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# Composite Airframe Cost Estimation Model Research: Report on Activity

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**Adedeji Badiru, PhD**
**J. Robert Wirthlin, PhD, Lt Col, USAF**
Department of Systems and Engineering Management

**Jeffrey D. Weir, PhD**
**M. Al-Romaihi, Col, Bahrain Air Force**
Department of Operational Sciences

Air Force Institute of Technology
2950 Hobson Way
Wright-Patterson AFB, OH 45433-7765

**Dr. Stephen Clay**
AFRL/RB
Wright Patterson AFB OH 45433

**Som Soni, PhD**
AdTech Systems Research, Inc.
1342 N. Fairfield Road, Beavercreek, OH 45432



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**1.0 Statement of Research Issues/Results:** Interest in estimating costs for aircraft using large composite structures is still in its infancy and there are no commonly accepted cost models for composite aircrafts. This lack of a universally agreed upon LCC model provides ample opportunities for further research into this area.

There are various manufacturing processes and parts counts associated with composites that are not currently incorporated in existing cost models used for procurement of aircraft systems comprising substantial composite materials. The proposed research is to improve the cost estimating models for composite material aircraft in comparison to historic metallic aircraft.

**2.0 Executive Summary:** The purpose of this overall research thrust is to improve the means for costing predominately composite material aircraft in comparison to historic metallic aircraft. There are significantly different manufacturing processes and parts count associated with composites that are not currently addressed with the procurement and life cycle management processes. Composite structure, when compared to metallic structure, is perceived to imply more risk than the associated cost advantages due to insufficient characterization of the life cycle benefits from optimal composite use. Their use has therefore been historically limited to components versus major structural assemblies. The goal of this research is to produce a validated cost model to predict the realistic cost of composite aircraft structures. The proposed research is a continuation of investigations carried out in 2009 and 2010 [1, 2] under sponsorship of AFRL/RB where effects of part count and labor touch hours on cost of composite aircraft were studied. In this effort, the proposed research will investigate the effect of realistic manufacturing process cost and total material cost on the cost estimates of composite aircraft.

**3.0 Statement of Work:** The emphasis on reducing defense related funding has been growing over the past twenty years, since the end of the Cold War. Though there was a spike in defense related funding after the September 11th terrorist attacks, a renewed focus has emerged from top congressional leaders that defense spending must decrease. This reduction in funding has caused military leaders to place an ever greater priority on the cost of major weapon systems. A leading philosophy behind many military scientists and aerospace officials is that composite materials can help lower the cost of military weapon systems. Composite materials are beginning to comprise an ever greater percentage of structural materials used in aircraft production. The increased usage of these materials has led several individuals within the Air Force community to revisit the life cycle cost (ACC) models that estimate the cost of weapon systems. The current life cycle cost models were developed when metals were the major material used in the production process. Since that time, actual data has been collected that shows that the current life cycle cost models may not be accurate when aircraft contain significant amounts of composite material.

The current life cycle cost models and procurement strategies do not take into account the different manufacturing techniques for composite materials. With the increased use of composite materials in aircraft production and the corresponding decrease in aircraft part count, the current cost models do not account for this potential cost savings due to reduced touch labor hours. Lack of research on the potential savings associated with reduced part counts due to the optimum use of composite materials has led consumers and industry officials to perceive composite use as more risky compared to use of traditional metallic materials. This perception has meant that the majority of composite use has been focused on component structures. Continuing research is leading prime contractors to investigate the possibilities of an increase in usage of composite materials.

Further, this lack of consideration leads to an inflated estimated life cycle cost when composites are incorporated into aircraft structures. This inflated estimated life cycle cost negatively impacts the average procurement unit cost (APUC), the procurement unit cost (PUC), and the cost per flying hour (CPFH) for structures containing a large percentage of composite materials. These estimated cost-ratios are some of the most important tools that decision makers use in determining whether to continue or start production of a new weapon system.

Although the primary use of composite materials has been for component parts, there are several arguments for large structural assemblies comprised of composite materials. Composites have several advantages over conventional aircraft production materials, including reduced weight, reduced number of fasteners, corrosion resistance, and an extended product life. In addition, composites can be designed specifically for certain aircraft parts to achieve desired stiffness and strength. This ability to custom design aircraft sections, key in the context of this research, reduces touch labor hours related to aircraft production and development. The main disadvantage and largest criticism of using composite materials is the raw materials cost.

**3.1 Objective of this Study:** The purpose of this research is to improve the method for evaluating life cycle cost of predominately composite material aircraft in comparison to metallic aircraft. The goal of this research is to modify the current life cycle cost model used by the Air Force community, which will better characterize the benefits and tradeoffs associated with composite aircraft development and production. The following are the research questions that this research will attempt to answer.

## 3.2 Research Questions

3.2.1 Does a relationship exist between reduced part counts and design, manufacturing and total material cost, design support, tooling, and testing costs?
3.2.2 If a relationship exists, how do we quantify that relationship?
3.2.3 If a relationship exists, how can the relationship be incorporated into current life cycle cost models?
3.2.4 How did the manufacturing process for the Advanced Composite Cargo Aircraft compare to the original manufacturing process in terms of touch labor hours?
3.2.5 What additional information is required?

## 4.0 Steps to Answering the Research Questions

Before a full-blown composite cost model can be developed, several avenues of research must be addressed. For instance, in order to answer question 3.2.1 above, several avenues of research need to be explored in order to develop a robust cost model. The following sub-sections represent our initial research results and constitutes the bulk of this final report.

## 4.1 Analysis of Z-Pinned Laminated Composites Fatigue Test Data:

***Introduction*: ** Because of their layered structure, polymer matrix composites (PMC’s) do not, in general, have the ability to deform plastically like metals, thus the energy absorption mechanism of composites is different from that of metals. In composites, energy is absorbed by matrix cracking and the creation of large fracture surfaces at the lamina interfaces, a phenomenon known as delamination. Delamination severely impairs the load-carrying capacity and structural integrity of composite structures and since composites naturally lack reinforcement in the thickness direction, delamination is a predominant failure mode. While composites have shown great promise achieving the performance and cost goals of future aircraft industry, their use may be limited by their susceptibility to delamination and the need to meet survivability requirements. Advanced processing techniques, interlaminar reinforcement technologies and innovative design concepts have been developed in recent years and provide significant improvements towards achieving survivable, all-composite structures while minimizing any increase in weight and cost. At the present time several 3D technologies are under investigation toward this end, namely: stitching, tufting, 3D weaving and z-pinning.

[2] describes the results of a combined experimental and analytical study to:

- Investigate mode I, mode II and mixed mode failure response of various composite specimen geometries with through-thickness reinforcement, and
- Verify the DYNA3D smeared property finite element model developed by Adtech Systems Research Inc. (ASRI) by comparing simulation and experimental results.

In references [2, 3, 4] specimen geometries tested include: T-section (T-SEC) components as well as double cantilever beam (DCB) specimens each with and without through-thickness reinforcement. Experiments were conducted “in-house” under low strain rate loading conditions using ASRI and AFRL test facilities.

***4.1.1 Problem Description*: ** The goal of this research work is to understand the fatigue response of co-cured composite laminate specimens with and without z-pin reinforcement. Table 1 shows representative z-pin configurations. For clarity, test data tables contain the specific details of configurations considered. The following parameters are considered:

- Test 9’’x1” specimens reinforced with 0.011” & .02” diameter Z-pins
- Compare response of 0.02” diameter z-pin reinforcement to that of 0.011” diameter z-pins
- Investigate the influence of maximum load as % of ultimate static strength of the laminate without z-pin fibers

Table 1: Z-pinned Specimen Configurations

| Configuration Type | Diameter of Z-pin | % of Reinforcement | Area of Z-pinning |
|---|---|---|---|
| A | 0.011 inch | 2.0 | 1 inch x 1 inch |
| B | 0.011 inch | 4.0 | 1 inch x 1 inch |
| C | 0.020 inch | 2.0 | 1 inch x 1 inch |
| D | 0.020 inch | 4.0 | 1 inch x 1 inch |
| E | 0.011 inch | 2.0 | 2 inch x 1 inch |
| F | 0.011 inch | 4.0 | 2 inch x 1 inch |
| G | 0.020 inch | 2.0 | 2 inch x 1 inch |
| H | 0.020 inch | 4.0 | 2 inch x 1 inch |

***4.1.2 Experiments and Data Analysis*:** We started conducting tests of specimens with 1" end tab material. To start we made 18 specimens. It was observed that specimens started failing at the end tab. To make the best use of the machined specimens, we decided to use those specimens by including stress raisers in the form of a hole (in case of laminate without z-pins) or three holes in the case of laminate with z-pins at appropriate locations. That helped avoid failure of specimens at end tabs. Tests provided additional insight in to the failure mechanisms in the Z-pin area or without z-pin area. Figure 1 shows the classes of parameters considered including stress raisers (hole diameters .1" and .2") and z-pin diameter (.011", .02"). Other parameters considered were % ultimate loads (60, 70 and 80%) and z-pin surface area (2% and 4%) [5]. All these parameters influence the lifecycle (number of cycles to failure) of the laminate.

For illustration purposes, Figure 2 shows the specimen after fatigue loading and failure of one specimen. The reader can easily visualize the specimen before loading. Each test was conducted by using 20 kips servo hydraulic test system using different loads with R=.1 and 4 hertz frequency. In order to understand the damage progression in the specimens at different cycles; specimens were fatigued till a specific number of cycles, unloaded and x-rayed. From the x-ray images a percent damage area was calculated by using imaging software developed by the Department of Health Researches [6]. Resulting cycle and load dependent damage data is given in Tables 2 and 7 below. In Table 2, Na, Nb and Nc represent numbers of cycles at which specimens were unloaded and x-rayed. Damage % area for each unloading is given in the next column. Tables 3-8 show other results of experiments conducted for fatigue loading of specimens with different parameters.

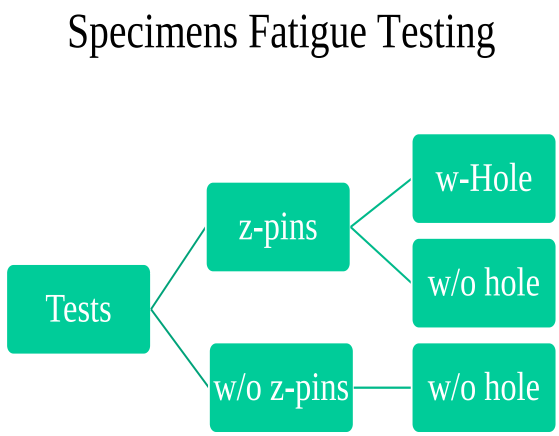



Figure 1:  Specimen Fatigue tests considered in this investigation

Figure 2: Specimen in The test fixture after fatigue failure.

Table 2: Example data for 9 specimens tested at different loads for given cycles and % damage till total failure

| Sample ID | Max Stress (ksi) | % Ultimate | Cycles $N_a$ | Damage Area a (%) | Cycles $N_b$ | Damage Area b (%) | Cycles $N_c$ | Damage Area c (%) |
|---|---|---|---|---|---|---|---|---|
| 5-1 | 92 | 80 | 8,907 | 100 | | 100 | | |
| 5-2 | 92 | 80 | 20,453 | 44 | 21,453 | 100 | | |
| 5-3 | 92 | 80 | 18,404 | 100 | | 100 | | |
| 5-4 | 80.5 | 70 | 188,636 | 64 | 253,931 | 90 | 267,122 | 100 |
| 5-5 | 80.5 | 70 | 187,376 | 65 | 202,224 | 61 | | |
| 5-6 | 80.5 | 70 | 242,378 | 62 | 251,428 | 61 | 296,450 | 100 |
| 5-7 | 69 | 60 | 328,026 | 53 | 407,688 | 60 | | |
| 5-8 | 69 | 60 | 121,710 | 37 | 246,552 | 61 | 311,000 | 100 |
| 5-9 | 69 | 60 | 242,165 | 47 | 325,148 | 63 | 400,000 | 100 |

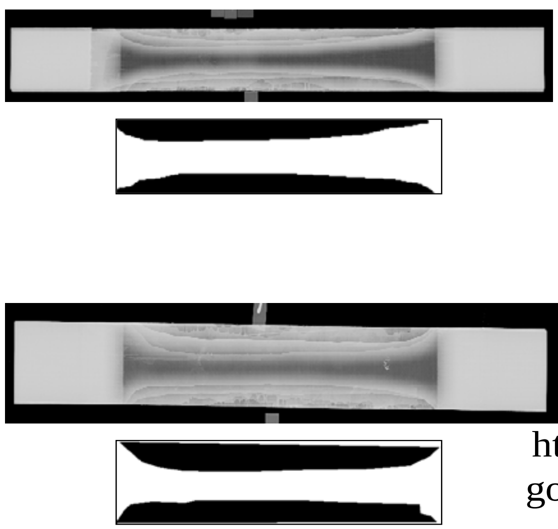
Table 3: Specimens with different loading conditions and cycles to failure and failure mode .011”, 4%, 1X1, 2T

| Sample ID | Stress Ksi | Nf Cycles | Failure Mode | Hole |
|---|---|---|---|---|
| B2-1 | 87.8 | 63,568 | Delam | No hole |
| B2-2 | 87.8 | 15,006 | Hole- Zpin | .2" dia |
| B2-4 | 87.8 | 40,565 | Hole- Zpin | .2" dia |
| B2-5 | 87.8 | 3,863 | Hole- Zpin | .2" dia |
| B2-6 | 87.8 | 15,120 | Hole- Zpin | .2" dia |
| B2-7 | 87.8 | 7,834 | Hole- Zpin | .2" dia |
| B2-8 | 87.8 | 378 | Tab | |

1X1 represent 1"x1" z-pin area, 2T represent z-pin area starts at 2" away from Tab both sides of the specimen.

Table 4: Specimens with different loading conditions and cycle to failure and failure mode, without Z-pins

| Sample ID | Max Stress Ksi | Nf Cycles | Failure Mode | Hole |
|---|---|---|---|---|
| XB-1 | 92 | 1,836 | Delam | N/A |
| XB-2 | 80.5 | 35,250 | Delam | N/A |
| XB-3 | 80.5 | 41,963 | Delam | N/A |
| XB-4 | 80.5 | 32,298 | Delam | N/A |
| XB-5 | 69 | 53,589 | Delam | N/A |
| XB-6 | 69 | 100,946+ | Delam | N/A |
| XB-7 | 69 | 122,031+ | Delam | N/A |
| XB-8 | 69 | 122,138+ | Delam | N/A |
| XB-9 | 80.5 | 32,729 | Delam | N/A |



XB-2

XB-7

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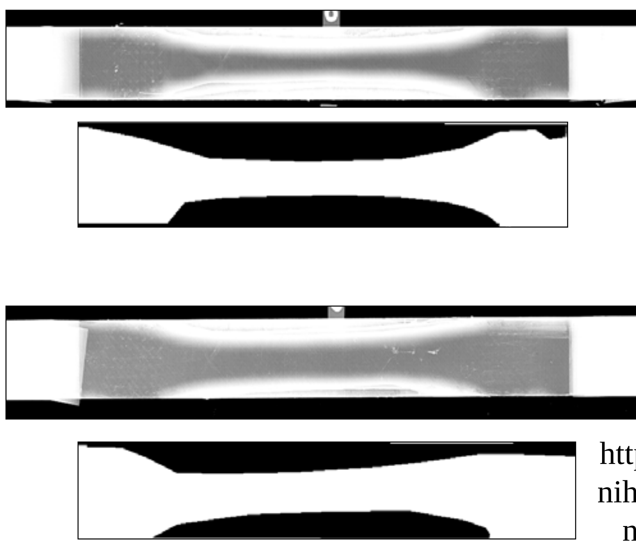
Figure 3: X-Ray of samples XB-2 & XB-7 with corresponding damage image computed using [6].

Table 5: Specimens with different loading conditions and cycle to failure and failure mode, , .02”, 2%, 1X1, 2T

| Sample ID | Max Stress Ksi | Nf Cycles | Failure Mode | Hole |
|---|---|---|---|---|
| XC1-1 | 69 | 100,500+ | Delam | N/A |
| XC1-2 | 69 | 111,005+ | Delam | N/A |
| XC1-3 | 69 | 109,931+ | Delam | N/A |
| XC1-4 | 69 | 114,108+ | Delam | N/A |
| XC1-5 | 69 | 121,669+ | Delam | N/A |
| XC1-6 | 80.5 | 112,861+ | Delam | N/A |
| XC1-7 | 80.5 | 160,621+ | Delam | N/A |
| XC1-9 | 80.5 | 114,821 | Delam | N/A |



XC1-6

XC1-9

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Figure 4: X-Ray of samples XC1-6 & XC1-9 with corresponding damage image computed using [6].

Table 6: Test Results summary

| Specimen # | Stress ksi | Nf Cycles Mean | Median | St Dev | Failure Mode |
|---|---|---|---|---|---|
| B2-1 to B2-8: | 87.8 | 16,477 | 15,006 | 14,300. | Hole-Z-pin |
| C2-1 to C2-4: | 86.0 | 283,658 | 294,845 | 90,263 | Center Ho |
| C2-6 to C2-9: | 87.8 | 70,286 | 58,063 | 39,203 | Hole-Z-pi |
| H1-2,7 & I1-1,2 | 92 | 2,388 | 2,595 | 2,162 | Failure near Z re |
| H1-3,6 &I1-3,4 | 80.5 | 6,745 | 6,653 | 5,282 | Same |
| H1-4,5 & I1-5,6 | 69 | 75,443 | 68,947 | 63,926 | Same |
| H2-5 to H2-9: | 84 | 95,327 | 91,541 | 13,313 | Hole-Z-pin |
| H2-5 | | censored at 113508 Cycles | | | |

Table 7: Damage Area for specimens with or with Z-pins

| Z-Pins | % Ultimate | Cycles | Damage Area |
|---|---|---|---|
| No | 60 | 328026 | 53 |
| | | 121710 | 37 |
| | | 242165 | 47 |
| | | 407688 | 60 |
| | | 246552 | 61 |
| | | 325148 | 63 |
| | | 100946 | 23 |
| | | 122031 | 50 |
| Yes | | 114108 | 3 |
| | | 121669 | 5 |
| No | 70 | 188636 | 64 |
| | | 187376 | 65 |
| | | 242378 | 62 |
| | | 253931 | 90 |
| | | 202224 | 61 |
| | | 251428 | 61 |
| | | 35250 | 43 |
| | | 41963 | 35 |
| Yes | | 112581 | 43 |
| | | 160621 | 51 |
| No | 80 | 8907 | 100 |
| | | 20453 | 44 |
| | | 18404 | 100 |
| | | 21453 | 100 |

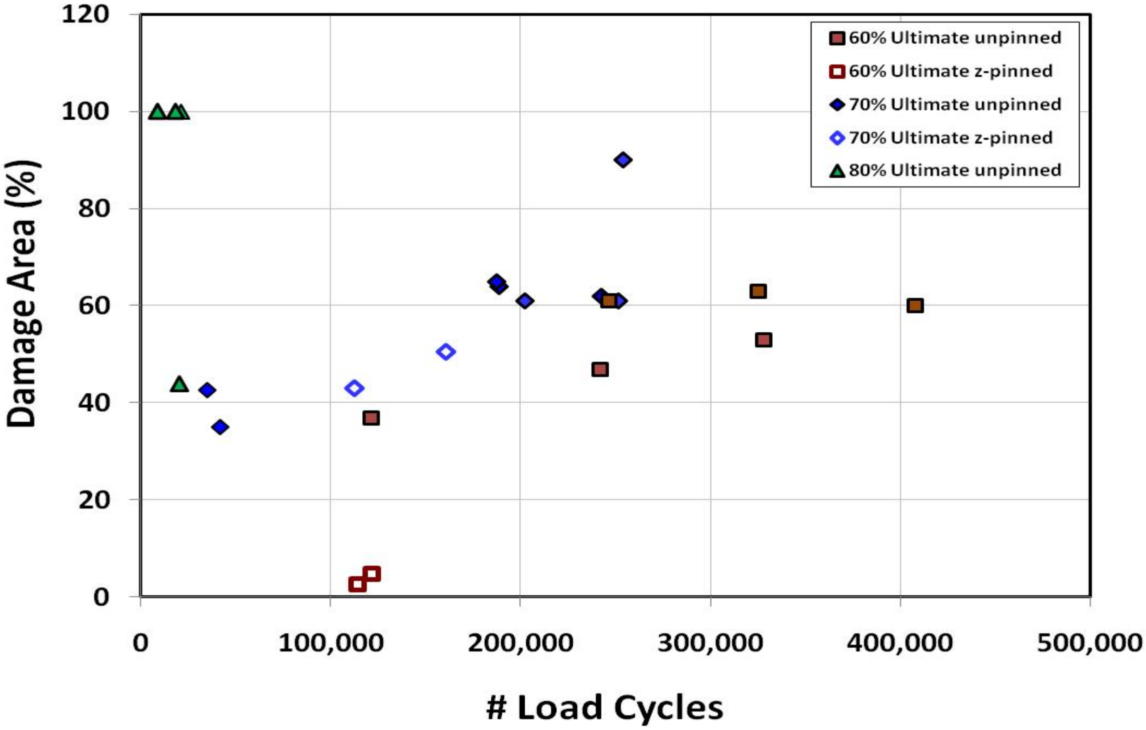


Figure 5: Damage Area in Specimens Tested at Different Cycles and % of Ultimate Load.

A set of specimens consisting of Series H1, J1 and I1 were tested for 80, 70, 60 and 50% of ultimate strength. Because of limited resources, we had tested 9 specimens in each class.

The following data (Table 8) for 60 and 50% of ultimate strength (i.e. 69 ksi and 57.5 ksi) speak that there is a benefit of using J1 configuration of specimen which uses .011" diameter and 2% z-pins.

Table 8: Fatigue cycles for H1, J1 and I1 specimen series

| Load | Specimens | Corresponding Cycles |
|---|---|---|
| 69 ksi | H1-4; 5; 9 | 1,845; 45,250; 66,508 |
| | J1-5; 6 | 161,150+; 181,150+ |
| | I1-5; 6; 7 | 71,386; 159,653; 80,515 |
| | 5-7; 8; 9 | 328,026+; 121,710+; 242,165+ |
| 57.7 ksi | H1-8 | 320,842 |
| | J1-7; 8 | 750,000+; 763,000+ |
| | I1-8; 9 | 458,219+; 563,760+ |

+ Represents the specimen censored at this cycle number.

This is work in progress and additional results will be provided in forthcoming publications[1].

[1] These subsections were originally published in Soni, S.R., Al-Romaihi, M., Wirthlin, J.R., Badiru, A.B., Weir, J.D. (2012) Analysis of Z-Pinned Laminated Composites Fatigue Test Data, *Proceedings of the International Conference on Agile Manufacturing-2012, Uttar Pradesh, India, December 16-19, 2012.*

**4.1.3 Future Research:** Development of additional data using Monte Carlo techniques is on-going. These further tests will be done to prove that the random data generated by Monte Carlo technique correctly represents the additional experimental test data. Further, we plan to investigate the cost implications of using Z-pins in composite airframe structure life cycle estimates.

**4.2 Doctoral Candidate Work:** Additional research is being continued by a Doctoral candidate at AFIT, Col. M. Al-Romaihi.

**5.0 Notes on the Research Team and Tentative Schedule:** Col M. Al-Romaihi, Dr. Jeffery D. Weir, Dr. Adedeji B. Badiru, Dr. Stephen Clay, LtCol. Joseph Wirthlin, and Dr. Som R. Soni all have contributed to and continue work on this research project.

The anticipated timeline of completion for this research is early 2014.

**6.0 Additional Research Emphasis:** As indicated earlier, much research remains. This is despite the fact that composite materials have been used in aircraft manufacturing for numerous years. As research continues in the area of composite aircraft, an area that requires additional research is the effects of automation on cost. Fiber placement machines are frequently being integrated into the manufacturing process to improve the efficiency of composite manufacturing in production scenarios. Further research is required to determine if a learning curve is present with the incorporation of fiber placement machines. Other research that is needed concerns the material cost factors currently used in cost models concerning composites. These material cost factors were developed by RAND in the early 1990's and have not been updated since that time. These efforts will lead to a more vigorous and accurate cost model that can aid the decision maker in determining the trade-offs in acquiring aircraft systems.

**7.0 Deliverables:** This report represents the first phase of this research. Other deliverables include scholarly publications and at least one dissertation.

**8.0 Publication and Presentation Plan:** Two to three scholarly papers in 2013 in preparation for a dissertation. A prospectus has already been authored and is undergoing review. The dissertation is expected in 2014.

**9.0 Long Term Acquisition Related Research:** Continued refinement of the composite cost model will need to occur over time, especially as stakeholder experience with composite aircraft increases. We envision this to be a rich field for continuing research.

**10.0 Statement of Research Status:** On-going as part of a long-term effort

**11.0 Current and Pending Project and Proposal Submission:** No additional projects and proposals are envisioned at this time. Based upon additional research results will determine the scope and number of additional submissions.

**12.0 References:**


1. Russell, John. "Composites Affordability Initiative: Transitioning Advanced Aerospace Technologies through Cost and Risk Reduction," *AMMTIAC Quarterly*, 1: 3-6 (2006).

2. Jason Freels, "Modeling Fracture in Z-Pinned Composite Co-Cured Laminates Using Smeared Properties and Cohesive Elements in DYNA3D", M.S. thesis, AFIT/GMS/ENY/06-S01, Air Force Institute of Technology, Wright Patterson AFB, Sept. 2006.

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4. Som Soni, Jason Freels and Jeff Kuhn, "Failure in Z-Pinned Co-cured Laminated Composites", SAMPE conference in Memphis, TN, Sept 7-11, 2008

5. S. Soni, M. Al-Romaihi, J.R. Wirthlin , A.B. Badiru, and S. Clay, "Reliability Aspects in Z-Pinned Composites", IEOM-2012, Istanbul, Turkey, July 5, 2012.

6. http://rsb.info.nih.gov/ij/download.html


About the authors:

**M. Al-Romaihi** is a Colonel in Bahrain Air Force. He is a PhD student in Department of Operational Sciences at the Air Force Institute of Technology. His specific area interest is cost modeling of composite aircraft structures. He obtained his M.S from Naval Post Graduate School, Monterey, CA.

**Jeffery D. Weir** is an associate professor in the Department of Operational Sciences at the Air Force Institute of Technology. He has a PhD in industrial and systems engineering from Georgia Tech. He teaches courses in decision analysis, risk analysis, and multi-objective optimization. His research interests are in the areas of decision analysis and transportation modeling. A former officer in the United States Air Force, he has worked on a wide variety of projects ranging from scheduling and routing aircraft, determining the value of future intelligence information, assessing the impact of FAA regulation changes to passenger and aircrew safety, and mode selection for multi-modal multi-commodity distribution networks. He has received grants from the Defense Intelligence Agency, United States Transportation Command, Air Force Materiel Command, the Joint Improvised Explosive Device Defeat Organization, Air Force Research Laboratory and Pacific Northwest National Laboratory, amongst others. Address: Department of Operational Sciences, Air Force Institute of Technology, Dayton, OH 45433; email: jweir@afit.edu

**Adedeji B. Badiru**, PhD, PE, PMP is Professor and Head of the Department of Systems & Engineering Management at the Air Force Institute of Technology, Dayton, Ohio. He was previously professor and department head of Industrial & Information Engineering at the University of Tennessee in Knoxville. Prior to that, he was professor of industrial engineering and Dean of University College at the University of Oklahoma. He is a registered professional engineer, a Fellow of the Institute of Industrial Engineers, and a Fellow of the Nigerian Academy of Engineering. He has BS in Industrial Engineering, MS in Mathematics, and MS in Industrial Engineering from Tennessee Technological University, and Ph.D. in Industrial Engineering from the University of Central Florida.. He is the editor of the Handbook of Industrial & Systems Engineering. He is a member of several professional associations including Institute of Industrial Engineers (IIE), Institute for Electrical and Electronics Engineers (IEEE), Society of Manufacturing Engineers (SME), Institute for Operations Research and Management Science (INFORMS), American Society for Engineering Management (ASEM), American Society for Engineering Education (ASEE), New York Academy of Science (NYAS), and Project Management Institute (PMI). He has served as a consultant to several organizations around the world. He has won several awards for his teaching, research, publications, administration, and professional accomplishments. He holds a leadership certificate from the University Tennessee Leadership Institute. Prof. Badiru has served as a Technical Project Reviewer, curriculum reviewer, and proposal reviewer for several organizations including The Third-World Network of Scientific Organizations, Italy, National Science Foundation, National Research Council, and the American Council on Education. He has more than 100 publications, books, peer reviewed journal papers, proceeding papers in his field of interest. Dr. Badiru has won numerous prestigious awards.

**J. Robert Wirthlin,** PhD, LtCol, USAF, is an Assistant Professor of Engineering Systems at the Air Force Institute of Technology. He holds a PhD and MS from the Massachusetts Institute of Technology, and a BS from the United States Air Force Academy. He is currently the Associate Head, Strategic Communications for the Department of Systems and Engineering Management. He is a member of the faculty of the Graduate Engineering Management Program and in the Systems Engineering graduate degree program at AFIT. His research interests include new product development of complex systems, risk, and application of lean to product development. Before his arrival at AFIT, Lt Col Wirthlin has led systems engineering efforts and program management of new and operational systems at Hill AFB, Los Angeles AFB, and Buckley AFB. He is a Certified Systems Engineering Professional and is an active member of INCOSE, AIAA, and the Design Society.

**Dr. Stephen Clay** is a senior aerospace engineer for the Air Force Research Laboratory. He earned a B.S. (1992) in Mechanical Engineering from the West Virginia Institute of Technology and an M.S. (1995) and Ph.D. (2000) in Engineering Mechanics from Virginia Polytechnic Institute. His research areas include advanced composite structures, structural joining, and progressive failure analysis of composites. Dr. Clay is an active member of the American Institute of Aeronautics and Astronautics (AIAA) and the American Society of Mechanical Engineers (ASME).

**Som R. Soni, PhD,** Associate Professor of Systems Engineering (Retired), Air Force Institute of Technology, Wright Patterson AFB OH 45433, BS (Hons), Punjab University, 1967; MS & PhD, University of Roorkee (renamed as IIT Roorkee) India, 1969 and 1972 respectively. Dr. Soni is involved in teaching and research related to systems engineering design, analytical and experimental mechanics of composite materials and structures. Before joining as an Associate Professor in Air Force Institute of Technology, Dr. Soni was involved in AdTech Systems Research Inc as President and CEO for more than 25 years (1984- 2005). He has worked as a Senior Scientist for the University of Dayton Research Institute (1981- 1984) conducting research in composite materials and structures. Dr. Soni is author/ co-author of 80+ research publications in the field of mechanics of solids and structures with special emphasis on composites. Dr. Soni is a Fellow of the American Society for Composites, an Associate Fellow of AIAA and a member of SAMPE.

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