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Paper MP-69-P-17-Starnes

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ABSTRACT

A summary of recent composite structures and materials research at NASA Langley Research Center is presented. Fabrication research to develop low-cost automated robotic fabrication procedures for thermosetting and thermoplastic composite materials, and low-cost liquid molding processes for preformed textile materials is described. Robotic fabrication procedures discussed include ply-by-ply, cure-on-the-fly heated placement head and out-of-autoclave electron-beam cure methods for tow and tape thermosetting and thermoplastic materials. Liquid molding fabrication processes described include Resin Film Infusion (RFI), Resin Transfer Molding (RTM) and Vacuum-Assisted Resin Transfer Molding (VARTM). Results for a full-scale composite wing box are summarized to identify the performance of materials and structures fabricated with these low-cost fabrication methods.

KEYWORDS: automation, tape, robot, tow placement, liquid molding

INTRODUCTION

The NASA Langley Research Center (NASA Langley) has been actively involved in the research and development of composite materials and structures for the past 30 years. In the 1970's, the research focus at NASA Langley was on hand lay-up fabrication processes, structural performance and flight demonstrations of secondary composite structures for transport aircraft. In the 1980's, the research focus changed to damage-tolerant design concepts, toughened-epoxy and thermoplastic resin development, advanced tape placement machines, and the further development of secondary composite structures for transport aircraft. In the 1990's, the research focus changed to cost-effective and damage-tolerant primary composite structures for transport aircraft. This more recent change in research focus has led to the development of automated fiber-placement machines, damage-tolerant textile material forms and liquid molding processes, such as the resin transfer molding (RTM), resin film infusion (RFI), and vacuum-assisted resin transfer molding (VARTM) processes. In addition, high-speed automated and robotic material-placement processes and low-cost out-of-autoclave tooling and processing concepts are being explored to address future economic challenges. Structural analysis and design methods are also being developed that reliably predict the response and failure characteristics of the composite structures fabricated by these advanced low-cost fabrication processes.

Liquid molding processes are being studied at NASA Langley because these processes offer the opportunity to use resins and fibers in their lowest-cost state by eliminating the pre-impregnation (prepreg) step in the fabrication process and by minimizing material scrap. Liquid molding processes have been used extensively in the boat building industry (Ref. 1), but until recently these processes have been highly labor intensive. The development of near-net-shape damage-tolerant textile preforms during

the last decade, and the development of innovative resin transfer tooling concepts, has led to an interest in textile-reinforced composite structures for transport aircraft applications. NASA Langley has evaluated several textile material forms including those made by weaving, tri-axial braiding, knitting, and stitching procedures. The use of through-the-thickness stitching of graphite preforms and the RFI process (stitched/RFI) were found to provide cost-effective increases in structural damage tolerance. This process was selected in 1995 to fabricate a12.8-m-long full-scale composite wing box that was recently tested at NASA Langley (Refs. 2 and 3).

Automated robotic placement processes for tow, ribbon, and tape forms of composite materials are being studied and developed at NASA Langley because these processes have emerged as promising low-cost fabrication processes for high-performance fiber-reinforced composite structures. Production-ready computer-controlled equipment has been used by industry to manufacture major portions of the Boeing 777 empennage, the F/A-18E/F stabilator and inlet ducts, and several V22 parts, among others. NASA Langley is studying the use of these cost-effective fiber-placement processes by using representative small-scale, experimental equipment that simulates the expected performance of larger manufacturing facilities. Such equipment is used to study, screen, and develop composite fabrication processes using new resins, new fibers, new intermediate materials forms, new in-situ curing mechanisms, and net-shape material placement procedures. A research laboratory has been developed at NASA Langley to study automated fabrication processes and to provide a means to address some of the research issues associated with these processes. The current research emphasis on heated-head automated tape placement (ATP) of thermoplastic prepreg materials, and on e-beam cure-on-the-fly ATP of epoxy prepreg materials (Refs. 4 and 5). This prototype research laboratory is used to study and perfect non-autoclave and in-situ ATP fabrication processes that have the potential of being scaled up to fabricate full-scale structures. Once these processes have been perfected, these processes are transferred to larger ATP equipment for the fabrication of full-scale structures. It is expected that these fabrication processes could be used for the successful in-situ fabrication of large, high-quality composite structures, such as cryogenic fuel tanks with diameters on the order of nine meters and lengths over 18 meters. Autoclave processing of such structures would be prohibitively expensive due to the need for appropriately sized autoclaves and related fabrication tools.

The present paper summarizes the research being conducted at NASA Langley to advance the state-of-the-art of low-cost fabrication processes for future composite structures. An overview of the research being conducted for the RFI, RTM and VARTM liquid molding processes and for the automated in-situ ATP and e-beam-cure processes will be discussed. An overview of the fabrication and testing of the 12.8-m-long stitched/RFI composite wing box will also be discussed.

TEXTILE PREFORMS AND FABRICATION TECHNOLOGY

Fabrication of Textile Preforms

High-quality fiber preforms are required to produce high-quality composite parts using resin transfer molding (RTM) processes. Various types of textile material forms have been used to fabricate near-net-shape preforms for resin transfer molding of composite aircraft structures. Multi-axial warp knitting is a highly tailorable automated process that produces multi-directional broadgoods for large area coverage. Two-dimensional and three-dimensional braids are used to create stiffeners, frames and beams with complex cross-sections. Through-the-thickness stitching is an effective way to

debulk preforms and to achieve improved out-of-plane strength and damage tolerance for composite structures.

Major advancements have been made in through-the-thickness stitching technology during the past 25 years. Early NASA Langley stitching studies used a limited-capacity, extended-arm, single-needle stitching machine to produce preforms for evaluating the damage tolerance of composite panels. The next-generation stitching machine was computer controlled and took advantage of quilting technology. This machine was limited in size and stitching speed, so a high-speed, multiple-head stitching machine was developed by Ingersoll Milling Machine Company and Pathe Technologies, Inc., to provide a stitching machine that is capable of stitching large wing skins for commercial transport aircraft. This second-generation stitching machine is shown in Figure 1. This machine has four stitching heads that can stitch a 3.0-m by 15.2-m by 38.1-mm preform at 800 stitches per minute. The stringer stiffened preform shown in Figure 1 was used to produce a wing cover panel for the NASA Advanced Subsonic Technology Composite Wing Program. The fabrication and structural evaluation of the resulting 12.8-m-long wing box is discussed subsequently in the present paper.

Resin Film Infusion (RFI)

The RFI process is being studied to develop cost-effective wing structures for commercial transport aircraft. The RFI fabrication process consists of an outer-mold-line tool, an epoxy resin film, a near-net-shape textile preform, an inner-mold-line tool, and a reusable vacuum bag. Thick film plates of resin (called "resin slabs") are placed on the outer-mold-line tool, and the preform and inner-mold-line tool are placed on top of the resin. The entire assembly is covered with a reusable vacuum bag (Fig. 2), and the part is placed inside of an autoclave. After the resin has been melted, vacuum pressure is used to infuse the resin into the preform. Once the resin has been infused into the preform, the part is cured under pressure and temperature in the autoclave. A completed wing cover panel is shown in Figure 3. The cured composite wing panel shown in the figure has no mechanical fasteners since all of the stiffener elements are stitched to the skin.

To produce aircraft quality parts with the RFI process, it is necessary to understand the compaction and permeability characteristics of the preform, the kinetics and viscosity profiles for the resin as a function of temperature, tool fit-up, sealing to prevent resin leakage, and the resin flow paths and infusion time. The stiffener elements and tooling blocks must be precisely located to achieve the required dimensional tolerances. The advanced stitching machine was used to locate structural details such as ply drops, stiffeners, interleaved spar caps, and rib clips.

Analytical models (Ref. 6) are used to predict the resin flow into textile preforms to eliminate trial and error process development. The models have been verified through precise experiments, and the three-dimensional models are used to represent the resin flow response for complex preforms, such as wing cover panels fabricated from stitched and knitted fabric skins and stitched and braided stiffeners. The analytical models predict the resin flow front position, resin viscosity, and degree of resin cure as a function of temperature and time and include resin flow, heat transfer, and thermo-chemical effects. These models predicted the temperature distribution for a two-stiffener stitched panel within 6 percent of the measured temperature and the predicted resin wet-out times were within 4 to 12 percent of the measured times.

3

Resin Transfer Molding (RTM)

The RTM process is a process that requires matched metal fabrication tools and precision preforms to fabricate composite structures with a high fiber volume fraction. The RTM process is well suited for fabricating complex shaped parts, such as curved beams or frames. This process has been used to fabricate braided frames for a curved fuselage keel structure. The process has also been used to fabricate braided and woven window frames for curved fuselage side panels. Because of the pressure requirements and tooling costs associated with this process, it is usually limited to relatively small parts that are difficult to fabricate by other composite fabrication processes.

Vacuum-Assisted Resin Transfer Molding (VARTM)

The VARTM process has been used for many years to fabricate fiberglass reinforced composite structures. The U.S. Naval Surface Warfare Center has been the major promoter of this technology for composite marine applications (Ref. 1). The major advantages of a VARTM process, compared to conventional autoclave processes, are lower tooling cost, reduced energy cost for curing composite parts, and an almost unlimited part size (i.e., no size constraints based on autoclave size). Until recently, the VARTM process was primarily used to fabricate glass reinforced polyester and vinyl ester composite structures. Because of recent developments in resin and preform technologies, aircraft manufacturers are beginning to be interested in the VARTM process for graphite-epoxy and graphite-bismaleimide composite material systems. One limitation of the VARTM process has been the low fiber volume fraction of parts fabricated by the process when compared to parts with higher fiber volume fractions when an autoclave process is used. However, stitching and debulking methods have been developed that provide near- net-shape preforms with little or no further compaction required during processing.

NASA Langley has been working with Seemann Composites, Inc. to establish the feasibility of using another VARTM process to produce aircraft quality composite structures. A proprietary process, known as SCRIMP™ (Seemann Composites Resin Infusion Molding Process) uses a resin distribution medium to achieve full wet-out of the preform. Seemann Composites, Inc. has developed a reusable bagging concept that eliminates most of the costs associated with conventional bagging procedures, and has demonstrated SCRIMP™ for lightly-loaded general aviation aircraft structures. A one-sided tooling concept and a graphite preform was used to fabricate a fuselage panel (Fig. 4) and a full-scale fuselage for a small aircraft using the SCRIMP™ process. NASA Langley is also investigating the feasibility of using the SCRIMP™ process to produce high-quality heavily-loaded primary composite structures for aircraft applications.

AUTOMATED TOW AND TAPE PLACEMENT TECHNOLOGY

Heated Head Automated Tape Placement

The NASA Langley prototype research laboratory for automated tow and tape placement has a robotic work cell that is configured for automated fiber placement. The work cell (Ref. 7) has a 6-axis Asea Brown Boveri (ABB) robotic arm with a modified Automated Dynamics Corporation (ADC) fiber placement head (Fig. 5) and supporting software that was developed by ADC and Composite Machinery Company. The initial research in this laboratory (Ref. 8) concentrated on the development of fiber placement heads, which were essentially stand-alone end effectors (Fig. 6) that feed, heat, cut, place and laminate unidirectional fibrous material.

4

The ABB robotic arm can support a payload of up to 150-kg force and has a maximum reach of 2.4 meters. The modified ADC head is capable of placing five 0.635-cm-wide ribbons or one 3.18-cm-wide tape of either a thermoplastic or a thermoset material, and it can be modified to place narrower tow material. Several heating methods are used to heat the tape as it is being placed, including two conventional nitrogen gas torches and a recently developed focused infrared lamp. A steel compaction roller is used to apply pressure to the heated tape. A heated flat tool and a cylindrical tools mounted on an ADC spindle satisfy most of the research requirements for the laboratory (Ref. 8).

The focused infrared lamp was developed first to augment and later to replace the nitrogen gas torches which had serious problems with reduced heat transfer in the nip region due to hot gas flow stagnation. The radiant heat source permitted operation at lower compaction roller temperatures that reduced resin adherence to the rollers and improved part surface smoothness. Photomicrographs of thermoplastic panels placed with the infrared heater indicated low void content and good ply-ply interfacial adhesion as demonstrated by wedge peel and double cantilever beam (DCB) tests which are discussed subsequently. The DCB initiation fracture toughness numbers were comparable to those reported for autoclave processed panels (Ref. 9).

Placement Quality Tape

The technology for fabricating a fully consolidated, dry unidirectional composite tape was developed at NASA Langley and at several industrial locations. The NASA Langley process involves the preparation of powder-coated tow material (towpreg) using a gravity-fed powder curtain process. The towpreg is passed through a tube furnace and onto air-cooled nip rollers to convert it to a dry, fully wet-out unidirectional tape (Ref. 10). A recent modification of this process starts with "wet" towpreg or tape prepared by coating fiber with an N-methylpyrrolidone solution of the polymer and drying it to 12 percent volatiles. The remaining volatiles (solvent and reaction products, if any) are removed in the tube furnace (Ref. 11). Specifications for dry ribbon and tape are given in Reference 12. Tight equipment tolerances and proper resin melt flow as well as many other factors are required to achieve tape with these specifications. The most important characteristic of a good quality dry material form is its ability to be placed robotically in a rapid manner ply-by-ply and to achieve consolidation with the previously laid ply. During the lay-down process, resin melt flow must be adequate to achieve intimate contact, reptation bonding and healing with the previously laid ply. To accomplish this requirement, placement is conducted at temperatures that correspond to a minimum in the melt viscosity, yet are below polymer thermal decomposition.

The standard method for measuring the interlaminar bond quality of a part is the DCB test. This test requires a specimen thickness that is between 3-5 millimeters or 24-40 plies, and specimen preparation and testing is time consuming. A rapid screening test, the wedge peel test, was developed which required a 2-ply-thick specimen that could be tested immediately (Ref. 13). Correlation between the two tests was made so that the wedge peel test could be used to screen candidate ribbons and tapes for in-situ processability, as well as to help identify process windows and conditions for efficient tow placement. A study was conducted in which PIXA thermoplastic polyimide ribbon was prepared under conditions that yielded material with varying degrees of processability (Ref. 12). Laminates were fabricated at the lower and upper temperature extremes of the placement processing window using the NASA Langley automated tow placement equipment, and DCB and wedge peel tests were used to determine the quality of the laminates and especially the interlaminar bond formed during the placement process. Ribbon made under conditions expected to be non-optimal (overheated) resulted

5

in poor placeability and composites with weak interlaminar bond strengths, regardless of placement conditions. Ribbon made under conditions expected to be ideal had good processability and produced well-consolidated laminates. The results demonstrated the importance of ribbon quality in heated-head placement of dry material forms.

In-situ consolidated flat panels have been prepared from high temperature polyimides such as AURUM™ PIXA/IM7, AURUM™ PIXA-M/IM7 and LARC™ PETI-5/IM7 and polyarylene ethers and sulfides such as APC-2™ (PEEK)/AS4, APC-2™ (PEEK)/IM6, PEKK/AS4 and PPS/AS4 (Refs. 6, 7, and 14). Lightly cross-linked materials, such as the LARC™ PETI-5/IM7 polyimide, required a high temperature post-cure to optimize performance. Open hole compression strengths at room temperature of some 24-ply PEEK, PIXA, and PETI-5 quasi-isotropic panels made by the ATP process ranged from 85 to 104 percent of the properties of composite panels made by a hand lay-up process and an autoclave cure. These results indicate that heated head ATP technology can be used effectively to fabricate quality, high-performance composite parts. Other flat panels, including 2-m by 3-m panels with three stringers and 2-m by 3-m sandwich panels with titanium core, have been fabricated from various polyimides using ATP equipment and 7.62-cm-wide tape to develop heated head robotic placement technology for aerospace applications.

Fiber placement with the focused-infrared heated head was used to fabricate high-quality, well-consolidated 8-ply quasi-isotropic cylindrical shells with a (+45/-45/0/90/90/0/-45/+45) laminate. These 61-cm-diameter, 9-cm-long shells were made with APC-2™ (PEEK)/AS4 material. The consolidated tape was supplied by Cytec Fiberite with a thickness and width of 0.015 and 3.17 cm., respectively. The tape had an average resin content of 32 percent by weight and an average fiber areal weight of 145 g/m². Acid digestion showed the tape to have a void content of 8.5 percent. Processing conditions were determined by varying the roller temperature, compaction load, placement speed and IR lamp output during fabrication of wedge peel specimens. Contoured trailing shoes were adapted to follow the compaction roller to reduce ply wrinkling, provide a smoother surface to the composite during fabrication and minimize roller fouling due to matrix and fiber adhesion. Placement processing files were altered as needed to account for tape width increase due to roller temperature and compaction force. A photograph of one of the shells is shown in Fig. 7. Roller sticking, which was a primary problem in earlier work with the gas torches, was eliminated because the roller could be operated at lower temperatures. Steel compaction rollers were machined to match the curvature of the shells for placement of the 0°, 45° and 90° plies. Photomicrographs of the shell-wall cross-sections indicated that very good interlaminar bonding had occurred. However, numerous dry, voidy and poorly wet-out intraply areas present in the as-purchased tape were also observed and could not be healed during the placement process. This observation confirms the need to prepare good quality starting material and not to depend on the lay-down process to correct material form imperfections.

E-Beam Cure-On-the-Fly Automated Tape Placement

NASA Langley has contracted with Boeing to design and build an electron-beam (e-beam) cure-on-the-fly (COTF) automated tape placement machine for materials and process development (Ref. 15). During the tape laying process, an electron beam gun initiates reaction of the matrix resin causing the cure of the prepreg in a layer-by-layer manner. This technique allows the fabrication of large structures without the large capital and tooling expenditures inherent in autoclave curing. It also provides significant consolidation pressure during curing which is not available during standard electron beam curing of composite materials. The machine is capable of automatically laying 7.6-

cm-wide composite prepreg for the fabrication of flat laminates up to 91-cm long by 91-cm wide with any combination of angle plies. The placement head was built by Applied Poleramic, Inc., and the electron beam gun was built by Electron Solutions, Inc. This new e-beam system is based on a two-axis, lower gantry motion design such that the e-beam head travels solely in one in-plane direction. Translation in the in-plane direction and rotation around the normal to the plane are achieved by motion of the flat laminate tool. This feature allows for the fabrication of large (91-cm by 91-cm) flat laminates, and for the expansion potential of the entire device. The two-axis, lower gantry motion design enables the fabrication of larger panels due to the functionality of the translation and rotation system on which the flat laminate tool is mounted.

The tape placement head is custom designed and engineered specifically for the e-beam cure-on-the-fly process. The design allows for compaction of the prepreg tape by compliant silicone compaction rollers. The compaction rollers make contact with the prepreg tape by means of a pressure piston with a variable force control system. In addition, the compaction rollers provide the flexibility to ensure adequate compaction force for laminates that are not perfectly flat. This feature is made possible by a hinge where the rollers are attached to the head. This design provides the necessary compaction prior to in-situ electron beam cure. An IR heating system has also been integrated into the ATP device, as well as an electronic control system. The electron-beam COTF process has been successfully demonstrated by fabricating an approximately 30-cm by 30-cm quasi-isotropic epoxy panel. The tape was placed and electron-beam cured as an in-situ process. The process was continuous from start to finish, and no operator intervention was required. Although the panel was of poor quality, it did demonstrate that the new equipment is capable of laying and curing composite tape in a simultaneous process.

EVALUATION OF STRUCTURAL COMPONENTS

Stitched/RFI Semi-Span Wing Box

A series of structural tests were conducted on the 12.8-m-long stitched/RFI semi-span wing box to demonstrate that the stitched/RFI fabrication process could produce aircraft quality structures. The wing box was attached to a strongback at NASA Langley and was subjected to the following tests: Brake-Roll at design limit load (DLL); -1.0-g down-bending at DLL; 2.5-g up-bending at DLL; discrete-source-damage in the upper and lower cover panels at 70 percent of 2.5-g up-bending DLL, and final failure in the 2.5-g up-bending condition. Prior to final failure, the wing box upper and lower cover panels were subjected to low-velocity impact damage to simulate foreign object damage. The wing box failed at 97 percent of design ultimate load (DUL). Failure initiated from a lower cover access hole and propagated through the lower cover into both spars. A photograph of the wing box just prior to failure is shown in Figure 8. Prior to the semi-span ultimate load test, analyses were conducted to determine the maximum wing tip displacement. The results of these analyses were within 5% of the 1.3-m maximum wing tip displacement measured during the test.

CONCLUDING REMARKS

Major advancements have occurred in the automated fabrication of composite structures made from textile preforms. Automated processes that produce multi-directional broadgoods for multi-axial knitting coupled with a second-generation stitching machine with four stitching heads have advanced the resin-infusion and resin-transfer-molding technologies to a high level of accomplishment. Wing cover panels

have been successfully fabricated from integrally woven preforms using an outer-mold-line tool, epoxy resin film, near-net-shape textile preforms, an inner-mold-line tool, and a reusable vacuum bag and the resin film infusion process. Low-cost vacuum-assisted resin transfer molding processes are now being applied to the fabrication of high-quality, heavily-loaded primary composite structures for transport aircraft applications. Structural tests of a full-scale composite wing box have demonstrated the effectiveness of these low-cost fabrication processes.

Significant progress has been made in developing automated heated-head tow and tape placement technology for the fabrication of high-performance composite structures. Methods have been developed for making quality thermoplastic ribbons and tape, machine design and operating requirements for in-situ placement have been determined, and a basic knowledge of the fundamental mechanisms involved in both ribbon and tape preparation and in-situ consolidation have been established.

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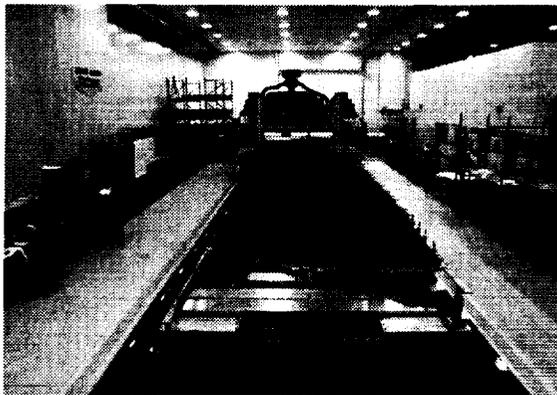
Figure 1. Advanced stitching machine.

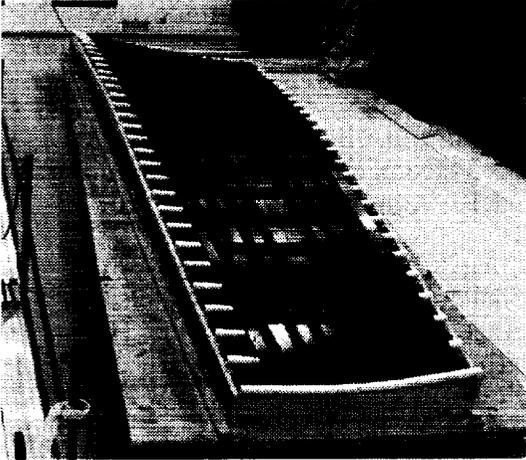Figure 2. Reusable vacuum bag for wing cover panel.

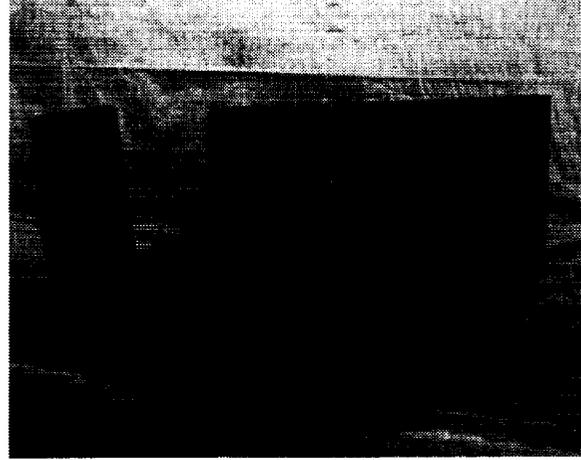Figure 3. Stitched/RFI composite wing cover panel.

Figure 4. Graphite-epoxy fuselage panel fabricated with the SCRIMP™ process
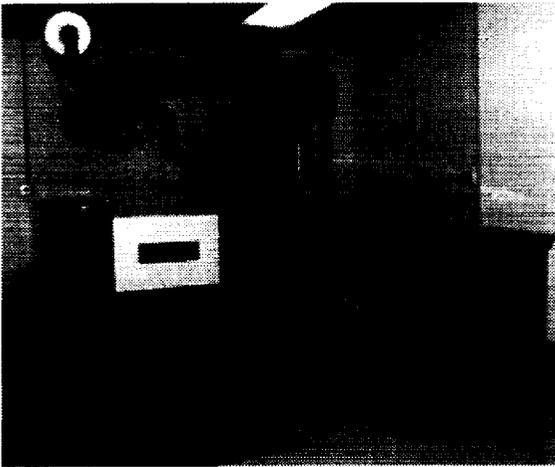

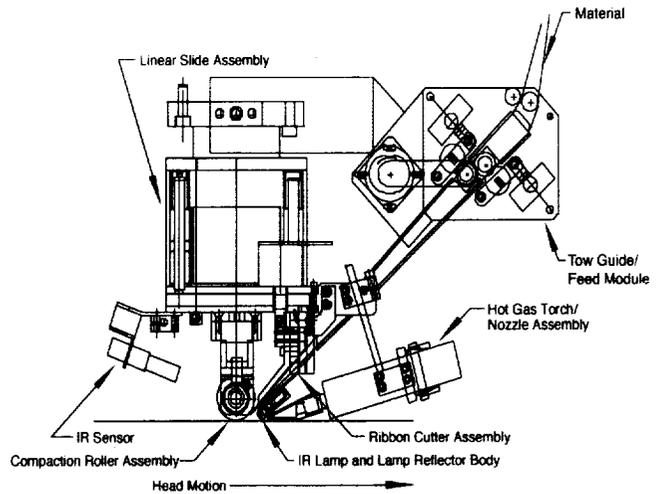Figure 5. NASA Langley robotic composite fabrication facility.
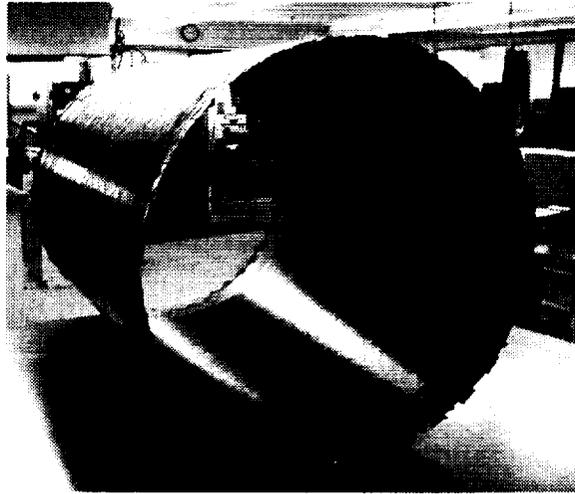

Figure 6. Schematic of the NASA heated head.

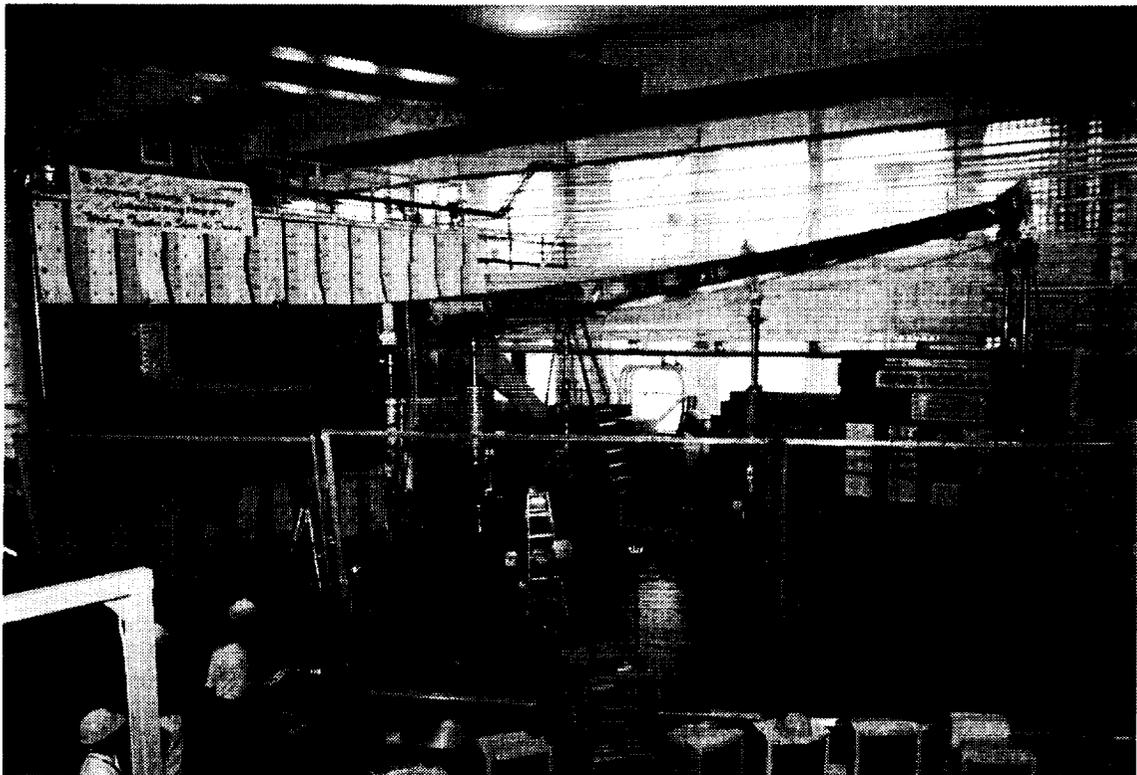Figure 7. Heated-head tape-placed composite shell

Figure 8. Stitched/RFI semi-span composite wing box just prior to failure.