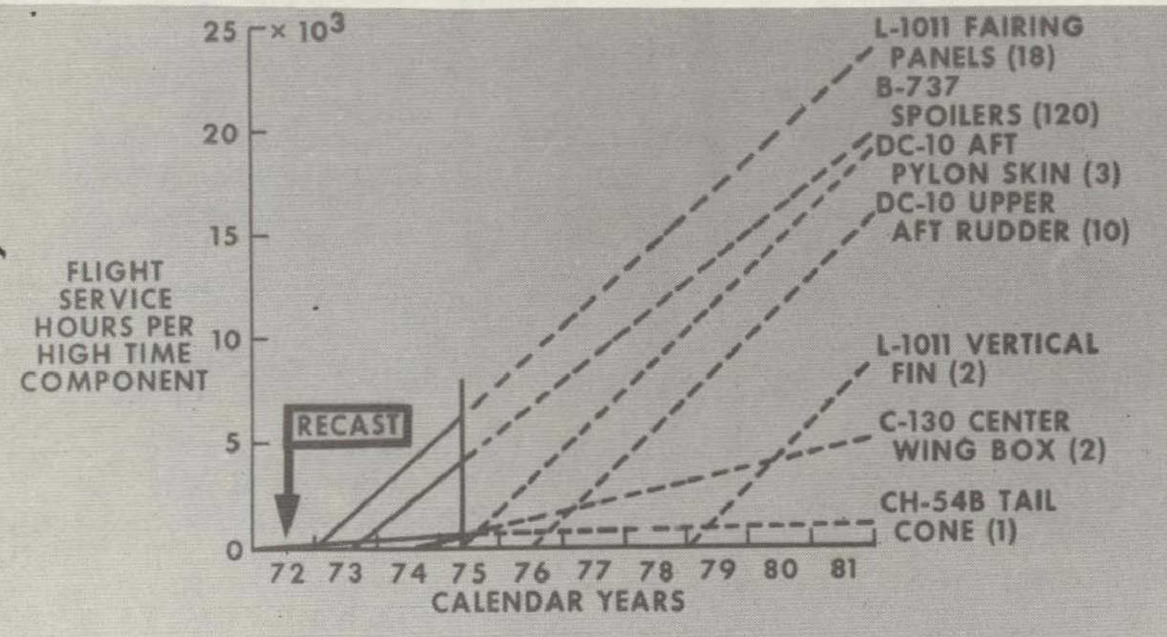


FIGURE 40. – NASA CONFIDENCE BUILDING, PAST AND PLANNED

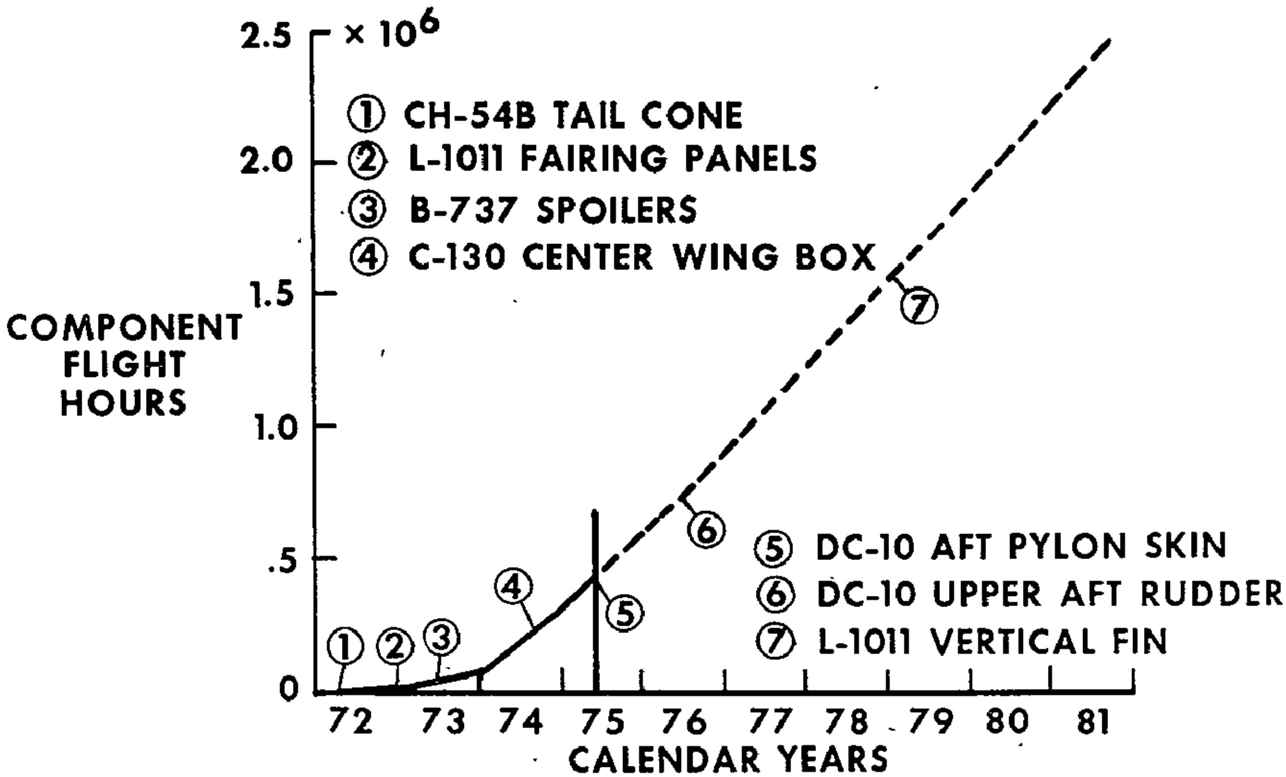


FIGURE 41. – CUMULATIVE FLIGHT SERVICE HOURS, ALL NASA COMPONENTS

NASA-Langley, 1975