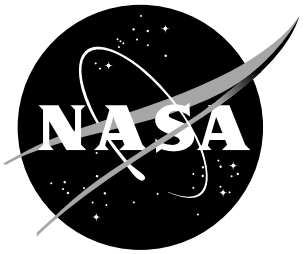
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An Assessment of the State-of-the-Art in the Design and Manufacturing of Large Composite Structures for Aerospace Vehicles

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April 2001

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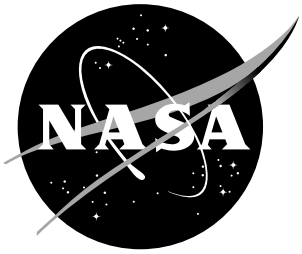
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Abstract

The results of an assessment of the state-of-the-art in the design and manufacturing of large composite structures are described. The focus of the assessment is on the use of polymeric matrix composite materials for large airframe structural components, such as those in commercial and military aircraft and space transportation vehicles. Applications of composite materials for large commercial transport aircraft, general aviation aircraft, rotorcraft, military aircraft, and unmanned rocket launch vehicles are reviewed. The results of the assessment of the state-of-the-art include a summary of lessons learned, examples of current practice, and an assessment of advanced technologies under development. The results of the assessment conclude with an evaluation of the future technology challenges associated with applications of composite materials to the primary structures of commercial transport aircraft and advanced space transportation vehicles.

Key Words

composite materials, graphite/epoxy, graphite, carbon, epoxy, thermosets, thermoplastics, aircraft, rotorcraft, general aviation, rockets, wing, fuselage, empennage, NDE, building-block approach, analysis.

Introduction

An assessment of the design and manufacturing practices for large composite structures has been conducted to determine the current state-of-the-art for these technologies. The background that motivated the assessment was a series of unexpected manufacturing and design problems with the composite structures of several NASA experimental vehicles currently under development. The focus of the assessment is on the use of polymeric matrix composite materials for large airframe structural components such as those in commercial and military aircraft and space transportation vehicles. The baseline for the assessment is the historical evolution of the use of composite materials in actual aerospace vehicles. The assessment emphasizes the application of composite structures in moderately to heavily loaded aerospace vehicles. Applications of composite materials are reviewed for large commercial transport aircraft, general aviation aircraft, rotorcraft, military fighter aircraft, and military transport aircraft. The baseline also includes the application of composite materials for unmanned rockets and space transportation vehicles. The assessment of the state-of-the-art includes a summary of lessons learned, examples of current practice, and an assessment of advanced technologies under development. The assessment concludes with an evaluation of the future technology challenges associated with applications of composite materials to the primary structure of commercial aircraft and advanced space transportation vehicles.

As a preamble to assessing the state-of-the-art in the design and manufacturing of composite structures, the design requirements for aerospace vehicles are briefly reviewed. Because of the universal design requirement to minimize the gross take-off weight of all aerospace vehicles, aerospace structural components are designed at or near zero margin of safety. While the margin of safety is not equal to zero for all the design criteria at each structural location, there is typically one criterion for each structural element that governs the design details of that element. The quest for the lowest weight structure then drives the design margin to nearly zero for the design limit load condition. (The Code of Federal Regulations [1] for Aeronautics and Space, Title 14, specifies that the structure shall undergo no permanent deformation at the design limit load (DLL). In addition, the structure shall sustain the design ultimate load (DUL) for at least 3 seconds before failing. The factor of safety between DLL and DUL is 1.5. Since most structural materials exhibit plasticity for metallic materials or microcracking for composite materials prior to structural failure, the factor of safety is mostly the difference between repeatable, linear, elastic behavior and structural failure. In other words, the 1.5 factor of safety will not provide a positive margin against unanticipated permanent deformation or damage to the structure. Therefore, aerospace structural designs will not accommodate any deleterious structural behavior.

In some cases, names of products commonly used in the public domain may be used herein. Any use of these company trademarks, trade names, or product names does not indicate a NASA endorsement of those products.

Part I. Historical Development of Structural Composite Materials

Large Transport Commercial Aircraft

The first composite components on commercial transport aircraft were designed and built as part of the NASA Aircraft Energy Efficiency (ACEE) Program and entered into flight service during 1972-1986 [2]. The primary objectives of the ACEE Program were to obtain actual flight experiences with composite components and to compare the long-term durability of flight components to data obtained from an environmental-exposure ground-test program. Boeing Commercial Airplane Company, Douglas Aircraft Company, and Lockheed Corporation agreed to participate in the program. A common feature of all three programs was the use of the Narmco T300/5208 graphite/epoxy material system. The T300 fiber is an intermediate modulus and intermediate strain-to-failure graphite fiber and the 5208 matrix is a thermoset epoxy that cures at 350 °F (177 °C) . In the early years of the ACEE Program, smaller components of lightly loaded secondary structure were designed and entered into service. These components included the Lockheed L-1011 fairing panels, the Boeing B-737 spoiler, the Douglas DC-10 aft pylon skin and the Douglas DC-10 upper aft rudder. In the later years of the program, larger, more heavily loaded control surfaces and empennage structures were designed and entered into service. Some examples of these components included the Boeing B-727 elevator, Figure 1a, the Boeing B-737 horizontal stabilizer, Figure 1b, the Douglas DC-10 vertical stabilizer, Figure 1c, and the Lockheed L-1011 aileron, Figure 1d. A Lockheed L-1011 vertical stabilizer was also developed during the ACEE Program. All three of the major flight components had premature failures before they were re-designed and successfully tested. These premature failures were related to an incomplete understanding of the differences in the failure characteristics of metallic and composite structures at the time that these structures were designed. By January 1987, 350 composite components had entered into commercial airline flight service.

As of 1993, the 350 components originally placed in service had accumulated over 5.3 million flight hours. The service performance, maintenance characteristics, and residual strength of numerous components were reported to NASA and compared to the data obtained from the 10-year, environmental-exposure ground-test program [3]. The data indicated an excellent in-service performance of the composite components during the 15-year evaluation period. The airlines reported damage such as ground handling accidents, foreign object impact damage, and lightning strikes. However, there was no degradation of the residual strength of the composite components due to fatigue or in-service environmental exposure. Furthermore, there was good correlation between the results of the ground-test program and the structural performance of the actual aircraft components.

A comparison of the applications of composite materials as a percentage of structural weight for large commercial transport aircraft is given in Figure 2. These data were obtained from several issues of *Jane's All The World's Aircraft* [4]. The plotted data show an increasing use of composite materials over the past three decades from lightly loaded secondary structure, to control surfaces, to more heavily loaded primary structure in the empennage of the Airbus aircraft (denoted by A3XX in the figure) and the Boeing B-777. The applications of composite materials in these aircraft are described in more detail in the next paragraphs. The current barriers to significant increases in the use of composite structures in primary structure are the

higher cost of composite structures relative to conventional aluminum structures, and the unreliability in the estimates of the design and development costs of composite structures.

Airbus was the first manufacturer to make extensive use of composite structures [4] on large transport commercial aircraft, see Figure 2. The Airbus A310 was the first production aircraft to have a composite fin torque box. Composite components on the A310 include the wing leading-edge lower access panels and outer deflector doors, nose wheel doors, main landing gear fairing doors, engine cowling panels, elevators and fin torque box, fin leading and trailing edges, flap track fairings, flap access doors, rear and forward wing/body fairings, pylon fairings, nose radome, cooling air inlet fairings and tail leading edges, wing leading-edge top panels, panel aft rear spar, upper surface skin panels above the main wheel bay, glide slope antenna cover, and rudder. The A320 was the first aircraft to go into production with an all-composite empennage. Also, about 13% of the weight of the wing on the A340 is composite materials. The fabrication responsibilities of the Airbus Consortium partners are as follows: Aerospatiale fabricates the cockpit, engine pylons and part of center-fuselage; British Aerospace fabricates the wings; Daimler-Benz Aerospace Airbus fabricates the most of the fuselage, fin, and interior; and CASA fabricates the empennage.

The Boeing B-777 makes extensive use of composite materials for primary structure in the empennage, most control surfaces, engine cowlings, and the fuselage floor beams. These components are shown schematically in Figure 3. About 10% of the structural weight is composite materials [4]. As the schematic shows, several different composite material systems were used. Graphite/epoxy composite materials were used for most secondary structure and control surfaces. A toughened epoxy material system, Toray T800H/3900-2, was used for the larger, more heavily loaded components including the vertical fin torque box and horizontal stabilizer torque box components of the empennage.

Rotorcraft and General Aviation Aircraft

Rotorcraft and general aviation aircraft have made extensive use of composite materials to achieve performance goals. The applications of composite materials as a percent of structural weight are plotted in Figure 4 for selected rotorcraft and general aviation aircraft to contrast the higher percent of composite materials in these aircraft relative to the large transport aircraft [4]. The V-22 tiltrotor aircraft designed by Bell and Boeing has a number of significant applications of composite materials. Bell and Boeing used an integrated product team approach to designing the V-22 airframe [4]. The approach is credited with saving about 13% of the structural weight, reducing costs by 22%, and reducing part count by about 35%. Approximately 41% of the airframe of the V-22, shown in Figure 5, is composite materials. The wing is IM-6/3501-6 graphite/epoxy material and the fuselage and tail are AS4/3501-6 graphite/epoxy material. The nacelle cowlings and pylon supports are graphite/epoxy material. The main cabin has composite floor panels and the crew seats are boron carbide/polyethylene material. The fuselage is a hybrid structure with mainly aluminum frames and composite skins. The wing box is a high-strength, high-stiffness torque box made from one-piece upper and lower skins with molded ribs and bonded stringers, two-segment graphite/epoxy single-slotted flaperons with titanium fittings, and a three-segment detachable leading-edge made of an aluminum alloy with Nomex honeycomb

core. The rotor also used significant amounts of graphite/epoxy (17%) and glass/epoxy (20%) composite materials.

Military Aircraft

Military aircraft have been designed with significant applications of composite materials in primary structure. While not all information on military aircraft is publicly available, the data in Figure 6, obtained from reference 4, compare the application of composite materials as a percent of structural weight for a number of fighter aircraft. For example, the Lockheed Martin F-22 Raptor, shown in Figure 7, is approximately 39% titanium, 16% aluminum, 6% steel, 24% thermoset composite materials, 1% thermoplastic composite materials, and 14% other material systems [4]. The fuselage is a combination of titanium, aluminum, and composite materials. The wing skins are made of monolithic graphite/bismaleimide materials. A view of the wings being assembled is shown in Figure 8. The wing front spars are made of titanium and the intermediate spars are made of a graphite/epoxy material. The horizontal stabilizer uses graphite/bismaleimide skins with an aluminum honeycomb core. The vertical stabilizers use graphite/bismaleimide skins over graphite/epoxy spars. The wing control surfaces are a combination of co-cured composite skins and non-metallic honeycomb core.

The Northrop Grumman B-2, shown in Figure 9, is constructed of almost all composite materials [4]. Development of the B-2 began in the late 1970's. The first flight test of the B-2 was July 17, 1989. The wing is almost as large as the Boeing B-747 with a span of 172 ft (52.4 m) and surface area of 5,140 ft^2 (477 m^2).The wing is mostly graphite/epoxy material with honeycomb skins and internal structure. The fuselage also makes extensive use of composite materials. The outer skin is constructed of materials and coatings that are designed to reduce radar reflection and heat radiation. Boeing Military Airplanes produced the wings and aft section of the fuselage. Northrop Grumman produced the forward center-sections including the cockpit. Boeing completed the outboard wing section of the twenty-first and final aircraft on May 3, 1994.

The original design of the McDonnell Douglas (now Boeing) C-17, shown in Figure 10, uses about 8% composite materials, mostly in secondary structure and control surfaces. In 1994, McDonnell Douglas proposed to re-design the horizontal tail using composite materials [4]. The tail was redesigned using AS-4 fiber in an epoxy resin for a 20% weight savings, 90% part reduction, 80% fastener reduction, and a projected 50% acquisition cost reduction. The prototype composite horizontal tail was successfully tested in 1998 to 133% of the design ultimate load. Orders have now been placed for 70 aircraft with the new composite horizontal tail.

Unmanned Rocket Experimental Aircraft

The USAF DC-X and the subsequent NASA DC-XA experimental flight vehicles [5,6] were developed to demonstrate vertical take-off and vertical landing (VTVL), aircraft-like turnaround times between flights, and advanced technologies that will be required for a single-stage-to-orbit reusable launch vehicle (RLV). The DC-XA vehicle, shown in Figure 11, is about

a 1/4 of the scale of the size of an RLV. The DC-X demonstrated autonomous VTVL flight operations by flying eight successful experimental flights, and also demonstrated operability and supportability of a complex, liquid fuel RLV by a small crew. Under a cooperative agreement with the USAF, NASA took over the DC-X program and created the DC-XA by implementing a number of important advanced technologies that will be enabling for an RLV. The advanced technologies implemented on the DC-XA included a composite shell intertank, a composite liquid hydrogen tank, an aluminum-lithium liquid oxygen tank, a liquid-hydrogen to gaseous-hydrogen conversion auxiliary propulsion system, enhanced avionics, and an in-situ health monitoring system. (The composite components will be discussed in more detail in the following paragraphs.) The DC-XA program demonstrated the use of rapid prototyping to design and fabricate the advanced technology components. The DC-XA successfully flew four flight tests and demonstrated the viability of significantly lighter-weight structural components, the auxiliary propulsion system, the in-situ health monitoring system, and the VTVL autonomous flight operation by a small operating crew.

The DC-XA composite intertank [5] resulted in a 44% weight savings over the DC-X aluminum intertank. The intertank, shown in Figure 12, was constructed in two semi-cylindrical halves and joined together by fore and aft aluminum attachment rings. The honeycomb sandwich shell was fabricated using graphite/bismaleimide (T650/5250-4) face sheets and an aluminum flex-core material. The face sheets were four plies of a fabric in a [0/+-45/90] lay-up. The aluminum flex-core material was chosen because of ease of fabrication and ready availability. Some fabrication problems occurred [6] in the form of ruptured core material as a result of out-gassing of the foaming adhesive used at the high processing temperatures, 440 °F (227 °C). The processing temperature was reduced to approximately 375 °F (191 °C), which is below the out-gassing temperature of the foaming adhesive, and no further problems were encountered. The intertank was ground-tested at NASA Marshall Space Flight Center (MSFC) to 153% of the DC-XA maximum load, and subsequently ground-tested at White Sands Missile Range (WSMR) before the flight test program. The intertank experienced no problems during the flight test program.

The DC-XA composite liquid hydrogen tank [5] resulted in a 34% weight savings over the DC-X aluminum tank. The liquid hydrogen tank, shown in Figure 13, was constructed in two cylindrical pieces that were joined together by a "belly wrap" bonded splice joint. The shell is a 24-ply IM7/8552 graphite/toughened-epoxy laminate. The shell thickness was somewhat over-designed to avoid leakage. An internal three-dimensional reinforced urethane foam was used to provide cryogenic insulation. Minor repairs were made to the tank due to a shop accident and the resulting separation of the insulation from the shell wall. The tank was ground-tested at MSFC for 29 pressure cycles filled with liquid nitrogen at 150% of the design limit load pressure, and filled with liquid hydrogen at 100% of the design limit load pressure. The tank was subsequently ground tested at WSMR for three engine firings and 16 pressure cycles. The tank then performed flawlessly during the four DC-XA flight tests with no observed leaks. The DC-XA is the first successful demonstration of a leak-free composite liquid hydrogen cryotank!

The objectives of the X-33 experimental rocket-powered vehicle is to demonstrate critical technologies for a reusable launch vehicle at hypersonic flight approaching Mach 13. Among these technology demonstration goals is a liquid hydrogen tank fabricated from composite

materials. The structural configuration, see Figure 14, is a complicated four-lobe (quadrant) conical shell with a noncircular cross-section and a non-spherical two-lobe end cap [7]. The tank shell is a sandwich construction with IM7/977-2 graphite/epoxy inner and outer face sheets with a Korex honeycomb core. The internal stiffening substructure is fabricated from textile preform graphite/epoxy composite materials. The all-composite, all-bonded tank is assembled using a complex, nine-step 350 °F (177 °C) curing and bonding procedure. In addition to the internal pressure required to maintain the liquid hydrogen in its liquid state, the vehicle is loaded by thrust loads during launch that are transferred directly through the liquid hydrogen tank. Unfortunately, the tank failed during the protoflight ground structural test [7]. The failure was primarily due to an incomplete understanding of the permeability of liquid hydrogen through composite materials and a lower than expected, as-manufactured bond strength between the honeycomb core and the inner face sheet of the tank shell sandwich structure.

Part II. Assessment of the State-of-the-Art

NASA uses a technology readiness level (TRL) scale from 1 to 9 to indicate the level of maturity of a technology. (The NASA TRL definitions are given in Table 1.) TRL values of 1 to 3 indicate research levels, with TRL 1 being fundamental research. TRL values from 4 to 6 indicate technology development levels. TRL values from 7 to 9 indicate advanced development levels, with TRL 9 signifying mature technology that is ready for actual aerospace vehicles. As a developer of advanced technology, NASA usually targets its technology development programs to advance the technology to TRL 6, and then transitions the technology to the aerospace industry. The description of TRL 6 is as follows: "system/subsystem validation model or prototype demonstrated in relevant environment (ground or space)." The NASA TRL scale will be frequently used in this section to indicate the level of maturity of technologies currently under development.

Structural Design, Analysis, and Testing: Lessons Learned

Vosteen and Hadcock [8] conducted a study of past composite aircraft structures programs to identify lessons learned and best practices. Interviews were conducted with 56 people from 32 organizations that were directly involved in design, fabrication, and supportability of composite structures. The Vosteen and Hadcock [8] survey identified the following lessons learned relative to structural design, analysis, and testing:

1. Design and certification requirements for composite structures are generally more complex and conservative than for metallic structures.

2. Successful programs have used the building-block approach with a realistic schedule that allows for a systematic development effort.

3. The use of basic laminates containing 0/90/+45/-45 plies with a minimum of 10% of the plies in each direction is well suited to most applications.

4. Mechanical joints should be restricted to attachment of metal fittings and situations where assembly or access is impractical using alternative approaches.

5. Large, co-cured assemblies reduce part count and assembly costs, but may require complex tooling.

6. Structural designs and the associated tooling should be able to accommodate design changes associated with the inevitable increases in design loads.

7. Understanding and properly characterizing impact damage would eliminate confusion in the design process and permit direct comparison of test data.

The Building-Block Approach is the Industry Standard Practice

Successful programs have used the building-block approach to design development and manufacturing scale-up, illustrated in Figure 15, to develop and verify the structural design details and manufacturing processes necessary for large composite structure. The complexities of light-weight, built-up structure led the industry to develop a building-block approach, which is the standard practice for both metallic and composite structures. The building-block approach relies on tests of elements and subcomponents to establish the effects of local details and internal load paths on structural behavior. The building-block approach also must include development tests to address manufacturing scale-up issues. This requirement is particularly critical in processing polymeric matrix composite materials where it is particularly challenging to scale-up accurately the curing kinetics to large-scale component fabrication. The lessons learned by the industry provide strong motivation for practicing collaborative engineering to design composite structures that can be reliably manufactured. Experienced materials and processing engineers should be included in the design phase and must be readily available to correct problems in production processes when they occur. The building-block approach must be used to avoid over-designed structure and high-risk structural designs.

The building-block approach relies on tests of coupons, elements, and subcomponents to establish the effects of local details and internal load paths on structural behavior. These tests are illustrated in the schematic shown in Figure 16 for a wing structure. By testing at each hierarchical level of detail, the interaction between the local elements are accurately represented in the structural design. These development tests can only be omitted if a design-by-analysis philosophy is supported by reliable, verified, high-fidelity design tools or by adopting a conservative design philosophy with large factors of safety. Since over-designed (heavier than necessary) structural components are not desirable and design tools are still under development, the building-block approach must be used to avoid high-risk structural designs.

While significant improvements have occurred to structural analysis methodologies over the past two decades, the current structural design and analysis methodologies used by the aeronautics industry are still largely semi-empirical. Very accurate finite element methods and sophisticated computer codes are used routinely to calculate the stress, strain, and displacement fields in complex structural geometries. Superior graphical interfaces have significantly improved pre- and post-processing of data files. Automated mesh generation, mesh refinement,

and automated adaptive remeshing have resulted in major efficiencies in model development time, analysis time, and accuracy of the numerical solutions. Post-processing algorithms and graphical interfaces have significantly improved the ability of the analyst to interpret the results of the stress analysis. However, the prediction of structural failure modes, ultimate strength, residual strength of damage-tolerant structure, and fatigue life has remained elusive for the structural engineer. A rigorous structural analysis suitable for predicting structural failure requires the generation of high-fidelity local stresses that can be used with failure criteria and damage models. The global/local method, illustrated in Figure 17, is one method currently under development to predict structural failure. At the present time, global/local analysis methods for metallic structures [9] are more mature and rigorous than are the corresponding methods for composite structures. This observation is primarily attributed to the fact that the failure modes for metallic structures are less complicated and, therefore, more deterministic than is the case for the failure modes for composite structures. In addition to damage and structural failure, nonlinear structural response characteristics such as buckling, postbucking and pressurized structural deformations are more difficult to predict for composite structures than they are for metallic structures. This difficulty is attributed primarily to the fact that composite materials are not isotropic, as are metals. Therefore, computational methods that rigorously account for material orthotrophy and anisotropy should be used for composite structures, and designers should understand the use of these methods.

Materials, Processes, and Manufacturing: Lessons Learned

The Vosteen and Hadcock [8] survey identified the following lessons learned relative to materials, processes, and manufacturing:

1. Materials development in conjunction with product development creates undue risks.

2. Experienced materials and processing engineers should be included in the structural design phase, and must be readily available to correct problems in production processes.

3. Manufacturing process scale-up development tests should be conducted to optimize the production processes.

4. Co-curing and co-bonding are preferred over secondary bonding which requires near perfect interface fit-up.

5. Mechanically fastened joints require close tolerance fit-up and shimming to assure a good fit, and to avoid damage to the composite parts during assembly.

6. Dimensional tolerances are more critical for composite structures than for metallic structures to avoid damage to parts during assembly. Quality tools are essential for the production of quality parts.

7. Selection of the tool material depends on part size, configuration, production rate, quantity, and company experience.

8. Tool designers should anticipate the need to modify tools to adjust for part springback, for ease of part removal, or to maintain dimensional control of critical interfaces.

State-of-the-Art in Materials, Processes, and Manufacturing

Significant improvements in the properties [10] and processability of polymer matrix composite materials have occurred over the past 30 years. New epoxies, as indicated in Figure 18, have been developed to improve significantly the toughness of composite materials. New thermosets and thermoplastics, as indicated in Figure 19, have been developed to increase significantly the use temperature of composite materials. Most epoxies cure at 350 °F (177 °C) and require an autoclave to insure proper fiber wetting, remove excess resin, minimize porosity, and promote the polymer cross-linking reaction. Material systems, such as T300/5208 with an intermediate modulus and intermediate strain-to-failure graphite fibers, have been used to manufacture structures such as the secondary structures and control surfaces for most commercial transport aircraft and the primary structures of the B-2. Higher performing material systems, such as the Toray T800H high-modulus, high-strain graphite fiber, and the toughened epoxy 3900-2 was used to manufacture the empennage structural components on the Boeing B-777. High performance military aircraft, such as the F-22, are manufactured out of materials systems such as IM-7/5250-4, high-temperature bismaleimide (BMI thermoset) and high-modulus, high-strain graphite fibers.

In recent years, the maturity of composite curing processes, such as resin transfer molding (RTM) and resin film infusion (RFI), have led to increased use of textile preforms such as braided, woven, and knitted fiber preforms, and through-the-thickness stitching[11]. These textile preforms have attractive features for low-cost manufacturing. Automated manufacturing has been facilitated by the development of high-speed fiber placement and stitching machinery which were adapted from those used in the textile industry. The development of advanced processing methods, such as powder-coated fiber tows, also contributed to the automated features of the processing of the material systems [12]. For example, fibers can be coated with a dry epoxy powder, braided into the desired form, and then cured so that near net-shape components can be readily fabricated. As an additional benefit, textile preform composite materials significantly improve the toughness of the composite material by providing through-the-thickness reinforcement. For example, the compression-after-impact (CAI) properties of stitched laminates using low-cost brittle epoxy materials satisfy or exceed the CAI properties of the significantly more expensive toughened epoxy systems. This technology is being used to fabricate the braided components of the internal stiffening substructure of the X-33 liquid hydrogen tank.

Quality Control, NDE/I, and Supportability: Lessons Learned

The Vosteen and Hadcock [8] survey identified the following lessons learned relative to quality control, nondestructive examination/inspection (NDE/I), and supportability:

1. Automated processes can help to reduce quality control costs.

2. Inspection and quality control should focus on aspects of the process and part that have a direct bearing on part performance.

3. Determine and understand the effects of defects on part performance.

4. Supportability should be addressed during design so that composite structures are inspectable, maintainable and repairable during service.

5. Most damage to composite structures occurs during assembly or routine maintenance of the aircraft.

6. Repair costs for composite structures are much higher than for metallic structures.

7. Improved Standard Repair Manuals are needed for in-service maintenance and repair.

8. Special long-life and low-temperature curing repair materials are required.

9. Moisture ingestion and aluminum core corrosion are recurring supportability problems for honeycomb structures.

The State-of-the-Art of NDE/I Technology

While the visual inspection method remains the method of choice for most airlines, nondestructive inspection (NDI) methods are also routinely used in both the manufacturing and flight operations environments. These NDI methods include thermal, ultrasonic, electromagnetic, radiography, and optical methods. Each method has strengths and weaknesses, depending on the specific inspection requirement. These NDI methods are listed in Figure 20 and the technology readiness levels (TRL) of the various methods are compared for applicability to metallic and composite structures with simple and complex configurations. (The comparative summary given in Figure 20 was prepared by the NASA NDE Working Group.) Referring to Figure 20, a TRL of 9 means that the technology is mature and is part of the industry standard practices. The gray boxes without a number mean that the corresponding NDI methods are not being developed for the specific application. The other colored boxes help to identify similar TRL levels. The distinction between "conventional" and "advanced" systems refers to the use of advanced computer-based numerical methods for signal processing. For example, conventional thermography refers to techniques where the temperature distribution of a structure is mapped using the image obtained from an infrared camera. The advanced thermography system relies on sophisticated computational software to analyze time-phased images of the infrared radiation given off by a structure and provides a map of the heat transfer or diffusivity of the structure. It has been found that the diffusivity of a structure provides much greater fidelity for determining the extent of damage than does the corresponding temperature distribution.

Boeing recently conducted an evaluation of current NDI methods for applicability to inspecting composite fuselage structure [13]. Several methods for detecting defects in stringer-stiffened structures and sandwich structures were compared. These methods include through-

transmission ultrasonic methods, lamb wave ultrasonic methods, pulse echo ultrasonic methods, and a specialized C-scan ultrasonic method for disbond detection. The through- transmission inspection system was found to be the most effective method and was able to resolve defects in both the skins and the core. However, the technique requires access to both sides of the component being inspected. Boeing concluded that improvements in current commercially available systems would be necessary to inspect composite sandwich structures reliably in the field.

Assessment of the Technology Requirements for an RLV

NASA has established a goal [14] of an order-of-magnitude reduction in the cost of launching a pound of payload to low earth orbit from current costs of about $10,000 per pound. A single-stage-to-orbit, reusable launch vehicle (RLV) is a leading concept for achieving this dramatic reduction in the payload launch cost. An RLV will have to operate much more like an airplane than the current Space Shuttle. The RLV must be robust, reliable, and require minimal inspection and maintenance between flights. Recent systems studies have shown that considerable reductions in the mass-fraction of the vehicle using conventional technologies must be achieved to reduce the gross take-off weight to a level where the vehicle can achieve orbit. (Mass-fraction is the ratio between the structural weight and the gross take-off weight.) Of all the technologies that may reduce the mass-fraction [6], the application of advanced composite materials for the primary structures and for the liquid hydrogen tank is projected to have the greatest potential for achieving the current take-off weight goals.

Over the past several years, NASA planning teams have evaluated various technologies, have estimated current technology readiness levels, and have prepared roadmaps for developing the technologies that will be enabling for an RLV. A summary of the enabling structures and materials technologies is shown in Figure 21 along with an estimate of the current TRL. (See Table 1 for the TRL definitions.) For current purposes, Primary Structures are defined as all load-bearing structures, exclusive of the integral cryogenic tanks. Cryotanks are defined as all elements of the cryogenic tank system, including the tank pressure vessel structure and the cryogenic insulation. Thermal protection systems (TPS) are defined as all elements of the vehicle thermal protection systems including both external TPS surfaces and internal insulation. Hot structural concepts applicable to vehicle leading-edges and control surfaces are also considered to be TPS subsystem elements. Some of the technologies listed in Figure 21 are much more complicated than others. The TRL is an estimate of the total technology, even though parts of the technology may be at a substantially higher TRL than the number listed in the figure. The technical evaluation summarized in Figure 21 indicates that extensive development of structures and materials technologies will be required to enable an RLV that will replace the Space Shuttle.

Lessons Learned from Technology Development Structural Tests

Over the past two decades, there have been a number of technology development programs that have designed, manufactured, and tested large composite structural components. Each of these tests is a source of lessons learned and provides valuable insight into further

developmental requirements. The following paragraphs summarize the results of six significant structural tests that have provided lessons learned.

Under contract to NASA, Lockheed-Martin designed and manufactured a large technology integration box beam [15]. The configuration, shown in Figure 22, resembled the C-130 center wing box and was about 150 inches (3.81 m) long, 50 inches (1.27 m) wide, and 28 inches (0.71 m) deep. The stiffness requirements were established to meet the commercial flutter requirements specified in FAR Part 25 [1]. The damage tolerance requirements satisfied both the FAR Part 25 requirements and the corresponding military requirements. Two graphite/epoxy material systems (AS4/1806 and AS4/974) were used. The ribs and spars were mechanically fastened to the cover panels. The test plan included a test of the box beam to design limit load with down-bending plus torsion loads, a test to design ultimate load with up-bending plus torsion loads, and a residual strength test to failure with impact damage at several locations with up-bending plus torsion loads. The box failed prematurely during the design ultimate load test at only 125% of the design limit load condition (83% of design ultimate load). An extensive failure investigation [15] determined that the failure initiated in the upper cover skin due to severe local bending of the skin in the region of the hat stiffener termination. It was found that a very small gap of unstiffened skin between the hat stiffener termination and the rib shear tie, as indicated in Figure 22, led to a local short-wave-length shear-crimping failure in the skin that resulted in the complete failure of the box. This unexpected failure highlighted the sensitivity (criticality) of composite structures to local structural detail features and complex local stress gradients.

A composite wing stub box was designed and fabricated by McDonnell Douglas, and tested by NASA Langley Research Center [16]. The wing stub box was the first of two major technology demonstration milestones in the NASA Advanced Subsonic Technology Composite Wing Program. The objective of the wing stub box was to demonstrate the viability of low-cost manufacturing technology. The wing stub box was fabricated using graphite/epoxy textile materials (AS4/3501-6 and IM7/3501-6) and stitched together using Kevlar thread. The IM7 graphite fibers were used only for the 0-degree fibers in the lower cover panel skin. The composite skin and stiffeners were composed of layers of dry fiber preforms that were prekitted in nine-ply-thick stacks with a quasi-isotropic stacking sequence. The resin film infusion (RFI) process was used to impregnate the dry fiber preforms with resin and the subsequent composite structures were cured in an autoclave. The composite test article, shown in Figure 23 (a), was attached to a metallic extension box to provide a load transition section so that loads representative of a transport wing structure could be applied to the stub box. The test plan included design limit load and design ultimate load tests with impact damage. The comparison of the test results to finite element model predictions, shown in Figure 23 (b), was excellent [16]. The model accurately predicted the onset of buckling in the cover panels of the box. The box failed at only 93% of the design ultimate load, which was slightly less than expected. The post-test investigation determined that the failure was initiated by nonvisible impact damage in the web and flange of a stringer that terminated near the front spar. This test highlighted the sensitivity of composite structures to nonvisible impact damage in regions of load redistribution such as the stringer termination. The cover panel was designed to account for compression-after-impact conditions, but the damaged stringer added a transverse shear load component to the

locally damaged area of the cover panel. This local shear load component created a local combined load condition in the damaged area of the cover panel.

June 1, 2000, marked the formal completion of the Advanced Subsonic Technology Composite Wing Program with the successful test of a 42-ft-long (12.8 m) Stitched/Resin Film Infused (S/RFI) Composite Wing Box [17]. The wing box, a manufacturing technology verification article, was designed and fabricated by the Boeing Company under contract to NASA to satisfy the requirements of a 220-passenger commercial transport aircraft. The S/RFI manufacturing process stitches together layers of multi-axial warp-knit graphite/epoxy fabric using Kevlar thread, and then impregnates and cures the resulting preform using the resin film infusion process. The tests of the wing box was conducted at NASA Langley Research Center and the wing is shown in Figure 24 for a design limit load condition. Prior to the failure test, the wing box was subjected to several design limit load (DLL) tests to measure structural response and to verify the accuracy of nonlinear finite element analysis procedures used to predict the wing box response. These tests included a 100-percent DLL test representative of a braked roll-out condition, a test with a 1-g down-bending condition, and a test with a 2.5-g up-bending condition. In addition, the wing was subjected to 7 inch (17.8 cm) long saw cuts, which are representative of discrete source damage, in the upper and lower cover panels. The wing box successfully supported the 70-percent DLL requirement with the saw cuts as required by the FAR Part 25 [1]. Prior to the design ultimate load (DUL) test, the saw cuts were repaired by an airline maintenance contractor to restore the wing box to its design ultimate load capability. Also, prior to the DUL test, the upper and lower cover panels were subjected to local impact damage events with impact energies ranging from 83 to 100 ft-lbs (113 to 136 N•m) to simulate foreign object damage. Sections of the wing structure were nearly 1-inch (2.54 cm) in thickness and were subjected to average running loads greater than 24,000 lbs/in (4.20 MN/m). The wing box failed at 97-percent of the DUL requirement with unrepaired nonvisible damage. This failure load is within the failure prediction accuracy of the finite element analysis used for this complex structure, and within the experimental scatter band for typical material properties. The ability of the wing box to sustain discrete source damage and foreign object damage demonstrates the robustness of the S/RFI composite manufacturing process and validates the accuracy of state-of-the-art damage-tolerance analytical methods for primary composite aircraft structures. (While few details were given, *Aviation Week & Space Technology*, March 20, 2000, p. 61, reported that DASA Airbus successfully completed a similar test program on a full-scale carbon fiber reinforced plastic wing and wing box.) These test programs clearly demonstrate that composite structures and materials can be scaled-up effectively to realistic, heavily loaded aircraft primary structures.

A full-scale segment of a reusable launch vehicle prototype wing was fabricated as a test article to demonstrate the integration of the thermal protection system (TPS) with large composite structural components and to validate the fabrication, design, and analysis methods for this wing [6,18]. A honeycomb sandwich construction was selected to provide broader design and fabrication experience. The upper and lower skin panels were fabricated using a graphite/bismaleimide (IM7/5250-4) material system. This material system was selected because it has good fracture toughness and good mechanical properties at elevated temperatures up to 350 °F (177 °C). The honeycomb core was glass/polyimide HRH-327 with a 3/16-in. (0.476 cm) cell size and a 4.5 lbs/ft^3 (72.1 Kg/m^3) density. The wing box is approximately 10

feet (3.05 m) long, 5 feet (1.52 m) wide, and 43 inches (1.09 m) deep with three ribs and three spars. While the wing box was not subjected to an elevated temperature test condition, three different types of TPS were installed on the upper skin to demonstrate the load carrying capability of the integrated structure. The test was conducted at NASA Langley Research Center and the test set-up is shown in Figure 25 (a). The wing box was loaded to design limit load and to design ultimate load with both up-bending and down-bending loading conditions. The box was then loaded to failure with the up-bending loading condition. Selected measured strain values recorded during the tests are shown in Figure 25 (b), and the results are in excellent agreement with the values calculated by the finite element analysis. The predicted upper skin buckling load was within 3% of the experimental value. The predicted shear failure load was within 5% of the experimental value. While considerable work is still required to develop manufacturing technology that can be scaled-up to an RLV size vehicle, the success of this test clearly indicates the viability of composite structures technology for primary structures applications to reusable launch vehicles.

A composite intertank design for the body of a reusable launch vehicle was developed, and a full-scale segment was fabricated and tested [6, 19]. The intertank was designed to contain the payload for the vehicle and, therefore, would have payload bay doors. The critical design condition is the compressive load due to maximum ascent acceleration; and the load transfer around the payload bay doors is a major design consideration. A design trade study resulted in the selection of a stiffened-skin configuration with internal frames. The graphite/bismaleimide (IM7/5250-4) material system was selected for the skin, stiffeners, and frames due to its good fracture toughness and good mechanical properties at temperatures up to 350 °F (177 °C). A curved section of the intertank design was selected as a structural test article. The test article, shown installed in the test facility in Figure 26 (a), was approximately 10 feet (3.05 m) long and 22 feet (6.71 m) wide, and includes about a 90-degree section of the intertank. The test was conducted in a structural test facility at NASA Langley Research Center. The test article failed prematurely when subjected to a compression load due to the separation of the hat stiffeners from the skin at approximately 70% of the predicted failure load. The failed test article is shown in Figure 26 (b) with a buckled skin. The premature failure was attributed to a poorly manufactured bond between the hat stiffeners and the skin. This test illustrates the critical need to include manufacturing scale-up development tests in the building-block approach to the design and fabrication of large-scale structural components.

The development of a lighter-weight liquid hydrogen tank is continuing with a series of ground tests at NASA Marshall Space Flight Center, see Figure 27 [20]. The objectives of the ground test program are to verify the structural integrity of an RLV flight weight tank, to verify the structural design and analysis methods, to verify the impermeability of the tank skin to liquid hydrogen, and to verify the lifetime performance of the insulation. The 8 ft (2.44 m) diameter tank, 1/4- scale for an RLV, was fabricated using fiber-tow-placed IM7/977-2 graphite/epoxy material and co-bonded stiffeners. A honeycomb core insulation was bonded to the outside of the tank. The tank shell wall thickness is only 14 plies in contrast to the over-designed 24-ply thickness of the DC-XA tank. To date, five pressure cycles in the test plan have been completed with liquid nitrogen. While the tank is structurally sound, a few minor leaks have required some repairs to the tank. The pressure testing with liquid nitrogen is continuing, and the test program will eventually begin testing with liquid hydrogen. While considerable work is still required to

develop manufacturing technology that can be scaled-up to an RLV size vehicle, the success of this test indicates the viability of composite structures technology for the cryotanks of an RLV.

Part III. Future Directions

Revolutionary Structural Concepts for Next Generation Aircraft

The quest for improved materials for aerospace vehicles is never ending. Design and market drivers include lower weight, improved corrosion and fatigue resistance, and lower acquisition and operation costs. It is interesting to contemplate the current use of composite materials on commercial transport aircraft and to try to extrapolate to the next generation aircraft. The most significant current barriers to increased use of composite materials are high manufacturing costs, poor reliability in estimating the design and development costs, and the inability to predict accurately structural failure. As illustrated in Figure 3, the advantage of composite materials in secondary structures and lightly loaded primary structures has been more-or-less fully demonstrated. Given the current state of the technology, a consensus has emerged within the community that the next step in the evolution of composite structures for commercial transport aircraft applications is a composite wing. Beyond this developmental step, the marketplace will decide the next opportunity for composite materials and structures. For example, composite materials may prove to be an enabling technology for a new class of aircraft that have superior performance characteristics compared to today's commercial transport aircraft. Several revolutionary aircraft configurations are illustrated in Figure 28. The aerodynamic performance of these vehicles may prove to be quite superior to conventional subsonic aircraft. The potential benefits that may be derived from these revolutionary aircraft include significant increases in flight range or performance, significant reductions in fuel consumption, significant reductions in engine emissions, and significant reductions in airframe and engine noise. However, major improvements in the current state-of-the-art for composite structures will be required to design and build these new aircraft reliably and economically. For example, the noncircular cross-section and compound curvature features of the blended wing body configuration will be a particularly significant challenge for structural designers.

Current research is expected to result in dramatic improvements in structural design and analysis tools. Reliable, advanced analysis methods will significantly reduce current dependence on the empirical design approach and provide better capability to optimize structural designs. High-fidelity, physics-based structural analysis tools are under development using both deterministic and non-deterministic computational methods. Rigorous, physics-based computational methods to predict accurately damage initiation and growth, structural failure modes, and the residual strength of damaged structure remains a grand challenge that is motivating considerable research attention in the structures community. Next generation structural design tools are under development that exploit the revolution in information technology. The use of intelligent systems to improve graphical user interfaces and three-dimensional immersive simulation of structural analysis results is illustrated in Figure 29. As illustrated in Figure 29 (a), the next generation design tools will use libraries of smart components to assemble finite element analysis models easily. Interface elements are under development that will provide seamless transitions between regions of a finite element model with different mesh refinements. These advanced methods not only automate model generation,

but also facilitate the implementation of global-local modeling strategies that are essential for the prediction of progressive damage and structural failure. Finally, advanced three-dimensional virtual reality capabilities, such as the system shown in Figure 29 (b), will greatly enhance our ability to interpret the results of structural analyses.

Breakthroughs in Materials Synthesis and Processes

Current manufacturing technology requires an autoclave to cure polymer matrix composite materials to provide high-quality, high-performance structural components. Eliminating the autoclave will dramatically lower the cost and complexity of manufacturing composite structures. Revolutionary new methods of curing composite materials are being developed to eliminate the autoclave from the curing process. These new out-of-the-autoclave processes, such as the electron-beam magnetic suspension process shown conceptually in Figure 30, may also facilitate the manufacturing of virtually final, near-net shape components. The concept illustrated in Figure 30 has the added advantage of eliminating the expensive tools currently required to make composite parts where precision controlled design tolerances are required. This technology could eliminate the need for mechanical fasteners and adhesive bonds, except for those associated with major airframe assembly splices.

The general field of nanotechnology offers the potential to be the next great industrial revolution. In the field of materials science, a paradigm shift may occur away from the traditional materials role of developing metallic, polymeric, ceramic, and composite materials to a revolutionary role of developing nanostructured, functionalized, self-assembling, and self-healing materials. Looking into the future, the theoretical potential of these revolutionary classes of new materials will create breakthroughs that will enable technology developments that are barely imaginable today. In the aerospace field, these new technologies may make space travel routine and enable human exploration of space beyond our current practical limitation of low Earth orbit. Imagine the possibilities if there was a material to replace aluminum that is an order of magnitude stiffer and two orders of magnitude stronger!

Breakthroughs in the methods used to synthesize new polymer chemistries will lead to highly tailored materials with significantly improved properties. For example, advanced computer software is being developed to exploit new knowledge in nanostructure-property relationships. Referring to Figure 31, "computational materials" is one of the emerging fields of "computer-designed materials" that is attempting to build a bridge between our knowledge of quantum physics and continuum-mechanics-based micromechanics. Computer models are generated at the atomic and molecular levels that model the relationship between the atomic structure of the material at the nano-scale and the physical properties exhibited at the macro-scale. First principles of quantum mechanics, molecular dynamics, thermodynamics, and continuum mechanics are being used to predict the properties of new material chemistries. These new computational tools have an extraordinary potential to optimize chemistries for specific performance goals and to conduct trade-off studies that quantify the effects of changes in chemistry on various material properties.

Computer simulation results and limited experimental studies show that small diameter, single-walled carbon nanotubes (CNT) may possess elastic moduli in excess of 145 Msi (1 TPa), and strengths approaching 29 Msi (200 GPa) [21-34]. If small-diameter, single-walled tubes can be produced in large quantities, and incorporated into a supporting matrix to form structural materials, the resulting structures could be significantly lighter and stronger than those made from current aluminum alloys and carbon fiber reinforced polymer (CFRP) composite materials used in conventional aerospace structures. Properties of single-wall carbon nanotubes (SWNT) and multi-wall carbon nanotubes (MWNT) reported in the literature exhibit quite a range in values. Theoretical properties have been determined from computer simulations using quantum mechanics [29, 30], atomistic simulation (molecular dynamics) [31, 32], and continuum mechanics [33, 34]. Experimental measurements of properties have been reported using atomic force microscopy and Raman Spectroscopy. The table below illustrates the variability in the data reported in the literature [21-28]:

Tensile modulus	44 to 260 Msi (300 to 1800 GPa)
Tensile ultimate strength	0.9 to 26 Msi (6 to 180 GPa)
Bending strength	0.9 to 3 Msi (6 to 22 GPa)
Elongation (strain to failure)	6 to 15%
Thermal conductivity	1000 to 3300 Btu ft^{-1} hr^{-1} $°F^{-1}$ (1750 to 5800 W m^{-1} K^{-1})

The specific modulus and specific strength of several aerospace materials currently used in structural components of aerospace vehicles are compared in Figure 32. The properties of sheet and plate forms of aluminum 2219 alloy were obtained from MIL-HDBK-5D [35]. The CFRP composite material indicated in the figure is a high-modulus, high-strength fiber in a toughened polymer matrix with a quasi-isotropic laminate stacking sequence and a 60% fiber volume fraction [36]. Theoretical properties of the carbon nanotube fiber reinforced polymer (SWNTFRP) composite were calculated using standard micromechanics equations. The modulus of the SWNTFRP was assumed to be 174 Msi (1200 GPa). The SWNTFRP laminate is assumed to be the same laminate as the CFRP laminate and the strength was limited to 0.9 Msi (6 GPa) (1% strain) to reflect current structures design practices. The single crystal bulk material (SWNT) plotted in Figure 32 represents the theoretical potential of nanostructured carbon that will require several breakthroughs in nanotube production technology to achieve. This highly perfect, single crystal, bulk material does not require a matrix binder material and is viewed as theoretically possible. As is evident from Figure 32, the polymer composite reinforced with nanotubes offers a significant advantage over conventional aluminum and carbon fiber reinforced polymer composite materials.

The theoretical properties of the SWNTFRP were used in a simple, systems analysis concept model of a reusable launch vehicle shown in Figure 33 (a). The computed vehicle dry weight results are shown in the accompanying bar chart in Figure 33 (b). Dramatic reductions in weight are achievable by replacing the aluminum components of the airframe structure and the cryogenic propellant tanks with the CFRP composite material or the SWNTFRP composite material, and then resizing the vehicle.

Evolution of Composite Structures NDE Technology

Nondestructive evaluation (NDE) techniques are currently used during component manufacturing, design certification, maintenance, inspection, and repair. Current research is focused on exploiting the role of computer simulations [37] to revolutionize the traditional NDE role, see Figure 34. It is generally understood that NDE issues that are not addressed during the component design stage must be addressed later in the manufacturing stage. This staging of the use of NDE procedures can be, potentially, at a much higher cost as maintenance and repair considerations increase with component age. If validated and robust NDE simulations are available during the initial design stage, then component configurations may be adjusted in "real-time" to lower the overall life cycle NDE costs while maintaining optimized system level benefits. Furthermore, these benefits are enhanced when manufacturing simulations make use of NDE process control simulations. Validated simulations of NDE for process control, when incorporated or embedded into the manufacturing process control, can reduce or eliminate manufacturing process steps, including conventional inspections, while further optimizing the yield of the manufacturing process.

For the foreseeable future, structural components will continue to incur operational service-induced damage and degradation. The requirement to evaluate component integrity and repair or replace damaged components will continue to challenge the NDE community. In the future, NDE simulations may be optimized to the point that they may be used to generate the plans for in-service maintainability and repair. Issues such as component design and functional specifications, work space geometry and component access, and accept/reject criteria or retirement-for-cause criteria will need to be incorporated into these NDE simulations. It is anticipated that NDE technology will evolve to a state-of-the-art where virtual reality NDE simulations in design, smart health monitoring systems, and telerobotic inspection and repair are commonplace. The challenge for the NDE community is to develop and validate virtual reality simulations that are robust and adaptable enough to function smoothly and autonomously.

Next Generation Design Tools and Collaborative Engineering

In order for NASA to meet its unique mission needs in space science, human exploration, earth science, and aeronautics, NASA has a new initiative to develop an intelligent synthesis environment (ISE). The ISE will utilize computational intelligence to synthesize existing, developing, and future relevant technologies to create a new product and mission development environment. In the ISE, synthesis will take place in three ways: synthesis of scientists, engineers, technology developers, operational personnel and training personnel all working in geographically distributed locations (collaborative engineering); synthesis of cutting-edge technologies and diverse, life-cycle design tools seamlessly integrated together both horizontally and vertically at all levels of fidelity; and synthesis of computers, intelligent hardware (robotics), synthetic (virtual reality) simulated designs, and design languages. The intelligent nature of the ISE will be derived from its concentrated use of non-traditional, intelligent computational systems such as intelligent product objects, intelligent agents, and intelligent computational methods. Computational intelligence will guide the utilization of vast resources of knowledge and predictive capability that is built directly into the design environment. Effective collaborative engineering, illustrated conceptually in Figure 35, will require that numerous

validated computer software modules, which represent a wide range of scientific disciplines, will operate together in a robust fashion to yield credible optimized mission performance [38]. The path to achieving credible optimized structural designs is dependent on the fidelity of the individual software modules and the optimization processes.

Summary of Assessment

Aerospace structural components are usually designed to be very close to a zero design margin. While the margin of safety is not equal to zero for all of the design criteria at each structural location, there is typically one criterion for each structural element that governs the design details for that element. The quest for the lowest weight structure then drives the design margin to be nearly equal to zero for the design limit load condition. The factor of safety between the design limit load and the design ultimate load conditions accounts for the difference between linear, elastic behavior and complete structural failure, and for uncertainties in other parameters such as loads and material properties. Therefore, aerospace structural designs do not have a large factor of safety to accommodate any unanticipated deleterious structural behavior.

Composite structures fail differently than metallic structures. The 65 years of successful experiences with the design of metallic structures cannot be directly transferred to the design of composite structures. First, composite materials are not isotropic like most metallic alloys. Second, the initiation and growth of material level damage and the failure modes of composite structure are not well understood and cannot always be predicted accurately. Due to these complications, the best design practices for composite structures are fully understood only by those engineers who are experienced at designing composite structures.

Composite structural design and manufacturing technology is not yet fully mature for all applications. There are three key factors that contribute to the lack of maturity of the design and manufacturing technology for composite structures. These factors are the lack of a full understanding of damage mechanisms and structural failure modes, the inability to predict reliably the cost of developing composite structures, and the high costs of fabricating composite structures relative to conventional aluminum structures. While the technology required to overcome these uncertainties is under development, these factors are barriers to expanding the application of composite materials to heavily loaded, primary structures. For those applications where development and fabrication costs are not a factor or where risks to aircraft structural integrity are low, there is extensive use of composite structures.

Successful programs have used the building-block approach to structural design and manufacturing process development with a realistic schedule that allows for a systematic development effort. The complexities of lightweight, built-up structure led the industry to develop a building-block approach, which is the standard practice for both metallic and composite materials. The building-block approach relies on tests of elements and subcomponents to establish the effects of local design detail features and load paths on structural behavior. The building block approach also must include development tests to address manufacturing scale-up issues. This observation is particularly critical in processing polymeric matrix composite materials where it is particularly challenging to scale-up the curing kinetics to large-scale component fabrication. The lessons learned by the industry provide strong

motivation for practicing collaborative engineering to design composite structures that can be reliably manufactured. Experienced materials and processing engineers should be included in the structural design phase of a project and must be readily available to correct problems in production processes when they occur. The building-block approach must be used to avoid over-designed structure and high-risk structural designs.

Maintenance, inspection, and repair technologies for composite structures are not yet fully mature for all applications. Technologies in everyday use today to support metallic structures do not apply to composite structures. Furthermore, the long-term, field experiences necessary to develop a support infrastructure does not yet widely exist for composite structures. Therefore, support issues must be anticipated in the design phase for composite structures to help facilitate effective maintenance, inspection, and repair procedures. Structures must be designed so that they that can be inspected and repaired in the field. In addition, NDE experts should be part of the collaborative engineering team so that inspectability is built into the structural design from the outset of the design.

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Table 1. NASA Technology Readiness Levels (TRL).

TRL	Description
1	Basic principles observed and reported
2	Technology concepts and/or applications formulated
3	Analytical & experimental critical function and/or characteristic proof-of-concept
4	Component and/or breadboard validation in laboratory environment
5	Component and/or breadboard validation in relevant environment
6	System/subsystem validation model or prototype demonstration in relevant environment (ground or space)
7	System prototype demonstration in an air/space environment
8	Actual system completed & flight qualified through test and demonstration (ground or flight)
9	Actual system flight proven through successful mission operations



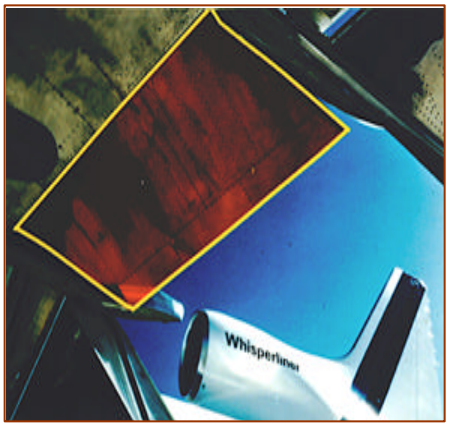
a. Boeing B-727 composite elevator



b. Boeing B-737 composite horizontal stabilizer



c. Douglas DC-10 composite Rudder and vertical stabilizer



d. Lockheed L-1011 composite aileron

Figure 1. Typical commercial applications initiated by the ACEE Program.

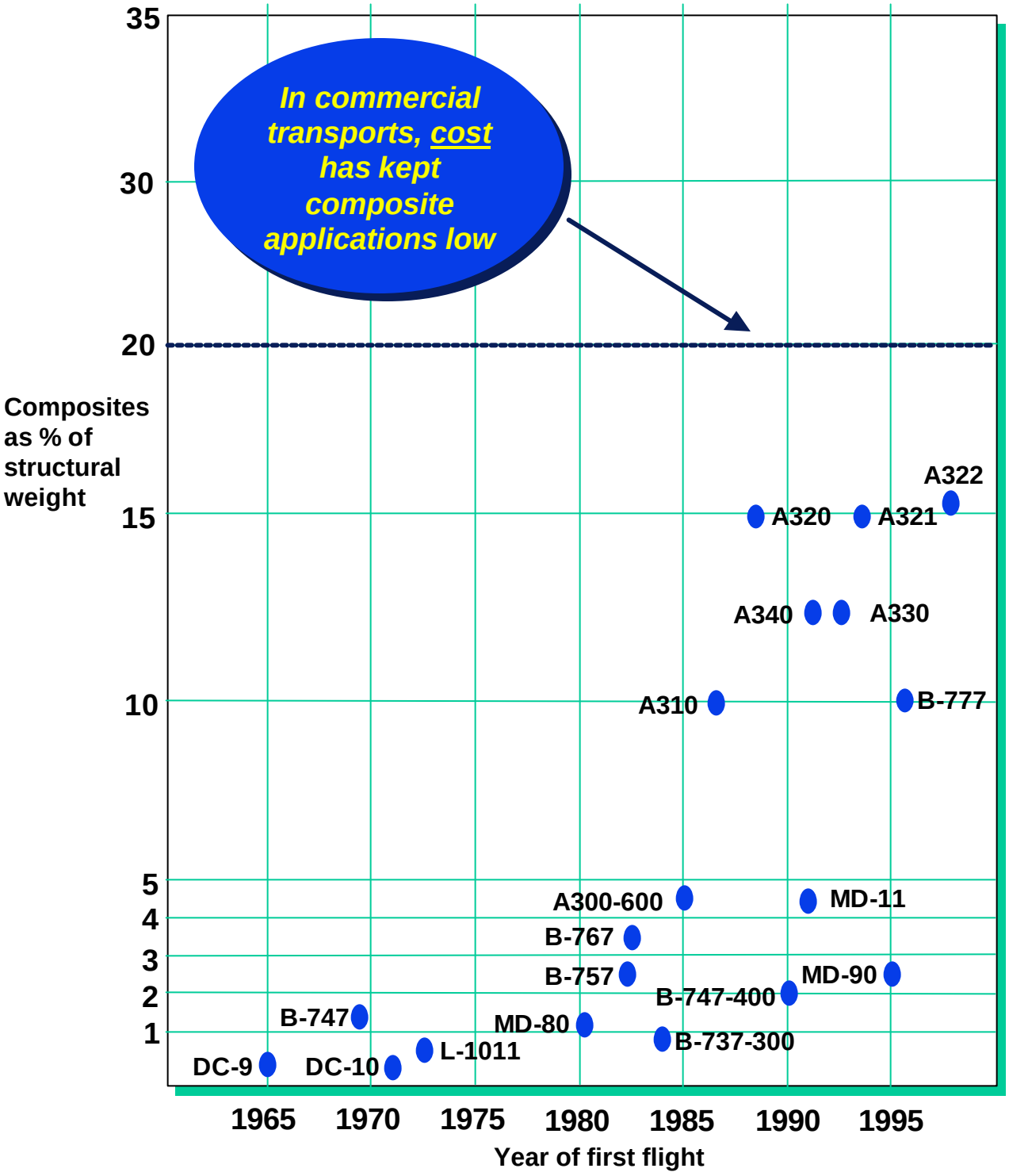


Figure 2. Composite structural applications in commercial transport aircraft.

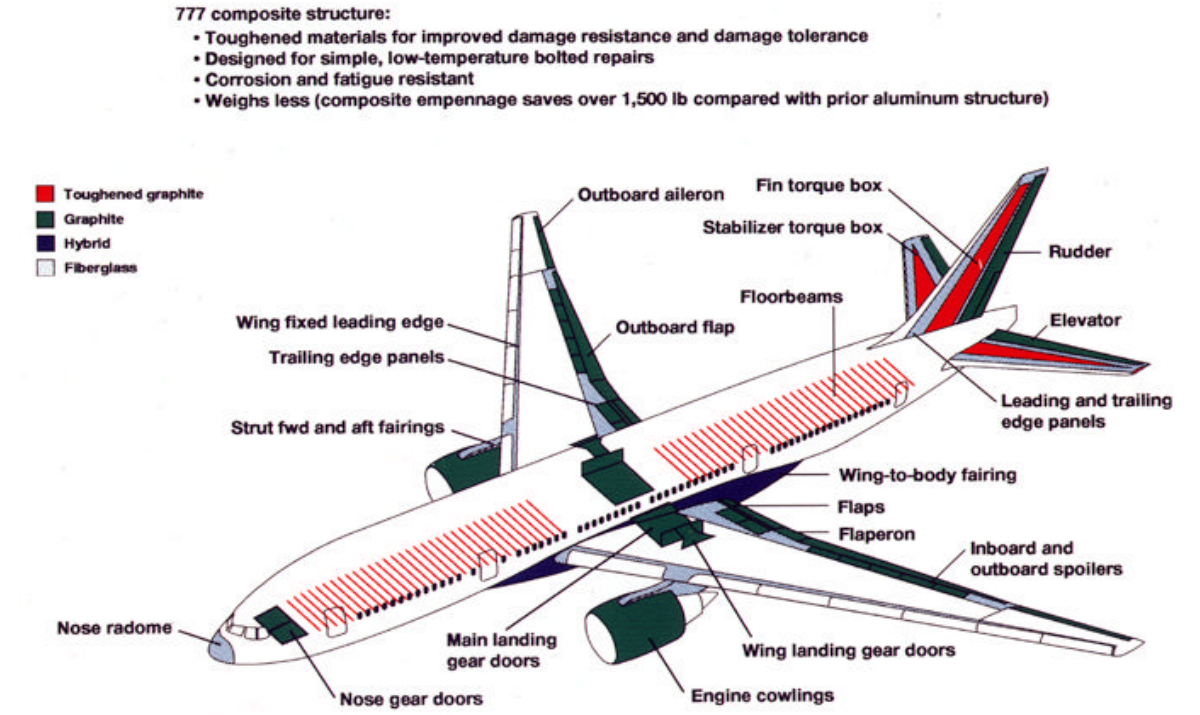


Figure 3. Structural composites on the Boeing B-777.

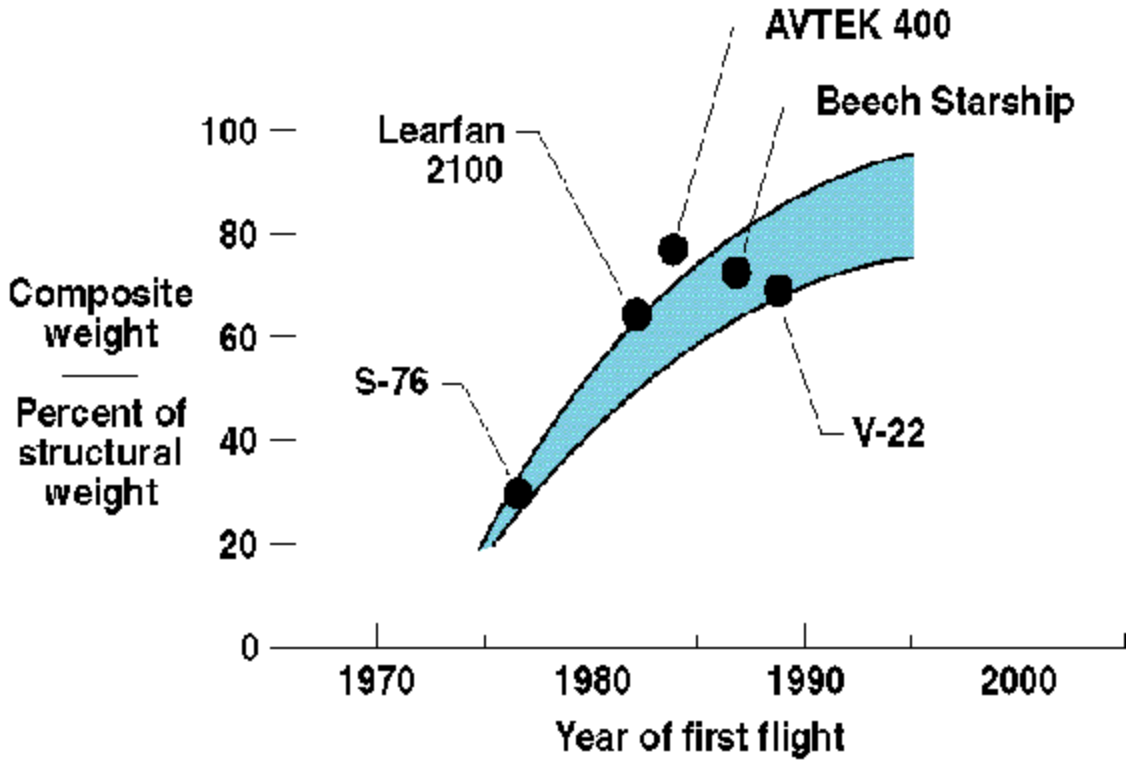


Figure 4. Composite structural applications in rotorcraft and general aviation.



Figure 5. Bell-Boeing V-22 Osprey.

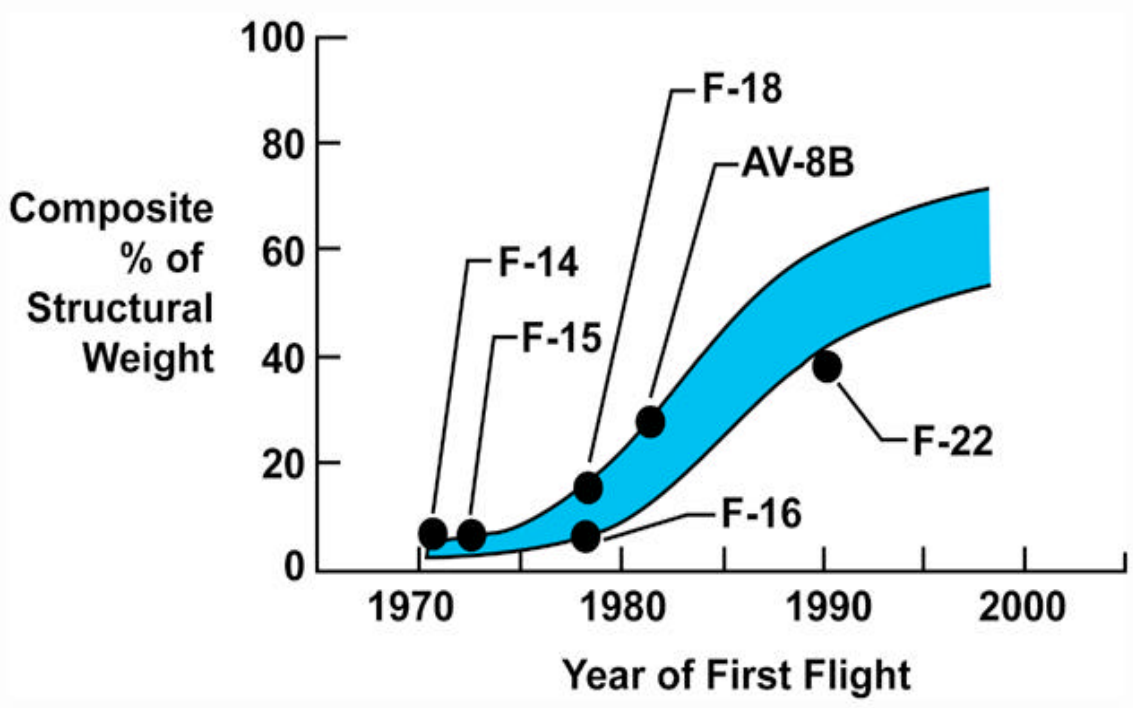


Figure 6. Composite structural applications in military fighter aircraft.



Figure 7. Lockheed Martin F-22 Raptor.

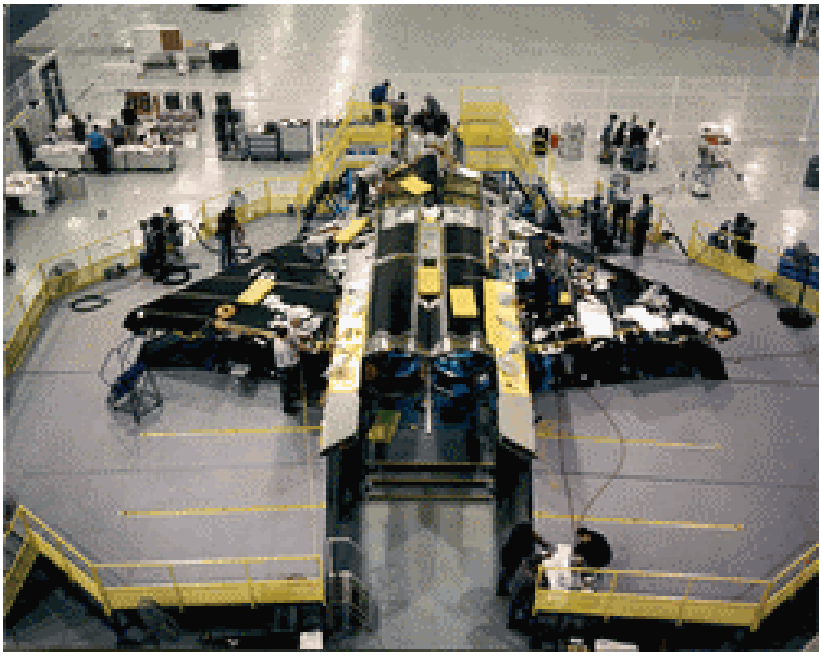


Figure 8. Assembly of F-22 composite wings.



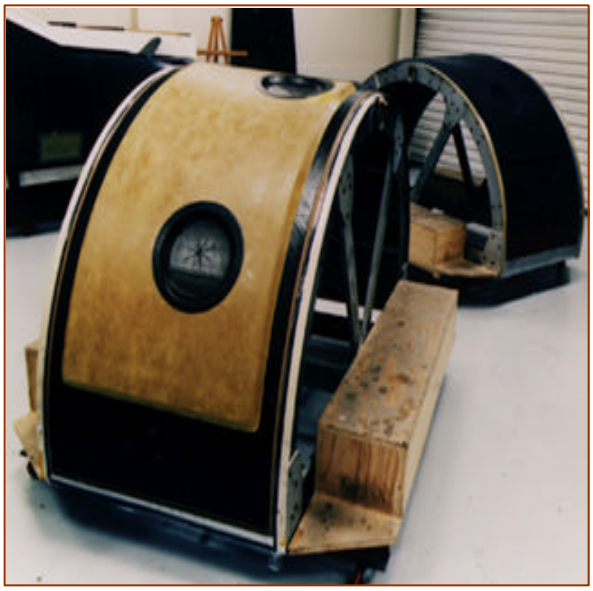
Figure 9. B-2 primary structure Is almost all composite materials.



Figure 10. C-17 horizontal tail redesigned using composite materials.



Figure 11. DC-XA experimental rocket aircraft.



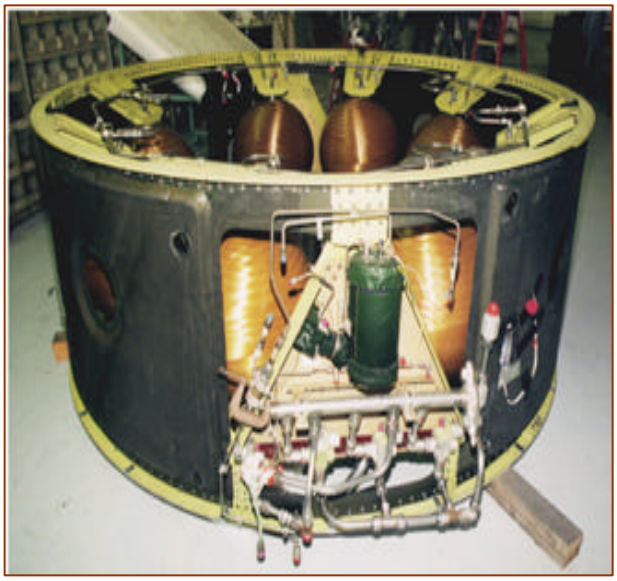


Figure 12. DC-XA composite intertank.



Figure 13. DC-XA composite LH2 cryotank.

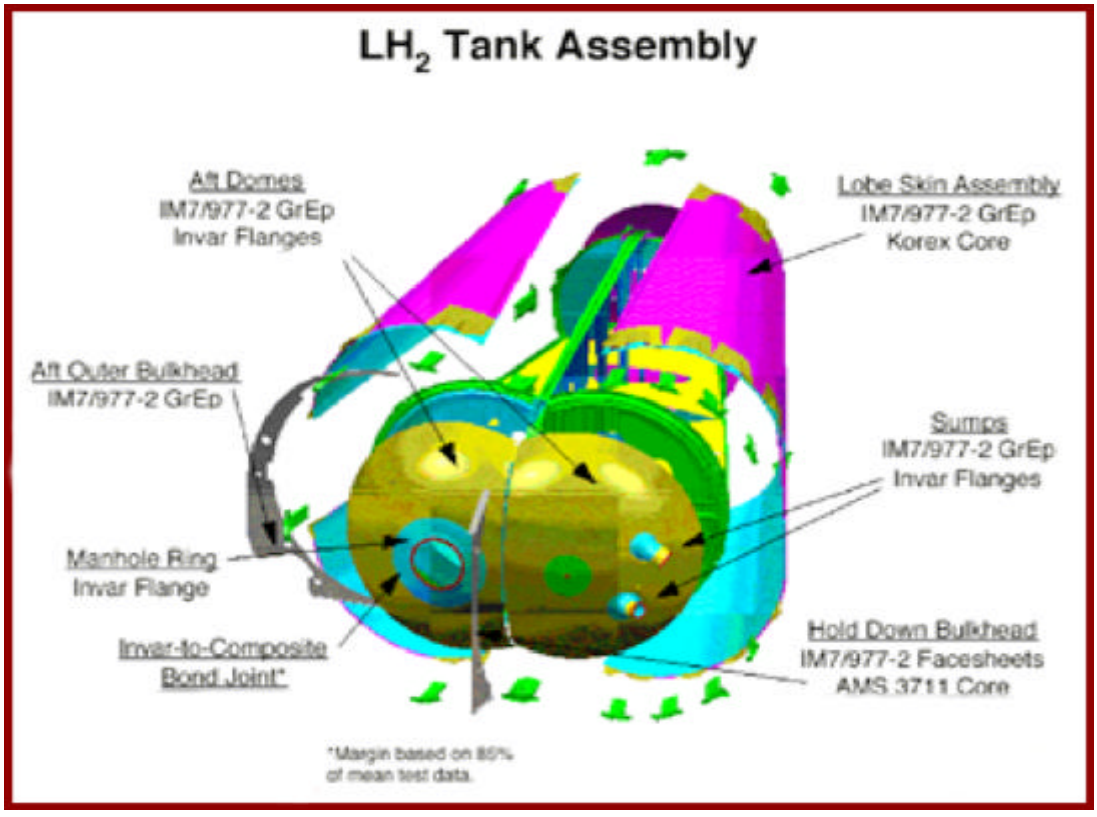


Figure 14. X-33 all-bonded, all-composite LH_2 tank.

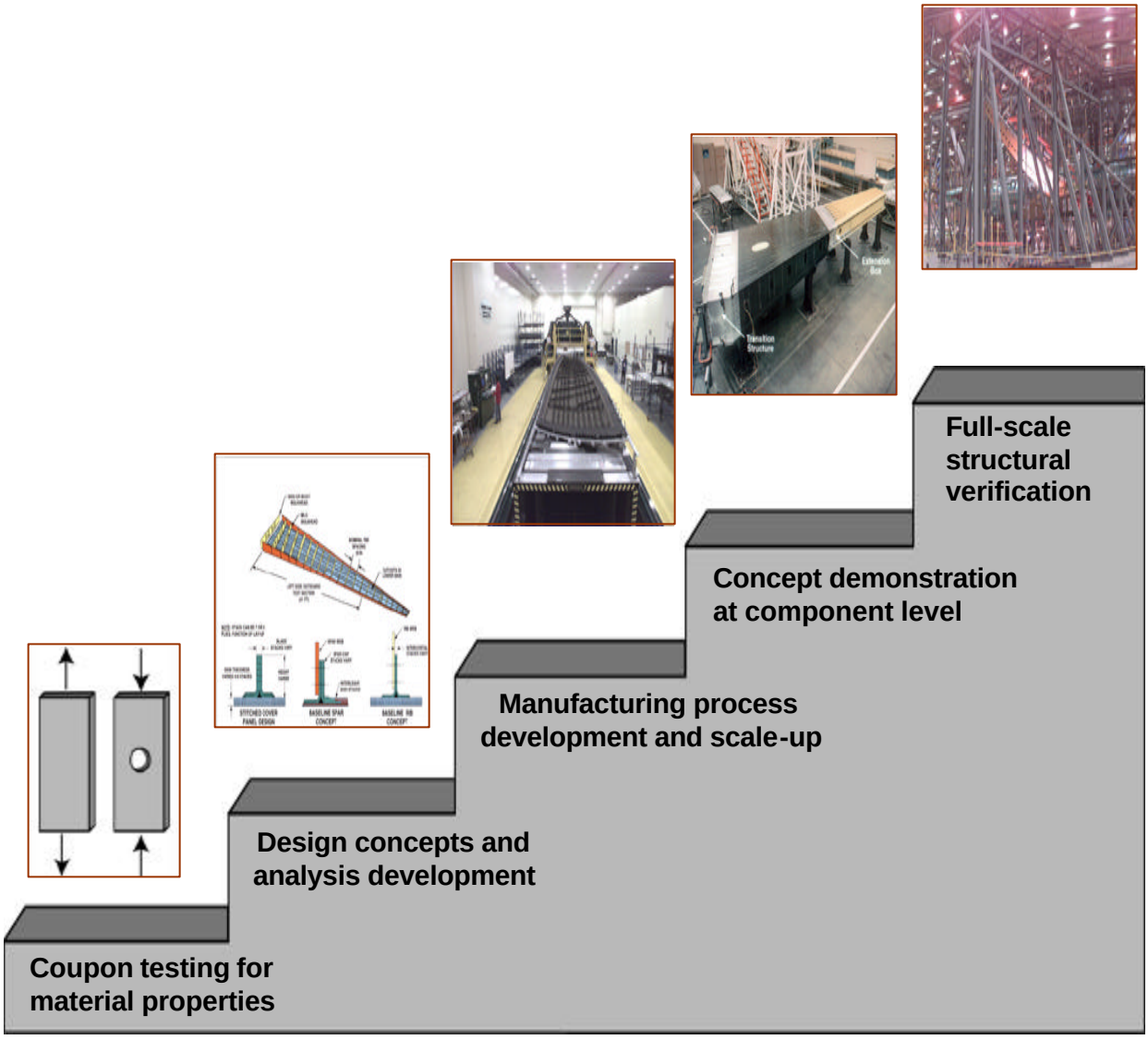


Figure 15. Building-block approach is the industry standard practice.

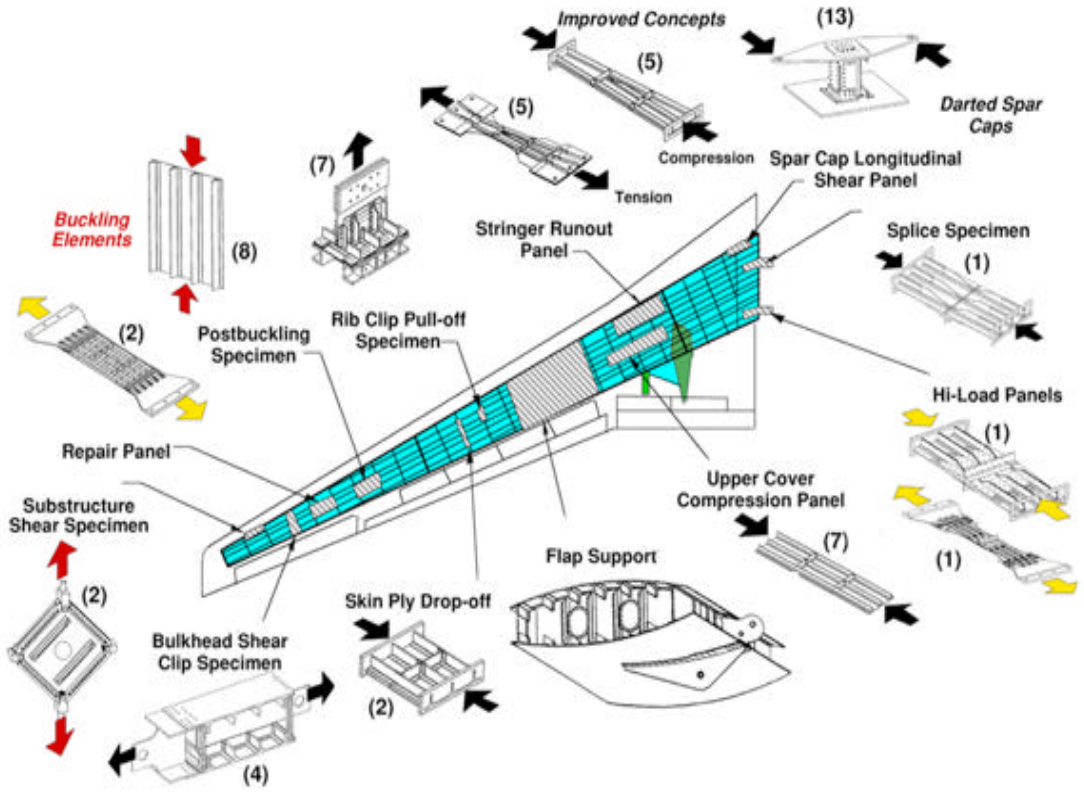


Figure 16. Design development tests in building-block approach.

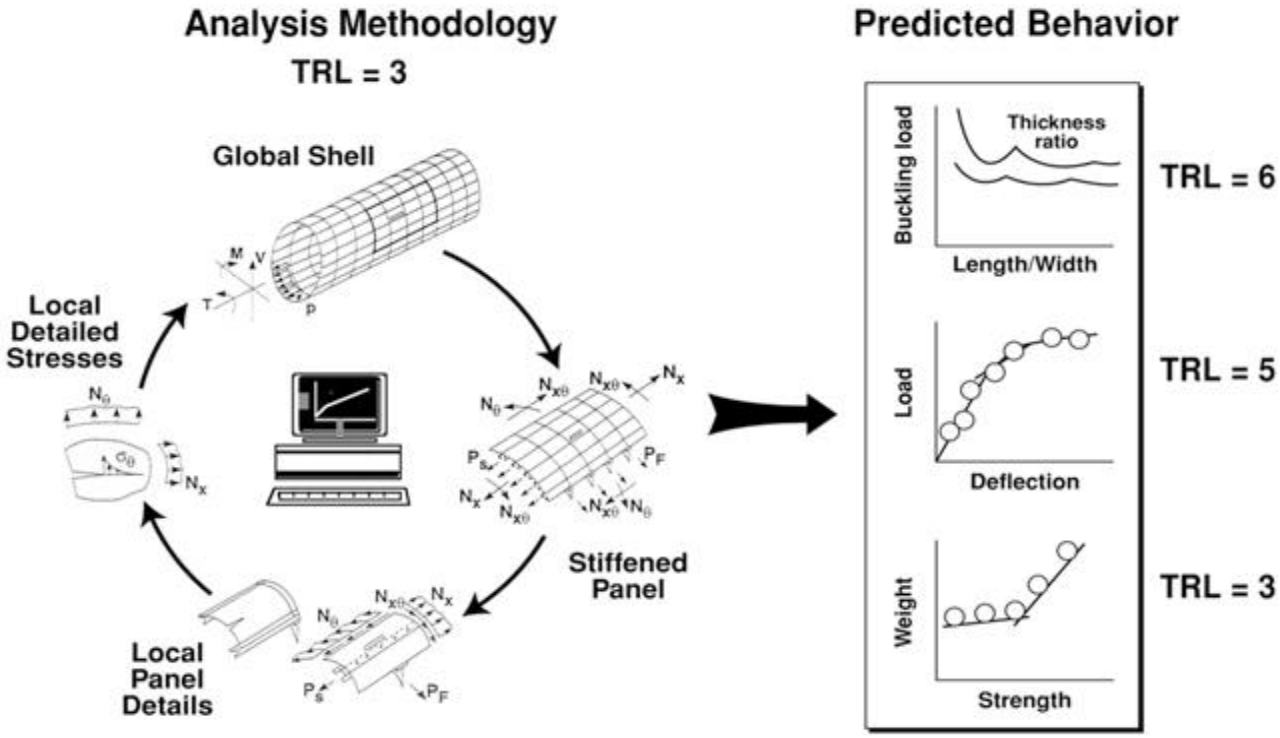


Figure 17. Global/local analysis for predicting structural behavior.

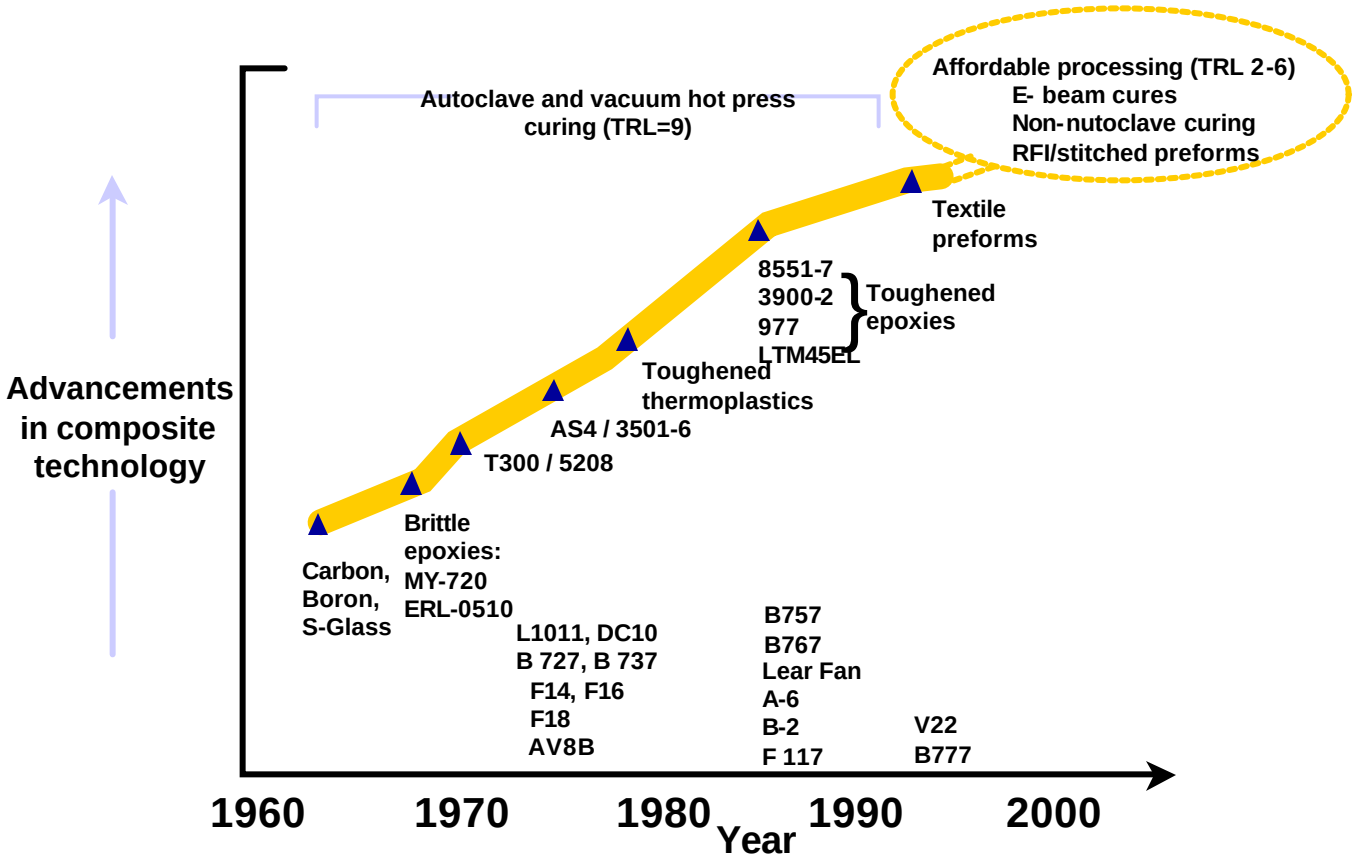


Figure 18. Evolution of composite resin development: *Epoxies.*

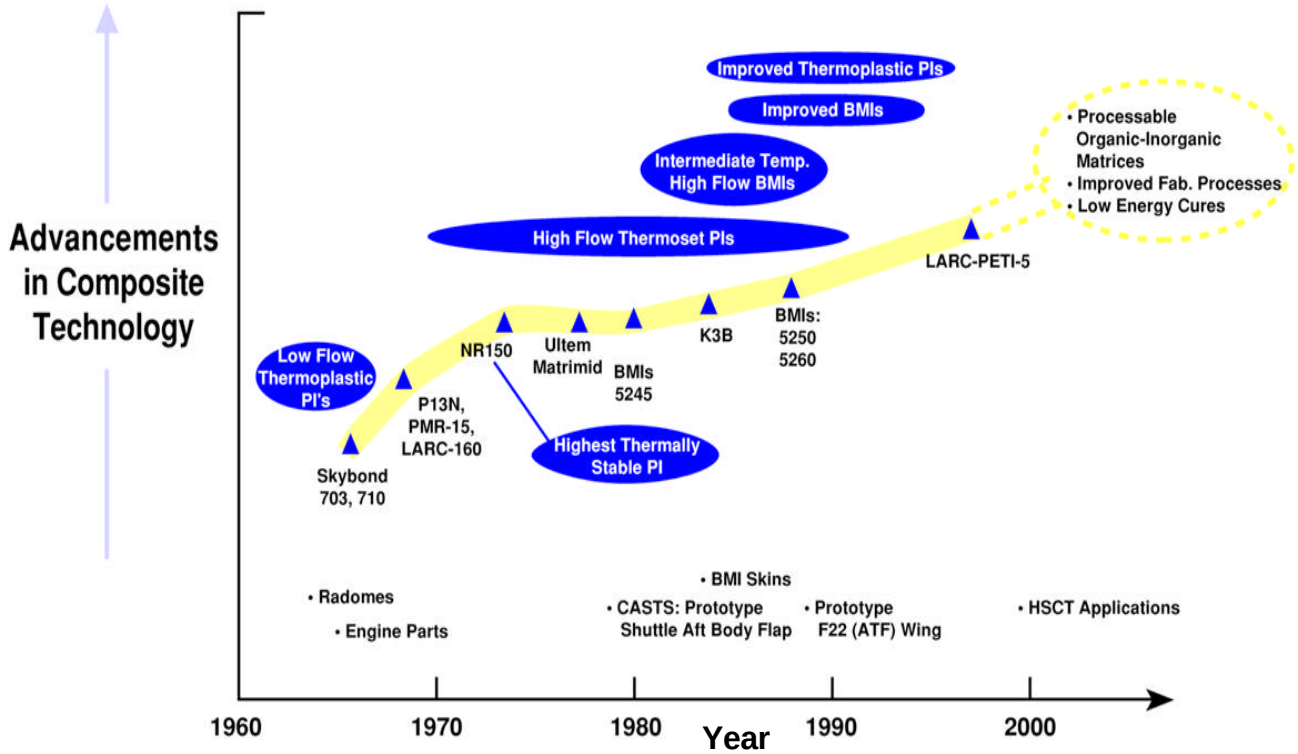


Figure 19. Evolution of composite resin development: *Intermediate and high temperature resins.*

NDE TECHNOLOGY	BASIC METAL STRUCTURES	BASIC COMPOSITE STRUCTURES	COMPLEX METAL STRUCTURES	COMPLEX COMPOSITE STRUCTURES
	Planar, slight curvature	Planar, slight curvature B-2, B-777	Irregular, curved, hybrid, bonded, honeycomb, built-up structure	Irregular, curved, hybrid, bonded, honeycomb, built-up structure, X-33, X-34
Conventional thermography	9			
Advanced thermography	9	9	4	4
Conventional ultrasonics	9			
Advanced ultrasonics	9	9	3	3
Conventional X-radiography	9	9		
Reverse geometry X-ray	6	5	4	3
Computed tomography	9	9	9	6
Backscattered X-ray	5			
Conventional eddy current	9			
Advanced eddy current	9		4	3
Optical shearography	6	4	3	3
Microwave	3	4	3	3
Conventional acoustic emissions	9	6		
Advanced acoustic emissions	9	6	3	3
Visual	9			
Penetrants (surface defects)	9		9	
Magnetic particle (surface defects)	9		9	
In-Situ vehicle health monitoring			3	3

Numbers in the table are TRL's

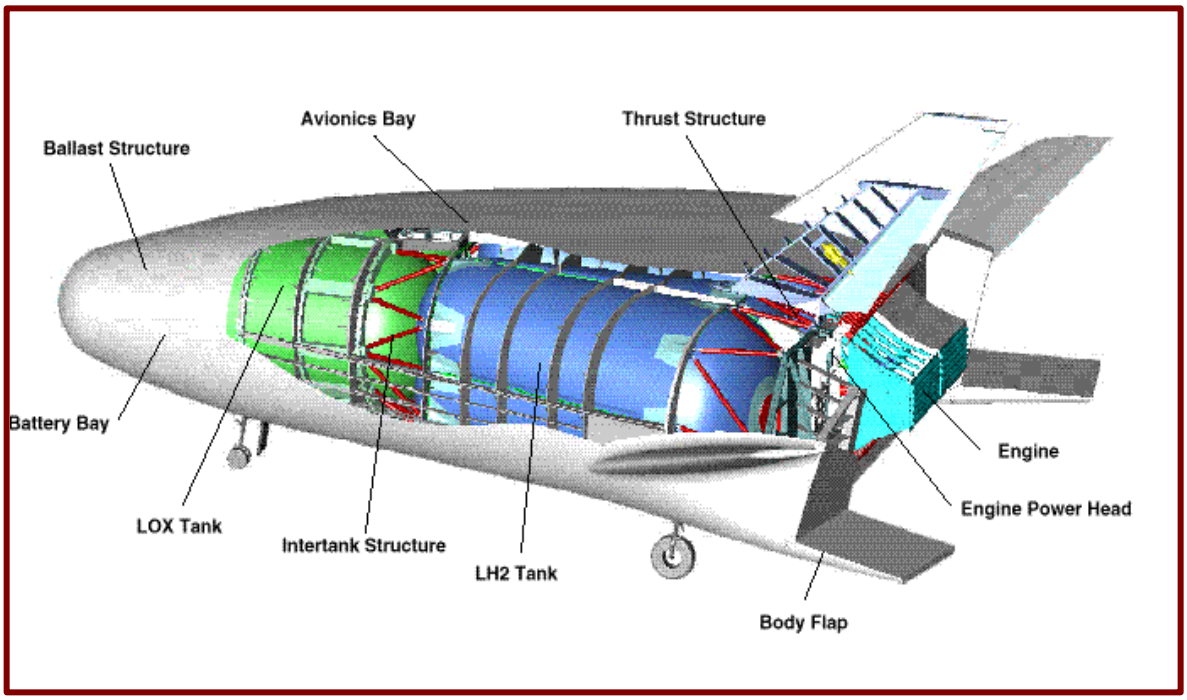
Figure 20. The state-of-the-art of NDE/I technology.

Leading edges / nose caps

- Refractory composites (TRL=9)
- Hot-structure control surfaces (TRL=5)
- High-temperature heat pipes (TRL=4)
- Active cooling (TRL=4)

Thermal protection system

- High-temperature metallics (TRL=5)
- Refractory composites (TRL=4)
- Rigid ceramics (TRL=9)
- Advanced flexible insulation (TRL=6)
- Validated thermal analysis/sizing (TRL=5)



Primary structure

- High-temperature polymeric composites (TRL=5)
- High-temperature metal composites (TRL=4)
- Noncircular composite shell structures (TRL=3)
- Joints and attachment techniques (TRL=4)
- Validated design and analysis methods (TRL=4)
- Nondestructive evaluation (TRL=4)
- Manufacturing technology (TRL=4)

Cryotanks

- Sandwich construction (TRL=4)
- Nonautoclave curing (TRL=3)
- Manufacturing scalability (TRL=4)
- Nondestructive evaluation (TRL=4)
- Vehicle health monitoring (TRL=3)
- Materials design allowables (TRL=4)
- Integrated TPS / cryo-insulation (TRL=2)

Figure 21. Assessment of technology needs for an RLV.

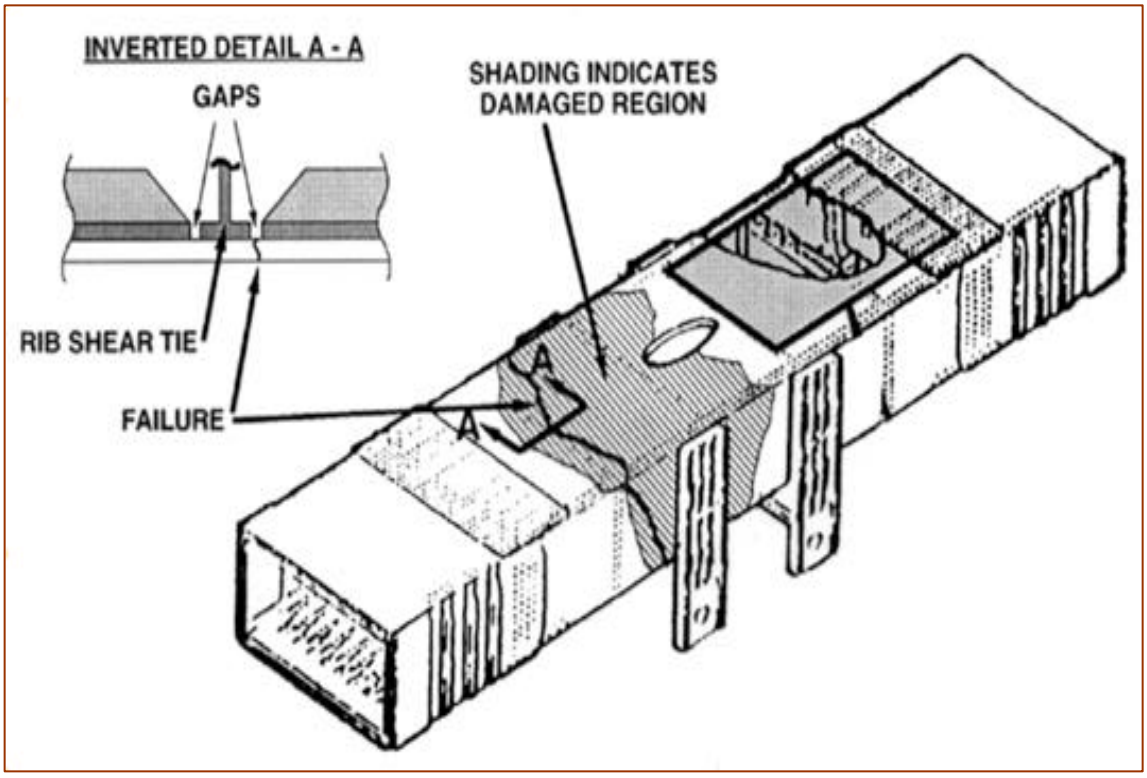
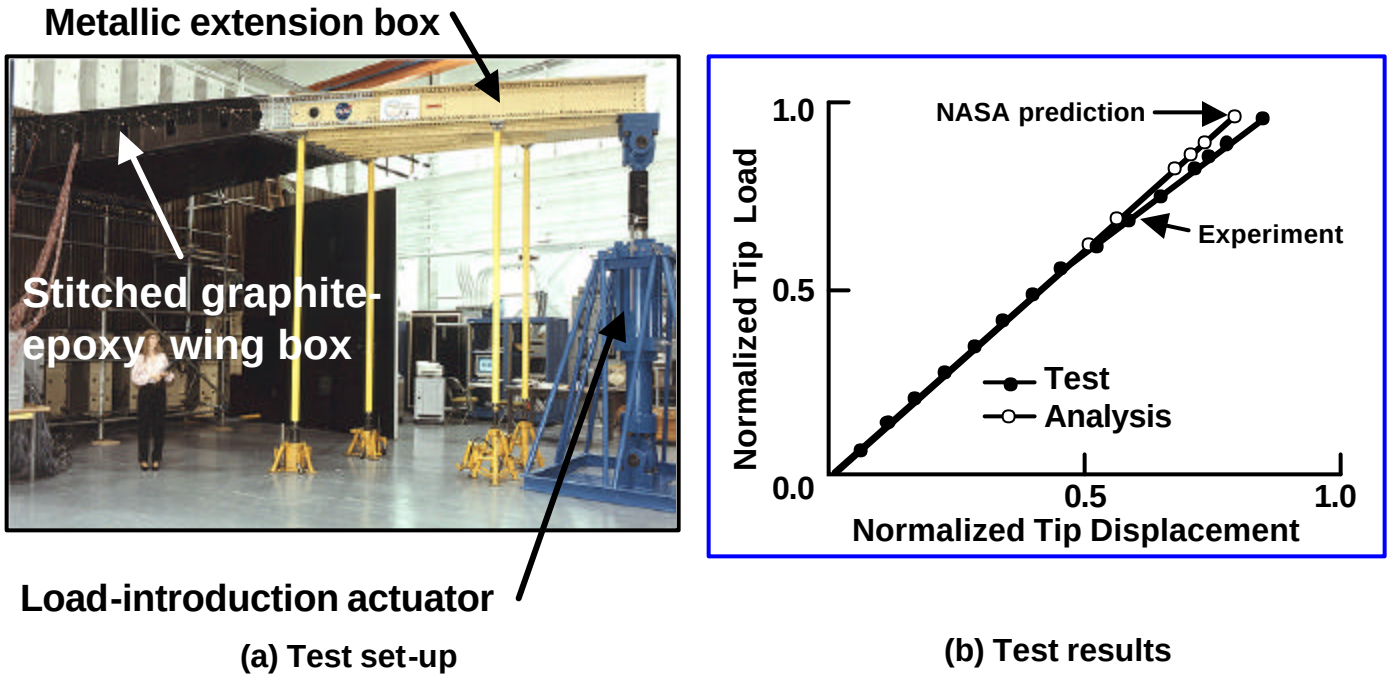


Figure 22. Technology integration box beam test.



(a) Test set-up

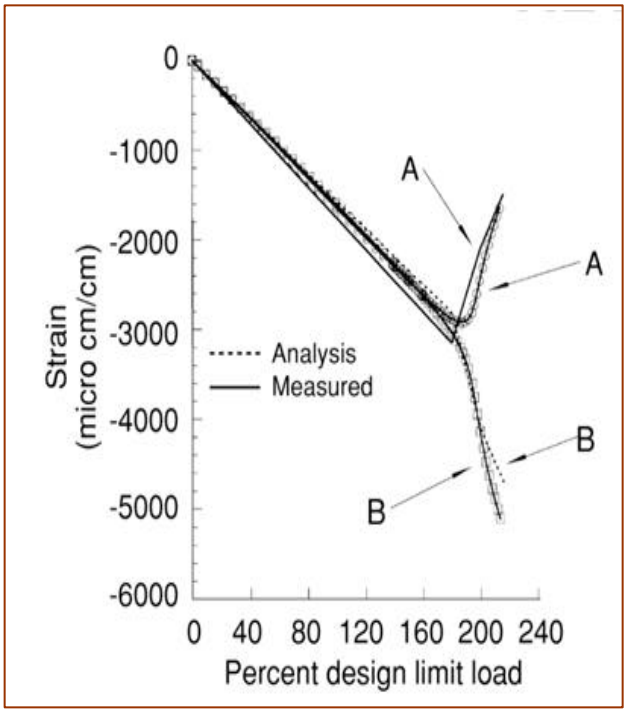
(b) Test results

Figure 23. Wing stub box test.



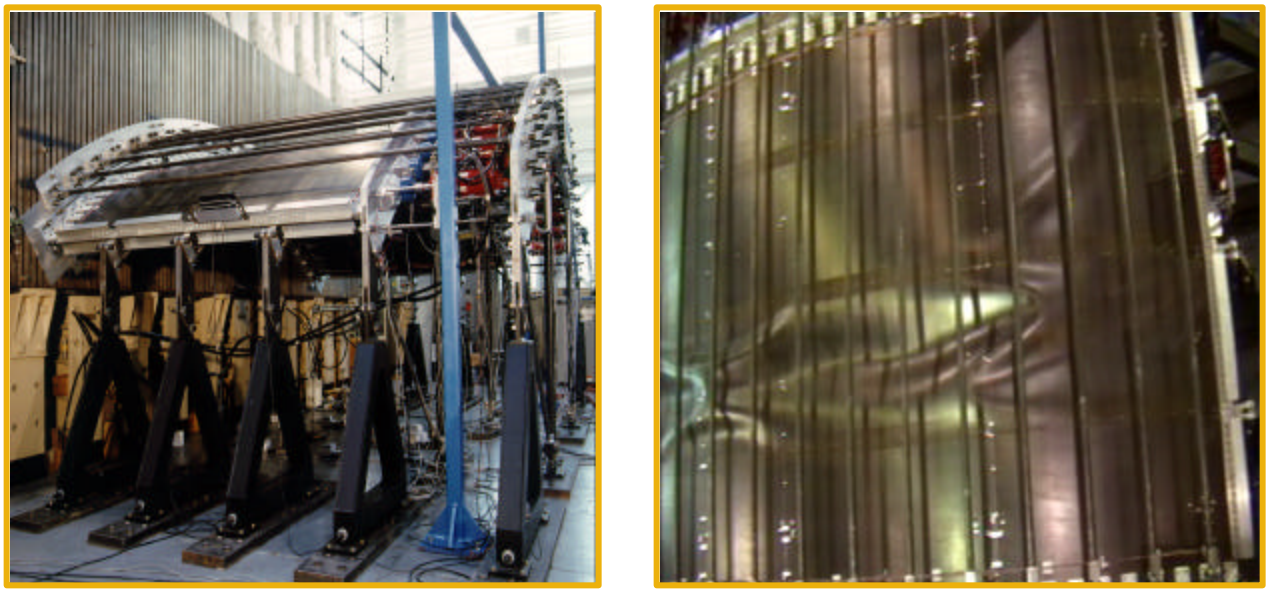
Figure 24. Test verifies cost-effective composite wing technology.





(a) Test set-up **(b) Test results**

Figure 25. Test verifies RLV wing box technology.



(a) Test set-up (b) Failed test article

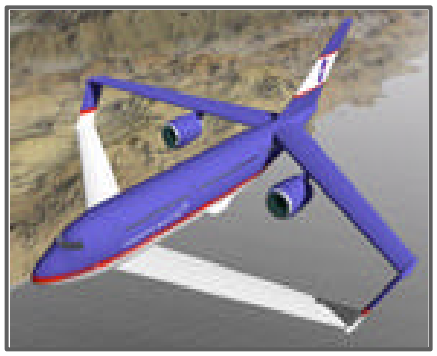
Figure 26. LaRC structural test of segment of an RLV intertank.



Figure 27. MSFC structural testing of RLV LH2 tank.



(a) Blended wing body

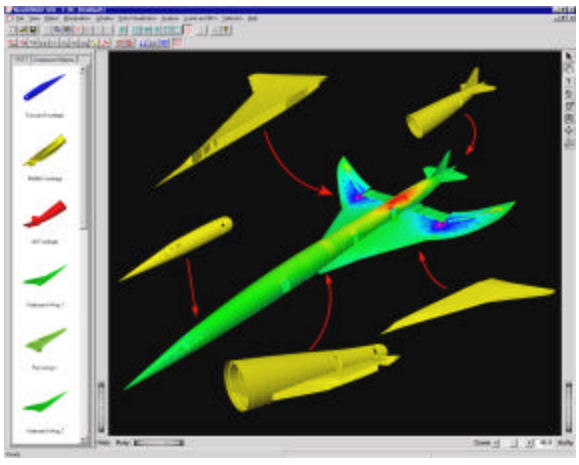


(b) Joined wing

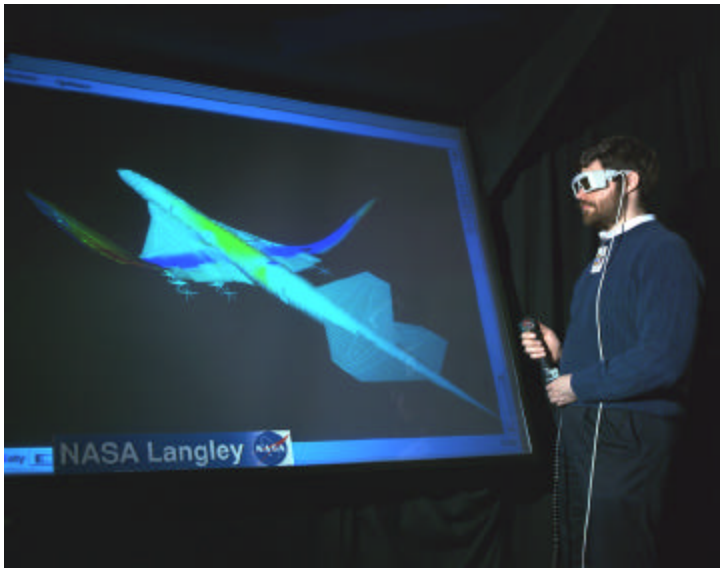


(c) Strut braced wing

Figure 28. Revolutionary airframe configurations.



(a) Next generation model assembly and graphical user interface



(b) 3-D stereo viewing of analysis results

Figure 29. Next generation structural design and analysis tools.

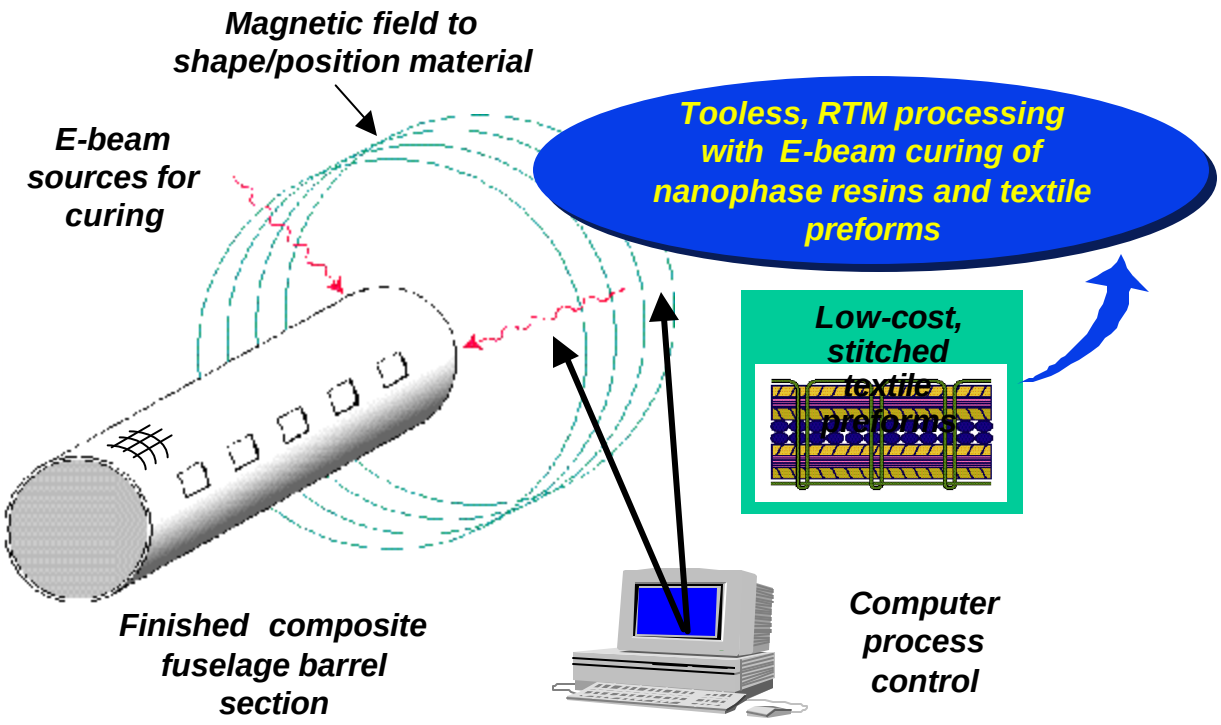


Figure 30. Low-cost composite materials processing of the future.

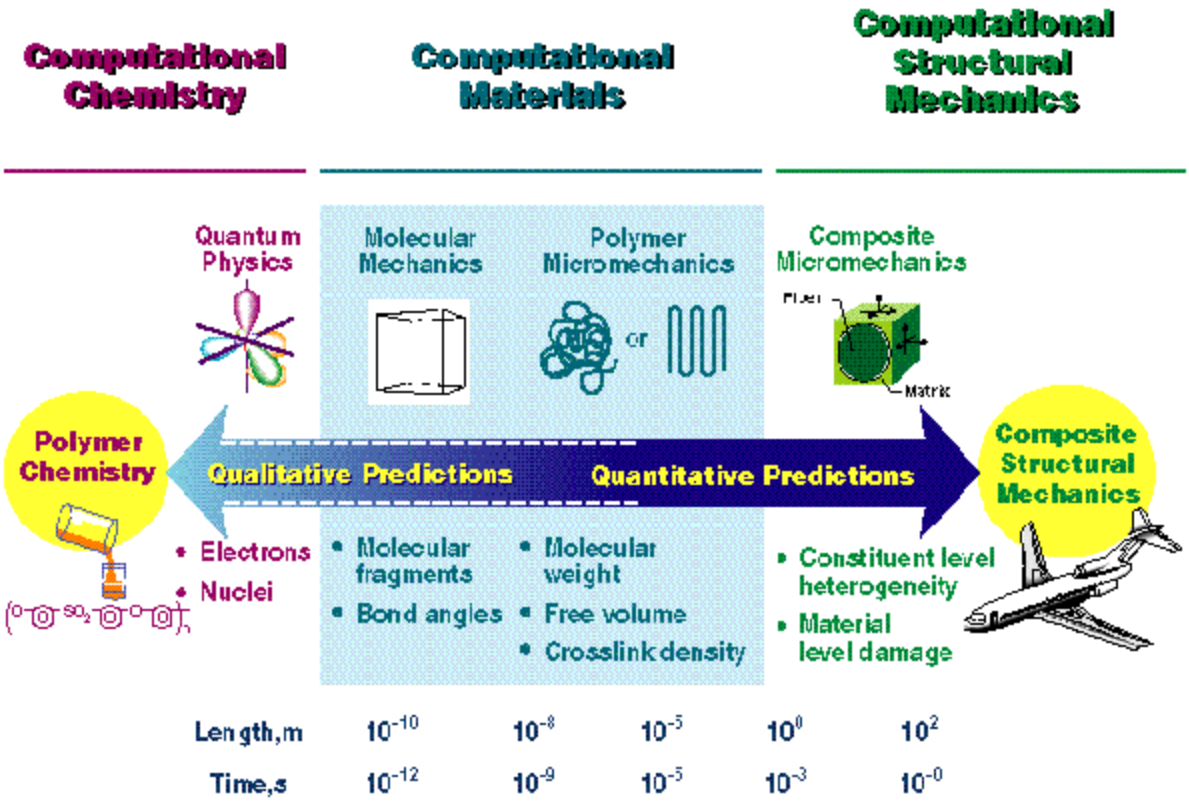


Figure 31. Computationally designed materials and structures.

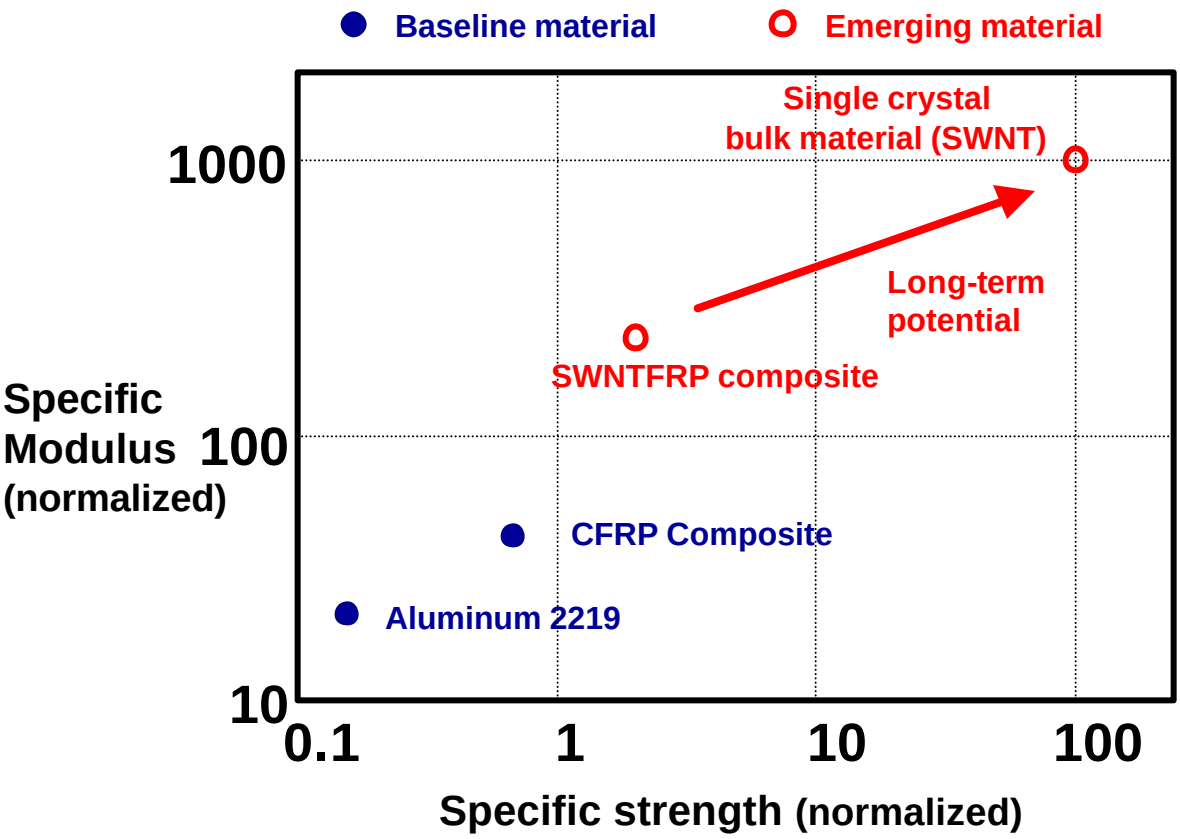
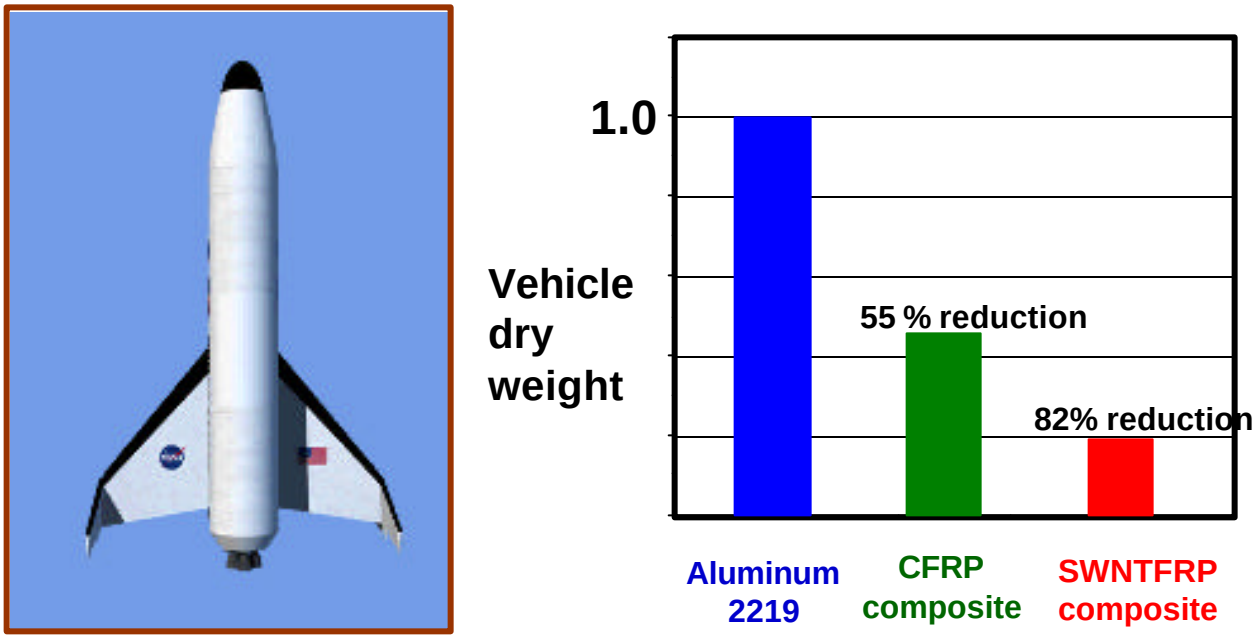


Figure 32. Properties of carbon nanotubes and composite materials.



(a) Reusable launch vehicle

(b) Systems analysis results

Figure 33 Systems analysis results for a reusable launch vehicle.

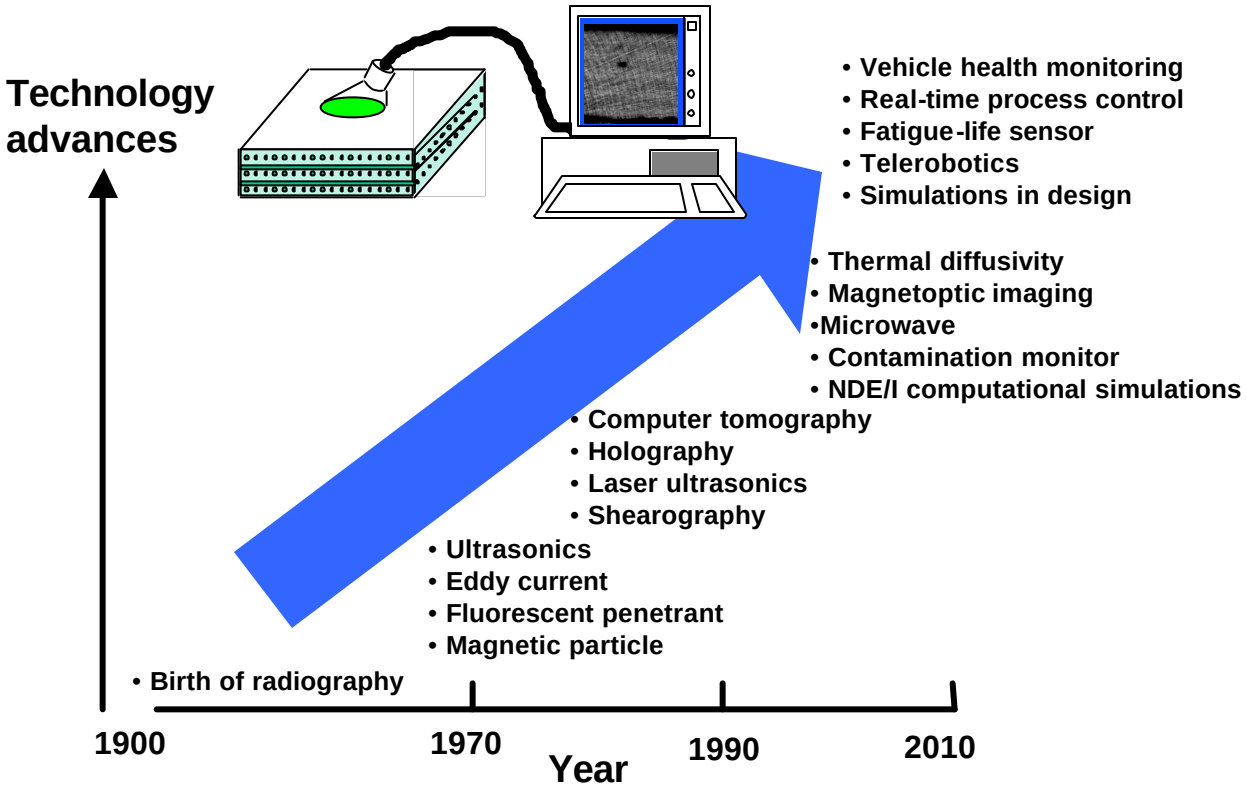


Figure 34. Evolution of composite materials NDE technology.

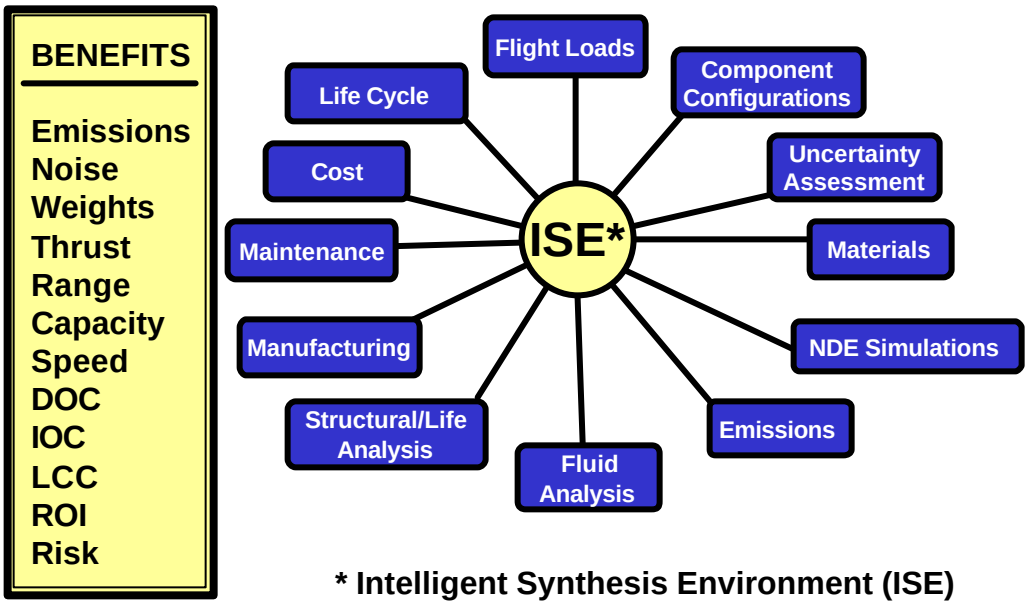


Figure 35. Collaborative engineering design environment.

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13. ABSTRACT (Maximum 200 words)
The results of an assessment of the state-of-the-art in the design and manufacturing of large composite structures are described. The focus of the assessment is on the use of polymeric matrix composite materials for large airframe structural components, such as those in commercial and military aircraft and space transportation vehicles. Applications of composite materials for large commercial transport aircraft, general aviation aircraft, rotorcraft, military aircraft, and unmanned rocket launch vehicles are reviewed. The results of the assessment of the state-of-the-art include a summary of lessons learned, examples of current practice, and an assessment of advanced technologies under development. The results of the assessment conclude with an evaluation of the future technology challenges associated with applications of composite materials to the primary structures of commercial transport aircraft and advanced space transportation vehicles.

14. SUBJECT TERMS
composite materials, graphite/epoxy, graphite, carbon, epoxy, thermosets, aircraft, rotorcraft, general aviation, rockets, wing, fuselage, empennage, NDE

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