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WL-TR-94- 4083
Volume I

CONFERENCE ON AEROSPACE TRANSPARENT MATERIALS AND ENCLOSURES
VOLUME I - SESSIONS I-IV

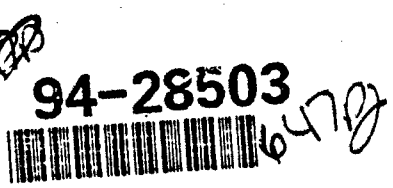
Compiled by Samuel A. Marolo

March 1994

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Conference Report for Period 9-13 August 1993

94-28503



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REPORT DOCUMENTATION PAGE

Form Approved
OMB No 0704 0188

Public reporting burden for this collection of information is estimated to average 1 hour per response, including the time for reviewing instructions, searching existing data sources, gathering and maintaining the data needed, and completing and reviewing the collection of information. Send comments regarding this burden estimate or any other aspect of this collection of information, including suggestions for reducing this burden, to Washington Headquarters Services, Directorate for information Operations and Reports, 1215 Jefferson Davis Highway, Suite 1204, Arlington, VA 22202 4302, and to the Office of Management and Budget, Paperwork Reduction Project (0704 0188), Washington, DC 20503

1. AGENCY USE ONLY (Leave blank)	2. REPORT DATE	3. REPORT TYPE AND DATES COVERED
	March 1994	Interim 9 AUG-13 AUG 1993

4. TITLE AND SUBTITLE
Conference on Aerospace Transparent Materials and Enclosures, Volume I: Sessions I-IV

5. FUNDING NUMBERS
F33615-92-C-3400

6. AUTHOR(S)
Compiled by Samuel A. Marolo

7. PERFORMING ORGANIZATION NAME(S) AND ADDRESS(ES)
University of Dayton Research Institute
300 College Park Avenue
Dayton, Ohio 45469-0110

8. PERFORMING ORGANIZATION REPORT NUMBER

9. SPONSORING/MONITORING AGENCY NAME(S) AND ADDRESS(ES)
Flight Dynamics Directorate (WL/FIVR)
Wright Laboratory, Materiel Command
Wright-Patterson Air Force Base, OH 45433-6533

10. SPONSORING/MONITORING AGENCY REPORT NUMBER
WL-TR-94-4083

11. SUPPLEMENTARY NOTES

12a. DISTRIBUTION/AVAILABILITY STATEMENT
APPROVED FOR PUBLIC RELEASE; DISTRIBUTION UNLIMITED

12b. DISTRIBUTION CODE

13. ABSTRACT (Maximum 200 words)

The purpose of this report is to make available the technical papers presented at the Sixteenth Conference on Aerospace Transparent Materials and Enclosures. Sixty-seven technical papers are presented in nine sessions that address transparent material for enclosures, coatings for transparencies, transparency design; bird impact resistance; human factors and optics; operational problems; design criteria on transparent plastics, glasses and elastomers; aircraft-structural integration of windshields and canopies; computed design; testing techniques; and cost of ownership reduction. The papers contained herein have been reproduced directly from the original manuscripts.

14. SUBJECT TERMS injection molding, optics, photochromics, polycarbonate, acrylic, interlayers, windshields, canopies, coatings, transparent materials, environmental resistance, computed design, design criteria, cost of ownership reduction, frameless trans-

15. NUMBER OF PAGES
647

16. PRICE CODE

17. SECURITY CLASSIFICATION OF REPORT	18. SECURITY CLASSIFICATION OF THIS PAGE	19. SECURITY CLASSIFICATION OF ABSTRACT	20. LIMITATION OF ABSTRACT
UNCLASSIFIED			

NSN 7540-01-280-5500

Standard Form 298 (Rev 2 89)
Prescribed by ANSI Std Z39 18
298 102

PREFACE

This report was prepared by the Materials Engineering Branch under Project 1926, "Aircraft Windshield Development." It was administered under the direction of the Wright Laboratory, Materials Directorate and Flight Dynamics Directorate, Air Force Material Command.

The technical papers enclosed in this report were presented at the Flight Dynamics/Materials Directorate "Conference on Aerospace Transparent Materials and Enclosures," held at the Sheraton Harbor Island Hotel, San Diego, California, on 9-13 August 1993. Mr. Samuel A. Marolo served as Conference Chairperson.

There were many who contributed to this very successful and timely conference which covered enhanced enclosure materials and advanced design technology. Special thanks are expressed to Mr. Russell E. Urzi, WL/FIVR, for providing a superior performance as Conference Technical Chairperson and as Conference Chairperson for Systems Technology. Gratitude and appreciation are expressed to Mr. Theodore J. Reinhart, WL/MLSE; Mrs. Louise Farrer, Mrs. Mary Wright, and Ms. Evie Beyers, all of the University of Dayton; and Mrs. Nancy Holland and Mrs. Christine Kindle, Wright Laboratory, for the outstanding job accomplished as Chairperson for Materials Technology, Conference Administration, Conference Secretary, Administrative Assistant, and Conference Aides, respectively. Gratitude is also expressed to Lt. David Hammershock, WL/FIVR, who served as Conference Display/Exhibit Chairperson.

Special appreciation and thanks are expressed to Mr. Ralph J. Speelman, WL/FIVR, for his timely help and contribution to the success of the Conference.

Appreciation and thanks are expressed to Col Robert L. Herklotz, Director, Flight Dynamics Directorate, and Dr. Vincent J. Russo, Director, Materials Directorate, for their support of the Conference and their expressed support of this technical area. Gratitude is also expressed to Col Herklotz and Lt Col Bruce Thompson, Deputy Chief of Fighter Requirements for Power Projection at HQ Air Combat Command, for their warm welcoming addresses and their expressions of importance of this technical area to the Air Force. Heartfelt appreciation is expressed to Mr. Eric E. Abeli, Chief of All Technical and Engineering Operations for the F-22 System Program Office, for taking time from an overloaded schedule to cliver a timely, challenging, informative keynot address which set the tone for the technical papers which followed.

This report was submitted for publication in March 1994.

Publication of this report does not constitute Air Force approval of the findings or conclusions presented. It is published only for the exchange and stimulation of ideas.

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SESSION I

SYSTEMS OVERVIEW - PART A

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ACQUISITION MANAGEMENT OF CONSUMABLE TECHNOLOGY UTILIZING A NOVEL TECHNOLOGY TRANSITION PROCESS

Russell E. Urzi
Flight Dynamics Directorate
Wright Laboratory

ACQUISITION MANAGEMENT OF CONSUMABLE TECHNOLOGY UTILIZING A NOVEL TECHNOLOGY TRANSITION PROCESS

Russell E. Urzi , USAF

ABSTRACT

The traditional approach towards acquisition of military hardware is inherently slow. By the time the weapon system is being delivered many of it's subsystems no longer meet the real needs of operational missions. Also, missions themselves evolve and yesterdays subsystem may no longer be adequate. Hence, the system is forced into expensive retrofit in order to get it up to the desired capability. A classic example of this occurs with the aircrew transparency subsystem. Typically, the aircrew transparency subsystem is of secondary concern in the aircraft design process and in the evolution of aircraft mission assignments during the life of the aircraft. This can result in an urgency driven, very expensive, redesign, and in this current tight fiscal environment, this process will clearly not do anymore.

A proven process for accelerated transition of technologies can be used to overcome these problems. This process to rapidly transition laboratory R&D to operational use evolved using principals which are now more commonly referred to as a TQM approach. This process can be used on any subsystem which is consumable, that is, it wears out and requires periodic replacement.

This paper explains this process, describes several examples of its application, and suggests applications in which it can be used in a broader sense.

INTRODUCTION

A common criticism is that the results of Laboratory Research and Development efforts are too slowly exploited into better/cheaper operational hardware. Rapid advancements are being made in the areas of cost reduction technology and to achieve technological superiority. These technology advancements are keeping pace with the ever changing missions demanded of the military.

Because some of these technology advancements have only recently been available and the rate at which the laboratory is progressing is much faster than before, some subsystems, while acceptable, may not be optimal. For their time, these subsystems were the best they could be, but now they could benefit from further technological advances. It is these subsystems that

could benefit from the proposed process to rapidly transition the Research and Development advancements to operational usage.

To illustrate, take for example the evolution from high altitude bombing to low altitude precision strikes. Among the various aircraft subsystems which must also evolve concurrently is the aircraft transparency. At lower altitudes, there is an increased risk of birdstrikes, as well as erosion from particulate matter suspended in air. The transparency system must be able to cope with this new environment. The transparency system of an existing fighter aircraft is a good example of this mission evolution and its resulting consequences. An early transparency system (canopy) was constructed from a single sheet of material and had a birdstrike resistance of less than 200 knots. It soon became apparent that at lower altitudes (below 3000 ft) an increase in birdstrike resistance was needed. A canopy was developed and provided protection to 350 knots. However more problems developed because of this low level environment, chiefly, abrasion of the canopy resulting in a loss of optical clarity. (Operation Desert Storm was a good example of this phenomenon being experienced on many aircraft.) Aircraft were also performing this mission at speeds higher than 350 knots, and with onboard equipment necessitating improved optical quality of the canopy. Consequently, the canopy had to provide even more birdstrike protection as well as abrasion resistance and improved optics. This requirement evolution took place over a 15 year period during which more complicated canopies were developed to meet these requirements as well as extending the service life of these parts. This was all accomplished by using the following technology transition process.

THE PROCESS - "PRELUDE"

The process can be explained much like the tires on your car. They wear out and need to be replaced. Your type of car and to some degree your personality, will determine what type of tires you will purchase. All must be durable and low cost. The sports car driver will insist upon good traction and manuerverablity for negotiating curves, the off-road four wheel drive vehicle will opt for more rugged construction and better traction, and finally the land yacht owner will opt for quiet and comfort. Each will evaluate the performance and durability of their current tire against these standards in selecting a new tire.

The tire manufacturers are constantly striving to provide better products in order to obtain more of the market share and consequently more profit. Hence, the consumer (driver) is faced with many more possibilities than say a year or two ago. He may consult with a auto mechanic or test agency before making any purchase decisions. Once these evaluations are completed, the consumer will make his choice of tires. He will drive these tires, form his own impressions about their performance and durability, and eventually wear them out. The process then starts all over again.

This oversimplified explanation illustrates the key concepts involved. One fundamental concept is that some weapon subsystems are of a consumable nature and eventually wear out. And, that

when they do wear out, the replacement should meet the requirements necessary to fulfill the mission.

The operational users, together with the system developers, maintainers, industry, testing and supporting laboratories form the basis in which this process works (i.e. the stakeholders.)

The second fundamental concept is that the stakeholders who have ownership of the weapon system want to deliver the best product they can possibly achieve. This is absolutely necessary for success! For without this personal commitment, the weapon system is doomed to mediocrity.

When these two fundamental concepts or conditions are met or exist, then the process may proceed as explained in the next section.

THE PROCESS - "FUGUE"

The first step in the process is to identify what performance and/or supportability features of the subsystem should be improved. The subsystem must be of a consumable nature and would be replaced periodically. Usually a standard criterion is used in order to make the determination of whether the current system is good enough or should be improved. However, this standard is of a dynamic nature and changes with the situation. To continue the transparency example, the Air Force uses the "444" concept as the standard to which transparency systems are compared. Simply stated, a transparency system should perform its mission for a minimum of four years after which it should be able to be replaced by no more than four technicians in four hours. Embedded in this standard are both performance and supportability features.

Once the standard is identified, a decision must be made. That is, should the subsystem be upgraded, procured as is, or both? Usually, the subsystem being of a consumable nature can be upgraded, but the upgrade must be proven sufficient to warrant the risk of purchase.

This decision is best done by the stakeholders. The using commands will comment on the performance, the maintainers on the supportability, etc. If the current subsystem is not okay, then the stakeholders must proceed to the next step.

The next step relies upon the supporting laboratory to apply emerging technology to the subsystem. The laboratory is constantly developing, integrating, validating and transitioning technology to meet mission performance requirements at lower cost. Depending upon the maturity of the technology, the subsystem may benefit immediately. It is important to note that of the stakeholders, the laboratory is best suited for this task, and not say the system developer.

The reason for this is that quite simply the laboratory is where technology risks are taken at minimal cost. Taking new technology and inserting it into the Engineering, Manufacturing and Development phase is much riskier and more costly.

When the new technology is applied to the subsystem, a prototype must be evaluated. This evaluation could be in the form of a "critical experiment" or an "operational evaluation". What is important in this step is that the evaluation must include all the stakeholders. If for example the operational command was not present, then there could not be any feedback on it's suitability for field use. Likewise, a similar logic could be said of the logistics command. The basic point is that the evaluation must not occur "in a vacuum".

If the operational evaluation was a success, then the technologies embodied in the subsystem could immediately be procured in an upcoming spares buy. Again, if the stakeholders were properly working together from the beginning this would not pose a logistics problem. If the subsystem did not perform as desired, then back to the laboratory it goes for continued technology development. This ends up being a somewhat cyclic process paced to coincide with specific upcoming spares buys.

As a case example, let's examine the transparency subsystem for an existing bomber. The current production design windshield had a service life of less than one year (350 flight hours) due to delamination of the outer glass ply, forcing the Air Force to spend $10 million per year for spares. Addressing this service life issue was truly a joint effort between all the stakeholders. The stakeholders working together agreed to a service life interim goal of two years. Doubling the service life to two years was definitely not consistent with the first "4" (4 year service life) in the "444" concept. But, the stakeholders agreed that the technologies to support an interim goal of two years could be demonstrated, with reasonable risks, in time to coincide with an upcoming spares buy. And, that the savings from this procurement against the interim goal would be a welcome reduction to Air Force costs of ownership while the four year service life goal was pursued.

Technology transition to users was realized as the prototype windshield designs went into production. This stakeholder technology exploitation team was working so well that in less than one year the stakeholders again selected a prototype windshield design to replace the current production windshield.

The new windshield designs provide 100% increased service life at half the cost of the initial production design. The Air Force is expected to save $7.5M per year for spare parts as a result of this technology transition. This savings is coming from increased service life and a result of competition between the vendors. The unit cost of the windshield dropped from $62K to $25-$33K since the program started.

The key reason for this success, is the close relationship that has been developed and maintained between all the stakeholders. The users, the logistics community, the laboratory and industry were full team members from the earliest phase, and their active participation throughout the program ensured a smooth and rapid transitioning of technology as soon as it was successfully demonstrated. The team is continuing in its quest to transition emerging technologies to meet the "444" goal.

THE PROCESS - "VARIATIONS"

We have encountered many applications where this process could be used to focus technology development and then rapidly transition it to operational use. One area which is just beginning to use this process is aircraft engines. The Propulsion System Program Office at Wright-Patterson uses a similar process to replace engines in high performance aircraft. The only criteria for successful application is that the two fundamental concepts discussed earlier be in place. Other examples include tires, brakes, wheels, and ejection seats. This process is not just confined to subsystems associated with aircraft but could be applied to any weapon system or platform.

SUMMARY

This paper has explained a technology transition process to rapidly transition technology from the laboratory to operational use. The only criterion needed is that the subsystem be of a consumable nature and the two fundamental concepts be in place. Additionally, this process requires a "team approach" to accomplish the transition. This is probably the most critical element in the process, because if this teaming did not occur then technology transition would be just "over the wall". This team is actually the stakeholders in the weapon system. They have a personal commitment to producing/maintaining the best system for the operational commands. Together, a close working relationship among the stakeholders is the key to successful technology transition.

ENDNOTES

1. Draft Proposed Aeronautical Systems Center (ASC) Technical Planning Integrated Product Team Charter dated 4 Dec 92. This charter brings together the vital stakeholders in the systems acquisition process in a forum to meet projected operational user needs.

TECHNIQUES TO CREATE A SUPPORTABLE TECHNOLOGY PROGRAM

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TECHNIQUES TO CREATE A SUPPORTABLE TECHNOLOGY PROGRAM[1]

Thomas R. Bennett, DoD[2]

ABSTRACT

A driving factor for the development and acquisition of military systems is supportability. However, this has not always been the case. Take the United States Automotive Industry's experience, from leading the International Community in automotive quality, to falling behind Japan in building supportable technologies, and then back to investing resources to develop supportable systems. Why the turnarounds? Because the customer's feedback was that supportability was one of their main considerations in buying a new car. Likewise, the US Military has learned from the Automotive Industry the importance of supportable systems. In the 1970s, leaders in the Defense Department established acquisition logistics organizations to develop the tools and expertise to allow future systems to quantitatively counterbalance performance requirements with supportability requirements as design drivers. Now that the DoD has empowered the technology customers with the economic resources to buy into their own systems, versus putting up with what they were historically given, the acquisition community must take an Automotive Industry approach to developing weapon systems.

Our acquisition logistics organizations now have the experts and tools to meet our customer's needs and to work with industry to further improve our systems. This paper will address the methods and tools that are available to determine, develop, demonstrate, and transition supportable technology/systems to our customers. Specifically, it will address these methods as applicable to transparency systems technology.

PART I - THE METHODOLOGY

INTRODUCTION

The Japanese Industries use quality process improvements to quantitatively improve their overall way of doing business. The concept is that business processes and product development can be quantified, and overall output results can be measured. Product performance can be measured by collecting field results data. Process performance can be measured through product performance data, product performance data trends, customer feedback, and economic analysis. The Japanese have shown that implementing these collection methods and feedback loops is necessary for an organization to be competitive in our current environment.

How does logistics tie into this concept? By quantifying the development processes, determining what performance data to collect, how to collect it, collecting it, and then using results data to improve the development processes and products, you have the key to incorporating logistics into a technology program. Whether the program is automotive production, personal computer production, military aircraft

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[1] Prepared for presentation at the 1993 Aerospace Transparent Materials & Enclosures Conference, held in San Diego, California on 9-13 August 1993.

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production, or a new research and development program for a T-38 transparency system, the development of quantified processes, collecting field data, and implementing a results based feedback improvement process is how logistics must be incorporated into programs.

Let us consider the research and development (R&D) types of programs. R&D programs, excluding the very basic research programs (referred to as 6.1 and 6.2 programs by the military), are similar in process to the much more elaborate major system developments, such as the F-22 aircraft program. On large scale production programs, such as the F-22, logistics is a formal process and managed by the acquisition logistics department of the program office. They are focused on such development items as the technical orders (TOs), the support equipment (SE), the training for maintenance and support personnel, etc. In the R&D programs, these types of logistics elements also need to be taken into account, but more from a common sense engineering perspective versus developing all of these logistics requirements. For example, if a new transparency system is being developed in a laboratory program, it is not reasonable for the research engineer to develop the complete technical orders for retrofitting this system onto a particular aircraft. However, it is reasonable for the research engineer to study what the current technical order requirements are for maintenance on comparative transparency systems and then assess the R&D design to determine what impacts the new technology will make on the maintenance community. This concept seems very simple, but in the majority of historical research programs, the technology was only driven by the goal of achieving a performance requirement. The need to fit the technology into an operational system, and to develop the logistics support for it, was typically considered to be the responsibility of the system program office (SPO) for the production system, and the technologists "threw the problem over the wall" to the SPO.

This disconnect, and the associated expense to redevelop an optimized system with the new technology, was the basis to matrix reliability, maintainability, and supportability (RM&S) engineers into the Air Force Labs. Their job is to support the R&D program managers in developing supportable technologies. They do this by creating business processes for incorporating logistics requirements into the programs, establishing logistics requirements for the development effort, and using/developing the logistics tools required to support the programs.

RELIABILITY, MAINTAINABILITY, and SUPPORTABILITY

These three terms are often referred to as the "ilities". My definition of reliability is the inherent material and design properties of a system, or subsystem, that allow it to function failure free for a specific number of operating hours in a given environment. Reliability can be measured in quantitative terms, such as mean time between a specific failure, or in number of failures over a given period of time for a group of identical parts. Time is the common denominator for measuring reliability. Likewise, my definition for maintainability is a qualitative evaluation of a system or subsystem design based on how much maintenance effort is required to keep the system operational. Such concepts as accessibility, mean times to repair, number of maintenance actions per flight hour, etc. are all considerations for evaluating how maintainable a system is. And supportability encompasses all of the infrastructure and support requirements to maintain the system or subsystem. Supportability is also a qualitative measure of a design, and incorporates such concepts as maintenance facility requirements, maintenance personnel training requirements, storage facility requirements, unique parts required. consideration of rare elements being supplied by countries other than the USA, etc.

Considering the three "ilities" in the design, and looking at the R&D program from a common sense engineering perspective will produce good results in developing a supportable technology. One key is to balance the RM&S requirements for the system with the system performance requirements. From my experience, I have found that the system performance requirements will always rule, unless you are developing a totally supportability driven program, such as the high reliability fighter, where system performance capabilities were traded off to see just how reliable a system could be designed. However, the proper perspective on developing a supportable R&D program is to evaluate the impact of the design on

RM&S, and make trade-off decisions based on this. Common sense engineering usually prevails once the RM&S concerns have been identified, and after the trade-offs have been made, the system always seems to be better.

QUANTIFIED REQUIREMENTS

It is difficult, if not impossible, to set RM&S requirements if they are not quantified. However, I have seen instances were contractual requirements stated only that the contractor should consider RM&S. I'm not sure what it means to consider RM&S. In the cases where I observed this problem, the contractors produced nothing in the way of RM&S, and since there is no way to enforce the term "consider" (at least I'm unaware of any), the program usually produces less than desired RM&S qualities.

Therefore, it is necessary to quantify RM&S requirements for each program, just as it is necessary to quantify the performance requirements for each program. The Windshield Program Office of Wright Laboratory[1] has demonstrated some good concepts for quantifying RM&S requirements in their windshield development programs. One of their most notable examples is the 444 concept. It stands for a 4 year service life, with a changeout by no more than 4 technicians in not more than 4 hours. This is a simple concept, and is applied to their contractual design requirements. The transparencies can be quantitatively tested before final acceptance, with the 444 concept covering RM&S concerns such as maintenance man-hours to replace, and mean time between failure of the system. In addition to unique concepts such as the 444, there are a multitude of other tools and techniques to quantify RM&S requirements on a research and development activity.

What is the right amount of RM&S requirements to add to a program? This is not an easy question to answer. In fact, often times there may be many answers that will provide good results for a supportable technology product. I usually consider a trade between the overall cost of the contract, whether or not the product is a study or a prototype, whether or not the technology is directed at a particular weapon system or is more generic, and what the current RM&S concerns are on similar fielded technology. The real answer to the question is the same answer the Defense Department Leaders came up with in the 1970s, which was to establish acquisition logistics organizations to develop the tools and expertise to support the development of all future weapon systems. A point to make here is that there are many tools and resources available to analyze what RM&S factors should be considered in a development effort, and in turn there are many tools and resources available to determine the RM&S parameters for given designs, but it does not make sense to take a shotgun approach with placing RM&S requirements on a program. An inexperienced project engineer could easily overwhelm the scope of a development effort by blindly applying RM&S requirements to it.

For example, let's consider a new windshield development. The R&D will take a systems perspective. It may be appropriate to include a contractual requirement for one or two tasks from MIL-STD-1388-1A, Logistic Support Analysis (LSA), to be performed. However, placing a generic statement in the statement of work that LSA must be performed on the effort, or incorrectly tailoring the LSA requirements could cause the proposals to come back with cost estimate overruns well over an order of magnitude beyond the basic research costs. The reason is that the generic LSA tool evaluates facility requirements, maintenance personnel training requirements, all the tech orders, etc. None of these categories are typically appropriate for an R&D effort. However, some LSA categories are appropriate to R&D, such as a trade study of the technology use, or the maintenance task listings for the system, etc.

Common sense engineering is the key to effectively utilizing the RM&S tools and techniques to develop a supportable technology. The methodology of developing quantified RM&S goals/concepts (i.e. 444), measuring the results that these goals have produced, and implementing fact based improvements to the RM&S concepts, based on the pros and cons from the measured results, is a method I have found to work effectively for developing supportable technologies.

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[1]Windshield Program Office, WL/FIVR, Wright-Patterson AFB, Ohio.

PART II - THE TOOLS

MAINTENANCE DATABASES

Maintenance databases are one of the few tools that the RM&S engineer has available to provide definite results based data to support trade-off decision making on a technology program. The Department of Defense maintains a multitude of databases on every hing imaginable, including all of the weapon systems that the US owns. Many companies that produce particular weapon systems, or subsystems, also maintain their own maintenance databases on their systems. The databases store data on everything imaginable, including such things as: the number of flight hours for every single system per day, the daily maintenance performed in each base shop, the reliability values for all the various weapon systems and subsystems, the cost of spare parts procured for each subsystem broken into various time periods, etc. Some databases are rudimentary, meaning they store only raw data, such as a backshop log of what maintenance was performed on a given day, by each technician, and what work was completed. Other databases are higher order, meaning they not only contain raw data, but they will utilize this data in their internal programming to determine higher level information, such as quarterly supply expenditures per wing, or the number of F-15 windshields that were replaced in a given month by the USAF, etc.

One shortcoming of maintenance databases is that they are only as good as the data that is entered into them. In my experiences, I have found that the raw data coming from the bases is not always accurate. An example to prove this point is one where we knew that the Air Logistics Center (ALC) at Hill AFB procured a certain number of F-16 tires per year. As we were searching the USAF databases on F-16 tire wear, we checked to see the total number of tires that were replaced in a year. It was approximately one third of the number of tires that we procured each year. Unless the USAF is building a large number of warehouses each year to store all the left over tires, the numbers are inaccurate. But, for the most part, I have found the raw base data to be usable for making trade-off decisions, as long as I remember that the data I am basing a decision on is sometimes only a rough estimate of what is actually occurring on a particular system. And, unless we invest a large amount of resources to collect the data ourselves, the DoD databases are our main source of decision making information.

The following information is derived from a publication produced by the Supportability Investment Decision Analysis Center (SIDAC) under a USAF contract[4]. It is a good reference for learning more about the DoD maintenance databases. Here are a few of the main DoD maintenance databases that I feel provide beneficial information.

REMIS (Reliability and Maintainability Information System) "The objective of REMIS is to provide the Air Force with an on-line, comprehensive, integrated information system. The system will include maintenance management information, equipment inventory information, status information, utilization data, configuration data, Mission Capability (MICAP) data, and Awaiting Parts (AWP) data. REMIS will be an equipment maintenance management information source for aerospace vehicles, trainers, Communication-Electronics (C-E) equipment, Automatic Test Equipment (ATE), Support Equipment (SE), and other reportable items." "REMIS when fully implemented, will replace 28 existing data systems." Although the REMIS database was supposed to have an Initial Operating Capability (IOC) in 1989, the system has not yet become fully functional.

MODAS (Maintenance and Operational Data System) "The objective of MODAS is to provide an on-line, interactive data storage and access system. Its primary function is to provide a Data Base Management System (DBMS) with automated analytical capabilities to support R&M and Product Improvement and Product Performance Programs established by the Air Force. MODAS contains maintenance man-hour information for both airborne and non-airborne equipment, and operational data for airborne equipment." MODAS's raw data comes from the base data in CAMS (which is described below). REMIS is slated to replace MODAS once it reaches full capability.

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[4]SIDAC Data Source Catalog, published sometime around 1990.

CAMS (Core Automated Maintenance System) "The objective of CAMS is to provide an on-line, base-level maintenance reporting system. CAMS is not currently available at all base locations. CAMS is replacing the previous manual process of reporting base-level maintenance actions. CAMS can improve the quality of data in D056 and REMIS as it contains an up-front editing capability for the technician who is inputting the maintenance repair or service action. This system contains maintenance data for a specific base, only. This base maintenance data is transmitted to D056 so system-level statistics such as MTBF and MM/FH can be compiled, monthly." The CAMS database is not normally accessed by acquisition personnel. Since CAMS data is raw data and feeds other relational databases, the RM&S engineer typically queries a higher level database such as REMIS or MODAS. And, the REMIS or MODAS systems can provide higher level data by combining, manipulating, and calculating overall raw data from around the world.

TICARRS (Tactical Interim CAMS and REMIS Reporting System) "The objective of TICARRS is to provide an on-line, maintenance reporting system for the F-16 and F-15E aircraft which includes engineering, configuration, and logistics data. Data is also provided via tape-to-tape transmissions from each F-16 and F-15E base, on a daily basis. The tape information is verified and loaded into the TICARRS database, thereby providing a complete database for weapon systems." The TICARRS database is an example of where the SPO and ALC wanted more accurate information than they received in the standard USAF databases, so they invested additional resources to develop their own method to collect RM&S data on their systems. The F-117 program and the F-15 A/B/C/D program is currently establishing a TICARRS database for their systems.

LIFE CYCLE COST MANAGEMENT

Another RM&S trade-off that should be considered in the design process is the life cycle cost impacts that the technology will have on a subsystem and overall system. There are three methods that I currently use and feel provide good information to allow relatively accurate trade-off decisions involving life-cycle cost considerations. The first method is another DoD database tool, and I again reference the SIDAC Data Source Catalog. The second method is a USAF software program tool that allows RM&S engineers to calculate a life-cycle cost prediction for the system, or subsystem, based on the predicted system usage. The last method that I will mention, which is always a good fallback, is the simple back-of-the-envelope calculations.

VAMOSC (Visibility and Management Operating Support Cost) Database. "The objective of VAMOSC is to provide operating and support costs for Air Force ground communications and electronic equipment, aircraft, and aircraft subsystems." "On-line access to VAMOSC is limited to the USAF major acquisition organization personnel."

CASA (Cost Analysis Strategy Assessment) Models. This is a USAF written program that allows the user to input system/subsystem usage variables, (such as number of aircraft per base, mean time between a particular subsystem failure, etc.) and have the program calculate the Life Cycle Costs associated with fielding a particular technology on a current or new system. It will also allow the user to perform "what-ifs" on current systems to determine if a new technology replacement system/subsystem is even beneficial from a cost standpoint.

Finally, if the RM&S engineer needs to get some quick numbers to make trade-off decisions with life cycle costs considered, I recommend using some simple back of the envelope calculations. This method often will provide relatively accurate information if you have a rough idea of the production and support costs for a current or new technology system/subsystem. I have found that the large investment of resources required to predict life cycle cost considerations is not always worth the investment for an R&D technology development program. I believe the resources are much better spent on obtaining an accurate prediction on the impact of the technology to the maintenance community, and particularly on technology transition.

OTHER RM&S TOOLS

There are many other RM&S tools available to the RM&S engineer as resources to support his technology program. The following are a few of those that I have found useful as resources to help me in supporting advanced transparency systems technologies and the other vehicle subsystems advanced technology development programs.

R&M 2000 is a USAF publication. The version I have is from October 1987. The publication is cleared for public release. The objective of the publication is to define the R&M goals for the USAF, listed in order of priority, such as "Increase Combat Capability", and "Increase Survivability of the Combat Support Structure". The document describes the principles and buikling blocks to develop these goals within an acquisition program. It describes how to set the requirements, how to integrate the requirements into the development program, and it provides some information on R&M tools that can help the RM&S engineer with implementing the R&M 2000 goals. Even though this document is six years old, it is still timely and presents many good ideas.

The Logistics Needs program was developed, and is managed by the Aeronautical Systems Center at Wright-Patterson AFB. Logistics Needs (LNs) are one page write-ups, kept in a large database. They are the USAF, USA, and USN's prioritized technology needs wish-list for improving the RM&S of their weapon systems. The LN program receives LN write-ups throughout the year and then meets with the LN representatives and RM&S technical experts from the tri-services to validate the need for a new technology to be developed. The database provides a hypertext search capability for RM&S engineers to search for particular requirements for application of a technology development program. This tool is particularly effective in tech transition, with an emphasis on the maintenance community.

Re-Bluing Visits (RBVs) and Blue-Two Visits (BTVs) are techniques that I have found very useful in learning first hand what the RM&S issues are on a fielded system. The RBVs and the BTVs are similar by design in that both are business trips where acquisition personnel are taken as a small group to an operating location and spend a couple of days with the maintenance, supply, and operating personnel. The group gets hands-on maintenance experience and learns first hand how a system is used and maintained in the field. One of the benefits of the visits is that the acquisition personnel see for themselves what technology/design ideas work well, and which ones create maintenance problems. The group also gets invaluable experience from talking with the maintainers and learning about very creative work-arounds to design problems. The RBVs and BTVs are not for the designers to see operational problems and design quick fixes, but to gain a better understanding of how different designs work in the field and apply this knowledge to the next generation designs. The differences between the RBV and the BTV is normally that the RBV is usually a smaller group of only government personnel that visits one operating location and focuses on specific technology areas, versus the BTV, which is usually a larger group of high ranking government and industry acquisition personnel that often visit multiple bases on one trip and focus on an overall perspective of maintaining and fielding various systems.

Integrated Logistics Support Plans (ILSPs) are another tool that I have found useful in supporting a technology program. Whereas the RBVs and BTVs are first hand observation of how the system/subsystem is used in the field, the ILSPs are the documentation on a particular system of how the design is to be used and supported in the field. The ILSPs provide the details of how the system is fielded from the time it leaves the production plant until it is used in service at its operating location. Even though we often use our systems/subsystems other than for how they were designed, the ILSP of a system provides the RM&S engineer with the information to base ideas on what impacts a new technology will have on a fielded system. Combining a system ILSP with a RBV to observe first hand the RM&S issues of a fielded system, usually provides a comprehensive picture of what RM&S factors must be considered in a new technology development program.

UNIQUE RM&S TOOLS

The available RM&S tools described above are generic in nature, meaning they are designed to aid RM&S engineers in a wide range of technical disciplines. Many RM&S engineers, and program engineers, have invented unique tools to support RM&S technology development factors in their particular technical area. Once again, the Windshield Program Office is a good example of using unique RM&S tools to improve a technology area. They have invented, developed, and used their own RM&S tools to evaluate and improve the RM&S characteristics of their developments. The following are a few of these tools, which should illustrate the usefulness of developing unique RM&S tools.

The Aircraft Transparency Durability Research Facility (ATDRF) was invented, designed, and built in the mid 1980s. It is a 25,000+ square foot indoor facility in which full scale transparency systems are mounted on a front fuselage section of their respective aircraft. The windshield is in a test section that is subjected to extreme thermal and pressure changes, which are based on typical daily mission profiles and flight line storage for that aircraft. The entire service life of a transparency system can be simulated in this computer controlled facility in three months of 16 hour days (5 day week). After correlating failure data to field data collected on the same transparency systems, the facility can be used to evaluate field reliability of new technology systems. Other environmental factors are being added to the test chamber, such as humidity and ultraviolet radiation, which will simulate flight line environments even more realistically. This facility has proven invaluable in evaluating a recent high reliability coating research and development program.

The Imbedded Aircraft Transparency Environmental Data Recorder (known as the Data Recorder) is another unique tool invention of the Windshield Program Office. The Data Recorder is designed as a small chip with a tiny power supply, which can be imbedded in a transparency system frame at the time of production. The Data Recorder will constantly multiplex different data sensors, which are also embedded around the transparency system frame at the time of production. The data will be stored for a particular period of time and then downloaded into a central database system to be evaluated by the Windshield Program Office R&D program engineers and the RM&S engineer. The data will detail the parameters that an operational system experiences both in flight and on the ground during maintenance and flight line storage. The RM&S factors, (such as temperature, pressure, humidity, etc.) are already known, since this is what the Data Recorder is measuring. Thus, the data being recorded are the parameters (specific values at each given time increment) to each RM&S factor. You can see that this type of RM&S development tool can be used for many other technical areas besides transparency systems.

The Combined Environment Testing Methodology is another idea that has stemmed from transparency systems research. When evaluating methods to perform accelerated testing on RM&S factors, and having them correlate with field failure data, it was observed that the test data sometimes correlated, and sometimes did not, depending on the aircraft subsystem being tested. The problem that seemed to exist was that combined environmental conditions, (such as a lifetime of ultraviolet radiation exposure along with rain erosion) had a much more detrimental effect on some designs than others. The solution was to devise a RM&S technique to combine testing environments. The environmental tests could be run sequentially in some cases, and at the same time in others. The result is that the RM&S test techniques that have provided relatively good results in the laboratory as correlated with field failure results, are now producing much better correlation. Again, I think a point can be made that this type of RM&S technique of combined environmental testing can be applied to many technology areas.

RM&S STANDARDS

Additional sources of information to help the RM&S engineer in supporting his programs can be found in the multitude of Government, Industry, and Professional Society's standards, regulations, publications, and handbooks. The following is a short list of some of the DoD's design standards for RM&S that I find helpful.

- MIL-STD-0-2 Overall Index of Military Standards.
- MIL-STD-470 Maintainability Design Recommendations.
- MIL-STD-785 Reliability Design Recommendations.
- MIL-STD-810 Guidelines on the Natural Environment with Parameters.
- MIL-STD-1388-1A/2B Logistics Support Analysis/Record.
- MIL-STD-1472 Human Factors Maintenance Design Recommendations with Parameters.
- ACTS MIL-PRIME Aircraft Transparency Systems Design Recommendations.

There are various professional societies that publish standards on RM&S and support working groups to advance the RM&S techniques in system design. The Society of Automotive Engineers has a G-11 RM&S working committee that has active international membership from a multitude of industry engineers from the US, and abroad, and active participation by RM&S engineers from the USAF, USA, and USN. Their charter is to develop generic RM&S design guidelines, recommendations, and standards. They are currently completing a project to publish a comprehensive listing of all known RM&S Government and industry documents, (such as standards, handbooks, etc.) both from the US and abroad. Another of their publications about to come out is a recommendation and definitized listing of what should be the minimum number of core RM&S terms that are needed to comprehensively track all RM&S factors for any given system development program. This will be a milestone publication, since there are currently thousands of RM&S terms found in all of the known RM&S documentation. They are also chartered to organize and hold the annual RAMS international conference, and to initiate academic courses and curriculums at the collegiate level to further the engineer's understanding of RM&S needs on system development.

Focusing specifically on aerospace transparency systems, the American Society of Testing and Materials (ASTM) has subcommittee F7.08. This working committee, also comprised of international membership, is chartered to develop specific design guidelines, recommendations, and standards for the development of aerospace transparency systems. Many of their recommendations, standards, and test methodologies have been developed to address RM&S factors of transparency system developments.

CONCLUSIONS

The methodologies and tools to incorporate RM&S factors into any system/subsystem R&D program are available. The Government is currently placing high priority on developing supportable technologies. It has been the DoD plan since the 1970's to be proactive versus reactive in addressing RM&S factors of system design. In Wright Laboratory, there are RM&S engineers in each of the directorates, co-located to all the different program offices, to support the program manager in making decisions concerning RM&S design trade-offs.

Programs have begun to establish processes to balance supportability requirements with performance drivers. However, the efforts to build in a mandatory process to address RM&S factors in the technology programs are still in the rudimentary stages. Because the technology customer, mainly the using commands, now have to pay out of their yearly budgets to maintain and upgrade their systems, along with operating costs, the design paradigms quickly changing. As mentioned at the beginning of this paper, the DoD is having to take an automotive industry approach to developing systems and subsystems. The customer now has the financial control to direct the future of Government research programs, and they are, by investing in the R&D programs that will help their RM&S problems. So, as with the automotive industry, the importance of RM&S in future DoD research and development is apparent.

SESSION I

SYSTEMS OVERVIEW - PART B

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TRANSPARENCY SYSTEM (CANOPY) TECHNOLOGY CHALLENGES

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TRANSPARENCY SYSTEM (CANOPY) TECHNOLOGY CHALLENGES

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Abstract

Canopy systems, including transparencies, are being recognized at the military and prime contractor levels as being an essential component of the aircraft assembly. This recognition, which is welcomed by the struggling designers and the transparency suppliers, has highlighted certain voids in technologies mandatory to the proper design and construction of the complex systems required on modern aircraft and for updating/redesigning existing aircraft transparency systems.

This paper reflects on a small subset of technology voids encountered on the Mission Integrated Transparency System (MITS) Program and encourages the transparency community to assist in developing the technologies needed to fill these voids.

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INTRODUCTION

Lockheed Fort Worth Company (LFWC), formerly General Dynamics, Ft Worth, under contract F-33615-88-C-3402 to Wright Laboratory (WL/FIVR), is prime contractor for the Mission Integrated Transparency System (MITS) Program. The program is a multi-year task directed at technical assessment of transparency systems for military aircraft for the 1995 and on time period. During this program LFWC has conducted design trade studies, tested and revealed successful transparency technologies, developed design and performance criteria, and, through a Critical Design Review, has transitioned the technology from the industry to the using customer.

The MITS program has been supported by the major suppliers in the transparency and materials industry, resulting in bringing new transparency systems into service for the USAF.

FIVR promotes and sponsors technology development that improves aircraft performance and reduces cost of ownership to the user. Therefore, an important output from the MITS program is the identification of transparency technology voids or gaps.

This paper is a draft of a larger forthcoming report intended to fulfill the FIVR objectives by listing and providing a brief assessment of the transparency system technologies within a single document.

While the primary focus of the technology listing in this draft and the report to follow, is on transparency systems, it is necessary to overlap into the entire canopy systems to ensure complete and satisfactory integration to the aircraft. Transparencies should not be designed without detailed attention to the canopy frames, latches, loading and load distribution, emergency and normal actuation, as well as associated pilot weapons delivery optical functions. Today's aircraft, requiring stealth and direct threat protection features demand considerations from the transparency such as the use of non-circular cross sections for structural loading, surface coatings that must withstand the environments (including hostile) and yet be geometrically shaped to provide good optical quality. The input to General Dynamics' Trade Study report, as well as the report, itself,(References 1 and 2) provide detail insight into the integration of many of the transparency technologies listed herein.

These diverse requirements must be integrated into a workable, durable hardware system by the designer and manufacturer. To do this, technical tools and managerial freedom are needed. This paper can only address the technical tools. Credit for the first push to technology needs or voids is given to Mr. Harley Walker (Reference 2) of Wright Laboratories, who, in 1983 challenged the community to face the issues.

The following page format is used to present each of the technology subjects. It should be noted that some of the subjects listed are currently under development in sponsored programs, and that in some cases sponsorship may exist but the information is not available to the author. The reader should treat this paper and the report to follow, as an independently compiled shopping list, to be evaluated for opportunity, and subjects accepted or discarded as appropriate to the need.

Format for Technology Subjects

Title
Background/Objective
Current State of Technology
Potential Solution
Benefits of Solution

Acknowledgement

Many of the technology subjects covered in this paper were initially identified in Reference 1 by Mr. K. E. Thompson, Canopy Systems consultant. Mr. C. A. Webster of Lockheed Fort Worth Company, has also participated in the identification, refinement and documentation of these subjects in Reference 2. The contributions of these two gentlemen is gratefully acknowledged by the author.

Bird Proof HUD

Background/Objective

Interface HUD Systems. Heads up displays (HUD) mounted on the glare shield in front of the pilot have become an interface item with the advent of transparency designs that depend on semi-bagging or flexibility to defeat bird impact. Out of geometrical necessity relative to the pilot, and drag area considerations relative to aircraft performance, the HUD is usually mounted in close proximity to the center of the forward transparency.

Current State of Technology

Typically, the HUD assembly consists of a thick piece of semi-tempered glass mounted into metal side posts using flexible adhesives. The side posts are anchored to a large electronic housing within the instrument panel area. The HUD glass is required to be reflective for transfer of signal information in collimated form to the pilot by reflection and/or refraction. During impact with a bird at high speeds, the transparency contacts the HUD glass, breaks it and scatters glass fragments over the cockpit. If unprotected, the sharp glass fragments can cut the transparency during high strain rate loading, resulting in immediate failure. Transparency designs have not accorded this risk, nor has the HUD in it's configuration. The HUD has proven to be an asset for the F-16 in reducing the total amount of deflection occurring during bird strike by interrupting the traveling wave.

Potential Solution

1) Seek HUD designs that utilize shatter proof transparent material.

2) Apply a sacrifice inboard ply to the transparency.

3) Install a protective material over the contact edges of the HUD.

4) Any of the above, but require through specification criteria that the HUD and transparency be compatible for bird strike.

Benefits of Solution

Simply to properly design for the entire bird risk, not just a portion of it as has been the practice.

Full Scale Time Line Measurement of Deflection, Strain and Strain Rate During Bird Impact Testing

Background/Objective

This is one of the technology voids identified in the 1983 Transparency Symposium by Mr. Harley Walker (Reference 2). The void still exists. The need simply is for test technology that will record in real time, and in precise form, the dynamic results of a transparency during a bird impact test. The technical community has conducted hundreds of bird strike tests on the F-16, F-111, and others and there is very little real engineering data to assess the geometrical and physical result. Material needs, limits of current materials, dynamic modeling, geometrical effects, etc. are at best guesswork for lack of recorded data. These are big challenges, but the technical reward is big.

Current State of Technology

This is basically a blank. Deflections on the transparency are estimated by use of straws located at known distances from the inboard surface of the transparency as one method. Another method is a complex arrangement of cameras, geometrically resolved to follow marked spots on the transparency. The first method is not time lined and risks loss due to HUD debris and internal wind blast. The second is indirect and cannot be adequately calibrated and validated.

Strain measurements were attempted in the late 1970's by Douglas Aircraft Co. under contract through UDRI using state of the art strain gages. Two results were observed. First, the polycarbonate transparency was damaged by gage installation, causing failure during the impact. Second, the gages lost range and failed prior to completion of the impact event.

Computer modeling of the impact event has been used extensively in an attempt to reveal the dynamics of the transparency during bird impact. These models intend to predict a stress and deflection. The deflection can then be related to that estimated value using the methods described above. This usually results in the model requiring adjustment to fit the result, leaving any future variable, such as speed, thickness, geometry or material unaccredited.

Potential Solution

No solution is offered to this very complex set of problems. Pursuit of solutions should be a priority.

Benefits of Solution

It is known that polycarbonate and other plastics are strain and strain rate sensitive. The designer needs to understand these limits and be able to relate these to the full scale event of bird strike. The benefits come in purely financial terms of optimized designs and greatly reduced testing costs. A fallout of this technology should be a more realistic material specification, and should open the possibility of new materials.

Sub-Scale Forming Simulation

Background/Objective

This sub-scale simulation technology is intended to implement and help validate software to assist in the preproduction study of formability of geometrical shapes as well as to provide the manufacturer with an ability to optimize forming processes before committing to full scale tooling.

Current State of Technology

The primary approach to full scale forming is one of doing what has worked in the past: experience. What is missing is that the past is not fully transferable, and even so, it may not have been the best approach. Typically, any manufacturing process tends to continue to be done the way it was started if the product is profitable. Developing new processes is risky and can be expensive. Put another way, if it isn't broke.....

Potential Solution

Using the predictive thermoforming model, design and develop suitable subscale tooling for demonstration of forming processes.

Benefits of Solution

The target is improved optic quality through a more controlled process. This translates to reduced initial investment and lower selling cost.

Cockpit Glare and Light Control

Background/Objective

Provide for variable control of cockpit exterior illumination. (Dial-A-Tint)

Current State of Technology

The industry does not currently offer transparency designs that provide for variable (controllable) light transmission for optimal viewing of cockpit displays. One exception, not yet generally adapted to complex transparency designs, is the use of UV activated photo sensitive compounds integrated into interlayer materials. PDA Engineering has conducted extensive research on this for nuclear flash protection. Joint IRAD programs between Lockheed (GD/FW) and Texstar, Inc have also pursued similar objectives, achieving some success in subscale flat coupons. Pilkington has been involved in MITS testing of photochromic materials. WL is currently active on an approach referred to as Dial-A-Tint that reduces/limits light transmission, but not to the degree required for nuclear flash. Another approach under investigation by commercial industry (PPG Industries) is Electrochromic activated transparencies. However, this approach currently appears confined to glass designs of limited size and curvature. One possible other problem is that some systems require power to retain a state of transparency.

Potential Solution

A specific solution is not known. Some considerations for development of an exterior light control system could include;

Combining electrochromics and photochromics to automatically control overall light, and also provide manual zone control.

Fail safe. A loss of power should result in full light transmission.

Benefits of Solution

Most of the benefits are peacetime related;

Reduced glare on flight instruments.

Reduced heat load in the cockpit during high ambient lighting conditions, while maintaining full light levels during reduced ambient conditions.

Custom Optics Design

Background/Objective

Develop a generic software program for the purpose of customizing theoretical optical quality by defining the relationship between the outboard and inboard surfaces (or all optically coupled surfaces). As an example of the situation, the F-16 transparency is compound contoured as is needed for single piece designs to aid manufacturing. The elevation component of optic error due to curvature is negative, while the error due to forming (wedginess) is positive. The shape of the error curves is not fully offsetting, and therefore, the greatest gain was not achieved during original design. The potential, however, is that a significant portion of the theoretical error can be compensated for by a controlled surface relationship. Other software package development within the predictive analysis software tools category needs to consider this program for surface definitions.

Current State of Technology

The existence of any suitable program is unknown.

Potential Solution

No technical solution is offered.

Benefits of Solution

This is considered the umbrella optics design tool, subordinating the other programs under the category of predictive analysis software tools since their results are components of the custom optics program output. The design capability represented by this program will significantly affect not only the current practice of thermoforming, but more importantly will control the newer designs being considered under the "frameless" design approach.

Technology Library

Background/Objective

There exists a need for the transparency community to have ready access to the enormous volume of data that has been published but is scattered among all the technical society. Each company, government agency, or manufacturer has their own limited library of information. None of these is complete and updated except through the ASTM and Transparency Symposium minutes. There is no known cataloging of data by subject material, key word association, or other methods. There is no central source from which the information can be accessed or acquired. Every new design starts from a zero database with only local data along with that provided during aircraft development through competitive situations. We need a central library that can be accessed via computer, and a method of soliciting selected data, reports, procedures, specifications, etc.

Current State of Technology

There is none. The data is scattered, users are not aware of existence of data, mistakes are repeated and testing is duplicated. There is little knowledge advancement except as transmitted during symposia and by competitive solicitations. "Lessons learned" is not being practiced. On the bright side, it is believed that UDRI plans to tackle at least part of the problem under contract with WL/FIVR.

Potential Solution

The UDRI effort may be the answer, or part of the answer. The stated need should be compared to the UDRI plan. If gaps exist, we need to pursue them.

Benefits of Solution

Self explanatory, see above statements.

High Temperature Polymers

Background/Objective

Polycarbonate and acrylic are commonly used for transparency structural materials. The heat deflection temperature (HDT) for polycarbonate is approximately 265°F. The heat deflection temperature for acrylic is approximately 210°F. The aerodynamic heating of high performance aircraft can easily exceed these relatively modest temperatures. Although limited excursions past a material's HDT are possible, transparency material could be a limiting factor in an aircraft's capabilities (especially a supercruiser).

A high temperature polymer would be useful as part of a system to defeat directed energy threats.

A high temperature polymer might serve as a better substrate material for ITO, a conductive coating. ITO is desirable for it's superior light transmission properties.

The objective would be develop a material that would have the thermal performance of glass but retain the weight, formability and impact resistance of lower temperature polymers.

Current State of Technology

Developmental high-temperature candidate surface-ply materials were evaluated during the MITS Program. However, these materials have not been demonstrated with coatings on full scale parts.

What about structural materials (polycarbonate replacements) such as AEC?

Potential Solution

Continue the quest for the "holy grail," - a material with all the desired properties including high heat deflection temperature.

Benefits of Solution

Increased flexibility to the designer in material selection. Opening up of the flight envelope for selected aircraft. Weight savings for aircraft that would otherwise be forcad to use glass to meet thermal requirements.

Anti-Reflective Coatings on Polymers

Background/Objective

Multiple images are formed in the transparency by the internal reflections of distant lights, such as from a runway. If the transparency is less than perfect, the pilot may see these as ghosts, and because of the high efficiency of the eyes, they appear comparable in intensity to the primary images. The presence of these images is distracting and can pose a safety problem.

Anti-reflective coatings would be used to control multiple images. The objective would be to develop an effective anti-reflective coating that can be applied to a liner, hard coat, or directly to polymers such as acrylic or polycarbonate.

Current State of Technology

To date, anti-reflective coatings have not been effective in eliminating multiple images.

Potential Solution

No technical solution is offered. Fully effective anti-reflective surfaces should be studied.

Benefits of Solution

Aircraft with forward transparencies that are other than flat can experience multiple images during night time operations. The use of cylindrical or conical shaped transparencies does not eliminate the problem since the pilot views through compound curvature at all positions except along the centerline in a perfectly formed part. The elimination, through original design, of multiple images reduces the unnecessary burden to the pilot of selecting/discriminating the real image during potentially hazardous night operations.

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AIRCRAFT TRANSPARENCY TECHNOLOGY DEVELOPMENT AND VALIDATION FOR TRANSITION TO DoD FLIGHT VEHICLE SYSTEMS

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Aircraft Transparency Technology Development and Validation for Transition to DOD Flight Vehicle Systems

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Summary

The investment portfolio for aircraft transparency technology developed by the Aircrew Protection Branch (FIVR) in Wright Laboratory (WL) targets not only evolutionary development of conventional technologies, but also includes revolutionary concepts for order of magnitude improvements in transparency system performance and cost of ownership. This paper presents an overall illustration of the aircraft transparency technology investment strategy being pursued by WL. Roadmaps of each cluster of programs targeted to meet WL Corporate Goals are presented and discussed. Brief discussion of key efforts in each of the roadmapped areas is included. Opportunities are outlined through the turn of the century to achieve new levels of mission performance while providing lower cost of ownership. In particular, transition opportunities are highlighted for technologies enabling directly formed and frameless aircraft transparencies. Suggested business strategies are presented through which industry might collaborate with WL in successful application of technology for directly formed and frameless aircraft transparencies.

Changes in the World of Systems Acquisition

DOD Streamlining

Within the Department of Defense (DOD), rapid change is being driven by powerful political and economic forces. The Cold War has ended, and the US is faced with the challenge of competing effectively in the world marketplace. The warfighting force is being drawn down drastically, and the Defense Budget is being slashed. In the arena of research and development, the DOD is pursuing strategies such as Project Reliance to enable collaboration across the Service Agencies, and to establish lead organizations for key areas.

Air Force Research & Development

Within the Air Force, leadership in streamlining processes for systems acquisition is reflected by major Command mergers. In July 1992, the Air Force Systems Command and Air Force Logistics Commands ceased to exist - both replaced by the joint command called Air Force Materiel Command. New policy and processes have been established by the Air Force to formally implement acquisition streamlining, and the former annual Planning, Programming, and Budget System has been replaced with the Biannual Planning, Programming, and Budget System (BPPBS). Major system development cycles now embrace the Integrated Weapon System Management (IWSM) concept with its single system point of responsibility from cradle to grave.

WL Mission

Within the Wright Laboratory (WL), one of the four Air Force super labs which have been established, emphasis has been placed on the acceleration of technology transition to system customers/users. Formal processes such as the Senior Engineering Technical Assessment Review (SENTAR) have been embraced by WL to interface with Aeronautical Systems Center engineering offices for effective transition of new WL technology.

Aircrew Protection Branch Vision

Opportunities

Within the swirling whirlwinds of change described above, the Aircrew Protection Branch (FIVR) members of the DOD "We Do Windows Gang" envision not only evolutionary progress, but also true breakthroughs in the development, integration, and validation of materials, design concepts, assessment techniques, and manufacturing methods for aircraft transparency systems. As shown in Figure 1, this new technology will permit balance to be achieved between performance and affordability in future aircraft transparency systems.

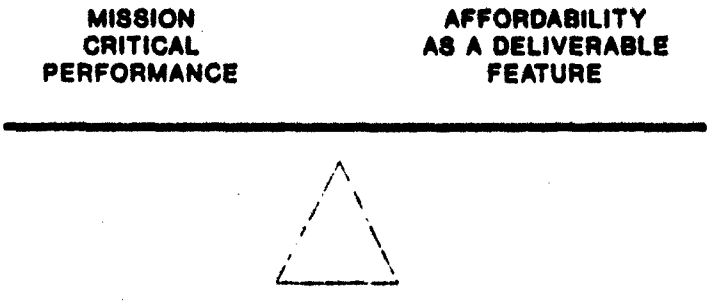


Figure 1. Balance Between Performance and Affordability

FIVR is pursuing both evolutionary and revolutionary solutions to problems which today still prevent the acquisition of affordable, mission compatible transparency systems. Figure 2 lists the key problem areas associated with current transparency systems.

- SHORT SERVICE LIFE
- HIGH UNIT COST
- INADEQUATE MISSION PERFORMANCE

Figure 2. Transparency System Problems

Goal

The goal of the FIVR program can be most succinctly stated as Validated "444" Transparency Technology. The meaning of the term "444" is illustrated in Figure 3.

- MORE THAN 4 YEARS OF MISSION COMPATIBLE SERVICE LIFE
- NO MORE THAN 4 TECHNICIANS REQUIRED FOR CHANGEOUT
- NO MORE THAN 4 HOURS REQUIRED FOR CHANGEOUT

Figure 3. "444" Transparency Technology

Increased Emphases

Through the remainder of the 1990's, and into the 21st century, FIVR sees the need for increased emphasis in a number of transparency system technology areas. These needs are driven by the opportunities and goals already discussed. Figure 4 lists key areas requiring increased investment of resources.

Decreased Emphases

Some technology development areas which have received significant FIVR resource investment in the past, will need decreased emphasis over the next decade. These areas include the development of stand-alone computer assessment tools, and the "hands on" acquisition of service life information through inspection of transparencies which are either installed in aircraft or have been removed from service.

- TRANSPARENCY RESTORATION/RECYCLE
- CONTROL OF COCKPIT AMBIENT LIGHT LEVEL
- BIRD IMPACT HAZARD DEFINITION FOR COMPUTER MODELING
- ELECTROSTATIC BEHAVIOR
- IN-HOUSE BASIC RESEARCH
- IN-HOUSE RESEARCH FACILITIES
- REMOVAL FOR CAUSE CRITERIA
- DIRECTLY FORMED FRAMELESS TECHNOLOGY TRANSITION
- AUTOMATED SERVICE LIFE DATA ACQUISITION
- DURABLE COATINGS FOR COMBAT HAZARDS
- INTEGRATED COMPUTER AIDED DESIGN/ENGINEERING/MANUFACTURING

(Note: Order of listing does not indicate priority).

Figure 4. Transparency Technology Areas Requiring Emphasis

"Business Expansion"

In summary, the Aircrew Protection Branch vision of the 21st century is that aircraft transparency technology represents a high leverage area which is ripe for increased investment in an era of declining resources and shrinking warfighting force. In short, what is required is a significant "business expansion" in this high payoff technology area.

Flight Dynamics Directorate Technical Area Plan

The annual publication of the Flight Dynamics Directorate (FI) which documents program plans/investment strategy is called the Technical Area Plan (TAP) for Air Vehicles Technology. The FI TAP is broken down into thrust areas, and further in core technology areas. Aircraft transparency technology efforts fall under the Vehicle Subsystems Thrust in the FI TAP, and within the Core Tech Area of Aircrew/Aircraft Survivability and Safety.

Aircrew/Aircraft Survivability and Safety

Within the Core Tech Area of Aircrew/Aircraft Survivability and Safety, the Vehicle Subsystems Division (FIV) has established three separate goals involving aircraft transparency technology. For each of the three FIV transparency technology goals, a separate program roadmap has been prepared for publication in the Fiscal Year 94 (FY94) FI TAP. Figure 5 lists the three FIV transparency technology goals.

- TRANSPARENCY DURABILITY ASSESSMENT METHODS
- TRANSPARENCY DESIGN TOOLS/CONCEPTS
- TRANSPARENCY TECHNOLOGY OPTIONS

Figure 5. Vehicle Subsystems Goals for Transparency Technology

Wright Laboratory Investment Portfolio

Transparency Durability Assessment Methods Roadmap

Figure 6 shows the FI TAP Roadmap to develop and validate analytical and experimental durability assessment methods. These methods will extend transparency system durability in operational mission environments, and as a result substantially reduce the cost of transparency system ownership.

The first exploratory development (Program Element 62201F) program shown in Figure 6, "Failure Analysis for Polycarbonate Transparencies," and its FY95 follow on, "Failure Analysis for Transparencies," are both efforts to sponsor research in university and industrial centers. Multiple awards are in place under the current effort, and are planned for the future effort. Current studies in progress focus on preventive and predictive criteria for polycarbonate fracture. The responsible FIVR project engineer is currently Mr. Richard Smith. Other papers planned for this conference which document individual efforts under this program include, "Fatigue Analysis of Polycarbonate Transparencies" from Purdue University Calumet, "Physical Aging of Polycarbonate by Free Volume Considerations" from Battelle Columbus Laboratories, and "An Experimental Evaluation of the Effect of Hole Fabrication/Treatment Techniques on Residual Strength and Fatigue Life of Polycarbonate Specimens with Holes" by the University of Dayton Research Institute.

The second exploratory development program shown in Figure 6, "Transparency Crazing/Fracture Mechanics," is an on-going effort to build in-house expertise in evaluation and assessment of transparency systems. Current capabilities include pressing and annealing samples, artificial aging exposure with ultraviolet light and humidity, dynamic mechanical thermal analysis, digital microscope with image processing for coating erosion and surface quality, and angular deviation mapping for (up to) very large transparencies. Planned capabilities include gel permeation chromatography, mechanical testing, and dust erosion testing. FIVR project engineer, Mr. Richard Smith, Palace Knight Research Engineer, Mr. Steven Clay, and technician, Sgt John Williams are

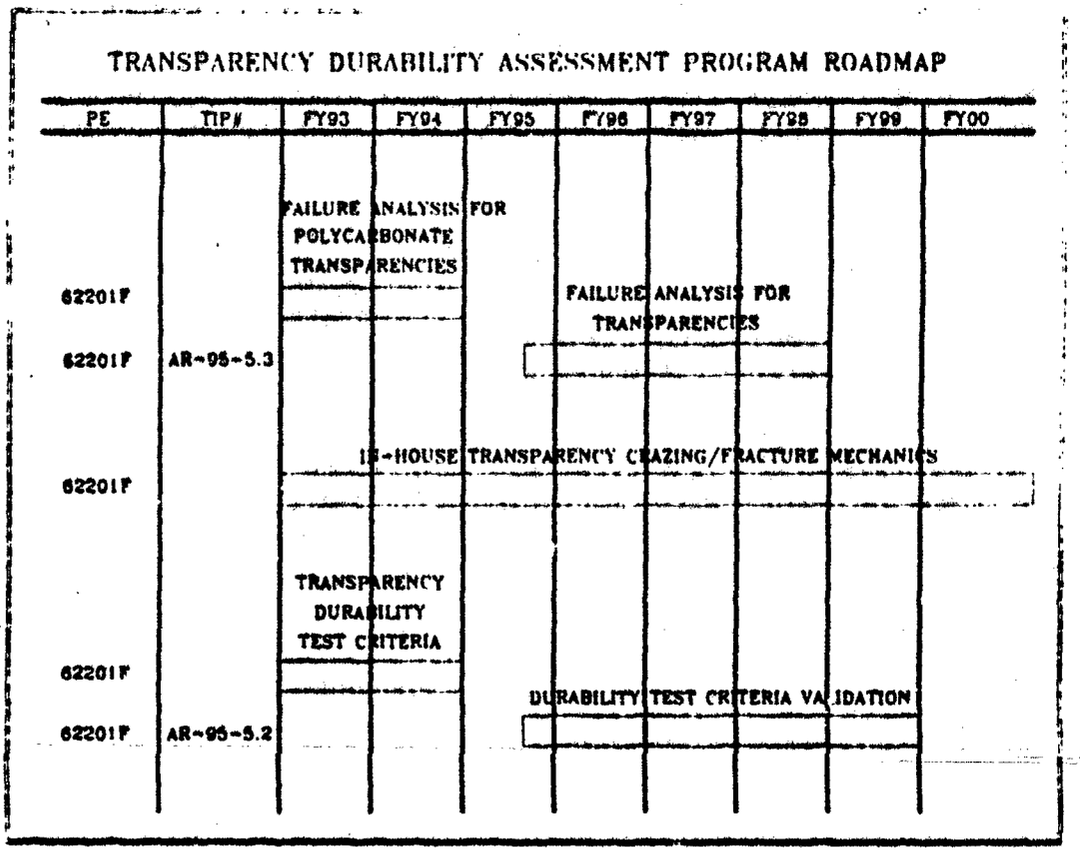


Figure 6. Transparency Durability Assessment Program Roadmap

responsible for the in-house program.

The third exploratory development program shown in Figure 6, "Transparency Durability Test Criteria," and its FY95 follow-on program, "Durability Test Criteria Validation" will provide a coupon scale test methodology to predict service life of new transparency systems. Application of this methodology and associated requirements/criteria will serve as the first "yardstick" available for use by the acquisition community to enable selection of systems for production based on lowest cost of ownership. FIVR project engineer Mr. Richard Smith is responsible for this program. Other papers planned for this conference which document efforts under this program include, "Development of a Transparency Durability Test Criteria: Coupon-Scale Testing and Field Service Data Analysis," and "Pressure Burst Testing of KC-135 Celestial Navigation Windows," both from the University of Dayton Research Institute.

A fourth and fifth program are not shown in Figure 6, but will be discussed here. One of these programs is not shown in Figure 6 because funding was completed prior to FY93, and the other is not shown because funding was obtained subsequent to development of the FY94 FI TAP.

The fourth program which is associated with Figure 6 is a Phase II Small Business Innovative Research (SBIR) program for stress measurement in structural plastics using acoustic waves. A field portable device will be delivered which can be used to non-destructively assess the level of outer fiber stress in transparencies either during manufacturing, installed on the aircraft, or removed from service due to some failure. Application of this device will permit selection of transparency systems which afford longer service life with respect to crazing, a common mode of failure driven by the level of stress present in structural plastics. The FIVR project engineer is Mr. Richard Smith, and the paper planned for this conference to document this effort is "Stress Measurement in Structural Plastics by Lambda-Critical Waves" from the Analytic Engineering Company.

The fifth program which is associated with Figure 6 is another Phase II SBIR program, this one for an imbedded or in-place data recording system which will be demonstrated for an F-16 canopy system. The microprocessor based system will include numerous transducers to sense key environmental factors in manufacturing, operations, and storage such as temperature, humidity, stress, and ultraviolet flux. Implementation of this device will permit greater understanding of the environment "seen" in service. This knowledge will allow design of more durable transparency systems better able to survive the service environment. Great potential for dual use spinoffs are envisioned from this work, such as solid state memory for personal computers to replace rotating floppy discs. The FIVR project engineer is Mr. Richard Smith, and the paper planned for this conference to document this effort is "Transparency Recorder for Obtaining In-Place Environmental Life History" from Nonvolatile Electronics Incorporated.

Transparency Design Tools/Concepts Roadmap

Figure 7 shows the FI TAP Roadmap to develop and validate design tools and concepts. Application of these "rules and tools" will reduce the vulnerability of transparency systems in the hostile natural and combat environments, as well as significantly reduce cost of ownership. The "rules and tools" under development here represent the second generation of methods produced by the Wright Laboratory. In the 1980's, WL developed and transitioned a number of stand-alone assessment tools to support transparency system design such as structural analysis and aerothermodynamic analysis codes. Now in the 1990's, these tools are being upgraded and combined into fully integrated systems for concurrent design which run on affordable engineering workstations. Project personnel at various levels such as program manager, designer, and engineering analyst can all employ the new integrated systems to accomplish

their individual roles as members of the concurrent design and development team.

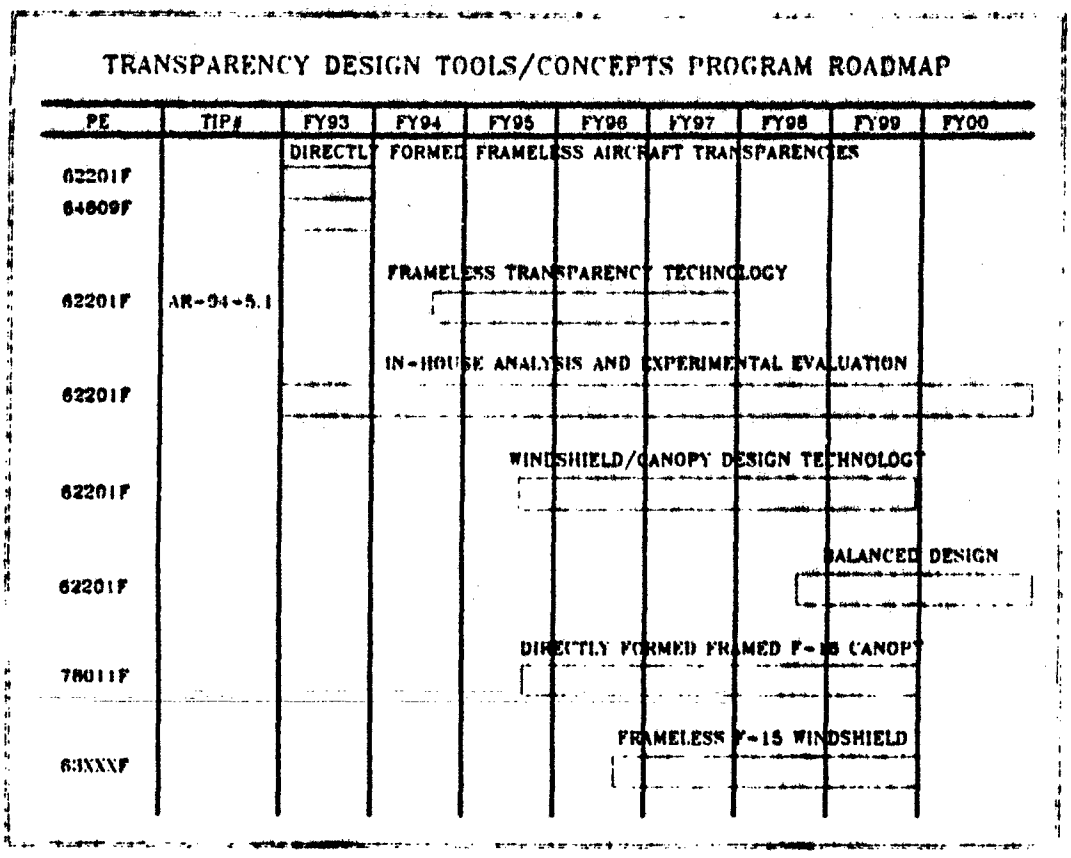


Figure 7. Transparency Design Tools/Concepts Program Roadmap

The first program illustrated in Figure 7, "Directly Formed Frameless Aircraft Transparency," is an effort jointly funded from Program Elements for exploratory development (62201F), and for engineering development (64609F). The principal product of the effort is an integrated computer aided tool for the design, engineering, and manufacturing (CAD/CAE/CAM) of directly formed frameless aircraft transparencies. This CAD/CAE/CAM tool is called the Analytical Design Package (ADP), and will be validated/confirmed via fabrication and extensive ground based testing of a Confirmation Frameless Transparency (CFT). The CFT moldlines were derived from the forward portion of the F-16 production canopy, and testing will address portions of the production specifications for the F-16 canopy. The technical Program Manager for this program is Mr. W. R. (Bob) Pinnell; the Program Manager for new business is Capt R. (Becky) Wagner; the technical lead for the ADP is Mr. Mike Gran, all of FIVR.

Many papers have been planned for this conference to document individual efforts under this program and they include, "Analysis of Transparency Panel Shrinkage Characteristics and Their Application to Transparency Mold Design" and "Dimensional Mapping and Shrinkage Characterization of Large, Thick-Walled, Directly-Formed Transparent Panels" from Florida International University; "Mechanical Properties Evaluation of Frameless Transparency Materials," and "Bird Impact Resistance Evaluation of Frameless Transparency Materials" from the University of Dayton Research Institute; "Optical Evaluation of Transparencies Utilizing New Test Apparatus," "Development of Directly Formed and Frameless Aircraft Transparency Technology, An Overview," and "Analytical Simulation of Low Pressure, Thick Walled Injection Molding of Aircraft Transparencies" from the Wright Laboratory; "Injection Molding as a Direct Forming Method for Aircraft Transparencies" from Envirotech Molded Products; and "The Use of Computerized Materials Data in ADP2," and "Analytical Design Package - ADP2 A Computer Aided Engineering Tool for Aircraft Transparency Design" from PDA Engineering.

The second exploratory development program shown in Figure 7, "Frameless Transparency Technology," is planned as a competitive procurement scheduled for FY94 award. Because it is foreseen that injection molds for complex transparency shapes will need to be segmented, this program is planned to provide risk reduction having to do with the "witness lines" which will remain on the transparency surface as a result of mold segmentation. This program will explore technologies to either circumvent the witness line issue through a one piece mold liner for example, or deal with witness lines through innovative methods for mold design for example. The FIVR project engineer is currently Mr. Mike Gran.

The third exploratory development program shown in Figure 7, "Analysis and Experimental Evaluation," is an effort to facilitate the development of WL in-house expertise in the application of concurrent design tools, not only for directly formed and frameless transparencies, but also for more conventional transparency systems. It provides maintenance, and upgrading of software and hardware systems for concurrent transparency design, as well as training for users/customers of this technology. The FIVR project engineer is Mr. Mike Gran.

The fourth exploratory development program shown in Figure 7, "Windshield/Canopy Design Technology," is a competitive procurement scheduled for FY95 award. This program is essentially a follow-on program for further development of the Analytical Design Package (ADP). It will produce an expanded capability for concurrent design with features like optical performance being included. The FIVR project engineer is Mr. Mike Gran.

The fifth exploratory development program shown in Figure 7, "Balanced Design," is a competitive procurement scheduled for FY98 award. This program is planned to "roll up" the Durability Assessment Methods produced by the programs illustrated in Figure

6, into the Design Tools produced by the programs illustrated in Figure 7. The result would be a superset of methods, criteria, rules, and tools enabling the design of transparency systems providing acceptable balance between mission compatible performance and supportability. The FIVR project engineer is Mr. Mike Gran.

The sixth program shown in Figure 7, "Directly Formed Framed F-16 Canopy," is planned to transition technology for direct forming of transparencies to the F-16 system. The program will develop and qualify a form, fit, and function F-16 canopy which would be directly formed (injection molded), but which would bolt into the production metal F-16 canopy frame with no modifications to the frame. This program will provide a preferred spare F-16 canopy offering lower cost of ownership and turn of the century mission compatible performance. The directly formed framed F-16 canopy spare will represent the next opportunity to transition new transparency technology to the F-16 fleet. subsequent to the 540 Knot F-16 Canopy program currently being conducted by the FIVR/Ogden ALC team and discussed later in this paper. A candidate for funding is the Manufacturing Technology Directorate of the Wright Laboratory (PE 78011F).

The seventh program shown in Figure 7, "Frameless F-15 Windshield" is also known within the Air Force as the "Next Generation Transparency" program. This program will develop and flight demonstrate a truly frameless F-15 windshield which offers lower cost of ownership, and meets turn of the century requirements for mission compatible performance. The frameless F-15 windshield system will represent the next opportunity to transition new transparency technology to the F-15 fleet, subsequent to the 540 Knot F-15 Windshield program currently being conducted by the FIVR/Warner-Robbins ALC team and discussed later in this paper. A candidate for funding is the FY96 6.3 POM.

Transparency Technology Options Roadmap

Figure 8 shows the FI TAP Roadmap to develop and demonstrate transparency technology options. These options will reduce vulnerability in the hostile natural and combat environment to enhance durability and reduce life cycle costs. The return on investment in this particular FIV goal area has been extraordinary. The value of airframes alone which survived what would otherwise have been catastrophic windshield bird impact incidents exceeds $750M.

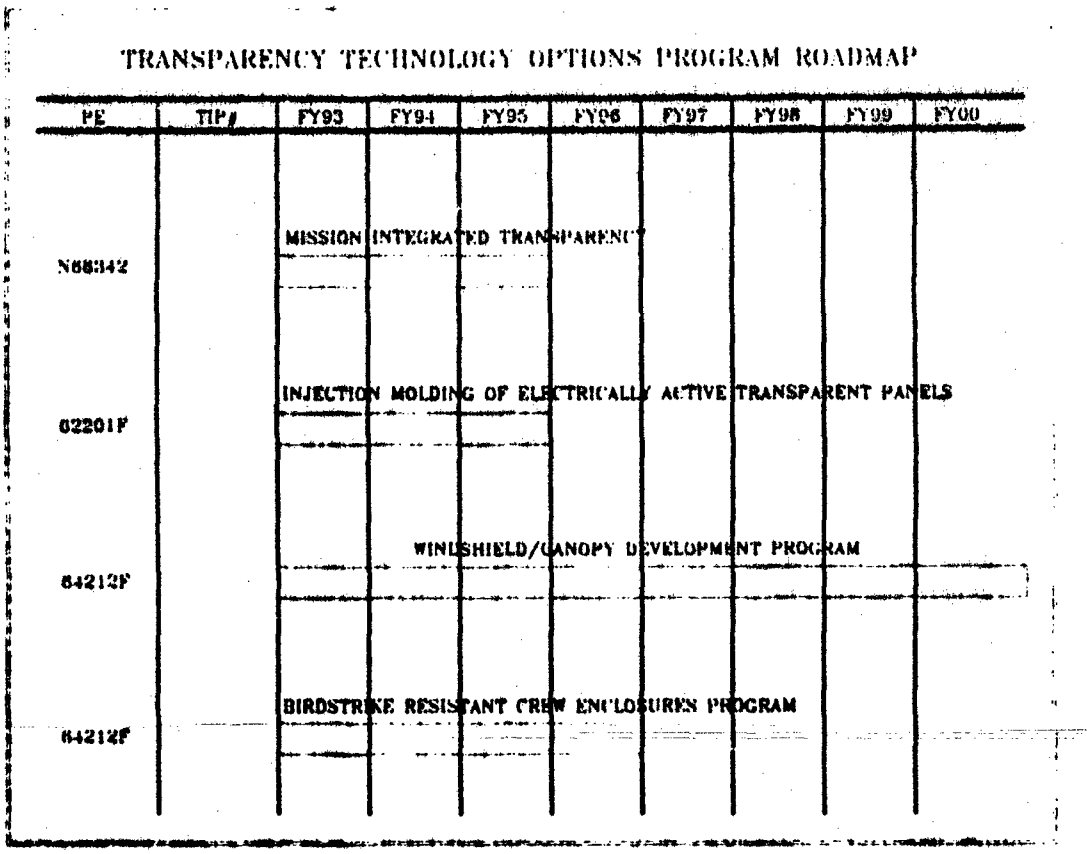


Figure 8. Transparency Technology Options Program Roadmap

The first program shown in Figure 8, "Mission Integrated Transparency," will transition technologies for bird impact protection and mission based design to the F/A-18 system for the Navy. Mr. James Terry is currently the program manager for this effort.

The second program indicated in Figure 8, "Injection Molding of Electrically Active Transparent Panels," is a research effort to explore the feasibility of directly forming transparent materials which are electrically conductive using a low pressure injection molding process. Lt Erik Joy is the current program manager for this effort.

The third program in Figure 8, "Windshield/Canopy Development Program," comprises multiple projects teaming FIVR with respective Air Logistics Centers, each developing and validating through flight test technology options for a specific system. Projects

currently include an upgraded F-16 canopy, T-38 student windshield, F-15 windshield, and B-1B windshield. Current project managers are Lt Joe Davisson, Lt Joe Coogan, Lt Guy Graening, and Lt Joe Coogan respectively. The overall program manager is currently Lt Guy Graening.

The fourth program in Figure 8, "Birdstrike Resistant Crew Enclosures Program," is a contracted program to provide technical support for the in-house "Windshield/Canopy Development Program." The current program manager is Lt Guy Graening.

Transition Opportunities

A number of transition opportunities are being pursued through the turn of the century to achieve new levels of mission performance while providing lower cost of ownership. From the programs illustrated in Figure 8, transition opportunities include a T-38 student windshield system with an all composite frame, an F-16 canopy system, an F-15 windshield system, a B-1E windshield system, an F/A-18 windshield system, and abrasion resistant windshield systems for helicopters.

Two transition opportunities from programs illustrated in Figure 7 will be highlighted in this paper because they involve directly formed and frameless aircraft transparencies. The first of these two transition opportunities is for a directly formed (low pressure injection molded) framed F-16 canopy. The second transition opportunity is for a frameless F-15 windshield.

Directly Formed Framed F-16 Canopy

Plans for a FY95 start for a program to develop and qualify a directly formed framed F-16 canopy have been endorsed by critical transition team members. Figure 9 illustrates the acquisition strategy developed to date for this program. The F-16 Systems Program Office (IWSM) has provided written commitment to implement the system after qualification. The Ogden Air Logistics Center has provided written commitment to acquire the qualified system as a preferred spare. SENTAR approval of the formal Technology Transition Plan has been obtained with WL/FIVR identified for overall technical program direction.

Sacramento Air Logistics Center has provided written commitment to acquire in-house engineering design expertise, and to provide program management. Potential investors indicating interest in providing program funding include ASC/SMT (RAMTIP), and WL/MT (Manufacturing Technology). The qualified spare canopy resulting from this program will offer $200M cost of ownership savings over the remaining life of the F-16 fleet. Consistent, high quality optics; state-of-the-art abrasion resistant coatings; and 21st century mission compatible performance will be provided.

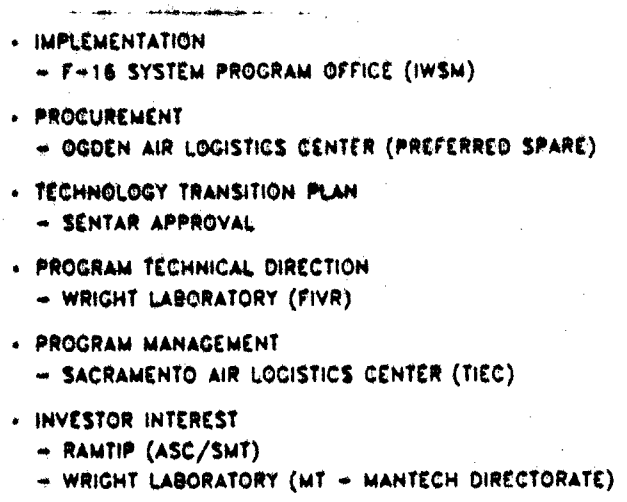


Figure 9. Strategy for Directly Formed Framed F-16 Canopy

Frameless F-15 Windshield

Plans for a FY96 program start to develop and qualify a frameless F-15 windshield system are gaining support. Figure 10 illustrates current acquisition strategy for this program. SENTAR approval of the Technology Transition Plan (TTP) has been obtained with WL/FIVR identified for program management. Potential investors include SAF/AQ providing 6.3 funding for the effort. Headquarters Air Combat Command has invited briefings on the program, and has been given the opportunity to sign the TTP. The qualified spare F-15 windshield resulting from this program will offer $75M cost of ownership savings over the current production design for the remaining life of the F-15 fleet, along with turn of the century mission compatible performance.

Transition Strategy

Wright Laboratory Technology Transition Management Process

Figure 11 illustrates the novel technology transition management process developed within FIVR for aircraft transparencies. This management process is discussed in detail in another paper planned for this conference titled, "Acquisition Management of Consumable Technology Utilizing a Novel Technology Transition Process," from Wright Laboratory. This process is product oriented, customer driven, and hinges on a team approach to solving user problems.

- IMPLEMENTATION/PRODUCTION
  - WARNER ROBBINS AIR LOGISTICS CENTER (IWSM)
- TECHNOLOGY TRANSITION PLAN
  - SENTAR APPROVAL
- PROGRAM TECHNICAL MANAGEMENT
  - WRIGHT LABORATORY (FIVR)
- INVESTOR INTEREST
  - SAF/AQ (63XXXF)
- USER ENDORSEMENT
  - HEADQUARTERS AIR COMBAT COMMAND (ACC/DRL)

Figure 10. Strategy for Frameless F-15 Windshield

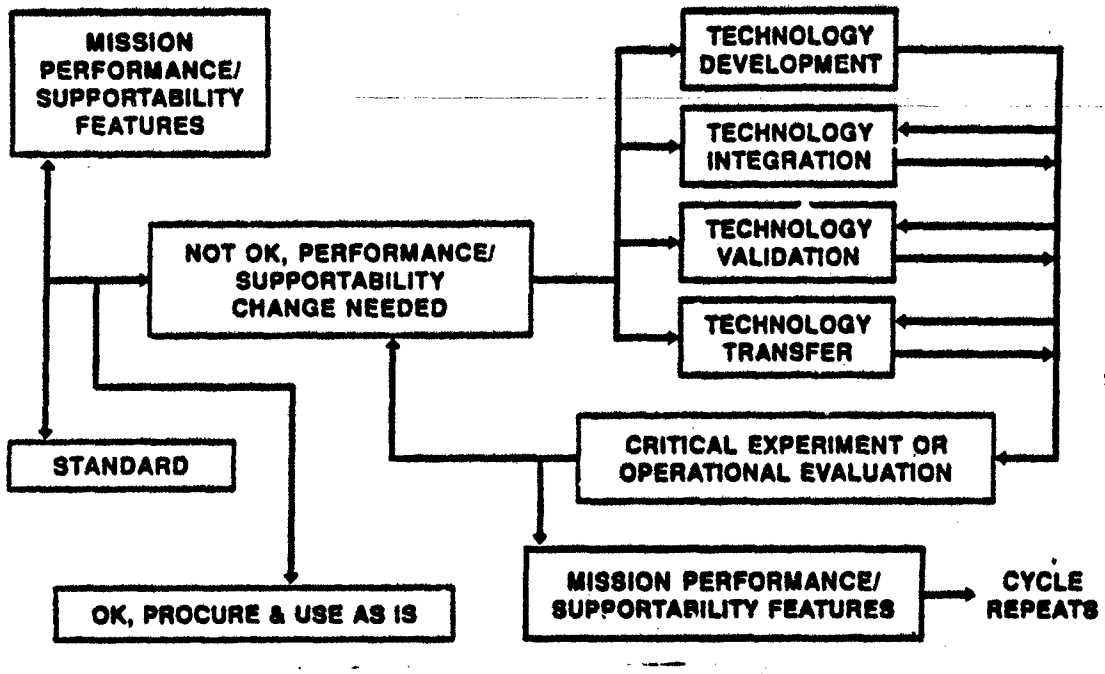


Figure 11. FIVR Technology Transition Management Process

Existing Teams

Effective teams are in place and functioning for a number of programs validating conventional or evolutionary technologies. These teams are working programs already discussed such as the T-38 windshield, F/A-18 windshield, B-1B windshield, F-16 canopy, and F-15 windshield.

Future Teams

Teams are currently in the process of being formed and strengthened for other FIVR programs validating truly revolutionary technologies such as direct forming processes, and frameless design concepts. Candidate systems for transition of direct forming technology include the F-16 canopy as already discussed, and the F-22 for which a non-uniformly thick injection molded canopy could enable significant weight savings. Candidate systems for transition of frameless technology include the F-15 as already discussed, and new acquisition programs.

Suggested Business Strategies

Considerable dialogue regarding possibilities for new business strategies within the aircraft transparency community is resulting from WL efforts in directly formed and frameless transparency technology. More dialogue will be necessary to put effective teaming arrangements in place to achieve the significant payoff offered by this revolutionary technology to the Air Force, and ultimately to the US taxpayers.

Conversations for possibilities to date regarding teaming for directly formed and frameless transparency technology application have suggested the business strategy illustrated in Figure 12.

Engineering organizations at airframe companies, research centers, and Air Logistics Centers may become the principal users of the inexpensive but powerful integrated computer aided "rules and tools" for design, engineering, and manufacturing. The transparency vendor industry may provide integration expertise to develop directly formed structural shells into complete systems meeting balanced requirements for mission performance and supportability. New team members which have not historically been part of the transparency technology community may emerge as subcontractors to traditional transparency vendors. Subcontracted tasks might include injection mold design and fabrication, as well as production injection molding.

A significant portion of the investment for this new technology will be borne by the Air Force, and provided for use by development teams in the form of Government Furnished Equipment and Support. The Computer Aided Design, Engineering, and Manufacturing tools, along with training and maintenance for these tools forms the "backbone" of the FIVR investment portfolio for the future as

already discussed. The goal in this arena is to employ only Commercial Off The Shelf (COTS) software, fully supported by the software industry.

- INTEGRATED COMPUTER AIDED DESIGN/ENGINEERING/MANUFACTURING
  - AIRFRAME COMPANIES
  - AIR LOGISTICS CENTERS
  - RESEARCH CENTERS
- SYSTEM INTEGRATION
  - TRANSPARENCY VENDORS
- INJECTION MOLD DESIGN/FABRICATION/MOLDING
  - MOLD DESIGN/MOLDING HOUSES
- COMPUTER AIDED DESIGN/ENGINEERING/MANUFACTURING RULES/TOOLS
  - AIR FORCE (COMMERCIAL OFF THE SHELF SOFTWARE)
- HARDWARE (INJECTION MOLDS/HEATING & COOLING PLANTS)
  - AIR FORCE (GOVERNMENT FURNISHED EQUIPMENT)
- RISK REDUCTION TECHNOLOGIES (MOLD LINERS)
  - AIR FORCE (EXPLORATORY DEVELOPMENT)

Figure 12. Suggested Business Strategies for Application of Directly Formed and Frameless Transparencies

Other examples of Government furnished equipment include large scale heating and cooling systems for thermal control of injection molds. The current heating and cooling system available from FIVR has been sized for large, one-piece transparencies such as the F-16 or F-22 canopies. Injection molds for systems which are successfully qualified such as the F-16 canopy, and F-15 windshield will be available for production phases of work.

In addition, continuing investment in risk reduction technologies are also part of the FIVR investment portfolio for this technology area. Concepts such as one-piece mold liners for use with segmented molds to avoid witness lines on molded articles will be explored. Alternate direct forming processes such as coinjection molding for laminated structures, and casting will be studied. Useful results will be folded in on-going system development and qualification programs.

Acquisition Plan

The FIVR vision for application of directly formed and frameless aircraft transparency technology centers on collaboration with industry to refine preliminary business strategies. Figure 13 shows a number of key issues which need to be dealt with to achieve win/win solutions to potential problems.

- DEVELOPMENT OF WORKABLE SCHEMES FOR TEAMING
  - EXPERIENCED VENDORS FOR CONVENTIONAL SYSTEMS
  - NEW TEAM MEMBERS FOR MOLD DESIGN/MOLDING
- AIR FORCE SMALL LOT BUYS VS LARGE MOLDING PRODUCTION RUNS
  - IMPACT OF STORING LARGE NUMBERS OF SPARES
  - ECONOMICS OF MOLDING SMALL NUMBERS OF ARTICLES

Figure 13. Directly Formed and Frameless Aircraft Transparency Technology Application Issues

One issue needing further resolution is the development of workable schemes for teaming among airframe houses, transparency vendors, and new team members such as injection mold designers or fabricators and molding companies.

A second issue revolves around the aircraft transparency community practice to date of producing relatively small lots of new or spare systems; e.g., a recent B-1B windshield system spares buy was for about 30 windshields. Historically, it has been common practice in the injection molding industry to produce relatively large numbers of a particular article in any one production run. Many issues related to this apparent conflict of scales need to be explored fully, such as the impact of storing large numbers of spares, or the economics of injection molding relatively small numbers of production articles in a single run.

FIVR is eager to work with other team members to acquire customer or user endorsements for planned programs, and to obtain investor commitments. The ultimate goal of the "We Do Windows Gang" in Wright Laboratory is to produce superior weapons systems which enable US forces to fly, fight, and win.

FIVR extends invitations to every organization which has an important role to play in achieving these goals: key companies and organizations which have been long standing members of the aircraft transparency system development community, as well as capable new team members who can contribute to the development of revolutionary new transparency systems for 21st century.

SESSION II

CURRENT SYSTEMS - PART A

Chairman: R. Speelman
Flight Dynamics Directorate
Wright Laboratory

Co-Chairman: G. Stone
Lockheed - Ft. Worth

Coordinator: G. Graening
Flight Dynamics Directorate
Wright Laboratory

ADVANCED TRANSPARENCY DEVELOPMENT FOR USAF AIRCRAFT

Michael P. Bouchard
University of Dayton

1 Lt Joseph C. Davisson
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ADVANCED TRANSPARENCY DEVELOPMENT FOR USAF AIRCRAFT*

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ABSTRACT

The development of advanced aircraft transparencies which meet USAF 1995-2000 mission requirements is described and illustrated. The key 1995-2000 mission requirements are first identified. A review of available birdstrike test data is then presented. This review demonstrates that the birdstrike requirement can be met with current technology. Based on the review, a set of design guidelines is given for improved bird impact resistance while meeting the additional future requirements. Subscale coupon testing to screen potential laminates for impact resistance and to obtain mechanical properties for use in finite element analysis (FEA) is then given. The mechanical properties of transparency polymeric materials have been found to be sensitive to strain rates, necessitating the tests to ensure accurate results in subsequent FEA. Nonlinear dynamic FEA for final analytic bird impact evaluation is described next. This task utilizes the recently-developed X3D explicit FEA code which provides strain rate-dependent material properties, direct modeling of the bird, element and ply failure criteria, and rapid turnaround on workstation and mainframe computers. Final design selection in view of their ability to meet all design requirements is described. Full scale fabrication and testing are underway, with initial findings presented herein.

I. INTRODUCTION

Many of the windshield and canopy (transparency) systems developed for current USAF combat aircraft were designed primarily for the high altitude air-to-air role. Projections for 1995-2000 missions forecast an increased birdstrike risk due to a greater frequency of high speed, low altitude flights. In response, programs have been initiated to develop advanced transparency systems capable of providing mission-compatible bird impact protection. These programs address additional 1995-2000 mission objectives, including meeting the USAF's "444" performance and supportability goal (maintain mission performance requirements for a minimum of 4 years, after which the transparency can be changed out in 4 hours by 4 technicians), retaining optical compatibility with emerging HUD (Head-Up Display) and night vision goggle technologies, and providing a path for incorporation of protection against emerging combat hazard threats, such as lasers.[1] The canopies must continue to meet all current specifications for mold line, weight, fit, and so forth.[2] Designs must also be affordable and competetively procurable.

A thorough engineering approach toward advanced transparency development is presented.[3] In brief, the approach consists of: (1) review of available birdstrike test data to obtain insight into the current state of the art and to develop guidelines for improved bird impact resistance; (2) design laminates which follow the birdstrike design guidelines of step (1) and provide the flexibility to incorporate technologies which will meet the other design requirements; (3) fabricate and test subscale coupons to screen potential laminates for impact resistance and to obtain mechanical properties at high strain rates for use in finite element analysis (FEA); (4) perform nonlinear explicit dynamic FEA for final analytic bird impact evaluation; (5) select the best designs in view of all design requirements; (6) fabricate prototype advanced transparencies, and (7) perform full-scale birdstrike, durability, and flight tests. Steps (1) - (5) have been completed, with Steps (6) and (7) currently in progress.

II. TEST DATA REVIEW

All of the available canopy bird impact test data were compiled into one volume along with an engineering interpretation of their significance for improved birdstrike protection.[4] The compilation and review considered canopy stiffness, bird impact

* Originally published as AIAA Paper No. 93-1391 with same title in proceedings for 34th Structures, Structural Dynamics and Materials Conference, April 19-21, La Jolla, CA.

capability, laminate construction, HUD effects, and canopy weight. The most impact-resistant laminates were found to incorporate at least 0.5 in of polycarbonate, ranging from one to three plies. Based on deflections at the pilot's head position, laminated polycarbonate-acrylic designs were found to be as stiff as monolithic polycarbonate designs of comparable thickness. The HUD proved to be a critical factor in canopy bird impact capability because contact with the canopy during bird impact could cause catastrophic fracture of the unprotected structural ply. HUD contact was found to limit the monolithic polycarbonate impact capability to 400 knots. Canopy weight was also determined to be an important issue, with the allowable being 139.5 pounds, and typical weights for monolithic and laminated canopies being, respectively, 136 and 130 pounds. Due to the HUD contact and weight limitations, the monolithic polycarbonate canopy was judged to be limited to its current 0.75 inch configuration, with 400 knot impact resistance, while laminated canopies appeared to allow for some growth in thickness and weight, permitting impact resistance of at least 500 knots.

Based on the review findings, design guidelines for the development of an advanced canopy capable of successfully resisting a high speed impact (in this case, 540 knots) with a four pound bird were then developed. In brief, these guidelines recommended a laminated polycarbonate canopy construction, which includes a single main structural polycarbonate ply (for improved optics over multi-structural ply designs) at least 0.5 inch thick, emerging advanced abrasion-resistant and static drain coatings applied to a thin outboard polycarbonate ply, and an inboard polycarbonate or cast acrylic inner ply to protect the main ply from direct contact with the HUD during bird impact. The current laminated canopy has the main polycarbonate ply but does not incorporate an inner protective ply. In addition, cast acrylic is used for the outer ply of the current production laminated canopy and is subject to crazing and cracking, which generates a substatial cost penalty in comparison to the desired four year service life.

Designs which follow these guidelines will not only meet the bird impact requirement, but will also meet durability, weight, optics, and fit requirements. In addition, the polycarbonate outer ply provides a means for later incorporation of combat hazard protection (application of coatings and/or substitution of new ply materials) as those technologies are developed. Finally, the increase in service life is expected to offset any increase in the cost of the advanced canopies.

III. LAMINATE IMPACT TESTS

At the time the program was initiated, successful determination and comparison of laminated canopy bird impact resistance using FEA technology entailed a fairly high uncetainty. Primary reasons for the uncertainty were the limited availability of high-priced computing power, and the lack of appropriate material properties and failure models for input into the FEA models, which would affect the accuracy of the results (see Section IV). Some experimental means was needed to pare down the potentially large analysis matrix to only the most promising designs.

The experimental method selected was a high-rate three-point beam test (Figure 1), which provided a relatively inexpensive means of evaluating

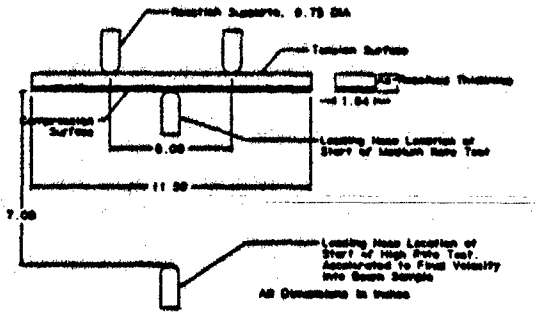


Figure 1. Three-Point Beam Test Setup.

the impact resistance of various laminated configurations. The method, which used beam geometry from ASTM-F736, employed MTS servohydraulic test equipment.[5] The test was performed at high loading nose velocities (2000 and 40,000 in/min), resulting in nominal strain rates of up to 130 in/in/sec in the ply materials, which was in the range of strain rates experienced by full-scale canopies subjected to birdstrike.[6-8] The test thus provided a measure of a laminate's ability to resist impact conditions akin to that of bird impact.

The test ranked impact performance of prototype laminates relative to current production laminates on the basis of peak sustained load, total absorbed energy, and failure mode. Figure 2 presents typical load-displacement curves for a current production laminate and a prototype laminate. The prototype laminate is seen to absorb 40% more energy (area under the load-displacement curve) and sustain

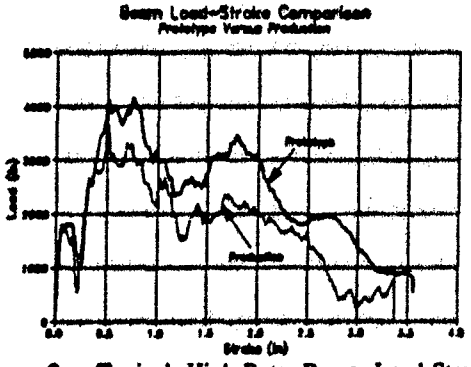


Figure 2. Typical High-Rate Beam Load-Stroke Curves.

13% more peak load than the production laminate. The tests demonstrated that five laminates exhibited significantly greater performance than the current production laminate. One laminate, which had utilized as-cast acrylic as the main structural ply sandwiched between two thin polycarbonate plies, was considered unacceptable due to fracture of the acrylic ply and subsequent loss of energy absorption. Another laminate exhibited outer ply delamination, with marginal impact performance. The test results also indicated that polycarbonate outer plies generally showed better impact resistance than acrylic outer plies when tension-loaded because the polycarbonate deformed plastically compared to the cast acrylic, which exhibited brittle fracture. In compression, the impact resistance of polycarbonate and acrylic outer plies was comparable because the acrylic, though it fractured, tended to have its pieces pushed together, providing load resistance. There was no significant difference in impact resistance between two-structural-polycarbonate-ply and single-structural-polycarbonate-ply designs. In view of this result, and due to the increased fabrication costs and more challenging optics associated with two-structural-ply designs, it was recommended that laminates incorporate a single polycarbonate structural ply.

IV. HIGH RATE MATERIAL TESTING

Tests to determine mechanical properties of individual ply materials were conducted next. The response to dynamic loading of the polymeric materials typically used to fabricate aircraft transparencies can be highly strain rate dependent.[9] Accurate FEA prediction of transparency response to bird impact therefore requires accurate strain rate-dependent material properties as input data, along with appropriate material models in the FEA code to properly account for the rate dependencies. Unfortunately, little rate-dependent testing of the materials of interest has been performed and very little data is available in the open literature. Such testing is not straightforward because of wave propogation phenomena and high elongations of some of the materials.

MTS servohydraulic test apparatus was used because high test rates (loading velocities) could be achieved, which, in conjunction with appropriate specimen sizing, resulted in material strain rates of up to 100 in/in/sec.[9] Polycarbonate, as-cast acrylic, and polyurenthane samples were tested. The goal of the polycarbonate and acrylic tensile tests was to obtain the entire stress-strain curve for various strain rates, while for the urethane tensile tests the goal was to obtain the tensile modulus for various strain rates. Tensile specimens for polycarbonate and acrylic (mini round tensile rod) and urethane (flat dogbone) are shown in Figures 3 and 4. Grips used were screw-in

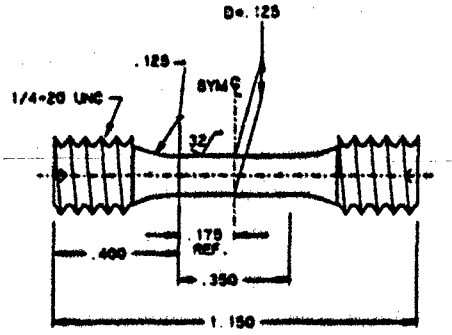


Figure 3. Polycarbonate & Acrylic Tensile Sample.

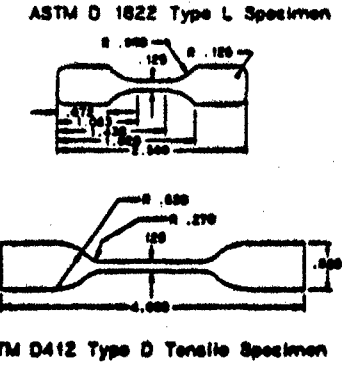


Figure 4. Interlayer Tensile Samples.

for the rods, friction for the D 412 dogbones, and shoulder for the D 1822 dogbones. The desired strain rates were achieved by using a short gage length

(since axial strain rate is essentially the pull velocity divided by gage length), which was designed into the rods and the D 1822 dogbones, while for the D 412 dogbones it was achieved by "choking up" on the test section with the friction grips. Very small (0.031 inch grid length) axially-aligned strain gages were bonded to the side of selected rod gage sections to monitor strain. Extensometers were similarly used for selected dogbone samples. The strain gages were effective at all strain rates. The inertia of the extensometers precluded their use for test rates above 10 in/sec, necessitating scaling of measured strain-displacement curves to the high rate displacement time histories.

The goal of the shear tests was to obtain shear modulus of each of the test materials as a function of strain rate. A torsional shear test configuration utilizing an MTS tension-torsion tester was chosen for these tests. Sample geometries, which appear in Figures 5 and 6, used a square planform for

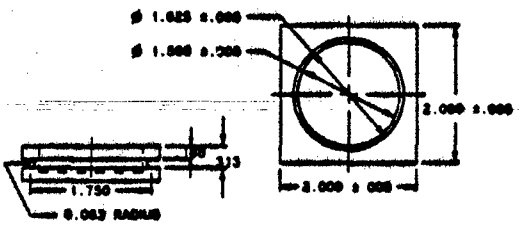


Figure 5. Polycarbonate & Acrylic Shear Sample.

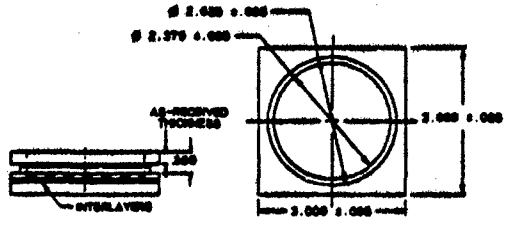


Figure 6. Interlayer Shear Sample.

insertion into the aluminum fixture blocks and a thin annular test section. The test section thickness was sized to obtain the desired high strain rates, while the diameter and width was sized to meet torque limitations of the test equipment. Data reduction for the interlayers utilized the simple torsion formula for a circular thin hollow tube while linear static finite element analysis was used to reduce the polycarbonate and acrylic results due to the more complex geometry (the radius to relieve stress concentration) and because the compliance of the grip section of the samples was not insignificant compared to that of the gage section. Due to the simplifications imposed by the linear static analysis (inertial effects are negligible, material properties are rate independent and homogeneous), the reduced data for the highest strain rates are first approximations of the true values.

Poisson ratio tests were conducted for polycarbonate and acrylic. A 2x version of the ASTM D 1822 Type L sample geometry was used (Figure 7), with a ± 90° rossette bonded to the center of the gage section to monitor strains. The samples were pulled using an MTS tension tester and shoulder-style grips.

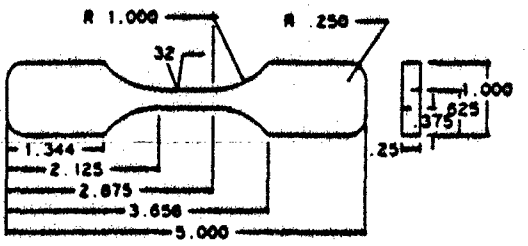


Figure 7. Poisson Ratio Sample.

Tests were conducted to determine the bulk modulus of the various canopy materials since some

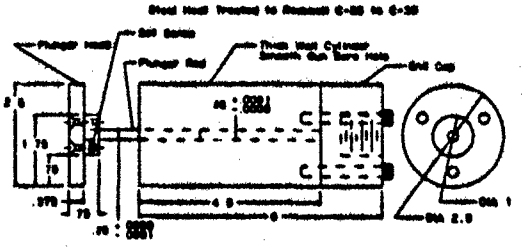


Figure 8. Bulk Modulus Fixture.

of the FEA material models require this input. The bulk modulus describes the resistance of a material to hydrostatic pressure. The lack of an appropriate bulk material model has in the past, under the action of bird impact pressures, allowed an FEA structural ply to squeeze an adjacent interlayer inside-out, permitting the structural ply to move down into the next structural ply. The test technique used was the Matsuoka-Maxwell cylinder and piston (see Figure 8).[10] The MTS machine pushes the rod into the cylindrical sample, causing it to deform against the end and side walls of the fixture. Bulk modulus is the slope of the pressure-dilatation curve. Pressure was taken to be the ratio of the MTS measured load to the cross-sectional area of the hole in the fixture. Dilatation is the change in volume divided by the original volume. Assuming the hole diameter remained constant during the test, dilatation was computed as the change in sample length (equal to the MTS measured displacement) divided by the original sample length.

Sample tensile stress-strain curves for polycarbonate at two strain rates are shown in Figure 9. As is expected for glassy polymers, the

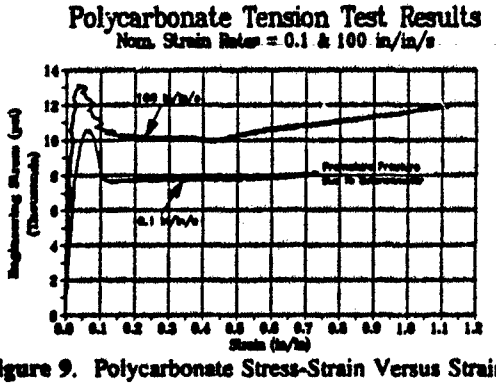


Figure 9. Polycarbonate Stress-Strain Versus Strain Rate.

polycarbonate tensile modulus is found to be mildly varying with strain rate and more sensitive to (decreasing with) strain level, while the yield and ultimate strengths are strongly sensitive to (increase with) strain rate.[11] The as-cast acrylic, in contrast to polycarbonate, showed strong dependence of modulus with strain rate as well as strain. No significant plastic deformation was observed, and the ultimate (fracture) point was essentially invariant with strain rate. Shear modulus data for polycarbonate and as-cast acrylic were found to follow trends similar to those of the tensile data. Poisson ratio was essentially constant for polycarbonate and higher than typical handbook values (0.39 versus 0.36 handbook).[12] Poisson ratio for acrylic decreased with increasing strain rate (from 0.37 static to 0.33 at 44 in/in/sec). This compares to a 0.35 handbook value.[12] The elastic modulus for the urethane interlayer materials increased with increasing strain rate. Note that interlayer materials are more accurately characterized as viscoelastic rather than elastic, so that material descriptions other than elastic modulus are more appropriate, as discussed below.

The final step was to fit appropriate FEA material models to the test data. (Note that other limited data for polycarbonate and as-cast acrylic were also used in determining the material constants for these models.) The FEA code being used (X3D - see Section V) provided two useful material models for the plate elements to be used. For acrylic and polycarbonate, the "elastic-plastic, rate-sensitive isotropic material" model was chosen. This is a simple bilinear elastic-plastic stress-strain model with strain-rate-varying yield point. For acrylic, the rate parameters were set to turn off the yield point strain rate dependency. The material models and material constants for polycarbonate and acrylic are given in Tables I and II. Urethane was treated as a "viscoelastic material with failure." This model allows specification of the bulk modulus and rate sensitive shear (deviatoric) response. The material

Table I. Polycarbonate Material Model.

Material Model: Elastic-Plastic, Rate-Sensitive Isotropic

ρ = density = 0.043 lb/in^3
E = tensile modulus = 325 ksi *
υ = Poisson ratio = 0.39 **
σ_y^0 = quasi-static yield = 7140 psi
H = hardening slope = 36.1 ksi
D = inverse of rate sensitivity scale factor = 196,500
p = inverse of rate sensitivity exponent = 12
σ_{ult} = ultimate stress = 13000 psi

where $\sigma_y = \sigma_y^0[1 + (d/D)^{(1/p)}]$

* At 1% strain; value varied considerably with strain (see text)
** At 0.1% strain

model and material constants are given in Table III.

The methods and results discussed above represent a "first-order" attempt at characterizing the strain rate-dependent behavior of aircraft transparency

Table II. As-Cast Acrylic Material Model.

Material Model: Elastic-Plastic, Rate-Sensitive Isotropic

ρ = density = 0.043 lb/in^3
E = tensile modulus = 450 ksi *
υ = Poisson ration = 0.35 **
σ_y^0 = quasi-static yield = 8920 psi
H = hardening slope = 8000 ksi
σ_{ult} = ultimate stress = 16000 psi

* At 2% strain; varies considerably with strain and strain rate (see text)
** At 0.05% strain; average of static and high strain rate values (0.37, 0.33 respectively)

Table III. Urethane Material Model*

Material Model: Viscoelastic Material with Linear Pressure-Volume Behavior

ρ = density = 0.037 lb/in^3
K = linear bulk modulus = 400 ksi
G_e = equilibrium shear modulus = 830 psi
G_i = instantaneous shear modulus = 3400 psi
β = decay constant = 10

where $G(t) = G_e + (G_i - G_e)e^{-\beta t}$, t = time,
G(t) = stress relaxation function

* Values can vary considerably with polymer formulation

materials. Other efforts are being conducted to refine the test methods, to verify and expand the material database, to develop more accurate material models, and to incorporate the effects of temperature. The results of these efforts will be documented and presented in future forums in a continuing effort to make such information available to the aircraft transparency community.

V. BIRD IMPACT ANALYSIS

Nonlinear dynamic impact FEA using UDRI's recently-developed X3D explicit nonlinear dynamic FEA program provided the final analytical evaluation of the bird impact resistance of the prototype canopies.[13] X3D employs the latest in FEA technology for nonlinear impact dynamics, including explicit time integration of the solution, large displacements, finite strains, advanced nonlinear (including viscoelastic) material models, element and ply failure, direct modeling of the bird, good integration with the PATRAN pre- and postprocessing program, and fast execution on the wide variety of UNIX-based workstation and mainframe computers.[14]

Birdstrike System Modeling

X3D's advanced features enabled engineers to model the entire birdstrike system, including the canopy, the HUD, and the bird, as shown in Figure 10. In the past, ad hoc estimations of bird loads

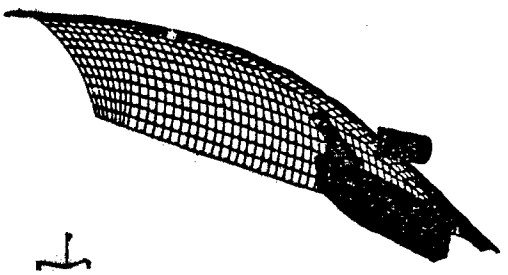


Figure 10. X3D FEA Model of Canopy, C/D HUD, and Bird.

based on bird impact tests on pressure plates were required.[15] The canopy model consisted of laminated shell elements with shear correction factors to account for nonlinear strain distribution through the canopy thickness due to the large differences in stiffness between structural and interlayer plies. The HUD model consisted of aluminum plate elements with adjustments made to match the static and free vibration bending response of the full-scale HUD (see HUD Validation below). The bird model consisted of tetrahedral elements which provided stable mesh behavior without the need to resort to an Eulerian description or free-Lagrangian model (disconnected discrete "ball" elements).[16] Symmetry and rigid wall conditions were imposed on the plane containing the canopy centerline, so that only half of the canopy, HUD, and bird were directly modeled.

To validate the system, a series of six full scale birdstrike test shots were conducted to compare with identical baseline simulations in X3D. The approach consisted of first focusing on each component model (canopy, HUD, or bird) individually and then validating the component models together as a system.

Canopy and Bird Model Validation

The objective of the first two shots in the test series was to validate the canopy and bird models. The test set up consisted of a production F-16 canopy

installed on the F-16 table top test fixture. A HUD was not included in the configuration so that canopy/bird interaction could be isolated. The impact velocity for both shots was 350 knots using a 4 pound chicken and impact locations on the canopy centerline at Fuselage Station (F.S.) 113.5 and F.S. 99 respectively. Validation was accomplished by plotting a time history of the experimental and FEA canopy deflection at selected points along the centerline. Experimental deflection at these points was captured by using two internally mounted cameras operating at 5000 frames per second and then reducing the data using a film analyzer and the triangulation technique.[17]

Deflection data was used to fine tune the canopy and bird models (primarily by adjusting the bird "failure strength," which affected momentum transfer to the canopy and therefore canopy deflection). Once confidence in the canopy and bird models was established, efforts turned to the addition of the HUD model to the system.

HUD Validation

A HUD, structurally equivalent to the production version, was used in the next two shots of the test series. Aluminum blocks were installed on the inside of the HUD chassis (avionics box) to simulate the mass, center of gravity, and moment of inertia of the production HUD. Anticipating that the combiner assembly would be damaged during the first shot, a spare combiner was fabricated using a design engineered to be structurally equivalent to the production item. The objective of these shots was to validate the HUD model and its effect on the system. Current production canopies were used in both tests with a 350 knot/4 pound bird impact at location F.S. 99 for the first shot and F.S. 113.5 for the second.

Prior to bird impact testing, the static and free vibration bending response of the full-scale HUD were measured. The static bending response of the HUD was obtained by considering the HUD as a cantilever beam, hanging a steel block of known mass at the free end, and measuring the vertical displacement at the load application point. The free vibration bending response of the HUD was then determined by manually exciting the HUD and reading the fundamental bending natural frequency from a Fast Fourier Transform analyzer recording the response of an accelerometer attached to the HUD. Similar analyses of the FEA HUD model were performed using NASTRAN, with density and modulus adjustments being made to the HUD to match the experimental response.

Further validation of the X3D bird model was also accomplished. The X3D bird material model requires input of bird strength, which was unknown. However, the bird strength affects the bird breakup pattern. Bird mass distribution data was therefore collected during the birdstrike tests by mounting a steel "bird catcher" device just aft of the pilot's head position. The 2' x 3' steel box was divided into 24 6" x 6" cells. Squares of aluminum honeycomb structure were placed in each cell to retain bird debris. The weight of each square was recorded prior to and after bird impact testing, and the net bird debris calculated. X3D analysis showed that a fairly weak bird best matched the experimental bird breakup pattern. The bird model, which was an elastic-plastic material with discontinuous pressure-volume behavior, is given in Table IV.

Table IV. Bird Material Model.

Material Model: Elastic-Plastic with Discontinuous Pressure-Volume Behavior

ρ_0 = initial density = 0.0343 lb/in^3
K_1 = linear bulk coefficient = 337 ksi
K_2 = quadratic bulk coefficient = 729 ksi
K_3 = cubic bulk coefficient = 2020 ksi
K_T = linear bulk modulus in tension = 1000 psi
G = shear modulus = 30 ksi
σ_y = yield strength = 3000 psi
H = hardening slope = 300 psi
σ_{ult} = ultimate strength = 3100 psi *

where $p = \Sigma K_i \eta^i$, i=1-3, p = pressure, $\eta = \rho/\rho_0 - 1$

* 3100 psi gave good deflection correlation; 3010 gave good bird debris distribution correlation

Deflection data was used to make final refinements in the HUD, canopy, and bird models. Figure 11 shows correlation between experimental deflection data and an identical X3D simulation.

System Model Confirmation

The final two shots of the test series served to validate the model of the entire system (canopy, bird, and HUD) while demonstrating X3D's reliability under different impact conditions, including the velocity, impact point, canopy thickness, and canopy laminate configuration. A three-ply canopy was used as the test article in the first shot and a two-ply canopy in the second.

The cockpit setup consisted of a HUD and a

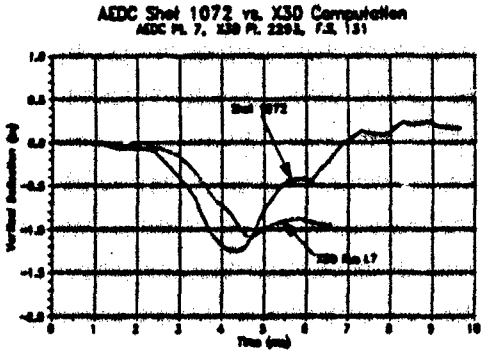


Figure 11. X3D Versus Experimental Deflection Time Histories.

"quick look" device installed in the cockpit at the pilot's head position (F.S. 140) to ensure that the requirement for no more than 2.25 inches of canopy deflection was met. The head location is critical as excessive deflection could result in an incapacitating pilot injury. Identical impact velocities (500 knots) and impact points (F.S. 113.5) were maintained for each shot.

Both canopies successfully withstood the four pound bird impact at 500 knots, and in both tests the HUD was shown to play a key role in attenuating canopy deflection.

Deflection data obtained from the tests was in good agreement with an X3D simulation using identical parameters as shown in Figure 12.

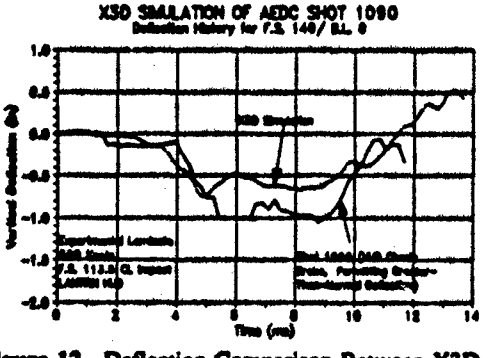


Figure 12. Deflection Comparison Between X3D F-16 System Model and Birdstrike Test Results.

This correlation confirmed the accuracy of the system model and served as the basis for proceeding with the evaluation of 540 knot prototype canopies using X3D as a design tool.

Critical HUD Determination

Before evaluating prototype designs, a determination of the critical HUD type (i.e., the HUD that allows the most deflection at the pilot's head during a birdstrike) was made using X3D. The C/D HUD is located farther away from the canopy making it less effective in attenuating canopy deflection at the pilot's head. The results of a comparative simulation between the two HUD types are shown in Figure 13. This comparison served as the rationale for using the C/D HUD in the X3D design analysis and in 540 knot birdstrike testing on prototype canopies.

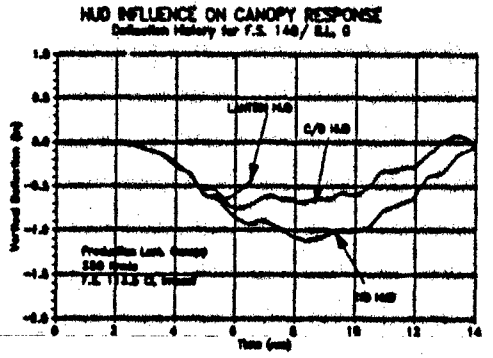


Figure 13. Influence of HUD Presence and Type on Canopy Deflection at Pilot's Head.

X3D Prototype Canopy Evaluation

X3D's relatively small memory requirements (due to explicit time integration) and laminated plate elements permitted overnight turnaround of complete analyses out to 20 msec (approximately 40,000 time steps) on UNIX workstations. Previous codes, using implicit time integration required 1-3 weeks of calendar time to perform the same task on a CRAY supercomputer. Because of the tremendous improvement in turnaround times, X3D was utilized as a design tool to evaluate the performance of several prototypes for fabrication.

The canopy design guidelines (Section II) and laminated beam test results (Section III) were used to select canopy prototypes for X3D evaluation. A total of ten simulations were conducted at an impact velocity of 540 knots. A mix of two-ply and three-ply canopies were evaluated along with the current production canopy. All three-ply designs and two of the two-ply designs passed the 540 knot birdstrike simulation. Two of the simulations showed

excessive deflection of at the pilot's head (one of those was a simulation of a 540 knot birdstrike to the current production canopy).

VI. PROTOTYPE FABRICATION AND TEST

After reviewing the results of the 540 knot/4 pound bird impact simulation, prototypes were selected for fabrication. The selection strategy balanced short and long term objectives to find a suitable mix between risks, costs, and technical requirements. To satisfy an immediate requirement for increased birdstrike protection and an extended service life, two-ply canopies with advanced environmental coatings were selected. Three-ply canopy designs were selected to meet the longer term objective of providing flexibility to incorporate combat hardening technologies as they mature (by applying advanced coatings to or substituting advanced materials for the outer ply).

To date, one set of three-ply prototypes has been received for bird impact testing. The first full-scale canopy fabricated passed the 540 knot/4 pound bird impact test as predicted by X3D.

The capabilities of X3D and the experience gained in this design process are currently being used in the development of other advanced combat aircraft transparency systems.

VII. CONCLUSIONS

The effort described herein advanced the state-of-the art in aircraft transparency design in four major areas:

1. The technical approach described in this paper increased the bird impact resistance of a fighter transparency system above 500 knots for the first time in fighter aircraft transparency design history.
2. The X3D explicit FEA code was demonstrated to be an efficient and accurate computational tool, with timely throughput using UNIX workstations, and therefore suitable for use as an integral part of the design process. This is in contrast to past experience with implicit codes which have required the use of supercomputers, and therefore were more suitable for final evaluation at the end of the design process.
3. An openly-available literature source of high strain rate mechanical and material models was initiated for increased accuracy in future FEA design work.
4. This effort highlighted the need to consider the HUD as an integral part of "birdproof" transparency systems.

In addition, the beam test method was demonstrated to be an effective means of screening potential canopy laminates for impact resistance.

VIII. RECOMMENDATIONS

The following items are recommended as a result of the work completed:

1. Full-scale prototype canopies should be fabricated and durability, birdstrike, and flight tests should be conducted (fabrication is already underway).
2. Continue development and validation of the X3D code to make it a thoroughly reliable and accurate tool for transparency design.
3. Continue to develop an openly-available literature source of high strain rate mechanical properties for transparency materials.

ACKNOWLEDGEMENTS

This work represents the combined efforts of a team that includes the USAF (Air Combat Command Fighter Requirements, Ogden Air Logistics Center, Wright Laboratory, F-16 System Program Office, and Arnold Engineering Development Center), Industry (General Dynamics, Texstar, PPG, Sierracin, and Pilkington), and Academia (University of Dayton Research Institute).

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INTEGRATED SYSTEMS APPROACH TO STRATEGIC TRANSPARENCIES

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INTEGRATED SYSTEMS APPROACH TO STRATEGIC TRANSPARENCIES

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ABSTRACT

The Strategic Transparency Technology (STT) program is the benchmark of transparency system integration. This paper presents the STT program which will integrate combat hazard protection into a durable strategic windshield. Supportability issues were identified early in the program, and treated equal to performance. The balance between performance and supportability was maintained through effective use of concurrent engineering principles. STT requirements are the most comprehensive ever defined for a transparency system. Early validation of critical technologies minimized program risk prior to technology integration. The STT program will integrate validated technologies to create transparencies that will meet the strategic needs of the Air Force into the 21st century.

INTRODUCTION

The goal of the Strategic Transparency Technologies (STT) program is to validate and integrate new technologies which will advance strategic transparencies, and strike a balance between performance and supportability. Current strategic transparencies suffer from extremely short service life and lack the combat hazard protection necessary to protect aircrews through the end of the century. Recent developments in individual transparency technologies have been made by industry, academia, and military laboratories. These developments will make it possible to provide multiple forms of combat hazard protection while making vast improvements in transparency system durability.

As with any design program, it was important to understand what similar systems already exist, and how well they perform. No matter what system an engineer is designing, there is always an existing baseline system from which the new system will evolve. This system will define the baseline for performance, reliability, maintainability, and life cycle costs, from which the new system will depart. The system will also provide important lessons learned. For the STT program, the baseline windshield was the B-1B transparency system.

All systems are designed to meet the user's requirements. If a system fails to meet the user's requirements, it ultimately fails to be good system. It is important that all requirements be defined and understood before system design begins. The best source for such requirements is the users themselves. In the Air Force, strategic transparency users are the flying units, the air logistics center, and the using command. A combination of interviews, operations and maintenance data, and requirements documents yield the majority of the system requirements. Balancing the requirements takes a team of experts more interested in the success of the system than its ability to excel in their particular areas of expertise.

With the requirements defined, it is necessary to identify the technologies which will be available to help the team achieve them. Some technologies will be ready for simple insertion into the system,

while others require further development or validation before they can be incorporated. The team must only count on technologies which are sufficiently mature. At this point, the team must make a decision about how much risk is acceptable and identify ways to reduce or control it. An important factor often overlooked on new technologies is the risk associated with integrating the new technology with other technologies.

To be useful, the technologies must be integrated into a system and transitioned to use in the field. With declining defense budgets, it is increasingly important the laboratories produce and transfer technologies directly to the user. For this reason, this program was geared directly toward proving the technologies on an aircraft which is currently flying in numbers large enough to substantiate the development investment.

HISTORY OF STRATEGIC WINDSHIELDS

Windshields for strategic aircraft are the most complex and difficult to design. Aircraft such as the F-111 and B-1 were designed to penetrate at very high speeds and low altitudes. In order to maintain a sleek aerodynamic shape, the windshields were sloped back and contoured to the airframe, causing a series of trades between birdstrike resistance, service life, optics, and aerodynamic performance. Initially, aerodynamics and optics won out on the F-111, until a few were lost in Vietnam due to birdstrikes. The B-1 learned from this mistake and placed heavy emphasis on birdstrike resistance at the expense of optics and service life. The low technology of past combat hazards allowed the windshields to meet the threats, but as high technology weapons find their way into more and more national inventories, we are more likely to encounter them.

When the B-1 windshield was designed, in the early 1970's, it was the most sophisticated windshield ever designed. Originally designed to meet the requirements of the B-1A, which was intended to fly supersonic, emphasis was placed on performance. The installation angle made it necessary to make the windshield extremely large to provide visibility. It can withstand a four-pound birdstrike at 560 knots, and has an electric heater built-in to prevent icing.

To meet the requirements, a polycarbonate structural ply was used with a glass face ply, polycarbonate spall ply, silicon interlayers, and metallic coatings which provide anti-static and heater capabilities. Numerous tests were conducted to ensure that the windshield would meet the operational requirements. The windshield passed the birdstrike tests, although in a birdstrike near the top of the center post may result in failure of the airframe in the area. In general, the performance requirements for the windshield exceed the performance requirements for the rest of the airframe.

From the very beginning of the B-1 program, there was much concern about whether to use a glass or plastic face ply. While the first B-1A rolled off the assembly line with acrylic face ply windshields, glass faced windshields had already been ordered to replace them. There were three primary factors in the decision to switch to glass: abrasion resistance, metallic coatings, and the heater. For resistance to abrasives, such as nuclear dust, glass was superior to the acrylic and liners of the time. Metallic coatings required for the anti-static coating and heater were difficult to apply to plastic and exhibited very poor durability. The most critical factor was that glass allowed the heater to operate at much lower temperatures than a plastic face ply, due to their respective thermal resistivities. The all plastic windshield suffered from localized hot spots making the windshield susceptible to failure.

Since the beginning of the B-1B program, the windshield has been a poor performer and adversely affected the operational availability of the aircraft. A few symptoms of the problem are delamination, cracking and melting from heater use, multiple imaging, poor post-birdstrike visibility, excessive change-out times, and lack of combat hazard protection necessary to meet the 1995-2000 threat. The windshields had a service life of less than one year, and required four days to change out. This cost the Air Force $10M per year for spares alone. With two forward windshield panels per aircraft, each of the just under 100 aircraft was MICAP twice a year for four days to change-out a windshield. Approximately 72 hours of the down time was waiting for sealants to cure. In terms of operational availability, this was disastrous.

In 1985 a Tiger Team was assembled to address the windshield issue. As a result, a team with the laboratories, B-1 SPO, OC-ALC, industry, and academia was established. A two-step get well plan was formulated. First, apply existing technologies to increase service life and reduce change-out time, to reduce the logistics burden. Through quick curing sealants, changing to chemically tempered glass, and sealing the edges, the service life has been doubled and change-out times significantly reduced. Next, develop and validate technologies which would further increase service life and provide combat hazard protection to meet the 1995-2000 mission threats. Combat hazard technologies have matured to the point that they are reliable and durable enough to be considered for incorporation. The most recent efforts have focused on switching to a plastic face ply and incorporating combat hazard protection directly into a windshield which has greater than a four year service life.

Many of the problems encountered in the B-1B have not shown up in the B-2. It has a similar cross section with the exclusion the heater and incorporation of an EMP screen. There is also a polyurethane liner in the exterior surface of the glass. An effort is under way to improve the liner technology, and the B-2 would gain greatly from the application of combat hazard technology validated in the STT program. Incorporating the nuclear thermal flash protection into the windshield and replacing the glass face ply with plastic could reduce transparency system weight by as much as 150 pounds.

DEFINING MEANINGFUL REQUIREMENTS

Strategic aircraft operate in an environment where it is no longer acceptable for windshields to "just keep the birds out". The transparencies must minimize radar cross section, protect the aircrew against modern combat hazards, provide good optics, be compatible with systems like night vision goggles, and place the minimum burden possible on the logistics system. Before it is possible to determine how the transparency will do this, it is necessary to define precisely what it is to do, and to what level it will perform. To design a well balanced and affordable transparency, the requirements must define not only performance requirements, but supportability requirements as well. Once the requirements are established to effectively address the operational and support needs, technologies which satisfy the requirements were identified and validated for incorporation into the design.

To ensure comprehensive coverage of strategic transparency requirements across the board, a review of all requirements documentation for Air Force strategic transparencies was performed. Baseline requirements were established through evaluation of F-111, B-1B, B-2, and Mission Integrated Transparency System (MITS) requirements. These requirements were divided into five categories:

- Natural Hazards
- Crew Machine Interface
- Combat Hazards
- Supportability
- Fuselage Integration

Through a mixture of "Bottom-Up" and "Top-Down" analysis, requirements were evaluated and the most stringent requirements selected. The requirements definition methodology developed by Northrop generated requirements which met or exceeded baseline aircraft specifications. The requirements were driven by strategic aircraft composite mission profiles and updated to reflect the latest threat analysis and Statement of Need for strategic aircraft. By involving experts in the definition of requirements, lessons learned from previous strategic transparency efforts and operational use were fed into the requirements process.

The process was initiated with a "Bottom-Up" analysis. The basic framework was established from the categories and parameters from the MITS program. Baseline specifications were loaded into the matrix. Specifications for the F-111, B-1B, and B-2 were reviewed and new parameters added to the matrix as they were identified. The most stringent requirements were chosen from those available.

Northrop's "Top-Down" analysis developed new requirements from existing baseline aircraft requirements which were modified based on updated threat information, lessons learned, and composite mission profiles. Composite conventional and nuclear mission profiles were developed to define the overall strategic mission aircraft performance requirements. The results of the "Bottom-Up" and "Top-Down" analysis were simultaneously evaluated and the most stringent requirements were chosen from those available. Acceptance criteria were identified for each requirement.

INVESTIGATING TECHNOLOGIES

As the system requirements were reviewed, technologies were identified that could be utilized to meet the requirements. The technology areas identified as critical to the success of the program were supportability and combat hazard technologies. The primary driver for supportability among the strategic community is currently service life. Most notably, the tendency for delamination between the glass face ply and silicon interlayer. The other technology area requiring investigation was emerging combat hazard protection to meet the 1995-2000 threat.

Metallic coating and liner technologies for plastic face plies have matured over the last five years to the point that they are nearly ready for incorporation into strategic transparencies. Bonding capabilities between interlayers and plastic plies, and similar coefficients of thermal expansion and strain rates between plastic plies will significantly improve supportability through reduced incidents of face ply delamination. This approach has a significant degree of risk because it requires durable liners and metallic coatings for application on plastic substrates. Until recently, this has been a major obstacle for plastic transparencies.

Recent advancements in photochromics and metallic coating technologies make it possible to incorporate combat hazard protection directly into the windshield. These technologies must work together to protect against millimeter/microwave radiation, lasers, nuclear thermal flash, and electro-magnetic pulse (EMP), while minimizing radar cross section. The photochromic dyes, which are imbedded in interlayer material and placed between the face, structural, and spall plies, activates rapidly enough to protect the aircrew from the high energy thermal pulse. However, PLZT goggles are required because photochromics can not react fast enough to protect the eyes. If properly designed, one metallic coating can provide protection from more than one type of combat hazard.

Any research program involves a certain degree of risk. It is the responsibility of the development team to manage the risks, minimizing or eliminating them to the greatest extent possible. As a central part of the STT program, a technology risk management process was implemented. [illegible] the program

technology goals clearly defined, risks were identified, assessed, and controlled. The top five risks were listed and specific plans for their reduction were defined and agreed to by the development team.

One of the keys to minimizing risks was to perform coupon level evaluations of the technologies' maturity and limitations. As an example, the ability of metallic coatings and liners on a plastic substrate to perform after exposure to harsh environments was identified as a critical risk. Tests were designed to evaluate candidate combinations as they were exposed to combat hazards, extreme weather conditions, and cyclic loading. These tests identified three combinations from two vendors that warrant further study. The other competitors may be able to meet the requirements for a specific aircraft, but would be unacceptably risky for incorporation into other transparencies. A goal of the STT program is to validate technologies which can be applied to the majority strategic transparencies.

The test series was designed to validate the candidate materials ability to meet the strictest requirements in a minimum number of tests. Coupons were fabricated with a metallic coating protected by a liner. The coupons were a mix of substrates with gold, silver, or ITO type coatings, protected with various liners. The coupons were subjected to optics, bending/thermal, bending/cyclic, thermal, QUV, humidity, liner adhesion, and thermal flash tests. The durability and performance of the coatings and liners were measured. For the thermal flash tests, coupons were tested with and without liners to identify their effects.

The other key technology risk was the durability of the photochromics. Under an IR&D program, Northrop has worked with the various transparency vendors to validate photochromic dyes which are not sensitive to oxygen, and can be incorporated into an all plastic windshield.

TRANSITIONING A USEFUL PRODUCT

Technologies validated in the laboratories are transitioned directly to the field where they can be used. The B-1B was chosen as the transition vehicle for the STT program products. The B-1B is in the greatest need for the supportability and performance improvements. Also, because there are almost 100 B-1Bs and only 20 B-2s, it offers the best economic investment cost/benefit ratio.

Because the B-1 requirements are not as stringent as the STT requirements in each of the categories, a trade study will be performed to identify the specific technology mix which will best serve the B-1B. The technologies will be integrated into a single cross section to validate manufacturability and combined durability. Tests similar to those performed on the metallic coating coupons will validate that the integrated transparency cross section meets the needs of the B-1B and is ready to be scaled up. Full scale test articles will then be manufactured and tested. While undergoing qualification tests, the full scale articles will be flight and durability tested.

Once proven on the B-1B, the technologies validated under the STT program will be made available for use in the B-2 and other aircraft with similar needs for combat hazard protection, through the Joint Aircraft Transparency Technology Insertion Center (JATTIC).

CONCLUSIONS & SUMMARY

The Strategic Transparency Technology program is the benchmark for transparency system integration. The program has taken a systems approach to providing technologies and balancing supportability and performance requirements. This will ensure strategic aircraft have affordable

windshields which meet their mission needs for 1995-2000.

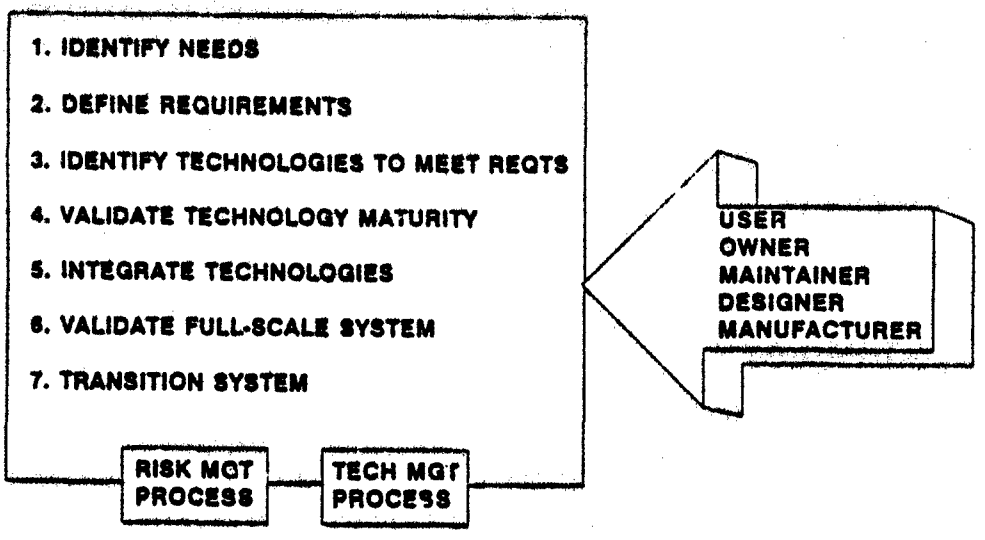
The history of strategic transparencies was reviewed and lessons learned were applied to define a program which will not repeat the mistakes of the past.

The most comprehensive and stringent set of requirements ever developed for a transparency were defined and levels of acceptance quantified. Supportability was treated as an equal with performance requirements to provide balance and affordability. A combined "Bottom-Up" and "Top-Down" approach was used to define the requirements, using composite mission profiles and strategic aircraft requirements.

Technologies have matured sufficiently to provide great improvements in supportability while incorporating combat hazard protection. Photochromics and metallic coatings on plastic have been validated. These technologies will be incorporated into an durable all plastic windshield. The ability to integrate the technologies will be demonstrated prior to production of full scale prototypes.

Current plans call for proving the technology on the B-1B first, followed making the technology available for application to the B-2 and other aircraft with the assistance of the Joint Aircraft Transparency Technology Insertion Center (JATTIC).

INTEGRATED SYSTEMS APPROACH
to Strategic Transparencies

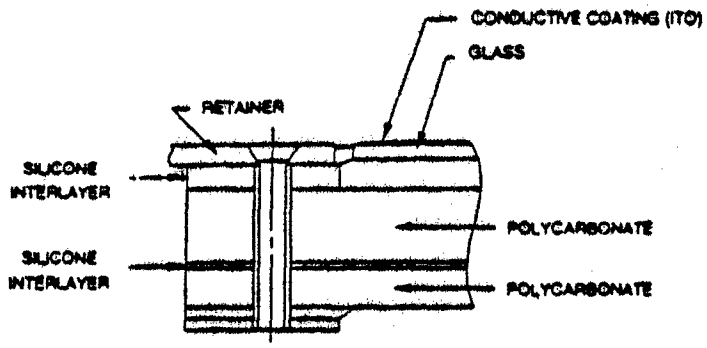


IDENTIFYING NEEDS

- MISSION NEEDS STATEMENTS
- AIRCRAFT MISSIONS
- OWNER
- USER
- MAINTAINER

NEED - A DURABLE STRATEGIC TRANSPARENCY SYSTEM WHICH MEETS AIRCRAFT MISSION NEEDS FOR 1995-2000 AND IS SUPPORTABLE.

B-1B WINDSHIELD CROSS SECTION



DEFINING MEANINGFUL REQUIREMENTS

- USED "BOTTOM-UP" AND "TOP-DOWN" METHOD.
- BASED ON F-111, B-1, B-2, AND MITS REQUIREMENTS
- DEFINED COMPOSITE MISSION PROFILES AND THREAT ANALYSIS
- BALANCED PERFORMANCE AND SUPPORTABILITY
- DEFINED QUANTIFIABLE REQUIREMENTS
- DEFINED ACCEPTANCE CRITERIA FOR EACH REQUIREMENT
- CHOSE MOST STRICT REQUIREMENTS

MOST COMPREHENSIVE, QUANTITATIVE, TESTABLE SET OF REQUIREMENTS FOR ANY AIRCRAFT TRANSPARENCY EVER DEFINED!

REQUIREMENTS CATEGORIES
for Strategic Transparencies

- NATURAL HAZARDS
- CREW-MACHINE INTERFACE
- COMBAT HAZARDS
- SUPPORTABILITY
- FUSELAGE INTEGRATION

"BOTTOM-UP" REQUIREMENTS MATRIX

CATEGORY - NATURAL HAZARDS - 3.1
PARAMETER -BIRDSTRIKE - 3.1.3

Description	VALUE or LEVEL	ACCEPTANCE CRITERIA
MFD		
4lb		
1		
B-2		
B-1B		
F-111		
SELECTED VALUE		
COMMENTS		

"TOP-DOWN" REQUIREMENTS MATRIX

CATEGORY - NATURAL HAZARDS - 3.1
PARAMETER - BIRDSTRIKE - 3.1.3

Description	VALUE or LEVEL	ACCEPTANCE CRITERIA
SELECTED VALUE	[illegible]	[illegible]

SAMPLE DESIGN & PERFORMANCE REQUIREMENTS

3.0 CATEGORY: NATURAL HAZARDS

CATEGORY	VALUE or LEVEL	ACCEPTANCE CRITERIA
3.1.3 BIRDSTRIKE	[illegible]	[illegible]

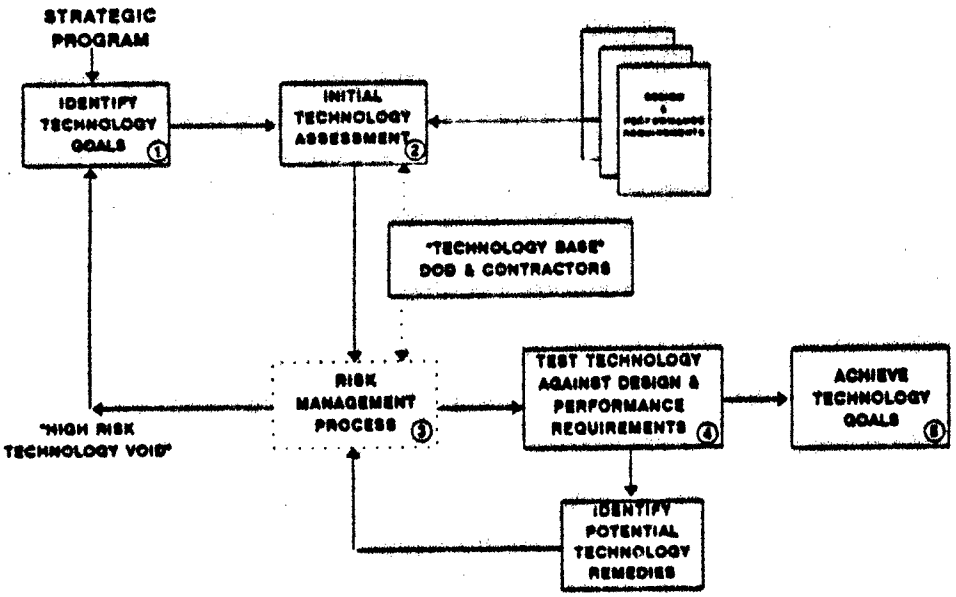
BASIC TECHNOLOGIES AVAILABLE
to Meet Requirements

- DURABLE METALLIC COATINGS ON PLASTIC
- PHOTOETCHED SCREENS
- OXYGEN INSENSITIVE PHOTCHROMICS
- DURABLE TRANSPARENCY LINERS

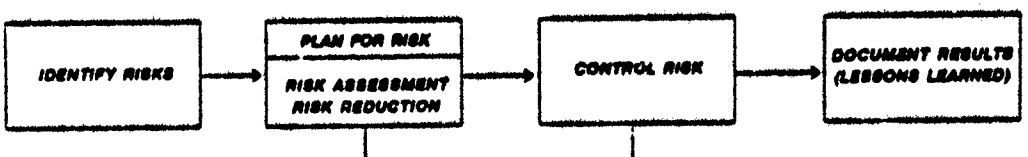
COMBAT HAZARD TECHNOLOGY AREAS
Under Validation to Meet Requirements for 1995-2000

- Radar Cross Section Control
- Laser Protection
- Photochromic Nuclear Thermal Protection
- EMP Protection
- Millimeter/Microwave Shielding

TECHNOLOGY MANAGEMENT PROCESS



RISK MANAGEMENT PROCESS



RISK ASSESSMENT

IDENTIFY RISK SOURCES
- TECHNICAL RISKS
- SUPPORTABILITY RISKS
- SCHEDULE RISKS
- COST RISKS

IDENTIFY RISK CONSEQUENCES
- MODELS
- ESTIMATES
  - PROBABILITIES
  - MAGNITUDE

PRIORITIZE RISKS

RISK REDUCTION

AVOID RISK
- REDUCE SCOPE
- PHASED DEVELOPMENT

IMPROVED SKILLS

REDUCE UNCERTANTIES
- PLAN ACTIVITIES
- VERIFY RESULTS
  - PROTOTYPES
  - DEMONSTRATIONS
  - TESTS
  - MODELS
  - ANALYSES
- USE PROVE TECHNOLOGY
- USE STANDARDIZED TECHNIQUES
- CHECK REFERENCES

REDUCE CONSEQUENCES
- PARLLEL ALTERNATIVE PATHS
- ALLOW FOR CONTINGENCIES

MONITOR RISK ITEMS

REASSESS RISK SOURCE AN CONSEQUENCES

VALIDATION OF TECHNOLOGY MATURITY

RECENT TESTS VALIDATED METALLIC COATED PLASTIC COUPONS FOR:

- OPTICS
- BENDING/THERMAL
- BENDING/CYCLIC
- THERMAL SHOCK
- QUV EXPOSURE
- HUMIDITY
- LINER ADHESION
- THERMAL FLASH

SUBSCALE TECHNOLOGY INTEGRATION

WILL VALIDATE INTEGRATED TECHNOLOGY:

- MANUFACTURABILITY
- MISSION PERFORMANCE
- DURABILITY

FULL-SCALE SYSTEM VALIDATION

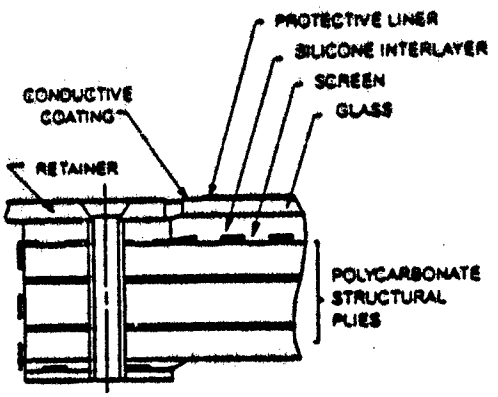
WILL VALIDATE ABILITY TO PRODUCE FULL-SCALE ARTICLES WHICH MEET REQUIREMENTS FOR:

- **MANUFACTURABILITY**
- **MISSION PERFORMANCE**
- **DURABILITY**

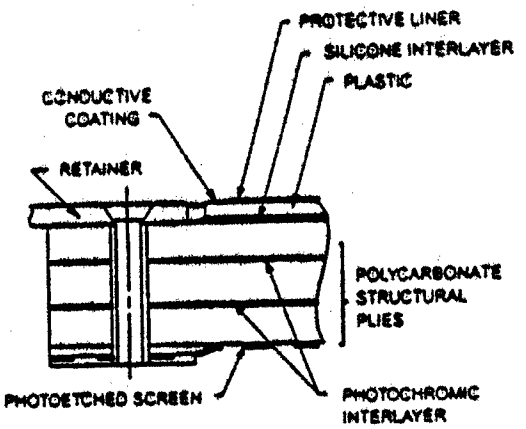
TRANSITIONING A PRODUCT

- **PREFERRED SPARE FOR B-1B**
- **WILL IMPROVE B-1B COMBAT HAZARD PROTECTION**
- **WILL IMPROVE B-1B OPTICS**
- **WILL IMPROVE SUPPORTABILITY**
- **TECH TRANSITION TO B-2 AND OTHER AIRCRAFT THROUGH JATTIC**

TECHNOLOGY APPLICATION ON B-2



COATED GLASS DESIGN



COATED PLASTIC DESIGN

SUMMARY

- INTEGRATED SYSTEMS APPROACH WAS USED ON STRATEGIC TRANSPARENCY TECHNOLOGY PROGRAM
- MOST COMPREHENSIVE SET OF TRANSPARENCY REQUIREMENTS EVER
- SUPPORTABILITY TREATED EQUAL TO PERFORMANCE
- DESIGN OPENED TO NEW TECHNOLOGIES TO MEET NEEDS AND REQUIREMENTS
- RISK MANAGEMENT INTEGRAL PART OF ENTIRE PROGRAM
- TECHNOLOGIES ARE READY FOR INTEGRATION AND SCALE-UP
- TECHNOLOGIES WILL TRANSFER TO OTHER AIRCRAFT

RAIN EROSION TESTING REQUIREMENTS - THE F-16 TRANSPARENCY SPECIFICATION REVISITED

Clifton A. Webster
Lockheed Fort Worth Company

Rain Erosion Testing Requirements
The F-16 Transparency Specification Revisited

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Fort Worth, Texas 76101, U.S.A.

Abstract

A review of the F-16 transparency specification was undertaken in response to a changed mission profile and the availability of new high performance materials and coatings. The F-16 transparency specification (revision C) requires that the outside surface withstand rain impingement of one inch per hour at an airspeed of 500 knots for a period of 5 minutes. This requirement was introduced into the Specification in 1981. Discussions were held with the author of the original Specification and the F-16 transparency manufacturers. No published analysis was forthcoming which would describe how or why the 500 knot velocity, 5 minute duration, and 1 inch per hour rainfall rate combination was selected.

This paper attempts to establish a basis for selection of levels of rainfall rate, velocity, exposure duration, and droplet size.

A literature search yielded 1.4 inch/hour rainfall rate as an operational criterion. However, the only available rain erosion test facility, the University of Dayton Research Institute (UDRI), has a fixed rate of 1 inch/hour.

Current mission profiles were used in combination with extreme rainfall rate and duration data from Florida to derive expected exposure durations appropriate for the F-16. The expected exposure duration for the 1 inch/hour rate correlated well with the Specification (4.18 minutes vs. 5 minutes respectively). The current test velocity also correlates well with a weighted average Air-to-Ground mission velocity (500 knots vs. 523 knots). The average mission velocity was taken only at altitudes at which rainfall may be encountered.

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Introduction

Air Force Specification 16ZK002C [1], defines the current requirements of F-16 transparencies for resisting wear and surface erosion when exposed to the impingement of rain. This specification requires that the outside surface of the F-16 transparency withstand rain impingement of one inch per hour at an airspeed of 500 knots for a period of 5 minutes, with visible surface damage limited to 10 percent or less of the exposed surface area. This requirement applies to both coated and uncoated exterior surfaces.

Rain erosion damage requirements in the current Air Force specification were introduced when the F-16 transparencies became Government Furnished Aeronautical Equipment (GFAE) in October of 1981. Discussions were held with the author of the original Specification and the F-16 transparency manufacturers concerning the basis for the rain erosion test parameters.

No published analysis was forthcoming which would describe how or why a rainfall impingement rate of one inch per hour at 500 knots for 5 minutes was developed. The following excerpts are typical of the literature search findings:

> "Rain erosion tests are realistic in evaluating how well a coating adheres to a plastic surface [2]."
>
> "There has been no correlation between results of this test and in-service data [2]."

F-16 forward transparencies are currently constructed with a top or outside layer of uncoated cast acrylic. This MIL-P-5425 acrylic material has been successfully tested for rain erosion resistance in accordance with Specification 16ZK002 requirements. Coupons are tested both unexposed and following exposure to the artificial equivalent of 3 years natural sunlight. The test specimens are mounted at 30° inclination to the direction of air flow.

Motivation

Why question the requirements of a specification for a transparency design that has been in service for over 10 years?

1) Change in Mission profile. In the late 1970's the expected mission mix for the F-16 was 50% air-to-air and 50% air-to-ground, with air-to-ground sorties being flown at 450 knots during ordinance delivery. The current mission mix is approximately 30% air-to-air and 70% air-to-ground. For the air-ground mission the average approach velocity is over 500 knots with a typical altitude in the 300-500 ft. range. Thus, there is both an increased chance of encountering rainfall at the lower altitudes of the air-to-ground mission (compared to the air-to-air) mission and an increased chance of damage at the higher velocities.

2) New coatings to evaluate. A new generation of environmental coatings is under evaluation. Qualification testing should be done using appropriate requirements.

3) New materials to evaluate. Improved materials that could take the place of polycarbonate and acrylic are being developed and should be tested using appropriate requirements.

Results of this study were used to make recommendations for changing the existing Air Force Specification, 16ZK002.

Discussion

During the review of available military standards, specifications, Air Force Wright Laboratory publications and other documents used to compile this report, it became readily apparent that it would be very difficult to quantify what could be considered "normal rainfall encounters."

Four variables are required to fully define the rain erosion requirement: rainfall rate, aircraft velocity, exposure duration, and droplet size.

Test Rainfall Rate

This study started with a search of F-16 literature. The Prime Item Development Specification for F-16 Aircraft, 16PS005 [3], directs the investigation to Specification 16PS011 [4]. Specification 16PS011, states that F-16 aircraft shall be operational under in-flight rainfall rates of 0.59 inches per hour as defined in MIL-E-38453 [5]. However, MIL-E-38453 was canceled on 7 April 1986. MIL-E-38453 has been superseded by MIL-E-87145 [6]. MIL-E-87145 lists MIL-STD-210 [7] as the document which defines flight operational rainfall rates at a 0.5 percent extreme.

MIL-STD-210C, gives a rainfall rate which would occur 0.5 percent of the time as 0.6 mm/min.

$$\frac{0.6\ \text{mm}}{\text{minute}} * \frac{1\ \text{inch}}{25.4\ \text{mm}} * 60\ \frac{\text{minutes}}{\text{hour}} = 1.4\ \frac{\text{inches}}{\text{hour}}$$

All rain erosion testing for U.S. Air Force equipment is performed at the Wright Laboratory facility by the University of Dayton Research Institute (UDRI). The Wright Laboratory apparatus has a calibrated rate of 1.0 inches per hour in contrast to the required rate of 1.4 inches per hour [8].

Test Velocity

Up-to-date mission profiles for both Air-to-Air and Air-to-Ground missions were obtained from the Lockheed Fort Worth Company F-16 Aero Performance group. Tables 1 and 2 summarize these missions. These mission profiles were used to calculate a weighted average airspeed at which extreme rainfall might be encountered and the rain impact would potentially be damaging. Calculations are presented in Table 3.

It was assumed that rain above 15,000 feet would be avoided due to the possibility of encountering ice and extreme air turbulence. It was further assumed that the rainfall rate and probability of occurrence would be uniform from ground level to 15,000 feet. Studies of vertical radar reflectivity profiles indicate that when high precipitation rates reach the surface, the rates and hence precipitation water content, stay fairly constant with altitude up to about 6 kilometers then decrease above [9].

A velocity of 500 knots was selected as that which would potentially cause the rain impact to be damaging.

Table 1: MISSION SUMMARY AIR-TO-GROUND MISSION F100-PW-229 ENGINE			
Mission Phase	**Time Interval (Min)**	**Airspeed (KTAS)**	**Altitude (Feet)**
T.O. & Accel (Max Power)	0.5	340	0
Climb	7.9	479	32,080
Cruise	23.3	477	33,010
Descend	10.3	542	200
Dash (500 KTAS @ 200 ft)	8.4	500	200
Turn (Max Power)	0.4	500	200
ESgain (1-2K ft)	0.1	500	200
Turn (NIL power)	0.8	500	200
ESgain (1-2K ft)	0.2	500	200
Turn (Max power)	0.4	500	200
ESgain (1-2K ft)	0.1	500	200
Turn (NIL power)	0.7	500	200
ESgain (1-2K ft)	0.2	500	200
Drop	0.0	500	200
Accelerate	0.2	550	200
Dash (550 KTAS @ 200 ft)	7.5	550	200
Climb	6.5	465	39,465
Cruise	24.9	463	40,399
Descend	18.5		
Total Mission Time (Min)	110.90		

Table 2: MISSION SUMMARY AIR-TO-AIR MISSION F110-GE-129 ENGINE			
Mission Phase	**Time Interval (Min)**	**Airspeed (KTAS)**	**Altitude (Feet)**
T.O. & Accel (Max Power)	0.5	340	0
Climb	8.7	487	35,770
Cruise	37.5	484	37,160
Descend	6.0	580	15,000
Accel (M 0.9-1.1)	0.5	708	15,000
Turn (1 at M1.1)	1.3	708	15,000
Drop Tank	0.0	708	15,000
Decel (M 1.1-0.9)	0.2	580	15,000
Turn (3 at M 0.9)	2.9	580	15,000
Dwell (M 0.9)	7.0	580	15,000
Accel (M 0.9 - 1.1)	0.3	708	15,000
Turn (1 at M 1.1)	0.8	708	15,000
Decel (M 1.1 -0.9)	0.2	580	15,000
Turn (3 at M 0.9)	2.6	580	15,000
Climb	6.0	507	44,468
Cruise	30.7	505	45,525
Decend	21.7		
Total Mission Time	126.90		

For mission segments from Table 1 in which the velocity starts or ends under 500 knots, the average velocity and half the time interval for the segment was used in Table 3. For example, velocity = (542 + 477) knots/2 = 510 knots and interval = 10.3 minutes/2 = 5.15 minutes.

Table 3: AIR-TO-GROUND MISSION BELOW 15K FEET

Airspeed (KTAS)	Flight Time (Min)	Fraction of Flight Time Under 15K Feet	Weighted Airspeed (KTAS)
510	5.2	0.1891	96.44
521	8.4	0.3055	159.17
500	.4	0.0145	7.25
500	.1	0.0036	1.80
500	.8	0.0291	14.55
500	.2	0.0073	3.65
500	.4	0.0145	7.25
500	.1	0.0036	1.80
500	.7	0.0255	12.75
500	.2	0.0073	3.65
525	.2	0.0073	3.83
550	7.5	0.2727	149.99
508	3.3	0.1200	60.96
	Total 27.50	1.0000	Total 523.09

Approximately 89% of the typical Air-to-Ground mission airspeeds, for altitudes under 15,000 feet, were in excess of 500 knots. A majority of the Air-to-Air mission is also flown at airspeeds greater than 500 knots. However, the altitude during these mission segments is typically 15,000 ft or greater. Therefore, the Air-to-Ground mission would be the worst-case environment. From Table 3, the weighted average mission airspeed at altitudes where rainfall encounters are expected was 523 knots for the Air-to-Ground Mission. A test velocity of 523 knots is within capabilities of the Wright Laboratory/UDRI facility where impact velocities to a maximum of 650 miles/hour or 564 knots can be attained [8].

Expected Exposure Duration

Expected exposure duration is dependent upon three variables:

1. The fraction of the total lifetime (flight and ground) of the transparency that is flight time. For this study it is assumed that a typical F-16 will fly 22 hours each month and that forward transparencies have an expected life of 4 years.

2. The fraction of the flight time that is both below 15,000 feet and 500 knots or faster. From Tables 1 and 3, the Air-to-Ground Mission fraction is 27.5 minutes out of a 110.9 minute total mission time. The Air-to-Air Mission fraction is 10.9 minutes out of a 126.9 minute total mission time.

3. Typical durations of extreme rainfall rates. MIL-STD-210C, gives a rainfall rate which would occur 0.5 percent of the time as 0.6 mm/minute (1.4 in/hour).

Air-to-Ground Mission:

$$\frac{27.5 \text{ minutes}}{110.9 \text{ minutes}} * 22 \frac{\text{hours}}{\text{month}} * 12 \frac{\text{months}}{\text{year}} * 4 \text{ years} * .005 * 60 \frac{\text{minutes}}{\text{hour}} = 78.6 \text{ minutes}$$

Air-to-Air Mission:

$$\frac{10.9 \text{ minutes}}{126.9 \text{ minutes}} * 22 \frac{\text{hours}}{\text{month}} * 12 \frac{\text{months}}{\text{year}} * 4 \text{ years} * .005 * 60 \frac{\text{minutes}}{\text{hour}} = 27.2 \text{ minutes}$$

These exposure times, being considerably higher than the 5 minute baseline of the Specification, were suspect. Another reason to be suspicious of this approach is the statistical modal for calculating the 0.5% extreme of MIL-STD-210 utilized 1-minute percipitation intensities only. It seems logical that testing with a 0.5% extreme rainfall rate should be done for a 1-minute duration only. Therefore another approach to calculating exposure duration was investigated.

Rainfall Intensity-Duration-Frequency Curves were obtained from the National Weather Service, U.S. Department of Commerce, Fort Worth, Texas [10]. These curves were compiled using accumulated rainfall data and a frequency analysis method of extreme values developed by Gumbel. A detailed discussion of this method of extreme values can be found in report AD766210 [11].

Extreme rainfall rates are available for such areas as India and Northern Brazil. However, this author chose to use data available for rainfall rates in Florida. Florida was chosen as typical of an area where extreme rainfall rates might be encountered during peacetime operations.

Duration for two rainfall rates, 1 inch per hour and 1.4 inches per hour, were taken from the above referenced charts for Jacksonville, Key West, Miami, Pensacola, and Tampa. These rates are tabulated in Table 4 for a return period of forty years. The forty year

return period was derived on the basis of "withstanding" an extreme rainfall occurrence. Report AD766210 defines "withstanding" as a calculated risk of failure of 10 percent. The four year expected duration of exposure is one-tenth or 10 percent of forty years.

A return period can be explained as the span of time during which at least one event of specific extremes is expected to occur. For example, in Tampa Florida, rainfall of one inch per hour or greater that lasts for at least 7.5 hours is expected at least one time during forty years. Thus, this rainfall extreme for Tampa has a forty year return period.

Table 4: FLORIDA RAINFALL DURATIONS

LOCATION	DURATION (HRS) 1.0 inches/hour	DURATION (HRS) 1.4 inches/hour
Jacksonville	8.2	4.0
Key West	9.0	4.7
Miami	11.6	6.3
Pensacola	10.6	6.4
Tampa	7.5	4.4
AVERAGE	9.2	5.2

For the Air-to-Ground Mission the encounter durations for the two rain fall rates selected are as follows:

For a 1 inch per hour rainfall rate the duration during the expected 4 year service life is:

$$9.2 \text{ hours} * 22 \frac{\text{flight hours}}{\text{month}} * \frac{27.5 \text{ min. under 15K ft.}}{110.9 \text{ min. total mission}} * \frac{1 \text{ month}}{30 \text{ days}} * \frac{1 \text{ day}}{24 \text{ hours}} * \frac{60 \text{ Min.}}{1 \text{ hour}} = 4.18 \text{ Minutes}$$

For a 1.4 inch per hour rainfall rate the duration during the expected 4 year service life is:

$$5.2 \text{ hours} * 22 \frac{\text{flight hours}}{\text{month}} * \frac{27.5 \text{ min. under 15K ft.}}{110.9 \text{ min. total mission}} * \frac{1 \text{ month}}{30 \text{ days}} * \frac{1 \text{ day}}{24 \text{ hours}} * \frac{60 \text{ Min.}}{1 \text{ hour}} = 2.4 \text{ Minutes}$$

Droplet Size

The droplet size distribution for a 0.6 mm/min (1.4 inches/hour) rainfall rate is found in Table 5. The distribution was estimated from a gamma-function fit to drop-size distributions observed during heavy rain in tropical cyclones [7]. The liquid water content is 1.6 g/m^3.

At the WL/UDRI rain erosion test facility, drop diameter ranges

from 1.7 to 2.3 mm with an average size of 2.0 mm at the capillary orifice. This drop size distribution corresponds with the standard 1 inch per hour rainfall rate [12]. Drop size and drop rate are controlled by the water temperature, capillary orifice diameter and head pressure of the water storage tank [8]. It should be noted the drop size that actually impinges on the test coupon is unknown due to the turbulence inside the test chamber. Adler [13] suggests that the drop size distribution at the WL/UDRI erosion facility is bimodal with a peak at 1.0 mm and the other centered around 4.5 mm.

Test droplet size for a 1 inch per hour rate is somewhat larger than that expected for a 1.4 inch per hour natural rainfall event. Considering the equation,

$$\text{Kinetic Energy} = .5 * \text{Mass} * \text{Velocity}^2,$$

test conditions should present a more extreme condition due to the larger test droplet size and therefore mass.

For the purpose of this study, droplet size as a test variable was ignored due to:

The difficulty in controlling droplet size as a test parameter.

Test conditions that should be more extreme than environmental conditions.

The F-16 Specification does not mention droplet size as a test condition.

Table 5: DROPLET SIZE DISTRIBUTION	
DIAMETER SIZE INTERVAL (mm)	DROPLETS PER CUBIC METER
0.5-1.4	1154
1.5-2.4	260
2.5-3.4	26
3.5-4.4	2
4.5-5.4	<1
5.5-6.4	<1

Recommendations

As a result of this study, the following recommendations were made to Ogden Air Logistics Center (OO-ALC):

1. Rainfall rate to remain at 1 inch per hour.

2. Duration to remain at five minutes.

3. Impact velocity to increase from 500 to 523 knots.

4. Coated transparency surfaces be tested to the same specifications as are the uncoated surfaces.

Any new specification should also include QUV exposure to coupons prior to testing. QUV machines expose coupons to a combination of UV light, humidity, and temperature. Previous testing [14] [15] indicates a high correlation between simulated environmental exposure and premature coatings failure during rain erosion testing.

Areas for Further Investigation

The discrepancy between the 1.4 inch/hour rainfall rate required by MIL-STD-210 and the 1.0 inch/hour rainfall rate capability of the WL/UDRI facility is unresolved. An approach might be to convert from a higher rainfall rate to a lower rainfall rate by increasing duration.

The number of raindrops that collide with the transparency within a unit of time is logically proportional to the product of rainfall rate and flight velocity. For example, a one inch rainfall rate for 4.18 minutes would result in more collisions than a 1.4 inch rainfall rate for 2.4 minutes (product equals 3.36). The 4.18 and 2.4 minute values were determined in the Expected Exposure Duration section. If the number of collisions is proportional to the amount of damage, then the rate/duration trade is valid.

Even if this relationship is valid, it likely has upper and lower bounds. The lower bound would reflect a threshold duration at which a specific rainfall rate and velocity combination would not cause damage. The upper bound represents complete surface damage (i.e. coating removal) or end of testing. In between the bounds as testing duration increases, surface damage increases. A critical assumption to the rate/duration trade and the ideas set forth in this paper is that the subject durations fall within these hypothetical bounds. Further investigation is required to prove or disprove this hypothesis.

The number of collisions for a given rainfall rate at the UDRI facility is also in question. Adler [13] suggests that run times be increased by a factor of three or four to better compare with representative natural rainfalls.

Much work remains in the field of rain erosion testing. As Gilligo [16] states, the mechanism of rain erosion is a complex phenomenon that is a function of test parameters and characteristics of the specific polymeric material.

Acknowledgement

This report was originally published by General Dynamics Fort Worth Division (now Lockheed Fort Worth Company) as report 16PR10780. The original title was: Assessment of Expected Damage To F-16 Transparencies From Exposure to Rain Over a Four Year Service Life. This report was co-authored by C. A. Webster and W. L. (Lynn) Early of Lockheed Fort Worth Company.

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EVALUATION OF HARDENED B-1B WINDSHIELD MATERIALS

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EVALUATION OF HARDENED B-1B WINDSHIELD MATERIALS

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and
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University of Dayton Research Institute

ABSTRACT

There is an operational need for providing integrated protection from projected weapon threats to B-1B transparencies. Initially, coupon-type specimens were molded and/or machined from candidate material systems and subjected to exposure to chemical threats, simulated nuclear heat flux, and laser irradiation. Additional samples of candidate hardened transparency materials were subjected to the following test/evaluation: quantifying the energy passed through the material during thermal flash exposure; quantifying changes in optical density during laser irradiation; and evaluating durability of unconditioned exterior coatings in a simulated nuclear dust environment. Existing photochromic interlayers show promise for controlling the heat energy passing through a transparent laminate during nuclear flash exposure, but fail to respond in time to satisfy flashblindness requirements. When subjected to simulated nuclear dust environment, samples with an elastomeric liner on the outer surface experienced less optical degradation than samples with a hard overcoat.

Recently, coupon-type specimens which incorporate ITO on a chemical strengthened glass face ply and the EOII system on a high temperature polycarbonate face ply have been exposed to four equivalent years of accelerated weathering plus cleaning. Subsequent test/evaluation, including laser irradiation, is being conducted to substantiate durability of the face ply coating and structural integrity of the face ply to structural ply bond.

INTRODUCTION

During 1988, Phase I hardened transparent coupon-type specimens were procured and tested to evaluate material degradation. Candidate material systems, having no pre-conditioning, were exposed to chemical threats, simulated nuclear flash, and laser irradiation. During 1990, Phase II candidate transparency material systems, hardened to protect the aircrew, were subjected to the following test/evaluation: quantifying the energy passed through the laminate during simulated thermal flash exposure; quantifying changes in optical density during laser irradiation; and evaluating durability of abrasion resistant overcoats in a simulated nuclear dust environment. During 1992-93, Phase III coupon-type specimens which incorporate a laser resistant face ply were procured, exposed to four equivalent years of accelerated weathering (UV/moisture/cleaning), and then subjected to tests to evaluate coating durability.

PHASE I: PROTECTION OF THE MATERIAL SYSTEM

Coupon-type specimens were molded and/or machined from the two laminated material systems shown in Figure 1.

Three replicates, each 1-1/2" x 1-1/2" x t, of each material system were exposed to three agents: thickened soman (TGD), thickened mustard gas (THD), and agent VX. Haze and transmittance measurements were made before and after exposure; testing being conducted by Battelle-Columbus Division. All three laminates tested; i.e., coated glass faced, coated GAC-590 faced, and uncoated GAC-590 faced, performed satisfactorily; all three being relatively impervious to the chemical agents.

Three replicates, each 2" x 2" x t, of each material system were exposed to laser irradiation at the Laser Hardened Materials Evaluation Laboratory (LHMEL), Wright-Patterson Air Force Base. Samples were subjected to different power levels and times using a continuous wave (CW) carbon dioxide (CO_2)laser having a flat-top beam profile to determine the severity of resultant frosting. Test results are presented in Table 1. Note that one coated glass faced sample was tested again at the 1995 power level threat and passed.

Three replicates, each 4" x 4-1/2" x t, of each material system were exposed to heat flux to simulate nuclear flash. These tests were conducted at the Tri-Service Thermal Radiation Test Facility established by the Defense Nuclear Agency at Wright-Patterson AFB. A flux level of 55 cal/cm^2-sec for a duration of 3 seconds was used. Aerodynamic flow over the specimens corresponded to a Mach number of 0.6. Visual inspection after exposure proved adequate to subjectively determine comparative material degradation and/or pass/fail criteria. The laminates having a coated GAC-590 face ply exhibited similar behavior to the laminates with uncoated GAC-590 face plies. Near the end of one three-second exposure of 55 cal/cm^2-sec, each experienced bubbling and pitting on the surface, locally turning milky-white. There did not appear to be any visible degradation during the first one or two seconds of exposure. The coated glass-faced laminate did not experience any change in visibility after three three-second exposures. However, there were resulting areas of internal delamination and some cracking upon cooldown, possibly caused by interference fit expansion within the test fixture.

PHASE II: PROTECTION TO THE AIRCREW

The test specimen cross-sections shown in Figure 2 were subjected to flux levels of 20 cal/cm^2 and 40 cal/cm^2 using a one-second rectangular pulse to simulate nuclear flash. Calorimeter measurements were made to determine the heat energy passing through the material during exposure.

Visibility was maintained after multiple exposures for all specimens tested. Approximately 5 cal/cm^2 and 10 cal/cm^2 passed through the "B" samples when exposed to 20 cal/cm^2 and 40 cal/cm^2, respectively. Approximately 0.5 cal/cm^2 and 1.0 cal/cm^2 passed through the "A" samples when exposed to 20 cal/cm^2 and 40 cal/cm^2, respectively. All the visible light appeared to pass through both configurations.

Tests were conducted to evaluate the changes in optical density of candidate transparency material samples during laser irradiation. These tests were conducted in the Optical Laboratories of the Applied Physics Division at UDRI. The test samples were exposed to two wavelengths in the infrared region (10.6 μm and 1.06 μm) and one wavelength (0.53 μm) in the visible range. The test samples absorbed the light in the infrared region instead of switching. This is in agreement with the thermal flash test results; similar specimens blocking the heat energy passing through the material during exposure. All samples failed to attenuate the beam when exposed in the visible spectrum.

Tests to evaluate the durability of protective exterior coatings in a nuclear dust environment were conducted in the DNA Dust Erosion Facility located at PDA Engineering, Santa Ana, CA. The test specimens consisted of flat 2x2-inch tempered glass substrates in three different configurations as follows:

1. Chem-strengthened glass with ITO coating outboard and with an antimony-tin oxide coating over the ITO (identified as Sample C).

2. Chem-strengthened glass with ITO coating outboard and an elastomeric liner over the ITO (identified as Sample D).

3. Thermally tempered glass with laser-reflective coating and an elastomeric liner on the outer surface (identified as Sample E).

The transmittance and haze measurements were sensitive to erosion damage. Although the luminous transmittance dropped by only a few percent, the haze increases in most cases were significant. Post-test haze measurements are summarized graphically in Figure 3 in terms of a bar chart. Each bar represents the mean value of two identical samples tested at each condition. The samples are identified as C, D, and E. As shown in the figure, damage increased with velocity and dust loading. Case 1 was the low velocity case and resulted in the least damage with haze levels of less than 10% for all materials. As velocity was increased to 700 fps with a reduced dust load in Case 2, the haze levels increased. Maximum damage occurred in Case 3 with the higher velocity and dust load. For Case 3, haze values ranged from approximately 19% to 50%. As shown in Figure 3, the samples with an elastomeric liner on the outer surface (i.e., D and E) experienced significantly less optical degradation than the samples with the hard (ATO) overcoat.

PHASE III: COATING DURABILITY

Coupon-type specimens were procured in accordance with the cross-section shown in Figure 4. Face ply materials consisted of ITO coated chem-tempered glass and EOII coated high temperature polycarbonate.

The specimens were mounted on racks and placed in QUV machines. The QUV exposure consisted of alternating cycles of 8 hours of UV exposure at 70- degrees Celsius (158°F) using UVA-340 bulbs, followed by 4 hours of dark/condensation at 50 degrees Celsius (122°F). Two weeks of exposure (336 hours) in the QUV simulates one year of

natural weathering. The specimens will be exposed for a total of 8 weeks (4 equivalent years). Routing cleaning was simulated by wiping the surface of the specimens 50 times with a 50/50 mixture of isopropyl alcohol every other day (after 48 hours of artificial weathering), using a Kimwipe paper towel.

After exposure to four equivalent years of accelerated weathering, the specimens will be tested to substantiate (a) durability of the face ply coating, and (b) structural integrity of the face ply to structural ply bond. Figure 5 presents the proposed test matrix.

ACKNOWLEDGEMENT

The work described herein was performed under Contracts F33615-84-C-3404 and F33615-92-C-3400 with the Air Force Wright Laboratory (WL/FIVR), Wright-Patterson Air Force Base, Ohio.

TABLE 1
TEST RESULTS - PHASE I

CHEMICAL/BIOLOGICAL	
Coated Glass	Pass
GAC-590	Pass
THERMAL FLASH	
Coated Glass	Pass
GAC-590	Pass 1 & 2 sec
	Fail 3 sec
LASER	
Coated Glass	Pass at 5, 7.5, & 10 sec
	Pass at 7 sec (1995 goal)
GAC-590	Pass at 5 sec (minor craze)
	Fail at 7.5 sec

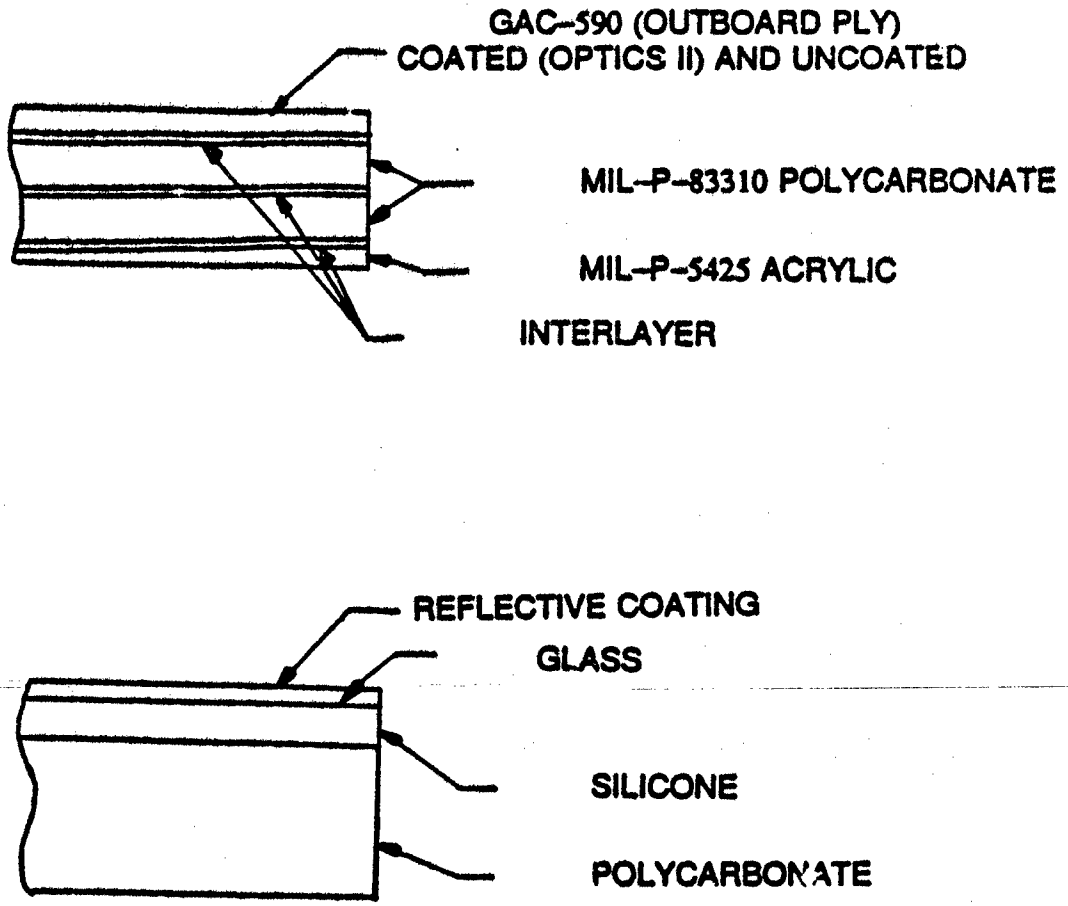
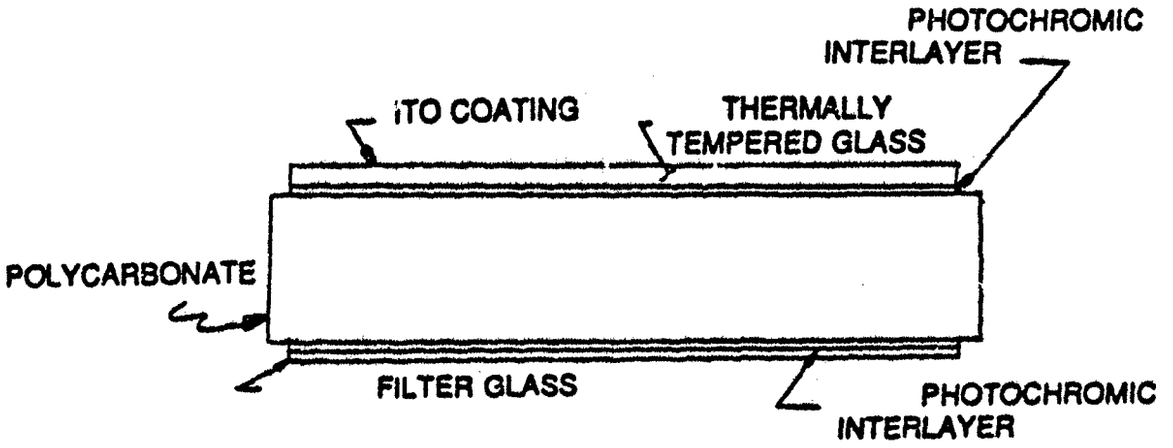


Figure 1. Cross-Sections of Phase I Transparent Laminates for Evaluation of Material Degradation



SPECIMEN "A"



SPECIMEN "B"

Figure 2. Cross-Sections of Phase II Hardened Transparent Laminates for Evaluation of Aircrew Protection

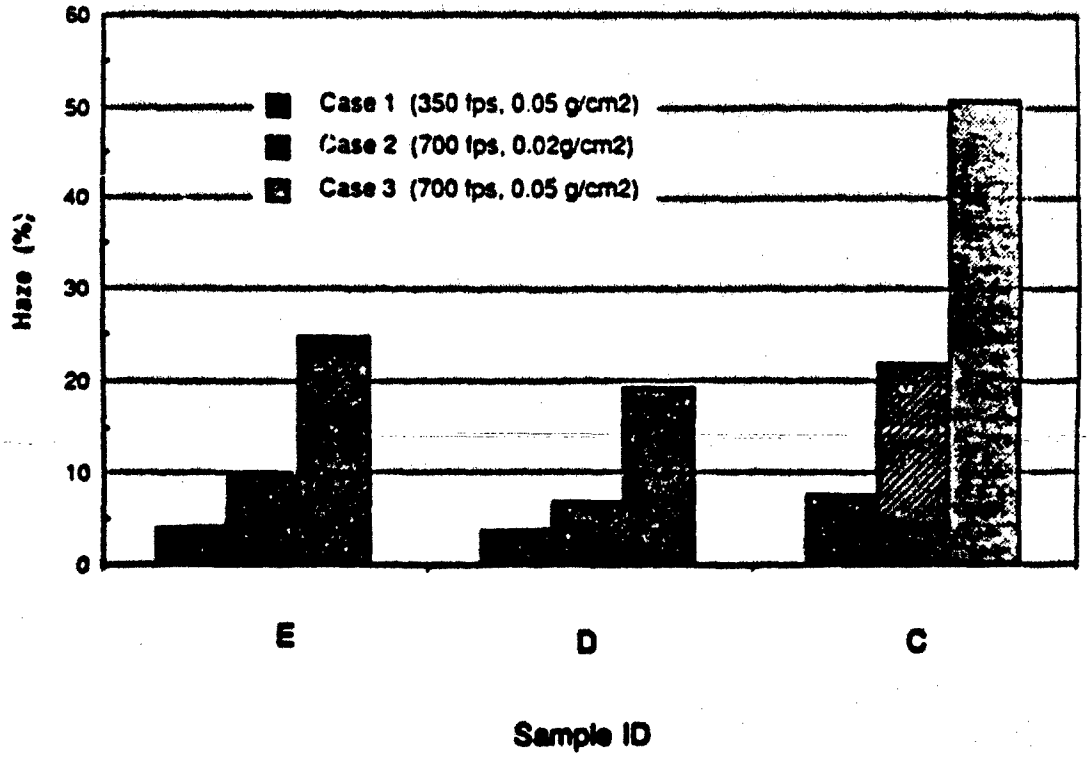


Figure 3. Post-Test Haze; Nuclear Dust

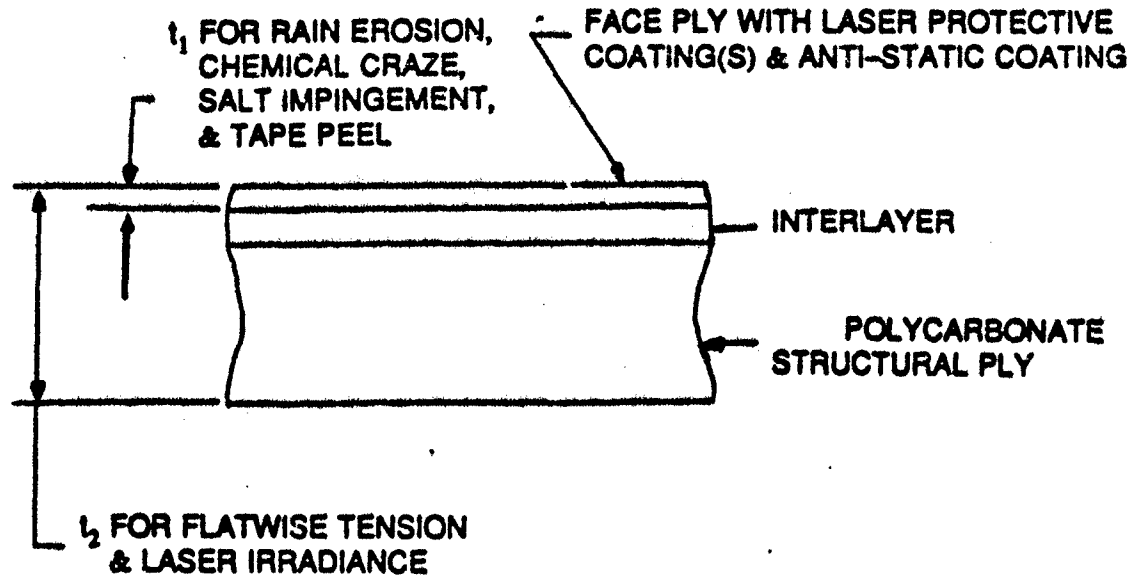


Figure 4. Cross-Section of Phase III Hardened Transparent Laminates for Evaluation of Coating Durability

Test	Parameter	Specimen Geometry	No. Required
Rain Erosion	Coating Durability	1" x .937" x t_1	6
Chemical/Stress Craze	Coating Durability	1" x 7" x t_1	15
Abrasion Electrical Resistivity	Coating Durability	4" x 4" x t_1	3
Tape Scribe Peel	Coating Adhesion	2" x 2" x t_1	3
Flatwise Tension Haze & Transmittance	Delamination	2" x 2" x t_2	5
Laser Irradiance @ LHMEL	Coating Durability/ Performance	2" x 2" x t_2	6
Total Specimens Required			38

Figure 5. Test Matrix for Evaluation of Coating Durability

ABRASION RESISTANT CANOPY (ARC) MATERIALS EVALUATION PROJECT

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Bell Helicopter Textron, Inc.

Peter G. Dehmer
ARL - Materials Directorate

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Abstract

A Tri-service test program (U.S. Army, Air Force, Marines) was conducted to evaluate recent technology developments in scratch resistant coatings for helicopter transparencies. Eight coatings were evaluated for application to polycarbonate and stretched acrylic flat panel coupons. Two metallic coatings, ITO and gold, were also evaluated in conjunction with the abrasion resistant coatings. A series of mechanical, environmental, and optical coupon tests were conducted on the coated and uncoated baseline configurations. Mechanical testing included three-point flexure and tensile properties. Environmental testing included fluid soak, humidity conditioning, sand and dust, and weatherometer. Optical testing included Bayer abrasion, haze, and light transmittance.

Two of the more promising coatings were incorporated in a flight test evaluation at Naval Weapons Center, China Lake, CA. The forward right gunner's windows on two AH-1W SuperCobra gunships were modified with the scratch resistant coating. Results to date show excellent scratch resistance and durability.

1. INTRODUCTION

In the U.S. military operational environment, mission readiness is predicated on a systems or subsystems ability to perform its function when required. Operation Desert Storm has shown that the modern battlefield of 10 years ago is no longer the dominant operational environment. The conditions of Southwest Asia create a severe and erosive environment that is detrimental to both our weapon systems and their mission readiness. This harsh environment is particularly detrimental to acrylic and polycarbonate transparencies.

The objective of this effort is to evaluate and demonstrate recent technology developments in abrasion resistant coatings and their applicability to helicopter transparencies. The goals associated with this program are to show an increase in component life through an increase in scratch resistance, improved chemical resistance and little or no optical degradation over time.

This report summarizes some of the initial coupon testing, producibility, and cost analysis contributing to the fabrication of two scratch resistant forward right gunner canopies and subsequent flight testing on two AH-1W SuperCobras stationed in a desert environment. The flight test evaluation has not yet been completed. Final approval and acceptance of the two prototype canopy configurations is to be made on the basis of flight test performance, on evaluation of the in-service effects with the maintenance crews and long term optical properties.

2. DISCUSSION

During Operation Desert Storm, a series of logistic support activities were initiated by the defense contractor industry. The impetus for this activity was based upon problems that became apparent because of the erosive desert environment. One problem that affected both land and air vehicles was the high demand for plastic transparency replacements. During this time, the four major suppliers of military canopies, PPG, Pilkington, Sierracin, and Texstar, were active in the development and application of erosion resistant coatings. Independently, each manufacturer supplied the U.S. Government either jet fighter or helicopter transparencies with the abrasion resistant coatings. Unfortunately, the U.S. Government spares procurement pipeline combined with the war effort did not readily allow for technical monitoring of the modified components. Because of this situation, there is no technical data that would show comparison of the modified windscreens to the uncoated systems. This realization motivated the development of a tri-service program, jointly funded by the Army, Marine Corps, and Air Force, under contract with Bell Helicopter Textron Inc., to test and evaluate the most promising of these materials. This program documents from coupon level through flight test the application of abrasion resistant coatings on helicopter transparency plastics.

2.1 Tri-Services Organization

In order to comply with the needs of each military service, the program was established with the following emphasis:

Army	Marines
Analysis - Producibility, Cost for the OH-58D POC Pete Dehmer, ARL Watertown, MA	Flight Test - AH-1W SuperCobra POC Lt. Col. John Boyd, PMA 276 Washington, DC

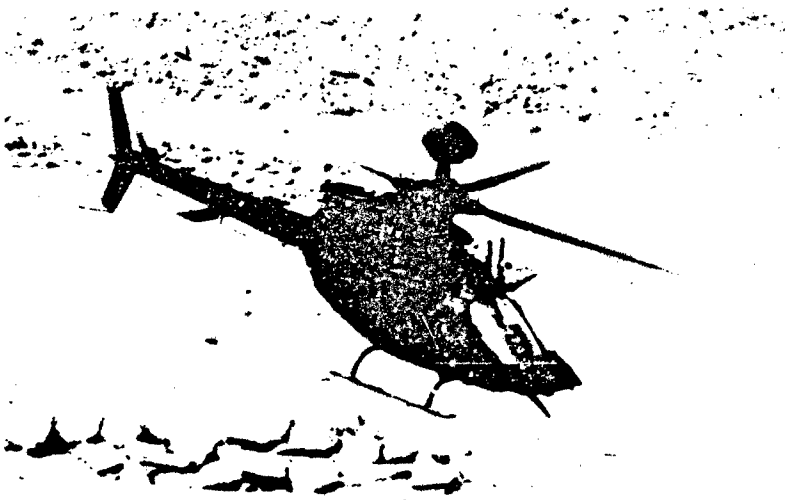
Air Force

Flight Test Data (Light Transmission and Haze vs. Time)
POC Kristen Alexander, WPAFP Dayton OH
POC Capt. Stephen Hargis, McClellan AFB, CA

Prior to the issuance of this program and in support of Operation Desert Storm, the U.S. Army modified 18 OH-58D helicopters with an ITO conductive coating protected on both sides with Pilkington Aerospace Inc. SS6590 hard coat. It was decided that because this hardware was in active service, the Army interest for this program would be placed in cost and producibility. Figure 1 shows the modified OH-58D with the hard coat/ITO windscreens.

The U.S. Marines, with two AH-1W SuperCobras stationed at the NWC, China Lake Desert, would be ideal for accomplishing desert sand environmental testing along with introducing the pilot and maintenance staff to the abrasion resistant coating technology. Figure 2 shows the AH-1W SuperCobra.

The U.S. Air Force Windshield Systems Program Office, WPAFB, Dayton, OH, and U.S. Air Force Advanced Composites Program Office, McClellan AFB, CA, offered the dedicated transparency engineering staff along with the ability to determine in-field optical measurements as a function of time.



31184

Figure 1. OH-58D with SS6590/ITO Transparencies



31194

Figure 2. AH-1W SuperCobra

2.2 Material Selection

2.2.1 Abrasion Resistant Coating

The abrasion resistant coating material selection was based upon a review of recent technology developments (Ref. 1) and interviews with the four major material transparency contractors. Table 1 outlines the coating choices and their applicability to base material of stretch acrylic and polycarbonate.

Table 1. Abrasion Resistant Coating Materials

Vendor	Coating	Type	Applicability		Metallic	
			Stretched Acrylic	Polycarbonate	ITO	Gold
Pilkington	Hard Coat	SS6590	Yes	Yes	X	X
Pilkington	Soft Liner	SS6831	Yes	Yes	X	X
Sierracin	Hard Coat	FX174	Yes	Yes	X	X
Sierracin	Hard Coat	S238	Yes	Yes	X	X
Sierracin	Soft Liner	S239	Yes	Yes	X	X
Texstar	Hard Coat	C678	No(1)	Yes	X	X
Texstar	Soft Coat	C659	Yes	Yes	X	X
PPG	Soft Liner	5300	No(2)	Yes	N/A(3)	N/A

Notes:

(1) At time of submittal, Texstar HC C454 was applicable to polycarbonate only; recent developments/submittal show capability to SA.

(2) At time of submittal, PPG 5300 was applicable to polycarbonate only; recent developments/submittal show capability to SA.

(3) Did not have metallic deposition compatibility with the 5300 liner at time of coupon submittal. PPG anticipates polycarbonate/liner/ITO capability late fall 1993.

2.2.2 Stretched Acrylic

BHTI predominately uses stretched acrylic for the forward and side transparencies of the OH-58D Kiowa Warrior and the AH-1W SuperCobra. The benefits of stretched acrylic over unstretched are an increase in resistance to crazing, higher impact strength, and improved resistance to crack propagation. The disadvantages are reductions in abrasion resistance and tensile and shear strengths.

2.2.3 Polycarbonate

The use of transparent polycarbonate on rotorcraft at BHTI is limited. The primary benefits of polycarbonate over acrylic are impact strength and thermal resistance. Deficiencies are poor chemical resistance, crazing, poor scratch resistance, and difficulty in repairing surface scratches. The overhead skylight bubble for the OH-58D is made from polycarbonate because of the potential for impact damage due to debris from rotor downwash.

2.3 Producibility Assessment

Many criteria must be considered in choosing the best coating or combination of coatings, including abrasion resistance, optical qualities, weathering, ballistics, and cost. It is also necessary to evaluate the techniques used to apply the coatings to determine the producibility, repairability, and field application aspects of each system.

It is helpful to recognize the fundamental distinction between liners and coatings before evaluating the techniques used in their application. For this program, coatings are relatively thin materials applied to the transparency substrate. They are typically less than about 0.003 inch thick. Liners are thicker than coatings and are usually greater than 0.010 inch thick. Coatings and liners may be either hard or soft materials. All coatings and liners may be applied to either of the two types of transparency substrates, acrylic or polycarbonate.

2.3.1 Application Methods

The proprietary nature of materials applied as coatings and liners as well as the equipment involved results in methods of application specific to each system. It is therefore difficult to provide an accurate description of each application technique. This is especially true of the complex vacuum coating methods, evaporation and sputtering, that are used to impart conductivity to the plastic substrates.

Typically, in the coating process, a coupon is often coated along with a helicopter transparency using the same processes and materials. The coupon is tested for solvent resistance, haze, abrasion resistance, or other properties important for the specific application. In addition to visual inspection, the helicopter transparency itself is often subjected to other nondestructive tests such as light transmittance. These inspection techniques are important in maintaining quality coatings and liners.

2.3.1.1 Flow/Dip Coating

Flowing and dipping are the simplest methods of applying coatings to transparencies and have been used for over 20 years. Both are usually performed on transparencies which have already been formed and both require that a primer be applied to the transparency before applying the coating in order to enhance bondability. The primers are usually applied by flow method and sometimes require a thermal cure. Both techniques are best performed in a clean room, preferably with humidity and temperature control. Flowing involves hanging or otherwise affixing the transparency in a vertical position and applying the liquid resin at the top of the part. It is usually accomplished using a hose with the liquid coating material under a small amount of pressure. The liquid material flows down from the top of the part and coats the entire surface evenly. Dipping is accomplished by immersing the transparency into a container of the liquid material. The part is then lifted out of the container and the excess liquid runs off as with the flow method. The pot life of the materials varies from about two hours for soft coatings to several weeks for hard coatings. Following application, both methods usually require cure at elevated temperature, although some coatings cure by exposure to ultraviolet light.

Most coating systems are one-part and require only thermal expocure to set up. An exception is the urethanes, which are usually two-part. The long pot life of the resins after application allows for any defects to be easily corrected by wiping off all or some of the uncured coating and re-applying as necessary. Defects include streaking or embedded dust particles. Both methods typically produce coatings of good visual quality.

The flow and dip coating techniques are used to apply hard and soft coatings. The hard coating materials include silicone, acrylic, and melamine while soft coatings are usually urethane. The dip method requires a great deal of material and its use is limited by the size of the container. Flow coating may be used if only one surface of the transparency is to be coated; however, dip coating is limited to parts which are to be coated on both sides.

2.3.1.2 Laminating

Laminating liners onto transparencies is a more complex process than the flow and dip methods discussed above. A proprietary primer is usually applied to the unformed transparency and cured if required. A cured, extruded thermoplastic liner is applied to the transparency and the two are formed together. The forming ensures full cure of the primer. Since the liners are thermoplastic, they are not unduly affected by the high temperatures or change in geometry caused by forming.

Although it is still a relatively new process, it is generally accepted that laminating has a higher potential for problems than flow/dip methods. Once the liner is bonded to the substrate, there is no opportunity to correct defects, which include imbedded particles and air pockets between the liner and substrate. The only option would be to remove the liner, if possible, and reapply a new one (see Repair section below).

2.3.1.3 Casting

Like laminating, casting is used to apply liners to transparencies. A primer is usually applied to the unformed transparency and cured, if required. The transparency is then placed in the casting tool. The uncured, liquid, thermoplastic liner material is poured into a cavity between the substrate and another material (usually glass). The cast liner is then cured thermally. After curing, the part is formed as required.

The casting process is complicated and particularly susceptible to contamination. Embedded particles or abnormalities in optical properties cannot be corrected after a liner is cast short of removing and replacing as described below in the Repair section.

2.3.2 Metallic Coating

As the interest in EMI, RF, and ESD protection continues to increase, the use of metallic coatings on transparencies will be a primary consideraton. The two metallic coatings considered for this program were ITO and gold. The conductivity range for the test coupons was held between 20 and 30 ohms per square.

2.3.2.1 Vacuum Sputtering

The vacuum sputtering technique is used to apply metallic coatings to transparencies. A primer is applied to the substrate material. The formed transparency is then placed in a vacuum chamber and the air is evacuated. The pressure and chemical composition of the gasses inside the chamber are closely controlled during the coating process. A heavy inert gas, usually argon, is injected into the chamber in order to provide a source of positively charged

ions. A cathode target composed of the material which will be deposited onto the transparency is attached to either a robotic arm applicator or a stationary fixture. The transparency is located between the target and an earthed anode. Application of voltage causes positively charged ions in the chamber to be attracted to the target, thereby dislodging atoms of the target material. The discharged atoms are forced against and bonded to the transparency. When reactive materials are being sputtered, a reactive gas, usually oxygen, is injected into the chamber along with the argon. The reactive gas will become a component of the coating (see Figure 3). If a robotic arm is used, the coordinates of the transparency are entered into the computer so that the arm moves above the surface of the part so as to apply an even coating. Use of a stationary fixture for the target requires that the part be rotated inside the chamber in such a way that the coating is applied uniformly. Usually, tooling requirements are minimal; all that is needed is a fixture to hold the part. Obviously, the chamber must be large enough to accommodate the transparency.

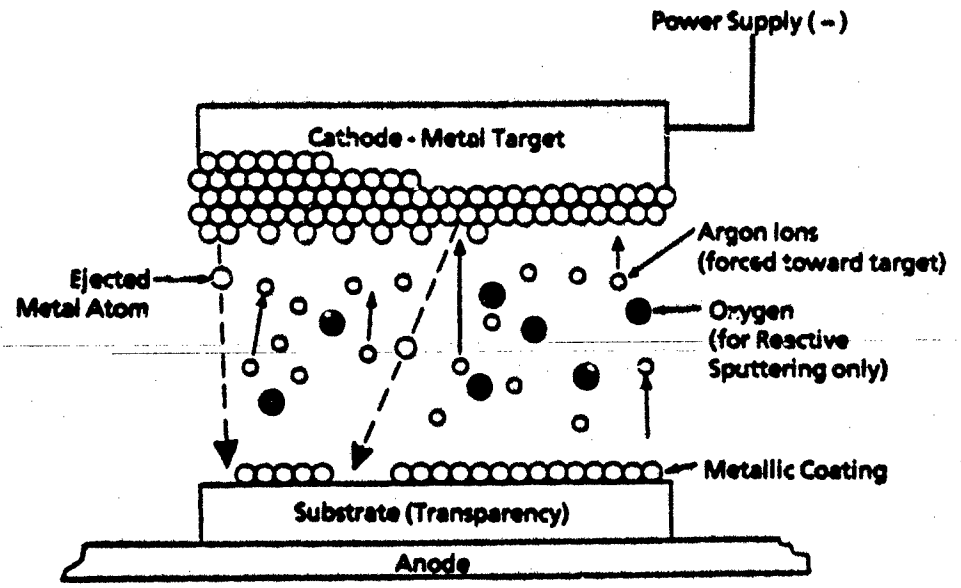


Figure 3. Applying Metallic Coatings by Vacuum Sputtering

Since vacuum sputtering typically uses close parameter controls, it is a very reproducible process. It produces uniform coatings of excellent optical quality. Defects include embedded particles and debonding of the coating from the transparency. The only way to correct defects incurred during application is to strip the entire coating and reapply. The process is time consuming because material is forced against the substrate one atom or molecule at a time.

Any metallic material may be applied using the vacuum sputtering process. However, vacuum sputtering is the only suitable technique for applying reactive coatings such as indium tin oxide.

2.2.2.2 Evaporation

Like the sputtering technique described above, evaporation is used to apply metallic coatings to formed transparencies. A primer is applied to the part. The part is placed in a vacuum chamber and the air is evacuated. The coating material in filament form is heated to

a sufficiently high temperature so that its vapor pressure is higher than the pressure in the vacuum nozzles. The coating material is then deposited upon the part. The part is stationary and the nozzles are fixed at appropriate locations so as to provide even distribution of the coating material. The vacuum chamber must be large enough to accommodate the part. The evaporation process is much faster than sputtering. The complexity of reactive evaporation typically precludes it from being used to apply reactive coatings. Therefore, evaporation is usually used to apply only nonreactive coatings, such as gold or silver. Evaporation produces coatings of good quality, although not as uniform in thickness or optical properties as those applied by sputtering. Embedded particles and debonded areas are typical defects. As with vacuum sputtering, defects can only be corrected by stripping the entire coating and reapplying.

2.3.3 Repair

The repairability of a coating system greatly enhances its worth. Inability to repair a relatively minor defect in a coating may result in scrapping the entire transparency, thereby increasing costs as well as affecting the readiness of the aircraft. The repairability of a coating/liner system should be considered in light of its increased resistance to damage. For example, repairability may not be as important for a system which has relatively high impact and abrasion resistance because it is less likely to sustain damage. The way in which coatings and liners are combined should also be considered. Inability of a metallic coating to be repaired in the field may be less important if it is covered by a hard coating, as is often the case. This section addresses the "shop" repair aspects of each coating or liner system as well as its field repair potential.

In order to repair a coating or liner system, it is usually necessary to remove some or all of the material from the transparency. As with the application methods described above, this is usually accomplished with proprietary materials and processes specific to each coating or liner system.

2.3.3.1 Flow/Dip Coatings

The field repair of coatings applied with flow and dip methods is generally limited to areas where optics are not critical (i.e., the edges). When required, a scratch can be polished down so that it is no longer visible. More coating material is then applied to the damaged area and cured thermally with a heat gun or portable ultraviolet light. The deleterious effect of such repair procedures on the optical properties of the transparency makes repairs away from the edges untenable.

It is possible to repair a flowed or dipped coating in a shop environment no matter where the damage is located on the part. The process begins with stripping away the entire coating using a proprietary material specific to the coating system. The agent is capable of stripping away the coating without damaging the transparency itself. The transparency is then cleaned and primed and the coating is reapplied.

2.3.3.2 Laminated and Cast Liners

It is not possible to sand or polish soft liners in order to effect repair to specific areas, either in the field or in a shop environment. The only option for those materials is removal and replacement of the entire liner. However, the urethane material from which most liners are made is difficult to strip due to its good chemical adherence. In the case of thick liners, it may be impossible to strip away the entire laminate or casting from the transparency; such systems are not repairable.

If the liner can be stripped, the proprietary agent specific to the coating is applied to the part. Following cleaning and priming, a preformed liner having the contour of the actual part is fitted against the transparency. The liner is then thermally cured to the part, using vacuum or positive pressure to ensure uniform bonding. Great care must be taken to avoid embedding dirt particles and air pockets between the liner and the part. Obviously, this operation is best performed in as clean an environment as possible.

2.3.3.3 Sputtered/Evaporated Metallic Coatings

The complex methods used to apply metallic coatings are not conducive to field repair. However, stripping and repair in a shop environment can be and are performed successfully. After stripping with the applicable proprietary material, the transparency is cleaned and primed. The metallic coating is then reapplied to the entire part. It is not possible to repair isolated defects on metallic coatings.

2.4 Coupon Testing

The mechanical and environmental coupon tests performed during this program were extensive. A minimum of 10 individual coupon tests were conducted on the eight abrasion resistant coated systems of stretched acrylic and polycarbonate. For brevity, the results from four of the coupon tests on stretched acrylic, and the subsequent follow-up test with ASTM 735 Bayer abrasion, will be presented in this paper. Figures 4 and 5 show the unconditioned ASTM 735 test results for the hard and soft coated acrylics.

2.4.1 Sand and Dust (MIL-STD-810C, Modified)

The Sand and Dust of MIL-STD-810C was selected to induce sand abrasion on the ARC coatings because it closely approximates the actual abrasive desert conditions of southwest Asia and other geographical areas; also, it presents a tailored program that is controlled and reproducible.

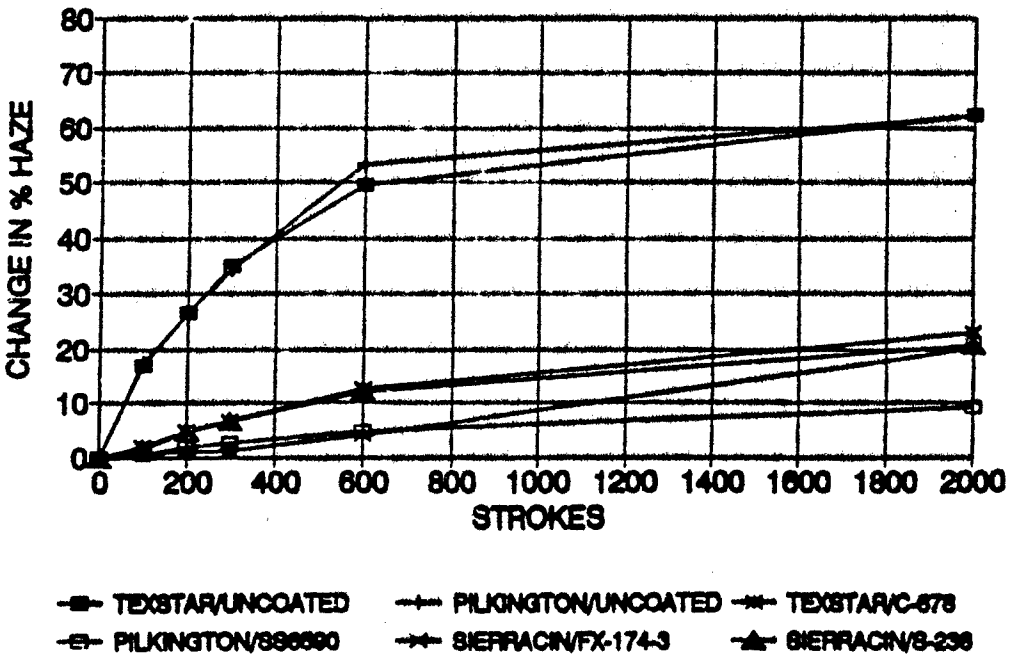
The Sand and Dust chamber utilized during the ARC program is a Model SD-27 constructed by Bemco Inc. of Pacoima, CA. It has a maximum wind velocity of 1750 cubic feet per minute. The chamber temperature is adjustable from room temperature to 160°F. The humidity is extractable to or below 22%, and the sand and dust dispenser/controller is programmable from approximately 0 to 100% (100% equals approximately 1.5 grams per cubic foot).

The ARC specimens were cut 6 inches × 12 inches and the backs and edges of each were protected with metal back tape prior to test. A section of the coated specimens was also protected from the sand abrasion to provide tested/untested visual comparisons.

The specimens were mounted for testing on a rack framework which allowed sand and dust exposure to occur at a 45-degree angle adjacent to the downward thrust of the medium. The 45-degree exposure angle was selected to simulate the approximate angle of an OH-58 windshield.

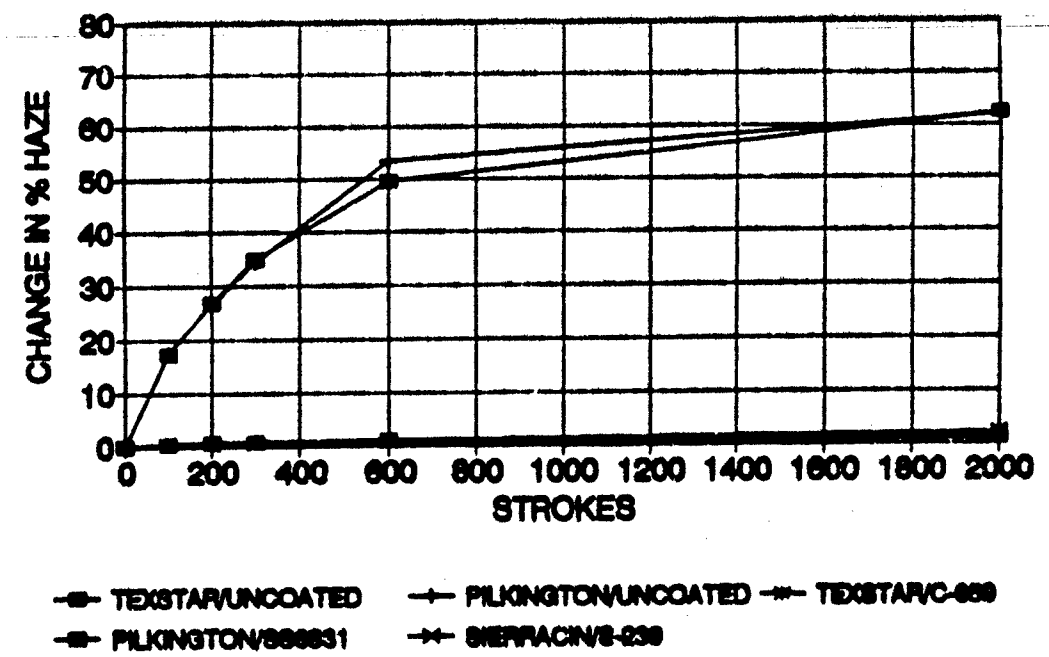
The specimens were rotated during the test to allow equal impingement of the sand and dust on the test surfaces.

The sand and dust dispenser was set to maintain 0.9 to 1.1 grams per cubic foot of abrasive media (MIL-STD-810C, 140 mesh Ottawa Quarts) during the exposure time. The wind velocity was set at 1750 cubic feet per minute. The temperature was controlled at 120°F and the humidity was kept below 22%.



SL174

Figure 4. ASTM F-735 Oscillating Sand Abrasion (soft coated acrylics)



SL175

Figure 5. ASTM F-735 Oscillating Sand Abrasion (hard coated acrylics)

The specimens were impinged for 300 hours, (considered to be approximately equal to 3600 5-minute, off-pad takeoffs/landings; estimates derived from OH-58 sortie information). Position rotation was accomplished approximately every 38 hours.

Following the 300-hour exposure, the specimens were demasked; tape adhesive was removed with IPA (TT-I-735) and washed with neutral soap and water. Visual observations were made concerning the optics, and then all specimens were subjected to ASTM D1003 haze and light transmission testing.

Figures 6 and 7 show the reduced sand and dust data for the hard and soft coated stretched acrylics.

2.4.2 Accelerated Weathering Carbon ARC (ASTM G23, Method 1)

The accelerated weathering was accomplished in a triple carbon-arc light-exposure apparatus (with water) in accordance with Method 1 of ASTM G23. The apparatus utilized for the test is an Atlas Electric Devices Model XW "Weather-o-meter" made in Chicago, IL. The unit has a 24-hour light source with water exposure capability and is equipped with a rotating specimen carousel that equalizes the light and water impingement on the specimens. The specimen temperature is adjustable and the exposure time is recorded by a run timer.

The ARC specimens were cut to 5 inches × 9 inches to accommodate the optimum carousel arrangement; 4-inch × 4-inch "cutoffs" retained from each specimen were utilized as untested control comparison specimens.

The ARC specimens were exposed to 300 hours on initial testing and follow-up specimens to 500 hours of continuous triple-arc light exposure with 18 minutes of water spray at the end of each 6 hours of runtime. The specimen temperature was controlled at approximately 125°F during the light exposure.

The ARC specimens were carefully cleaned with neutral soap and deionized water at the conclusion of the exposure time, and then subjected to ASTM F735 scratch testing to determine the effects of the accelerated weathering on the coatings.

After observations were made and the ASTM F735 data was reduced, the ARC coatings were compared by substrate.

Figures 8 and 9 show the ASTM 735 data for the stretched acrylic accelerated weathering specimens.

2.4.3 Moisture Absorption

The ARC moisture absorption testing was accomplished in a humidity chamber using procedures designated by MIL-STD-810C, Method 507.1, Procedure I. This procedure outlines a plan for 10 days of cyclic humidity exposure whereby the relative humidity remains approximately 95% while the temperature is varied from 85°F to 149°F within 2 hours, is then stabilized for 6 hours at 149°F, and finally returned to 85°F within 16 hours.

The ARC specimens were cut to 4 inches × 4 inches, weighed, and dried in a 160°F chamber until a constant weight was achieved. All uncoated areas of the specimens were masked with metal backed tape and weighed.

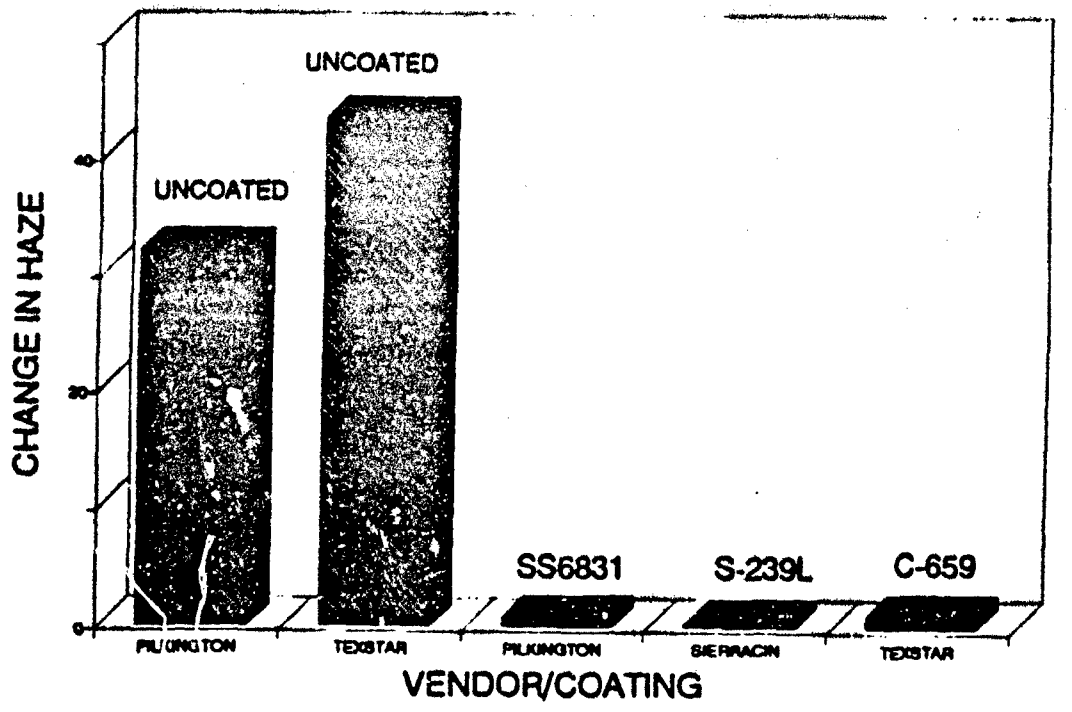


Figure 6. MIL-STD-810C Sand and Dust Exposure (soft coated acrylics)

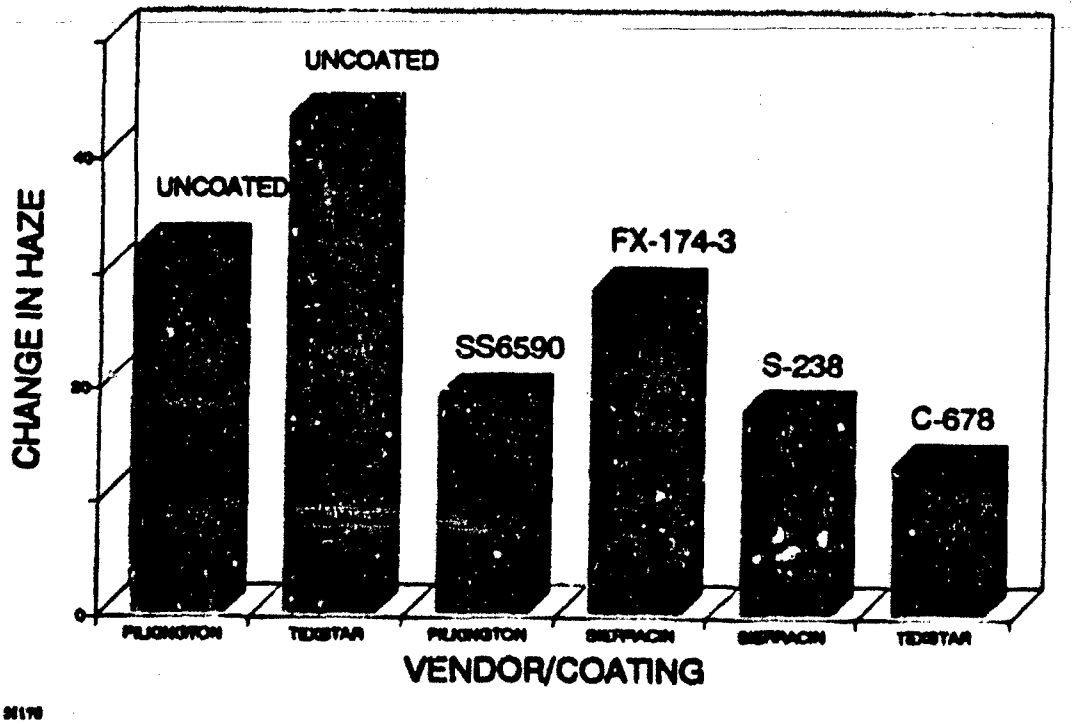


Figure 7. MIL-STD-810C Sand and Dust Exposure (hard coated acrylics)

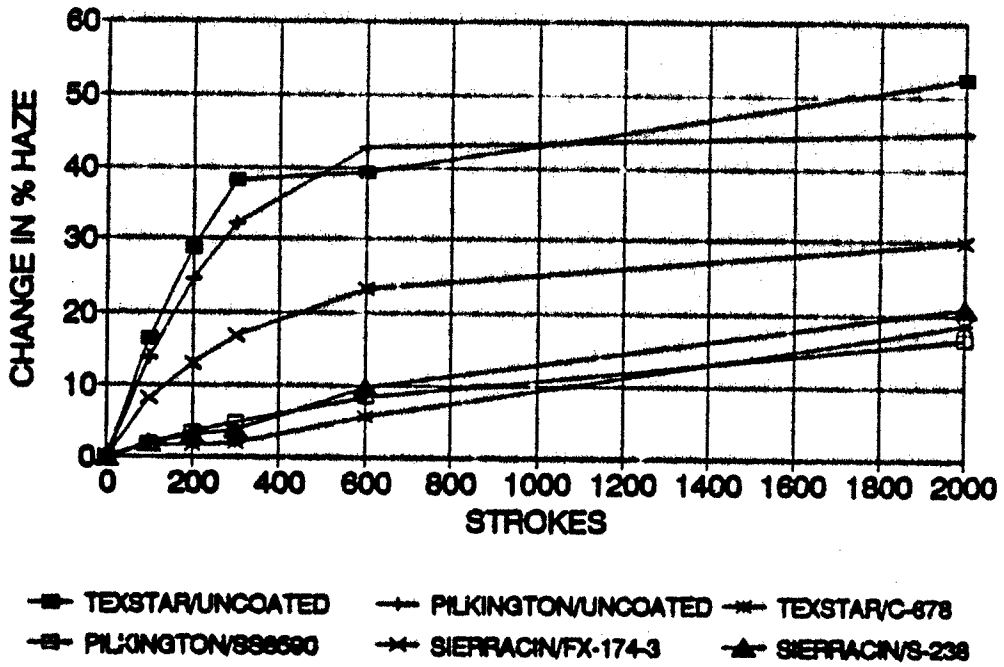


Figure 8. ASTM F-735 Oscillating Sand Abrasion (weathering on hard coated acrylics)

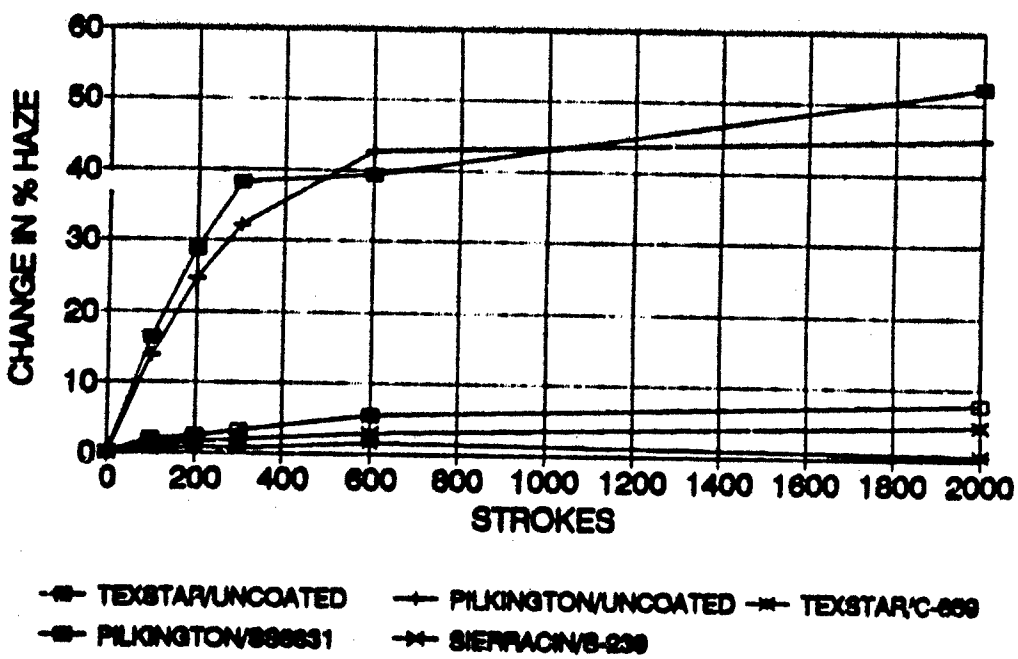


Figure 9. ASTM F-735 Oscillating Sand Abrasion (weathering on soft coated acrylics)

The ARC specimens were then exposed to 10 days of cyclic humidity exposure with observations and weight measurements made each day.

Following the 10 days of cyclic humidity exposure, the ARC specimens were subjected to scratch testing in accordance with ASTM F735 to determine the effects of the moisture absorption on the ARC coatings.

After observations were made and the ASTM F735 data was reduced, the ARC coatings were compared by substrate.

Figures 10 and 11 show the ASTM 735 cyclic humidity test results for hard and soft coated stretched acrylics.

2.5 Cost Analysis

The current configuration of the armed OH-58D Kiowa Warrior has polycarbonate sky lights and acrylic windshields, chin bubbles, and crew door windows. These windscreens are easily scratched causing visibility problems. Initial investigations at the beginning of the ARC Program indicated that the windscreens are removed and replaced every three years. There is significant cost associated with this action over the life of the fleet (500 aircraft).

The proposed configuration adds a scratch-resistant coating to the interior and exterior of the windscreens. The additional durability will increase the life of the windscreens. The percentage increase will depend on the combination of hard coats/soft coats/liners that are chosen to protect the windscreens. There are reports that state that the coated windscreens of fielded aircraft are lasting 4 times longer than the uncoated configurations (Refs. 2 and 3). This analysis conservatively assumes that the coated windscreens will last 3 times longer than the current configuration, no matter what combination of coatings/liners are chosen.

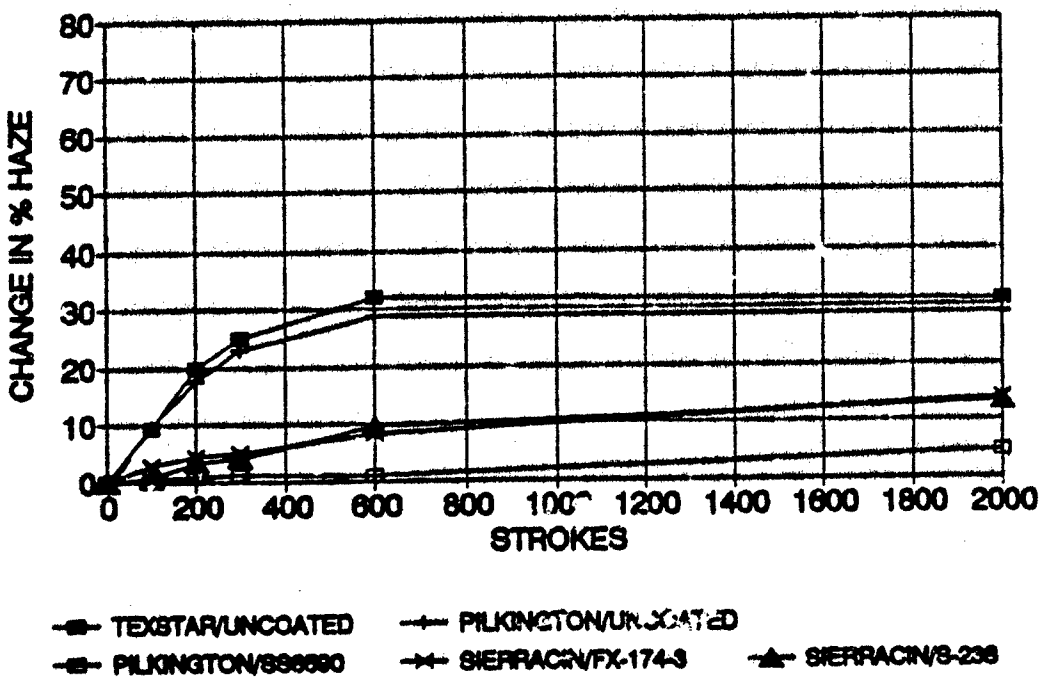
An analysis was conducted to determine the cost effectiveness of implementing the proposed configuration on the Kiowa Warrior fleet. The operating and support (O&S) cost over the life of the fleet was estimated for the current configuration, assuming the coated windscreens are not incorporated. Also, the nonrecurring cost to implement the proposed change and the O&S cost of the coated windscreen configuration were considered. The analysis showed that the incorporation of coated windscreens on the Kiowa Warrior fleet would significantly reduce the Army's O&S cost over the life of the program.

2.5.1 Operating & Support Cost of Current Configuration

Based upon the initial assumption, the current windscreens will be removed and replaced six times over the life of each aircraft (years 3, 6, 9, 12, 15, and 18). This equates to 3000 replacements over the life of the fleet (500 aircraft × 6 replacements/aircraft).

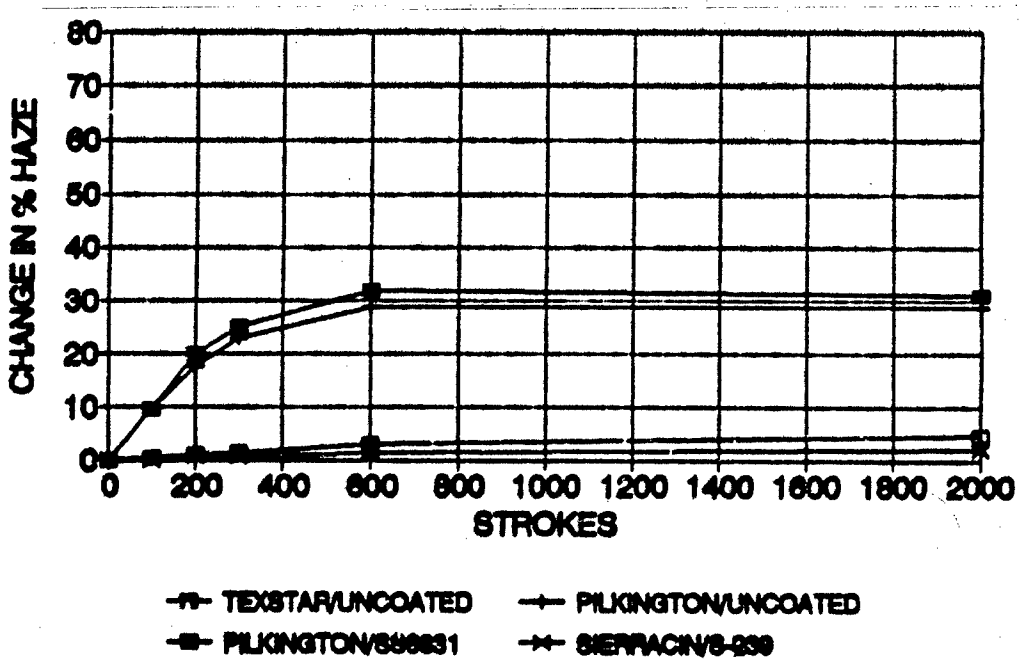
The assumed cost of the windscreens was $6,600. The assumed cost to remove and replace the windscreens is $5,400. Therefore, the total cost to remove and replace a shipset of uncoated windscreens is $12,000.

If the proposed change is not implemented, the O&S cost of the current windscreens is $36,000,000 over the life of the fleet ($12,000 to remove and replace a shipset of windscreens × 3000 replacements).



82100

Figure 10. ASTM F-735 Oscillating Sand Abrasion (humidity on hard coated acrylics)



82101

Figure 11. ASTM F-735 Oscillating Sand Abrasion (humidity on soft coated acrylics)

2.5.2 Implementation and Operating & Support Costs of Proposed Configuration

The nonrecurring implementation cost of the proposed configuration is $22,000. This includes the cost to change drawings, part numbers, and update both logistics spares provisioning and technical manuals.

The coated windscreens are estimated to last three times longer than the current configuration, or 9 years. Over the life of each aircraft, the proposed windscreens would be removed and replaced twice (years 9 and 15). This equates to 1000 replacements over the life of the fleet (500 aircraft × 2 replacements/aircraft).

The cost of the coated windscreens is projected to be $9,900. This cost will vary depending on production alternatives. The labor cost to remove and replace the windscreens is $5,400. Therefore, the total cost to remove and replace a shipset of coated windscreens is $15,300.

The O&S cost of the proposed windscreens over the life of the fleet is $15,300,000 ($15,300 to remove and replace a shipset of coated windscreens × 1000 replacements).

2.5.3 Estimated Operating & Support Cost Savings

The estimated O&S cost savings is $20,678,000. This was calculated by subtracting both the nonrecurring cost to implement the change and the estimated O&S cost of the proposed configuration from the O&S cost of the current configuration ($36,000,000 - $22,000 - $15,300,000).

2.5.4 Cost Analysis Conclusions

Recent data reveals (Ref. 2) that the current windscreens are only lasting 1.5 years. A higher potential cost savings to the Army can be achieved by substituting more realistic data for the durability of the current configuration. The assumptions in this analysis will be refined as the ARC Program progresses (Ref. 4).

2.6 Flight Testing

Prior to the application of any transparency coating to the AH-1W SuperCobra, the U.S. Marine Corps. required confirmation that emergency egress would not be compromised. As an added value to the current contract, and in cooperation with Sierracin/Sylmar, BHTI performed a canopy detonation on a coated transparency installed in an AH-1T. The test was conducted on the AH-1T prior to a modification upgrade to the AH-1W SuperCobra. The Sierracin abrasion resistant coating was FX174. Figure 12 shows the photograph of a successful detonation for emergency egress.

The enhanced performance obtained from each canopy vendor's abrasion resistant coating made the decision to choose a flight test configuration difficult. After many hours of independent evaluations and "discussions," Pilkington's hard coat, SS6590, was to be flown on aircraft no. 163933 and Texstar's soft coat, TXPU, was to be flown on aircraft no. 163937. Figure 13 shows aircraft no. 163937 during canopy installation.

The most common reason for removal of an aircraft transparency is the optical degradation due to haze, halation, or contrast loss as perceived by the pilot. In order to determine optical degradation as a function of time on the coated and noncoated transparencies, ASTM F943-90, Standard Test Method for Measuring Haze of Transparent Parts, was performed on the installed canopies. This test will be repeated on the test windows



Figure 12. AH-1T Canopy Detonation with FX-174

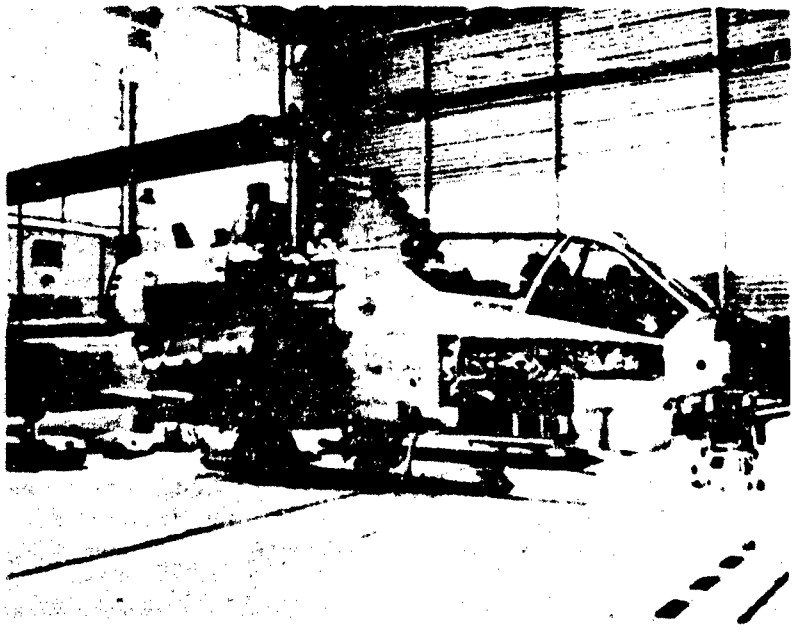


Figure 13. AH-1W Modified with TXPU

during scheduled hanger maintenance, and if necessary, prior to and following any flight training involving a severe desert/erosive environment.

At the time of this writing, both aircraft have been flying the modified transparencies for four months. Future flight operations should include rocket firings, low level flying (NOE), and hovering in the desert sand environment of China Lake, CA.

2.7 Summary

This program documents the benefits associated with the application of abrasion resistant coatings on helicopter transparencies. Recent world developments, as demonstrated through Operation Prime Chance, Operation Desert Storm, and Operation Restore Hope, have shown that the mission environment for military helicopters consists of an abrasive and high temperature environment. The use of these coatings will increase the transparency service life and ultimately mission readiness.

The results from the coupon tests have conservatively indicated an increase in transparency component life by a factor of 4 or more. This is complemented by the recent flight test data obtained from the OH-58D aircraft that were modified with the ITO/hard coat during Operation Desert Storm. From information obtained from Ft. Bragg, Government maintainers and the BHTI on-site technical representatives, the average transparency life of the uncoated forward windscreen was 6 to 9 months (Ref. 2). At the time of this writing, with a minimum of 2 1/2+ years since Desert Storm, none of the 17 modified aircraft (1 loss) has had a canopy replacement. This is further complemented by the fact that the Desert Storm urgency required the application of the abrasion resistant coating on used canopies. These aircraft are currently assigned to the 4/17th at Ft. Bragg, NC. The response from the maintenance crews and pilots indicate that this coating technology was well received.

When applied properly, the hard coat technology offers excellent chemical resistance and good abrasion resistance. In contrast, soft coat technology offers excellent abrasion resistance and moderately good chemical resistance. However, the hard coat technology, if applied improperly, will craze and the soft coat technology is susceptible to mark-off when covered improperly. Repairing isolated defects on coated systems has an adverse effect on the optical properties of the transparency. The coatings applied by flowing and dipping can be field repaired in noncritical optical areas of the transparency. With this exception, all other defects and coating systems require complete removal, stripping, and reapplication. These benefits and concerns must be understood by the manufacturers and users if the full potential of this technology is to be realized.

The use of abrasion resistant coatings on helicopter transparencies has reached the level of maturity to allow expanding the current flight test program to include other military rotorcraft. Efforts are currently underway to modify additional OH-58D aircraft at various locations in order to expand the environmental exposure and personnel involvement. It is through this expanded flight test effort that the transparency technical community will be able to determine the level of user acceptance for the various hard and soft coatings.

As the world situation continues to put more demands on our military rotorcraft, our ability to be more responsive in terms of mission readiness will depend upon the application of technologies that will survive in a desert/sand environment.

3. REFERENCES

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2. Moncure, M.V., Pilkington Trip Report No. 95-93-0090, May 12, 1993.

3. Plumer and McDonald, Evaluation of Scratch and Spall-Resistant Windscreens, AVSCOM Report No. 76-22, Contract No. DAAG46-75-C-0005.

4. Rogers, Marlene, Value Analysis 2395, Improved Windscreen Durability, September 8, 1992.

SESSION II

CURRENT SYSTEMS - PART B

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Lockheed - Ft. Worth

Co-Chairman: R. Speelman
Flight Dynamics Directorate
Wright Laboratory

Coordinator: G. Graening
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F-16 CANOPY TECHNOLOGIES: ACCOMPLISHMENTS, CHALLENGES, AND OPPORTUNITIES

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ABSTRACT

The US Air Force, in a team effort with industry and academia, initiated a program to demonstrate and transition technologies that will enable the F-16 canopy system to meet its mission requirements for the 1995-2000 time frame. The program goals are centered around the objective of balancing mission performance with supportability. This paper offers a "broad brush" look at the F-16 Canopy Team's successes to date, describes ongoing efforts, and explores the future direction of the program. Recent team accomplishments include a qualified 540 knot birdstrike resistant canopy that will meet the four year service life goal using advanced coatings. A fully qualified canopy dry seal system that will reduce canopy change out times to under four hours will be included as part of this system. Future opportunities to increase canopy performance include, combat hardening technologies to protect against laser and chemical weapons threats, tough HUD technologies for a complete system of birdstrike protection, and mission compatible optical performance with emerging systems such as Helmet Mounted Displays (HMD) and Night Vision Goggles (NVG).

INTRODUCTION

The current production canopy for the F-16 was developed to meet the needs of a lightweight air-to-air combat aircraft, providing 350 knot bird impact resistance. Today, F-16 missions require routine low altitude operations at flight velocities reaching 540 knots. The service life is limited between two and three years, and canopy change times can take as long as 96 hours. Both conditions result in a high cost of ownership of the system in comparison to the four year service life and four hour change out goals. The F-16 canopy is vulnerable to combat hazard threats such as chemical and biological weapons and out-of-band lasers.

Responding to the changing environment, the F-16 Canopy Team launched a program to improve the performance and supportability of the F-16 canopy in order to meet its 1995-2000 mission requirements. The F-16 Canopy Team included the Wright Laboratory Windshield Program Office (WL/FIVR), Ogden Air Logistics Center (OO-ALC), the F-16 Systems Program Office, Headquarters Air Combat

Command (HQ ACC), industry, and the University of Dayton Research Institute. In the near term, the Team planned to demonstrate a 540 knot/4 pound bird impact resistant canopy system that meets the 444 goal, which states that a transparency shall maintain mission performance requirements for 4 years at which time it can be changed out in 4 hours by no more than 4 technicians. For the long term, the Team planned to demonstrate combat hardened technologies and integrate them into the system as they mature.

DISCUSSION

Current Accomplishments

540 KNOT/4 POUND BIRDSTRIKE RESISTANCE

The 540 knot/4 pound birdstrike resistance goal emerged out of discussions with HQ ACC on F-16 mission requirements and a desire to protect the pilot throughout the low altitude flight envelope. 540 knot birdstrike resistance was an ambitious goal considering that the current production canopy had been qualified at 350 knots. Complementary laboratory tests and analytical tools were used to reduce the developmental risk. Three point beam tests were conducted on proposed laminate designs and the current production laminate (Figure 1). These tests were useful for identifying laminates with the potential to meet the birdstrike resistance goal based on peak load and energy absorbed during the test. From the beam tests, several laminates were selected for evaluation using a new explicit finite element analysis code called X3D (Figure 2). Quick turnaround times using X3D provided a tool suitable for evaluating the birdstrike resistance of several different laminates prior to selecting a design for full-scale fabrication.

The team's selection strategy for fabricating prototype canopies sought to cover both short and long term objectives. OO-ALC took on the responsibility for demonstrating a two ply design to satisfy the near term objective of 540 knot birdstrike resistance at minimum cost. WL demonstrated a three ply design that features a thin outer ply that will permit the incorporation of a combat hardened system as those technologies mature. Advanced coatings, selected from the Mission Integrated Transparency System (MITS) test program were applied to the canopies to confirm their compatibility with the birdstrike resistant system. Features of the designs are shown in Figure 3. Both designs have passed qualification birdstrike tests, and flight evaluations are planned for the Fall 93 time frame. OO-ALC is preparing to procure the two ply design on a preferred spares basis in FY94.

FOUR YEAR SERVICE LIFE

Inspections of failed F-16 canopies brought in from the field have shown that the service life is limited primarily because of crazing and cracking of the acrylic outer surface and electro-static discharge damage. To combat the short service life, the MITS test program, led by WL/FIVR, evaluated the durability of a new generation of advanced coatings in parallel with the 540 knot birdstrike resistant

canopy effort. Features of those selected for F-16 canopy application include soft urethane coatings designed to eliminate acrylic crazing, and a static drain system which consists of either a metallic outer layer or a conductive soft coating. Flight evaluations are underway for some coatings and others will begin in Fall 93.

Validation of the four year service life will be supported by full-scale durability tests designed to evaluate the coatings in an accelerated F-16 thermal and pressure environment. Quantitative methods to evaluate the ability of the coatings to maintain mission performance requirements include a measurement of contrast loss and haze index, and birdstrike tests before and after durability testing. The advanced coatings will be integrated as part of the 540 Knot Birdstrike Canopy system.

FOUR HOUR CHANGE OUT

Change out times for the current system can reach 96 hours primarily because of the cure times associated with wet sealants, and the labor intensive process of removing "tacky" tape from the frame in preparation for the replacement canopy. To cut change out times down to four hours, OO-ALC led an effort to demonstrate a new dry seal system. The dry "quick" seal system consists of a gasket type pressure seal that is installed between the canopy frame interface, and a rain seal that attaches to the edge of the canopy frame fairing. The new system can be installed in just a few hours, is easy to handle, and can be reused. Ogden ALC has several units flying on F-16s at Hill AFB and at the Kansas ANG. With the system fully qualified, Ogden ALC has taken procurement action and will begin fleet implementation in FY94.

Future Opportunities

COMBAT HARDENING

The threat of chemical weapons attack in Desert Storm brought home the fact that USAF transparency systems are vulnerable. Organic plastics react with the chemicals leaving an opaque surface which can disrupt mission performance and/or ground aircraft. The transparency system is also vulnerable to high powered, out-of-band lasers. Although laser systems are not presently a large threat, delivery systems are shrinking in size making the threat a future reality.

The F-16 Canopy Team demonstrated that a three ply system could defeat a 4 pound bird at 540 knots while meeting the current weight requirement and optical specifications. This system has left the door open to incorporate a combat hardened system that is capable of defeating chemical and laser weapons threats while maintaining 540 knot birdstrike resistance and a four year service life. Figure 4 shows a variation on the three ply design which could incorporate a sacrificial or hardened ply, that when coated with an inorganic film could provide an impenetrable combat hardened system.

The Air Force must work closely with industry to find affordable solutions that have minimal impact on the durability of the system. WL/FIVR will begin a coupon scale test program with UDRI and industry in FY94 to identify possible candidates for the combat hardened system.

TOUGH HUD

Although an F-16 canopy capable of withstanding a birdstrike at 540 knots has been demonstrated, the effort to improve total system safety continues. High speed films from birdstrike tests with impact velocities of 350, 400, 500 and 540 knots, revealed that contact between the canopy and HUD during the birdstrike event (Figure 3) sends a spray of glass toward the pilot's neck, face, and eyes, which could cause injury and potentially lead to loss of aircraft control. This highlights the need for a toughened HUD combiner system, using either a plastic combiner "glass" or coating the glass with a tough, transparent material. This would allow a controlled response of the system and minimize risk. Retaining optical quality in the HUD system will be the largest hurdle to demonstrating this technology. However, the Team is exploring options with the input of the avionics community and will initiate a formal program in Fall 93.

OPTICAL PERFORMANCE

The "bubble" canopy shape gives F-16 pilots the most continuous, unobstructed viewing area of any US fighter. But this complex shape makes the F-16 canopy one of the most optically challenging transparency systems to manufacture as well. The emergence of HMD and NVG systems, present new challenges in ensuring optical quality of the canopy. The F-16 canopy community must anticipate changes in optical requirements and apply innovative manufacturing solutions to meet the new challenges and ensure that F-16 pilots maintain the winning edge in combat.

One promising manufacturing solution under evaluation is Directly Formed (Injection Molding) technologies. This unique forming process offers the possibility of better optical quality, and the potential to reduce procurement costs by an order of magnitude by eliminating many of the steps used in conventional forming methods. WL/FIVR is currently demonstrating the process for the forward portion of an F-16 canopy. The Team's attention is focused on the optical quality and impact resistance of the injection molded parts.

SUMMARY

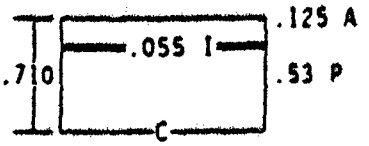
The Team has enjoyed numerous "wins" for the F-16 canopy system over the past few years. The transparency system is now capable of meeting the birdstrike threat in its low altitude flight environment. Reprocurement costs will go down as a results of a longer service life through durable coatings. The dry seal system means faster turnaround times and fewer work hours in the field.

New challenges are on the horizon as the Team continues to provide affordable technology solutions for superior canopy performance and supportability. Durable combat hardening, mission compatible optical performance, and total birdstrike system safety through tough HUD technologies are all challenges that will be met by a strong team dedicated to ensuring that the F-16 meets its mission requirements into the next century.

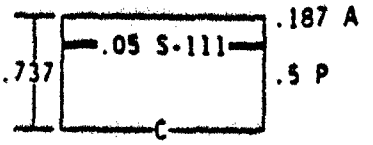
LAMINATES EVALUATED IN BEAM TESTS

CURRENT PRODUCTION CROSS-SECTIONS

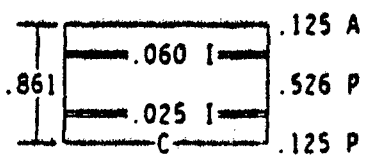
Texstar

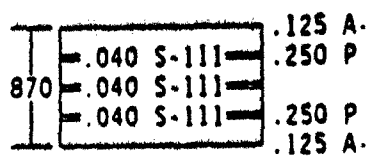


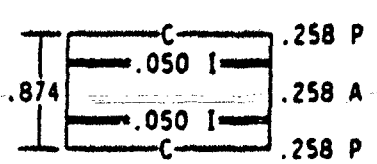
Sierracin

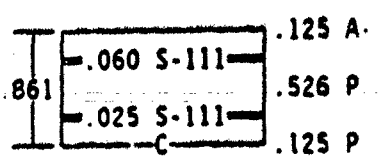


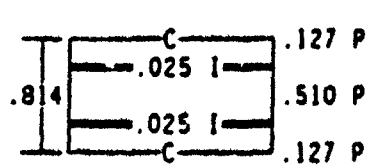
PROPOSED CROSS-SECTIONS

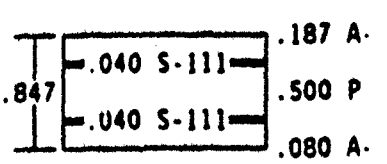


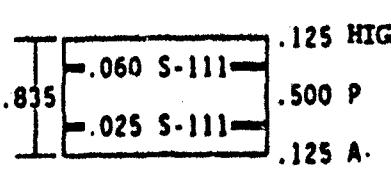












F-16 CANOPY TEAM



NOTES:

A = Acrylic
P = Polycarbonate
C = Coating
U = Urethane
S = Silicone
I = Interlayer

Schematics Not To Scale

Figure 1

F-16 X3D BIRDSTRIKE SIMULATION

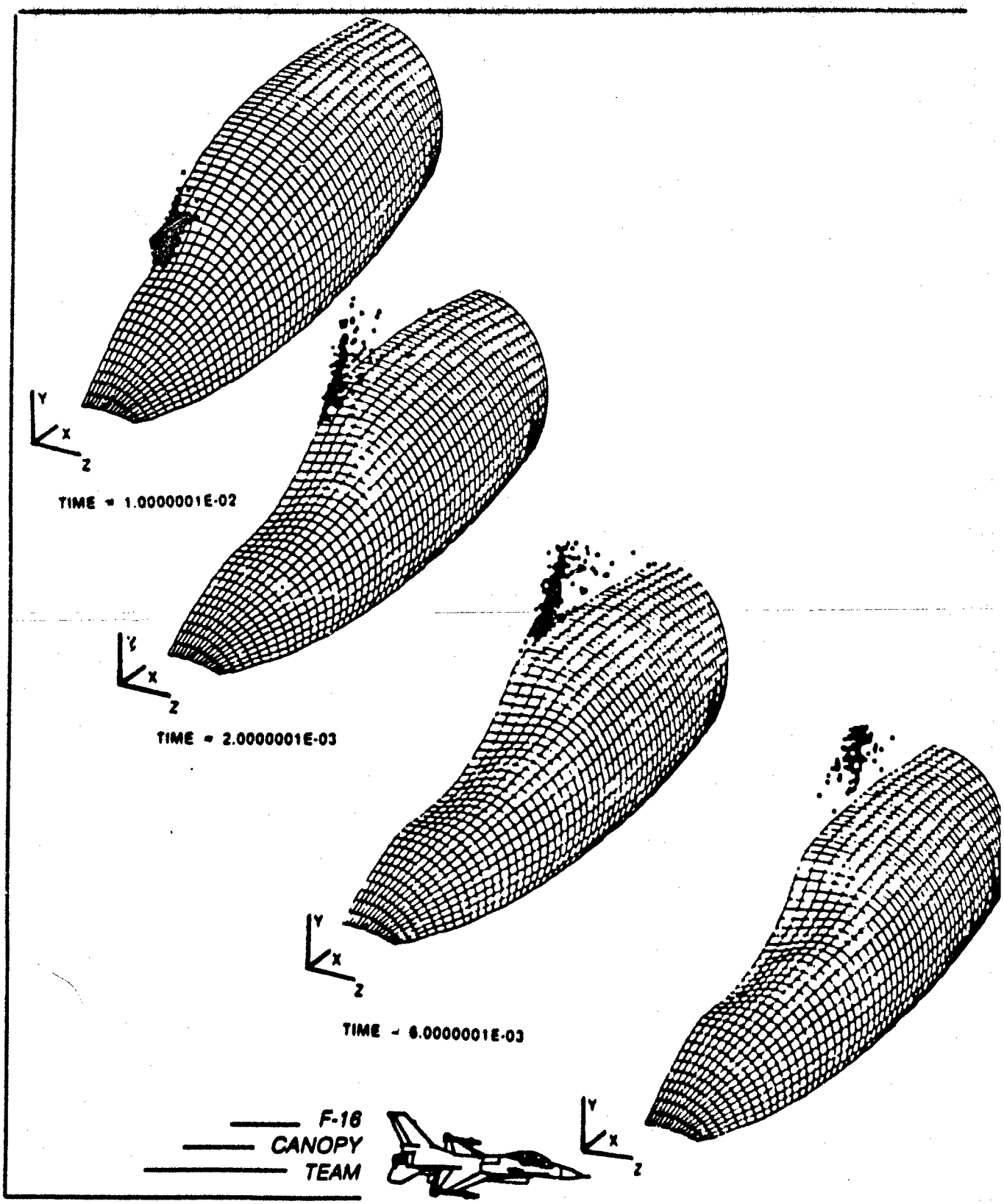


Figure 2

540 KNOT CANOPY DESIGN FEATURES

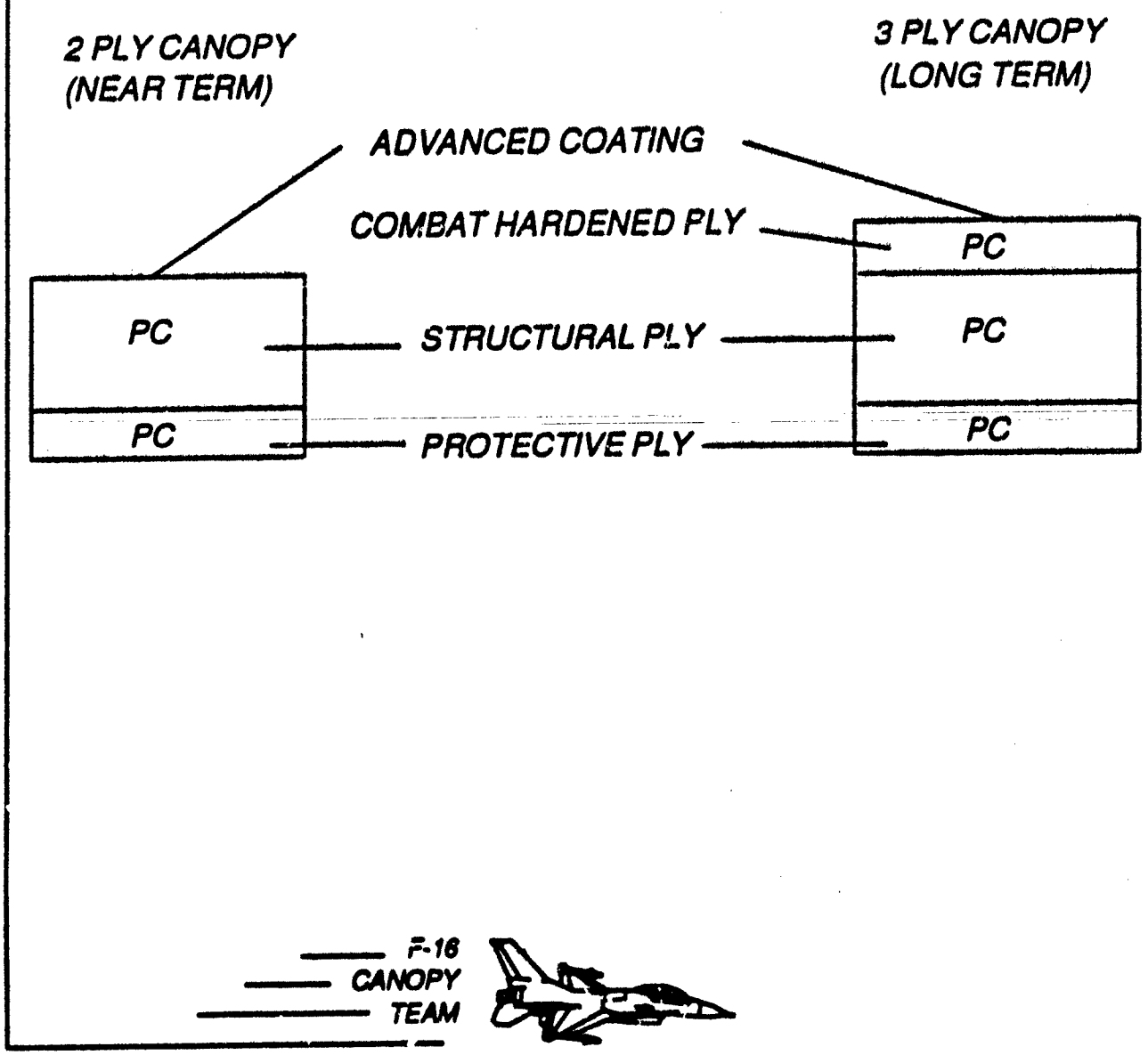


Figure 3

COMBAT HARDENED SYSTEM

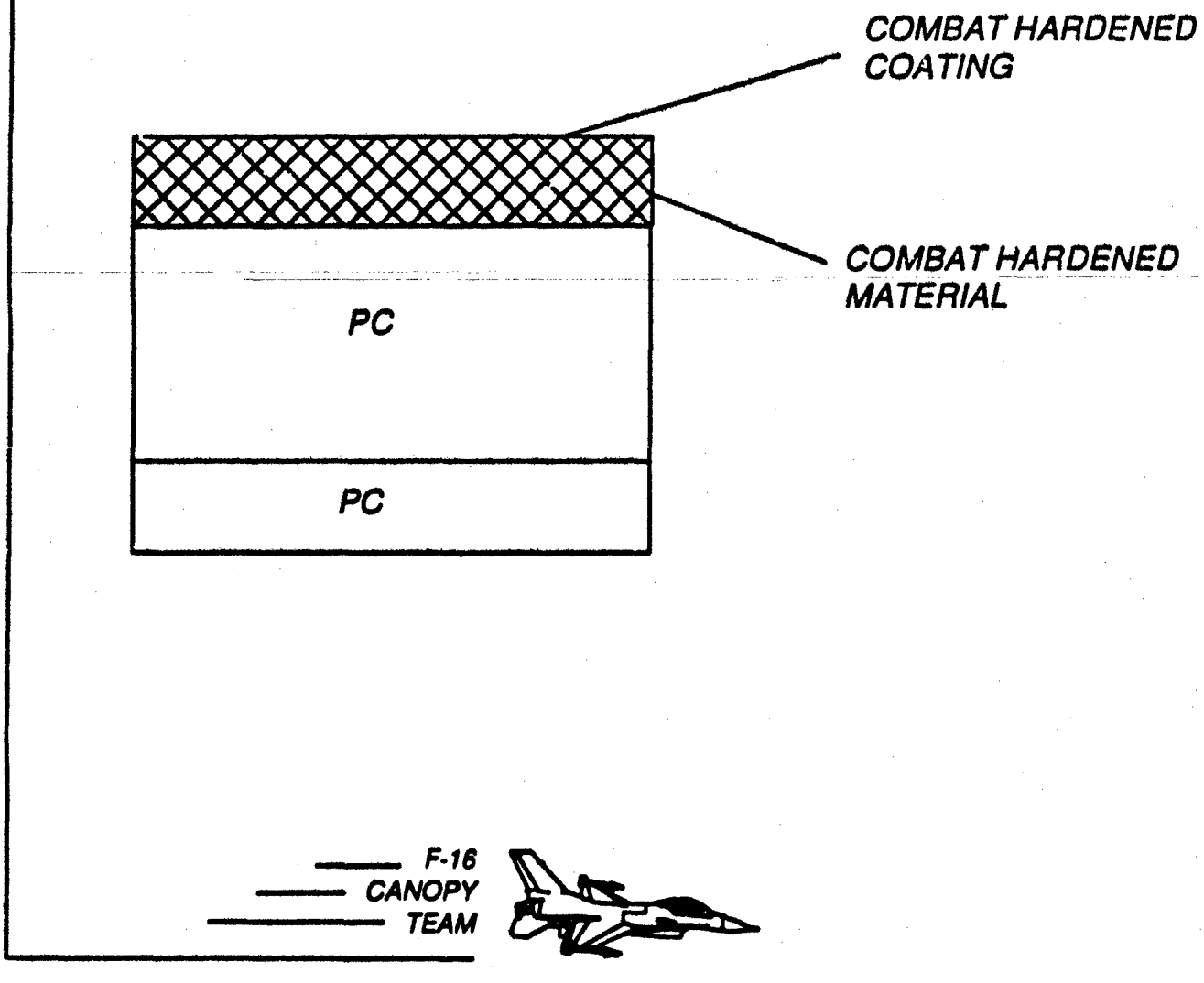


Figure 4

TOUGH HUD TECHNOLOGIES

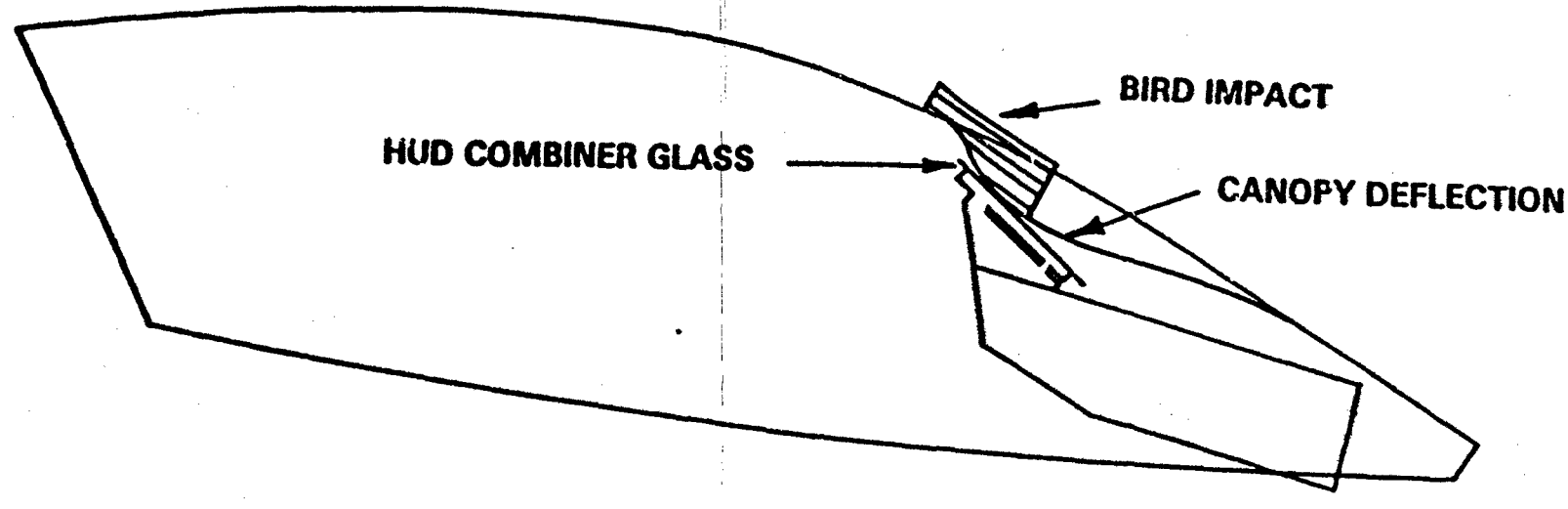
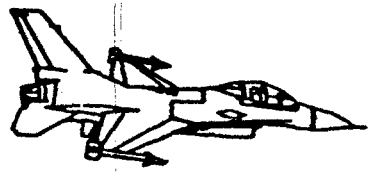


Figure 5
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F-16
CANOPY
TEAM



MISSION INTEGRATED TRANSPARENCY SYSTEM (MITS) ACCOMPLISHMENTS AND STATUS

Gordon C. Stone
Lockheed Fort Worth Company

MISSION INTEGRATED TRANSPARENCY SYSTEM (MITS) ACCOMPLISHMENTS AND STATUS

Gordon C. Stone, PHD

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LOCKHEED
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Fort Worth, Texas 76101, U.S.A.

Abstract

Under the auspices of the MITS Program, generic transparency system (canopy) design has been concentrated on at the prime contractor and subcontractor levels to identify, quantify and prioritize all pertinent design requirements. These requirements have been pursued by applying the most advanced technologies and analysis techniques while utilizing the most advanced materials.

Four MITS detailed designs have been generated which feature transparency systems which encompass all pertinent mission profiles with performance levels including Radar Cross Section reduction, birdstrike protection, aeroheating for Mach 2.5, four-year service life, four-hour change-out by two technicians and laser protection.

MITS has also facilitated the development of the latest multi-functional outer surface coatings that will be demonstrated on flying F-16 shaped advanced laminates and by full-scale ground testing.

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1.0 INTRODUCTION

The objective of the MITS Program is to utilize a generic approach to design, fabricate, and experimentally demonstrate (ground test) a Mission Integrated Transparency System for advanced tactical aircraft to operate in the last half of this decade.

In order to get a quick glimpse of the essence of the MITS Program, one should pursue the definition of "Mission Integrated Transparency System" The key to this definition lies in the following five words in the objective:

(a) Transparency System

(b) Generic

(c) Mission Integrated

First, what is a "Transparency System"? In the prime contractor community, this part of the airplane has historically been called the canopy. However, in this case, the canopy is not all that is being considered. Rather, all of those parts that either touch, attach, to, or coordinate with the canopy such as the cockpit side structure, the ejection seat, the head-up display, etc., have been included with the canopy. Thus, the term "Transparency System" is used to represent this combination of components.

Next, expanding "Transparency System" to "Generic Transparency System" broadens the emphasis to a transparency system not related directly to that of any existing tactical aircraft or any future aircraft design on the "Drawing Board" or, rather,in any computer aided design data base.

In order to design this "Generic Transparency System", one must begin with an airplane mission. Since the MITS Program is not related directly to any particular airplane, representative missions were fashioned from the different types of existing aircraft missions.

At some point in this generic design process all pertinent design goals or requirements are identified. The MITS Program began with a broad set of goals and emphasized particular goals by utilizing the representative missions. These mission-emphasized requirement goals have been pursued and traded by integrating the latest technologies into this generic transparency system design. Thus, a "Mission Integrated Transparency System" has been developed.

The theme of "Mission Integrated Transparency System" development has been expanded upon in Sections 2, 3, and 4 of this paper. We begin with a summary of mission generation and progress to descriptions of requirements evaluation, trade studies, selection of forebodies, and preliminary designs. We then report

the coupon testing used to evaluate the preliminary designs and describe detail designs. Section 5 describes an add-on effort to the MITS Program that focuses on two particular requirement goals and a specific airplane configuration. Section 6 summarizes the accomplishments and discusses payoffs of the MITS Program.

The Windshield Program Office (WL/FIVR) of the Flight Dynamics Directorate, Wright Laboratory (WL), sponsors the MITS Program. Mr. James L. Terry (WL/FIVR) is the Air Force Program Manager. General Dynamics Fort Worth Division (GD/FW), now known as Lockheed Fort Worth Company (LFWC), is the prime contractor with the author serving as Program Manager at LFWC.

The transparency manufacturers have been thoroughly involved since the outset of the MITS Program. Initial work was competed to the "Major 5" companies (Texstar, PPG, Sierracin, Pilkington and Loral) in the Fall of 1988. PPG and Sierracin were chosen in January 1989 to participate in Phases 1 and 2. Phase 3 called for Single-Source procuring a PPG Windshield, a Sierracin Canopy and a Pilkington Aft Fairing for the Phase 3 Demonstrator. Concurrently, the existing "Major 4" companies (Pilkington, Sierracin, PPG and Texstar) were chosen to participate in the Advanced Canopy Coatings Project. The MITS Program schedule is shown in Figure 1-1. The original MITS Program Plan is presented in Reference 1.

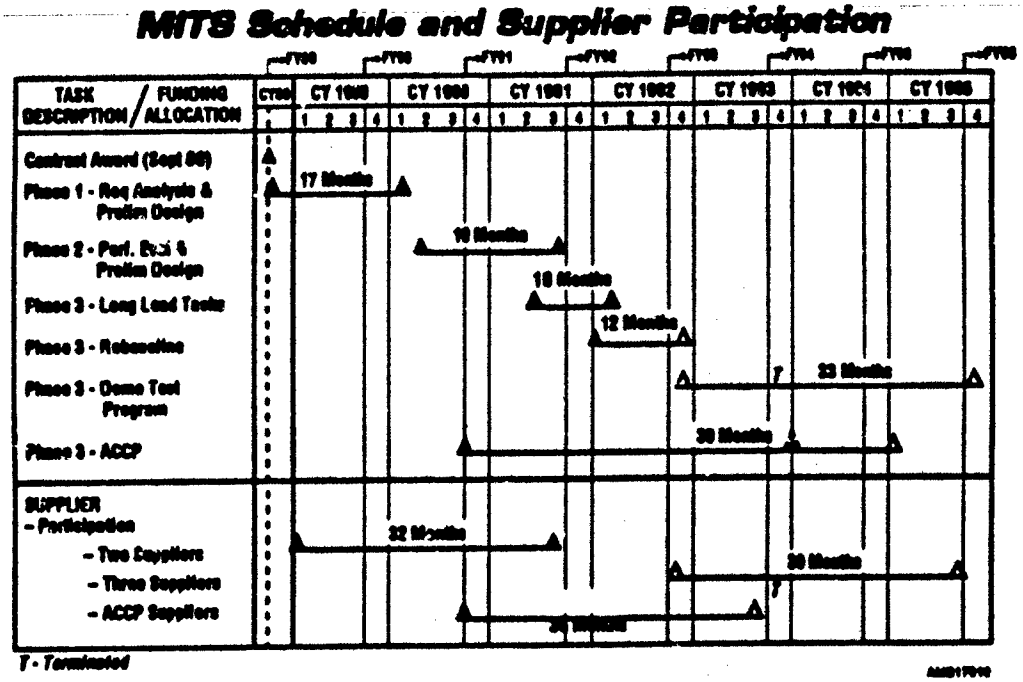


Figure 1-1 MITS Schedule and Supplier Participation

2.0 PHASE 1 - MISSIONS, REQUIREMENTS AND PRELIMINARY DESIGN

2.1 Missions

It is expected that the next generation of advanced tactical aircraft will require transparency systems far more complex than those that currently exist. The missions, functions and the flight profiles of these aircraft, coupled with the advanced threats that they are likely to encounter, have generated a large quantity of critical and conflicting transparency system design, performance and supportability requirements. Sets of these transparency system design and performance requirements have been postulated, categorized, and provided as baseline data by the MITS Air Force Customer. The requirements were grouped into five categories: natural hazards, crew/machine interface, combat hazards, supportability and fuselage integration. They were incorporated, along with current requirements, as design drivers for the purpose of this mission development. For more details see References 2, 3, and 4.

As mentioned in Section 1.0, in order to design a "Mission Integrated Transparency System" one must begin with an airplane mission. WL/FIVR, the MITS Air Force customer, provided GD/FW two mission profiles: a typical ground attack (GA) profile and a typical air superiority (AS) profile. GD/FW was then required to generate a multirole (MR) mission profile using these two mission profiles and incorporating all other pertinent considerations. The GA and AS mission profiles are shown in Figure 2.1-1 and 2.1-2, respectively.

A mission definition approach was employed to derive the mission related requirements to aid in establishing the MITS design goals. To perform a mission definition, two primary considerations are necessary: threat definition and mission analysis. It is necessary to develop a combat scenario and the associated threats, targets, and operational procedures to provide the background environment the mission profile will be flown in. The two primary categories closely analyzed during the mission analysis were combat hazards and natural effect hazards. Each of the additional critical categories of design parameter requirements was also reviewed for relevance to operational capabilities applications and the resultant implications.

The initial mission analysis step was to perform a systemized mission review and operational applications analysis of the typical or baseline AS and GA mission profiles provided by WL/FIVR. Establishing mission requirements and mission objectives for each of the baseline profiles was integral to interfacing with a preliminary MR profile. Included in the mission requirements were such operational environmental considerations as: Day/Night Operations, All Weather Operations, Penetration Altitudes/Speeds, Tactics Employment, and operational procedures (Rules of Engagement, Safe Passage Procedures, etc).

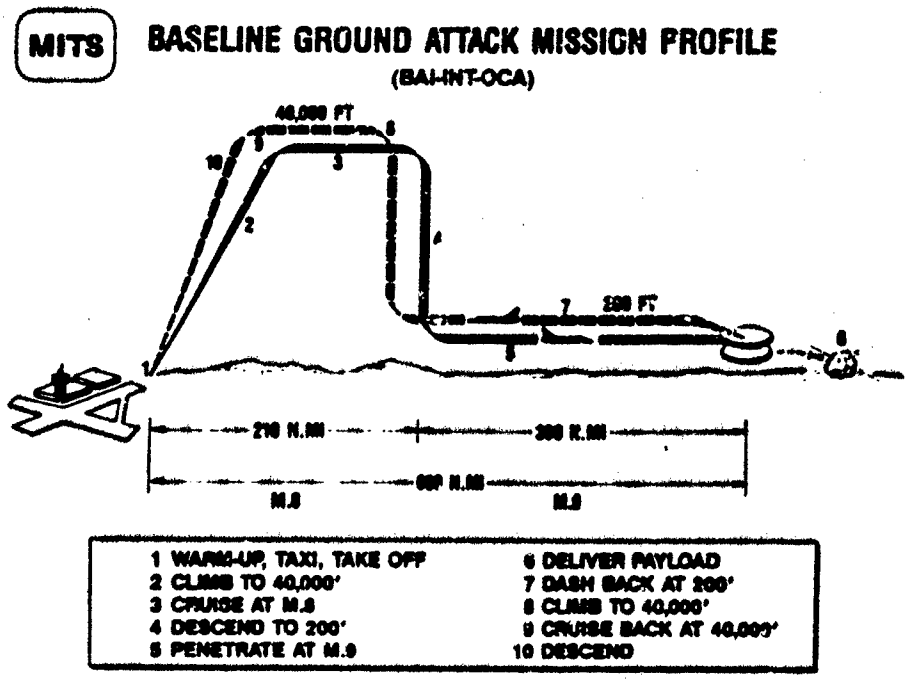


Figure 2.1-1 Ground Attack Mission Profile

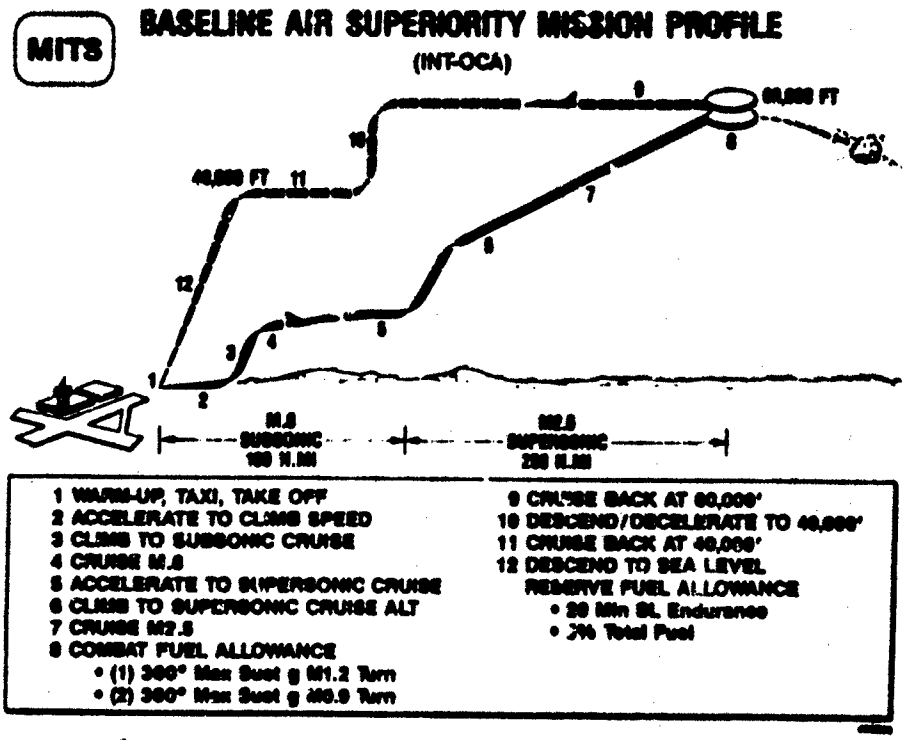


Figure 2.1-2 Air Superiority Mission Profile

Also included in the analysis are mission activities that are not specifically addressed but that are integral or inherent to the mission profiles. Examples are: Formation Flying and Rendezvous, Aerial Refueling, and Approaches and Landings.

2.2 Multirole Mission

Based on threat definition and postulated operational background, the development of the MR mission began with a description of the activity envisioned as taking place during this type mission and of a prioritized set of design requirement parameters. Then an analysis of the probable possibilities of mission combinations to form a MR mission plus a description of the likelihood of the combinations was performed. Transparency operational conditions and interactions, the probabilities of the projected threat density/detection distributions, and the anticipated expected encounters were also identified. The MR mission profile is shown in Figure 2.2-1. A leg-by-leg detail description of the MR mission and additional information are contained in References 2, 3, and 4.

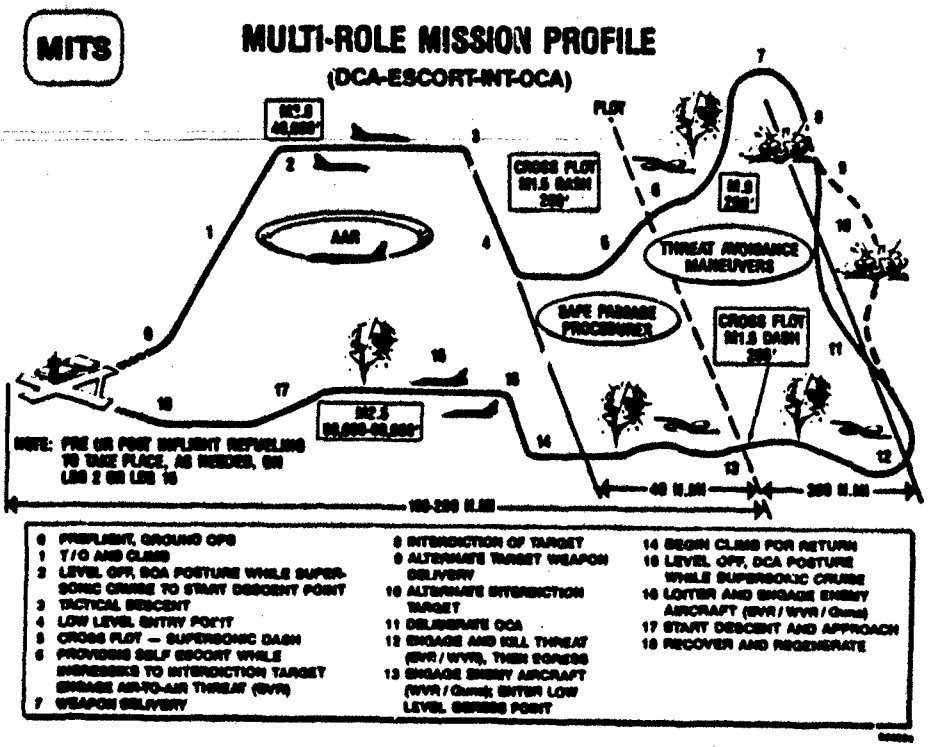


Figure 2.2-1 Multi-Role Mission Profile

2.3 MITS Requirements Evaluation

As was implied in Section 2.2, mission development and operational factors led to mission driver requirements. Mission definition is not complete without cognizance of mission impact on requirements. Therefore, mission development and evaluation of requirements go "hand-in-hand". In order to achieve clarity, however, mission development and evaluation of requirements have been presented sequentially. The process of mission definition, operational factors and mission driver requirements is depicted in Figure 2.3-1.

MISSION DEVELOPMENT AND OPERATIONAL FACTORS LEAD TO MISSION DRIVER REQUIREMENTS

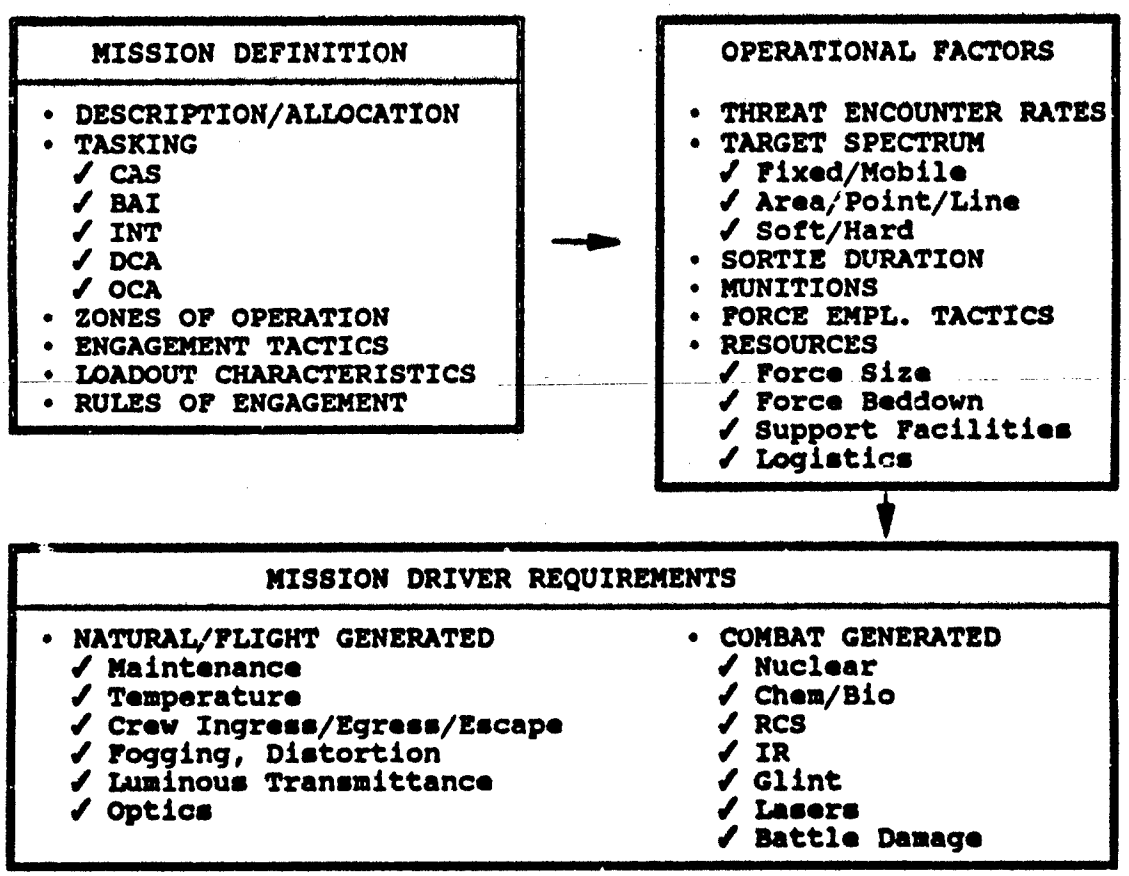


Figure 2.3-1 Mission Definition / Driver Sequence

The purpose of the MITS requirements evaluation was for GD/FW and the Government to establish a common understanding of the MITS systems requirements before doing preliminary design work. GD/FW evaluated these requirements to determine not only relevance, but attainability in the 1995 to 2000+ time frame.

Eighty-six (86) areas of concern were addressed and were grouped into five (5) categories: Natural Hazards, Combat Hazards, Supportability, Crew-Machine Interface, and Fuselage Integration. As was recognized early in this process, the missions determine key driver requirements. Several categories of the 86 areas of concern have substantial impact on mission success as is shown in Figure 2.3-2. Of the areas of concern, thirty (30) were initially designated as design parameters by the (WL/FIVR) customer.

THE MISSIONS DETERMINE KEY DRIVER REQUIREMENTS

86 PARAMETERS WERE ADDRESSED, BUT SEVERAL CATEGORIES HAVE SUBSTANTIAL IMPACT ON MISSION SUCCESS

MITS REQUIREMENT AREAS	MISSION PROFILE		
	AIR SUPERIORITY	GROUND ATTACK	MULTI-ROLE
GROUND OPERATIONS (Solvents, Interchangeability, Change-Out Time, Thermal)	✓	✓	✓
SERVICE LIFE	✓	✓	✓
OPTICAL QUALITIES	✓	✓	✓
SIGNATURE	✓		✓
HUD INTEGRATION	✓		✓
AERODYNAMICS	✓		✓
BIRDSTRIKE		✓	✓
LASER PROTECTION		✓	✓
BALLISTIC PROTECTION		✓	✓
NIGHT/ALL WEATHER VISION		✓	✓

Figure 2.3-2 Key Driver Requirements Determination

The design parameters were prioritized based upon mission requirements. AS and GA Mission requirement priorities were established in Reference 5 by the Government. MR mission requirement priorities were established by GD/FW and submitted formally as Reference 6. Also, GD/FW re-prioritized goals for the AS and GA Missions. The final prioritized goals for all three missions are summarized in References 3 and 4 in addition to Reference 6.

The priority values ranged from one to three with the priority indicating the importance to the Air Force in accomplishing the mission. Priority one is most important while priority three is least important.

The results were presented, discussed and approved at a Systems Requirements Review (SRR) in April, 1989. SRR attendees represented the Windshield Systems Program Office at WL, various technical disciplines at WL, several Air Logistics Centers, the three MITS vendors: Sierracin, PPG,and Lockheed Aeronautical Systems Company (LASC) and GD/FW in various technical areas. The objective of the SRR was adhered to closely for the Government to ascertain the adequacy of contractor efforts in defining system requirements as is stated in MIL-STD-1521A (USAF). Traditional system requirements are rather hard and fast, not to be compromised later in the program.

In contrast to traditional system requirements, MITS requirements were set at a level to optimize each particular parameter, neglecting conflicting requirements. The requirement levels were allowed some flexibility and were subject to change as a result of trade studies. The desired result, with this non-traditional approach, was to encourage the contractor and subcontractors to "reach" for technical innovation rather than settle for easily obtainable goals. GD/FW utilized diverse inputs from "in-house" technical specialties, mission analyses, design groups, materials groups and support groups as well as the transparency manufacturers and LASC.

The prioritized requirement goals are represented for one area of concern in Figure 2.3-3 which provides an overview of natural hazards parameters. The driver category (i.e. thermal, weather, solvent, etc) is broken down into the quantified parameters with the priorities for each of these parameters listed by mission. The additional parameters listed at the bottom of Figure 2.3-3 were considered but were not rigorously prioritized. Similar overviews of the remaining requirements categories (Combat Hazards, Supportability, Crew-Machine Interface and Fuselage Integration) are contained in References 3 and 7.

Ten common priority "one" design drivers emerged from the requirements evaluation for each of the three missions. These common priority "one" drivers are shown in Figure 2.3-4. These ten requirements became the primary drivers for the MITS designs.

The ten top-priority drivers were determined to have both competing and complementary performance parameters and are broken down into typical performance considerations as shown in Figure 2.3-5. Due to the importance placed on service life, some lower priority requirements were included.

For a thorough description of each design parameter, including value or level, priority by mission and acceptance criteria, please refer to References 3 and 6.

OVERVIEW OF NATURAL HAZARDS PARAMETERS

DRIVER CATEGORY	QUANTIFIED PARAMETERS	PRIORITIES		
		MULTIROLE	AIR SUPERIORITY	GROUND ATTACK
THERMAL	Aeroheating Flightline/Warehouse	1	1	1
WEATHER	UV/Moisture Rain, Hail, Ice Rain Removal Anti-Ice, Defog Fungus, Salt Spray	3	3	3
SOLVENT	Cleaning Solutions/ Materials	1	1	1
BIRDSTRIKE	Birdstrike	1	3	1
ABRASION	Cleaning Abrasion Rain, Hail, Ice, Salt Spray	2	2	2
GENERAL ENVIRONMENT	Lightning	3	3	3

ADDITIONAL PARAMETERS

✓ Vibration	✓ Paint/Solvents	✓ Sealants	✓ Acid Rain
✓ Static Charge	✓ Sunlight	✓ Fuel Spill	✓ De-Ice (ground)
✓ Atmospheric Pollutants	✓ Snow	✓ Runway Debris	

Figure 2.3-3 Overview of Natural Hazards Parameters

MANY COMMON #1 DRIVERS RESULTED

FOR EACH MISSION (MULTIROLE, AIR SUPERIORITY, GROUND ATTACK) THESE DRIVER CATEGORIES WERE FIRST PRIORITY:

✓ Service Life	✓ Structural Integrity
✓ Thermal	✓ Electrical Compatibility
✓ Optical	✓ Interchangeability
✓ Signature	✓ Change-Out Time
✓ Solvent Resistance	✓ HUD Integration

• THESE REQUIREMENTS WERE THE PRIMARY DESIGN DRIVERS

Figure 2.3-4 Common Priority #1 Drivers

THE TOP-PRIORITY DRIVERS HAVE BOTH COMPETING AND COMPLIMENTARY PERFORMANCE PARAMETERS

DRIVER CATEGORY	TYPICAL PERFORMANCE CONSIDERATIONS
SERVICE LIFE	Rain, Ice Crystals, Abrasion, UV, Moisture *
OPTICAL QUALITIES	Transmittance, Haze, Ang. Deviation, Distortion, Binoc. Disparity
SIGNATURE REDUCTION	RCS, Infrared, Visual
SOLVENT RESISTANCE	Cleaning Solutions and Materials, Fuel, Paint, Sealants
THERMAL CAPABILITIES	Aeroheating, Flightline, Warehouse Temperatures
ELECTRICAL COMPATIBILITY	Static Charge Drain, Continuous Conductivity to Airframe
STRUCTURAL INTEGRITY	Pressure, Temperature Cycles
INTERCHANGEABILITY	No Rerigging of Support and Clamping Devices
CHANGE-OUT TIME	Two Crew/Four Hours
HUD INTEGRATION	Wide Field-of-View

* Some lower priority requirements were included in service life due to its importance.

Figure 2.3-5 Compatibility Features of Top 10 Drivers

2.4 Trade Studies

As he synthesizes the design, the designer is ever mindful of the requirements that his final creation must meet. He is constantly challenged by what seems to be conflicting demands or requirements by the customer.

To fully satisfy one requirement, other requirements must be compromised. A final solution attempts to meet as many of the requirements as possible. Although it is almost impossible to fully meet all requirements, it is possible to do well with respect to most of the requirements through a series of compromises and trade-offs. In the MITS Program, the trade-off process was a key element of the design process and is documented in References 3 and 8.

Traditionally, the greatest emphasis on trade studies occurs in the preliminary and final design phases. However, many trade studies, at least at the conceptual level, are performed while synthesizing a preliminary design. The MITS Air Force customer directed the use of the three aircraft configurations and missions as described in Section 2.3. The only information known about these configurations was respect to mission to be flown. Nothing was known of the overall shape, number of engines, number of crew, inlets, avionics etc. Quantitative trade-off data cannot be generated without specific designs on which to perform analysis. Therefore, the trade-offs addressed in References 3 and 8, were primarily constrained to the conceptual or qualitative level. However, through an informal cooperative effort of both MITS Program Managers, (Air Force and GD/FW) the trade study document has been revised several times to reflect detail design trade-offs.

Three baseline assumptions were included in the basis for the trade study:

1) Only single seat, fighter-type aircraft were considered. Two seat aircraft were not included.

2) Through-the-canopy ejection was not considered.

3) Any ejection seat considered for reference in the Trade Study Task has capabilities comparable to the McDonnell-Douglas Aces II seat.

This trade study accomplishes MITS program objectives in providing a trade and assessment foundation to serve as a basis for future transparency system designs. It is complete to the extent that data was available to the GD/FW MITS Program Office in time for publication of references 3 and 8. The study provided the tools for:

a) Early development of transparency system concepts that can be evaluated through trade-offs.

b) Development of preliminary designs that can meet conflicting requirements.

c) Assessing the degree of compromise required for any technology to satisfy priority drivers.

2.5 Selection of Forebodies

Rather than exist in isolation, a transparency system is integrated with an air vehicle. Therefore, some knowledge of the baseline vehicle must exist. The characteristic design features of the vehicle, including the transparency system, should function in a manner to serve the stated mission of the vehicle.

To begin the Preliminary Design task, representative air vehicle baselines suitable for conducting subsequent technology evaluation were required. The mission profiles dictated a fixed wing fighter type aircraft. The baseline aircraft was selected using the following criteria:

a) The aircraft must currently be in either the United States Air Force or Navy inventory and should be expected to be in the inventory in the 1995-2000 timeframe.

b) The aircraft must currently be used for the subject mission.

c) The MITS program was committed to develop unclassified designs per customer guidelines,

Aircraft considered as baselines were the: F-15, F-111, F-14, F-16, F-18, F-4, A-7, AV-8B, and A-10. Noticeably absent from this list are several advanced tactical aircraft currently under development. None of these aircraft were selected as baselines in order to satisfy criterion (c), above. Aircraft were selected as baselines for the following reasons: Use of one of the advanced aircraft as a baseline would have dictated that the design work be conducted in a classified area by appropriately briefed personnel. Conversely, by selecting current fleet aircraft, a large amount of baseline data would be available. Readily obtainable baseline data was especially important in analysis of reliability, maintainability, and supportability parameters. Selection of the AS, GA and MR Forebodies is covered in Reference 3.

The McDonnell-Douglas F-15 "Strike Eagle" was selected as a baseline for the AS Mission. This aircraft is currently the premier AS fighter for the Air Force. The F-15 is capable of a Mach 2.5 cruise (leg 7 of the AS Mission Profile). However, what counts most in an AS fighter is not the highest speed but the capacity to maneuver while withstanding the maximum loads. Maneuverability is expressed in climbing performance, acceleration capacity and turning speed. A thrust-to-weight ratio of greater than one gives the F-15 the performance necessary for this type of mission.

The F-15 forebody design, as with the entire aircraft, is derived from its primary mission. The crew station configuration, single place or two-place tandem, and the allowance for a large radar dish form the size envelope for the forebody. High speed flight requirements shape the forebody to achieve acceptable transonic/supersonic wave drag characteristics.

The GD/FW F-16 "Fighting Falcon" was selected as a baseline for the MR Mission. This aircraft currently serves as the "low" component of the USAF's high-low mix. During any conflict, the F-16 would be expected to fly both AS and GA missions. A thrust-to-weight ratio greater than one gives the F-16 the performance necessary for this type of mission.

The F-16 forebody design, as with the entire aircraft, is derived from it's primary mission. The crew station configuration, single place or two-place tandem, and moderate size radar dish form the size envelope for the forebody. High speed flight requirements shape the forebody to achieve acceptable transonic/supersonic wave drag characteristics. A high priority on field of view results in a one piece wrap-around canopy.

The McDonnell-Douglas AV-8B "Harrier" was selected as a baseline for the GA Mission. The Harrier is a vertical takeoff and land (VTOL) aircraft. Because of the unique VTOL capabilities, the aircraft can be based close to the front lines, away from vulnerable airfields. The Harrier is a versatile aircraft that can operate at supersonic speeds and engage other fighter aircraft in combat in addition to its GA capabilities. Such capabilities will

ensure continued usage of this vehicle in the U.S. Navy (Marine) inventory.

The AV-8B forebody design, as with the entire aircraft, is derived from it's primary mission. The crew station configuration, single place or two-place tandem, and lack of radar form the size envelope for the forebody. Visual ground target acquisition requires that the forebody be shaped to allow good downlook, both over-the-side as well as over-the-nose.

As a baseline assumption the inlets and wings were neglected for the purpose of the MITS program. Inclusion of the inlets and wings would impact aerodynamic analysis results to the extent that changes to the transparency would have minimal effects.

2.6 Preliminary Designs

In order to avoid repetitious coverage of transparency system features, a brief description of each of the three preliminary designs, along with pertinent concepts considered, will be given in this section. More detailed coverage is provided in Section 3.2 which covers the three MITS detailed designs. The three preliminary designs are described in detail in References 3,4 and 9.

Multi-Role Mission Configuration

The MR mission calls for both supersonic cruise at high altitude and supersonic dash "on the deck." Air-to-air engagements are expected as well as weapons delivery to ground targets. The full range of combat hazards are expected (ballistics, laser, nuclear, chem/bio, and radar/infrared/visual detection) and a severe combination of natural hazards (birdstrike, rain, ice, temperature, etc.) are likely.

The MR transparency design, which utilizes a modified F-16 Forebody, (Section 2.5) features an aft-pivot windshield/canopy and a fixed aft canopy section. The windshield and forward canopy are separate sections attached by an integral bowframe. This facilitates producibility, improves optical features, and allows weight reduction by tailoring the transparency laminate thicknesses to the expected threat severity (temperature and birdstrike, in particular) for the respective transparency sections.

The windshield and canopy sections have an outer two-ply chemically-tempered glass laminate separated from polycarbonate (PC) structural plies by a gap filled with nitrogen or filtered air. This gap is subjected to the same pressure as the cockpit. The glass laminate plies are bonded with silicone and the PC plies are bonded with urethane.

The outer surface of the glass is coated with a conducting film of either indium tin oxide (ITO) or an ITO/ATO (antimony tin oxide) combination. The sheet resistance of this conducting film will be approximately 10 ohms/square (a good compromise between

microwave reflectivity and optical (visual) transmission).

A second conducting film layer is presently designed at either the outer surface of the second glass ply or the outermost PC surface. Plans were made to test for the best location and effectiveness of this film during the detail design phase of the program. Existing knowledge indicated that the silicone interlayer would likely degrade the performance of this film if it is placed between the glass plies.

The total PC thickness was between 0.5 and 0.75 inches in the windshield section and between 0.375 and 0.6 inches in the canopy section. The aft section was initially designed to be an outer ply of 0.12 inch chemically tempered glass bonded to a 0.188 inch thick PC ply with urethane.

An optional MR laminate design was maintained in order to provide a window of opportunity to infuse developing technologies, particularly materials, into the program as they became available. This optional laminate called for a high temperature outer ply constructed from one of several developing materials: (a) "HTPC" (high-temperature PC) such as that being developed by Mobay, (b) "S240", under development by Sierracin/Sylmar, (c) Acriview 590 from Pilkington (formerly GAC590 from Loral Defense Systems), (d) AEC from Dow Chemical, (e) Durel from Celanese and (f) PPC 4701 from General Electric.

The outer surface coating system was expected to be that developed under the WL/MLPJ Electro Optics II Contract with Loral Defense Systems, or a metallic film with a protective topcoat.

The total length is slightly over 11 feet and the total area is approximately 39 square feet. The shape is "Mod 2," i.e., the second modification to the starting generic design. Two other modifications would be made before reaching the final shape.

Air Superiority Mission Configuration

The strongest performance drivers for the AS mission are high temperature capability, excellent aerodynamic shaping, low signature characteristics and quality optics. Lower emphasis is placed, for this mission, on birdstrike resistance and hardening to conventional (ballistic) and laser weapons.

The AS design which utilizes a modified F-15 forebody, features a fixed bowframe with separate acting windshield and canopy sections. The windshield pivots forward for ease of maintenance access. The canopy assembly, which in itself has an integral bowframe, pivots aft. This design features ease of manufacturability for the glass laminates, minimization of weight of the individual pieces for ease of change-out and handling, successful compromises between optics, signature, and producibility, and tailorability of laminate thickness (and thus weight) in the three sections. As in the case of the MR design,

excellent service life is expected.

The AS design is similar to that for the MR design in that a glass laminate is used outboard of the PC structural plies and is separated from them by a pressurized nitrogen or clean-air gap. The silver or gold (or other metallic alloy) film is retained as optional for this configuration and the PC thickness is downscaled to reduce weight, since the priority for birdstrike protection is reduced for this mission. The aft transparency section featured a 0.188 inch thick PC ply, but it is anticipated the PC ply can be deleted since the glass laminate can carry the aerodynamic and pressurization structural loads.

The dimensional characteristics of the AS "Mod 2" shape includes an overall length of 13 feet and a total surface area of over 52 square feet.

Ground Attack Mission Configuration

Although the aerodynamic heating induced-temperatures are the least severe for this mission, the requirements for protection from birdstrike, conventional and laser weapons, and reduced signature are high.

The overall GA configuration utilizes a modified AV-8B Harrier forebody. The design features a single-piece canopy and subframe which pivots forward for excellent ingress/egress. For emergency egress/escape, the canopy/frame pivots aft.

The recommended laminates for the GA transparency system featured a thicker structural laminate to provide greater birdstrike protection for this low-level mission. Slightly improved ballistic resistance is also offered, although a secondary ballistic shield will be required. The outer layer of the design was targeted to consist of Loral's GAC 590 urethane coated with the EO II coating system to protect against combat and natural hazards.

It should be noted that although transparent ballistic shields are featured in this design, Kevlar-type materials with cut outs for visual sighting were also considered. A typical shield configuration of this type would be 0.1 inch aluminum facing with approximately 5.5 lb/ft^2 Kevlar composite.

The Mod 2 design for the GA transparency has an overall length of nearly 12 feet and a forward down-look capability of 19.6°: the best for any of the mission-driven designs.

A Technical Information Plan, a Logistics Support Analysis Plan and a System Safety Program Plan were written for the three MITS preliminary designs. These plans are covered in detail in References 10, 11, and 12, respectively.

3.0 PHASE 2 - PERFORMANCE EVALUATION AND DETAILED DESIGN

3.1 Evaluation of Preliminary Designs

The MITS design and performance requirements were quantified and prioritized as described in Section 2.3. The capabilities of each preliminary design were assessed as to whether it met or fell short of the MITS design and performance requirements. In some cases a quantitative assessment (percentage of requirement) was made. The assessments were typically based on test data (MITS-generated, subcontractor data, or open literature), analysis, experience, or similarity to another requirement. To illustrate, a small portion of the assessment results are shown in Table 3.1-1. This table covers the assessment of the capabilities of the MITS Preliminary Designs to satisfy just 10 of the 86 quantified and prioritized performance requirements.

Throughout the detailed design development of Phase 2, these assessments were refined in many cases by the various technical analyzes (birdstrike, aerodynamic heating, aerodynamic drag, optics, etc). Concurrently, these refinements were verified and/or enhanced, where applicable, by materials coupon testing. The materials coupon testing effort and results are described on Section 3.3. The preliminary test plan for the materials coupon testing is covered in detail in Reference 7.

The complete assessment results as well as a summary by mission of all of the prioritized design driver categories and the performance requirements are contained in References 3 and 9.

3.2 MITS Detailed Designs

Multirole Design

The MR transparency system consists of three main components: a windshield section, a canopy section, and a fixed aft fairing. Separate components enhance supportability and manufacturability. The laminates for the windshield and canopy feature two glass outer plies, exterior ITO coatings, and a pressurized air-gap between the glass plies and PC structural plies. The canopy frame assembly, featuring an aft pivot for ingress/egress and emergency escape, integrates windshield/canopy, latch hardware and emergency system components with the aircraft forebody. Dry seals are used for all transparency-to-structure interfaces. Common length fasteners are utilized along transparency edge attachment boundaries.

The MITS program involves not only transparency design, but the design of forward fuselage structure as well. The structural integration of the MR transparency/fuselage design involved concurrent engineering in areas such as sill plane and parting plane locations, crew station geometry and equipment, and latch hardware and resultant support structure.

Table 3.1-1 Capabilities Assessment of the MITS Designs

Requirement	Mission			Basis for Assessment
	MR	AS	GA	
1. Aerodynamic Heating	M	M	E	A
2. Fltline/Warehse Temp.	M	M	M	E,T
3. UV	M	M	M	E,T
4. Moisture	M	M	M	E,T
5. Rain	M	M	D	E
6. Hail	M	M	D	E
7. Ice	M	M	D	E
8. Rain Removal	M	M	M	E,S
9. Anti-Ice/De-Ice	M	M	M	S,E
10. Defog	M	M	M	S,E
↓	↓	↓	↓	↓
86. Latch/Lock/Seal	M	M	M	A

Capability:
E = Exceeds Requirement
M = Meets
F = Falls Short
% = Estimated % of Requirement
D = Deferred for Phase 2 Assessment

Basis:
T = Test
A = Analysis
E = Estimate, Experience
S = Similarity

Notes and Comments

Comments and explanatory notes are given below in the order of the requirements listed.

5-7. Assessment of weathering properties for the GA design was deferred pending selection of the coating system.

8-10. Methods to accomplish these requirements were evaluated in Phase 2. Jet blast will not be used for GA.

Figure 3.2-1 presents a cross-section of the MR fuselage roughly where the pilot's feet are located to show typical structure. Conventional frame and longeron construction was used for the MITS MR fighter.

The transparency system design heavily involved the most important component of an aircraft weapon system: the pilot. Consideration was given to vision obscuration due to frame position and Head Up Display (HUD) support structure. Mobility within the cockpit influenced helmet/transparency clearance, side console space, and ejection seat geometry. The location of structure did not interfere with the pilot's ability to perform a mission or the ability to escape during an emergency.

Figure 3.2-2 shows vision through the MR transparency as seen from the design eye position. The forward bowframe placement was driven to maximize forward uplook angle. The bowframe masks the transition from a single curvature windshield to a compound curvature canopy. A high temperature fiberglass straps required to secure the glass plies to the edgeframe cause a slight increase in vision obscuration.

The canopy frame assembly is the key to structural integration. The frame operates structurally, transmitting flight and internal pressure loads into the fuselage. The frame must also withstand loads caused by bird impact and emergency escape. The assembly operates reliably during ingress and egress. A side view of the canopy frame is shown in Figure 3.2-3. Some of the major detailed parts are labeled.

The MR transparency geometry was developed using 3-D computer graphic tools. Figure 3.2-4 shows the MR transparency with a laminate summary. The transparency contour was optimized for the forebody geometry using design/analysis iterations. Assessments of performance along with design methodology resulted in integrated outer mold lines for MITS designs. Initial modifications were driven by signature reduction, yet exhibited poor optical performance. Final modifications were driven to improve optical performance while maintaining good signature performance. Other issues involved during the trade analysis were aerodynamics and human factors. The transparency measures 11.5 ft. in length, 3.2 ft. maximum width, and 44.5 sq. ft. in total wetted area.

Laminates for the windshield and canopy consist of two glass plies laminated with a silicone interlayer. A pressurized air-gap separates the inner glass ply from the PC structural layer. The glass plies carry the pressurization load. The PC plies are not pressurized; operating pressures are balanced across the PC plies by the laod-carrying glass plies. Tests indicate increased service life potential for this configuration. The aft transparent fairing consists of a glass/PC laminate with a photochromic-doped silicone interlayer.

A glass-faced laminate was selected for the MR configuration primarily due to the durability of the ITO/Glass face ply under extreme service conditions. STAPAT thermal analysis predicts maximum surface temperatures in excess of 360° F. The air gap design insulates the structural PC ply from extreme temperatures. Tests conducted during Phase 2 indicate excellent resistance to rain, solvents, and abrasion.

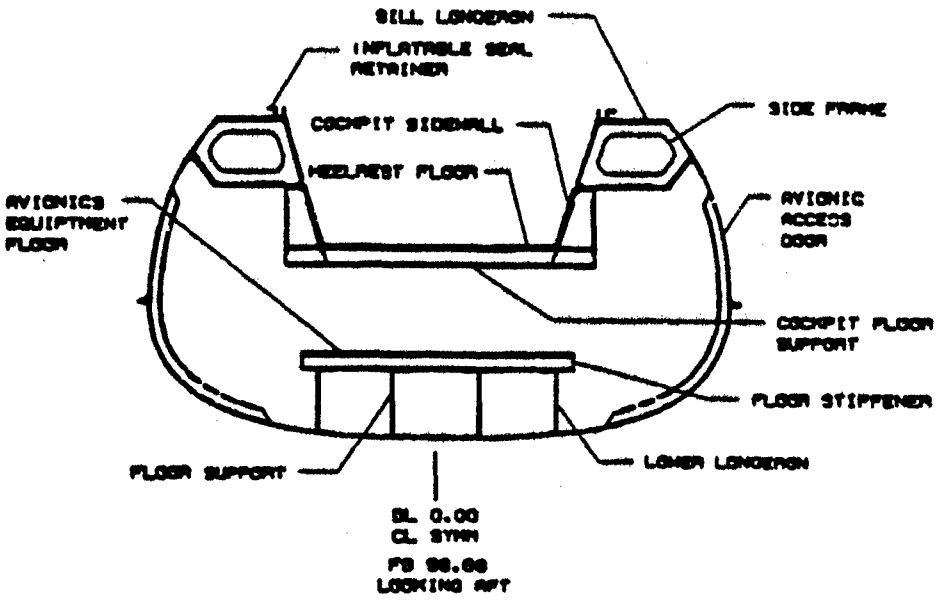


Figure 3.2-1 Cross-section Of Multirole Fuselage

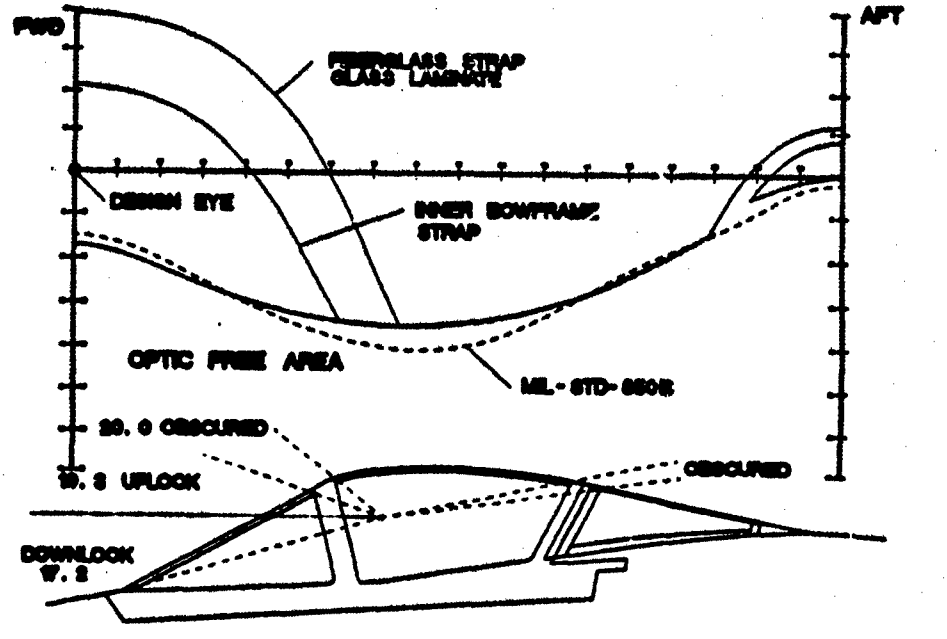


Figure 3.2-2 Vision Through The Multirole Transparency

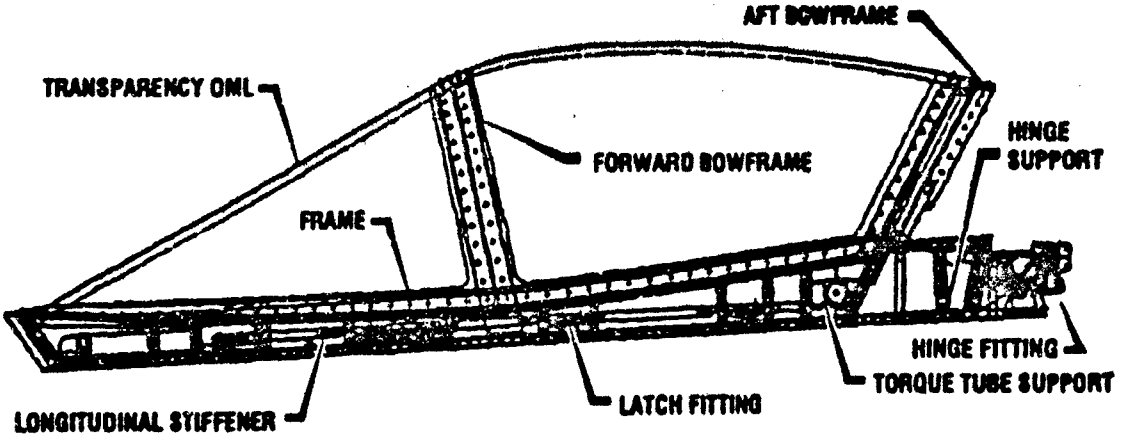


Figure 3.2-3 Multirole Canopy Frame

The MR canopy latching system consists of four rotating hooks on each side of the canopy frame operated by a push-pull linkage mechanism, and latch receptacle studs attached to fittings on the fuselage sill longeron. The linkage mechanism is operated by bellcranks on a transverse torque tube on the canopy frame. A pair of linear actuators provides the force to operate the latches as well as raise and lower the canopy assembly. A canopy latch locking mechanism is incorporated in the latching linkage near the second hook on the left-hand side.

Each hook features a curved surface of engagement to allow the canopy to be cinched down and accommodate large dimensional variations in the hook and latch receptacle positions. The hooks protrude through openings in the bottom of the canopy frame to engage the latch studs on the inboard flange of the longeron. The pilot can visually confirm latch engagement with ease.

For access to the cockpit, the canopy assembly opens by pivoting about hinges at the aft ends of the frame. The canopy rotates approximately 34° to the normal, fully-opened position. The actuation system allows the canopy to be stopped and held at any position between fully closed and fully opened.

The MR canopy actuation system consists of two linear screwjacks driven by an electromechanical gear motor through rigid driveshafts. The output end of each screwjack is attached to a bellcrank arm on the canopy torque tube. The length of the screwjack stroke is approximately 14 inches. Each screwjack is attached to an actuator support fitting mounted on a structural bulkhead at the rear of the cockpit. When the canopy is opened, the four-point support provided by the two hinges and the two screwjacks makes the canopy very stable under extreme wind-load and

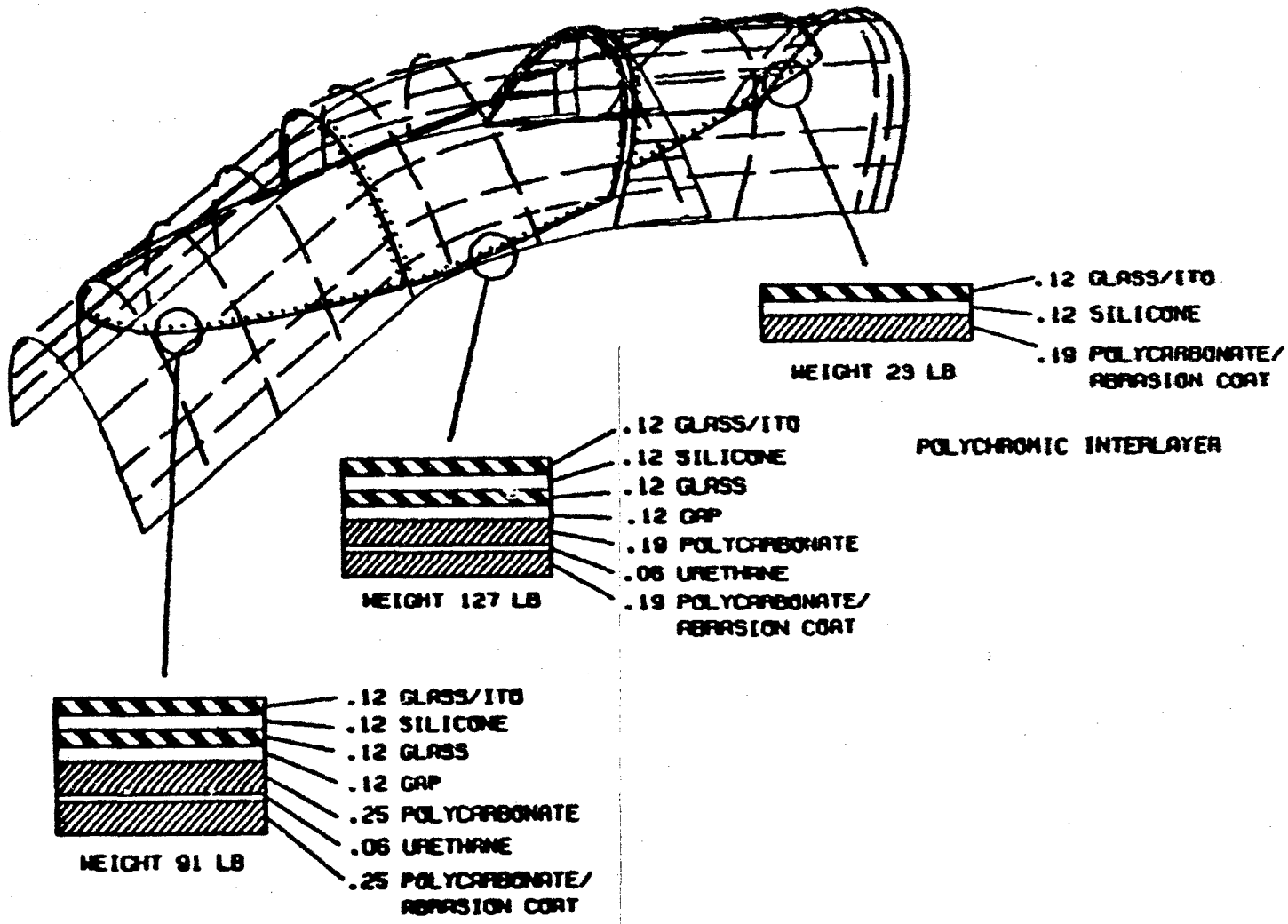


Figure 3.2-4 Multirole Transparency Geometry

bounce conditions. To prevent a person from being trapped inside the cockpit while the aircraft is on the ground, a manually-driven actuation subsystem (hand crank) is incorporated.

The MR emergency system consists of devices that rapidly disconnect the canopy assembly from its attachments and propel it from the aircraft, plus the devices to initiate these actions, and the interconnecting network of explosive lines. Thin layer explosive (TLX) lines are used to transfer the explosive energy between devices because they are very rugged and easy to handle. Because TLX lines are flexible, they easily accommodate challenging installations. Replacement of the explosive initiation/transmission hardware with laser initiators and fiber-optic transmission lines offers substantial supportability payoff when this technology matures and is qualified.

Installation of the canopy assembly involves removal of hinge support access covers from each side of the airplane. Eccentric bolt keepers located on the hinge support are removed from each side of the airplane. A sling is attached to the canopy assembly and is used to raise and position the assembly into place.

The canopy is tilted aft to an angle of 47° with respect to the fuselage sill longaron. The canopy is gently lowered until the aft hinge engages the hinge support fitting, then it is allowed to rotate about the bearing until the hinge is captured by the lower roller. Finally, the lower roller eccentric bolt is adjusted and the bolt keeper is re-installed.

Additional installation activities involve attaching the canopy actuator linkage to the torque tube and connecting the emergency system sequencing lanyards. Final canopy installation details involve rigging the latches to nominal clearances and conducting a cockpit pressurization check.

Ground Attack Design

The GA windshield/canopy frame assembly is shown in Figure 3.2-5. The windshield is designed to pivot forward for cockpit maintenance access. Two windshield retainer fittings are used to secure the windshield to the forebody. The windshield bowframe is also used to fasten the assembly to the fuselage. Skin panels allow access to the retainer bolts.

The canopy frame/fuselage sill interface is defined by two parting planes. A transition bulkhead is used to stiffen the frame assembly at the parting plane intersection. A close-out shelf is used to stiffen the aft frame and support loads from the hinge fitting.

The interface between the windshield and canopy bowframe is shown in Figure 3.2-6. The figure shows the windshield and canopy bowframes. Also noted are the weather seal and pressurized inflatable seal.

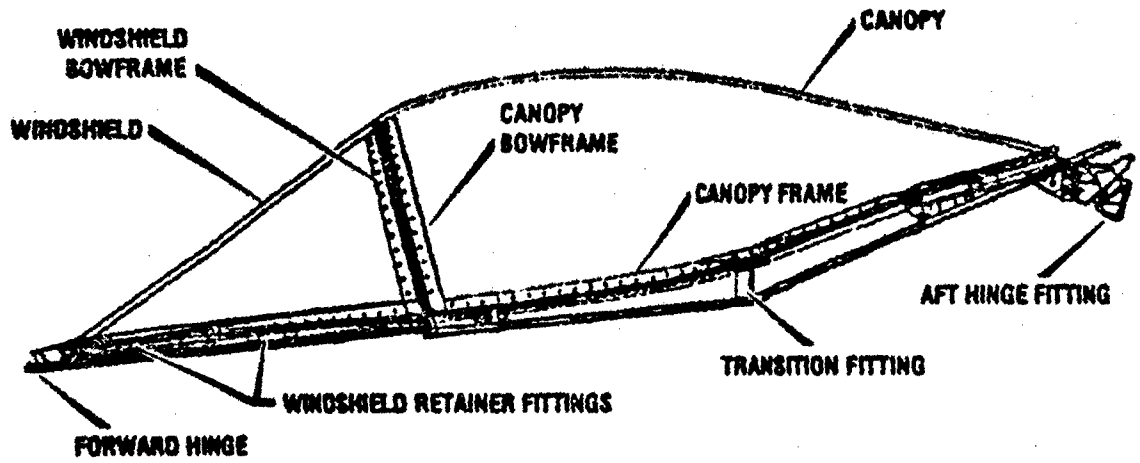


Figure 3.2-5 Ground Attack Windshield/Canopy Frame Assembly

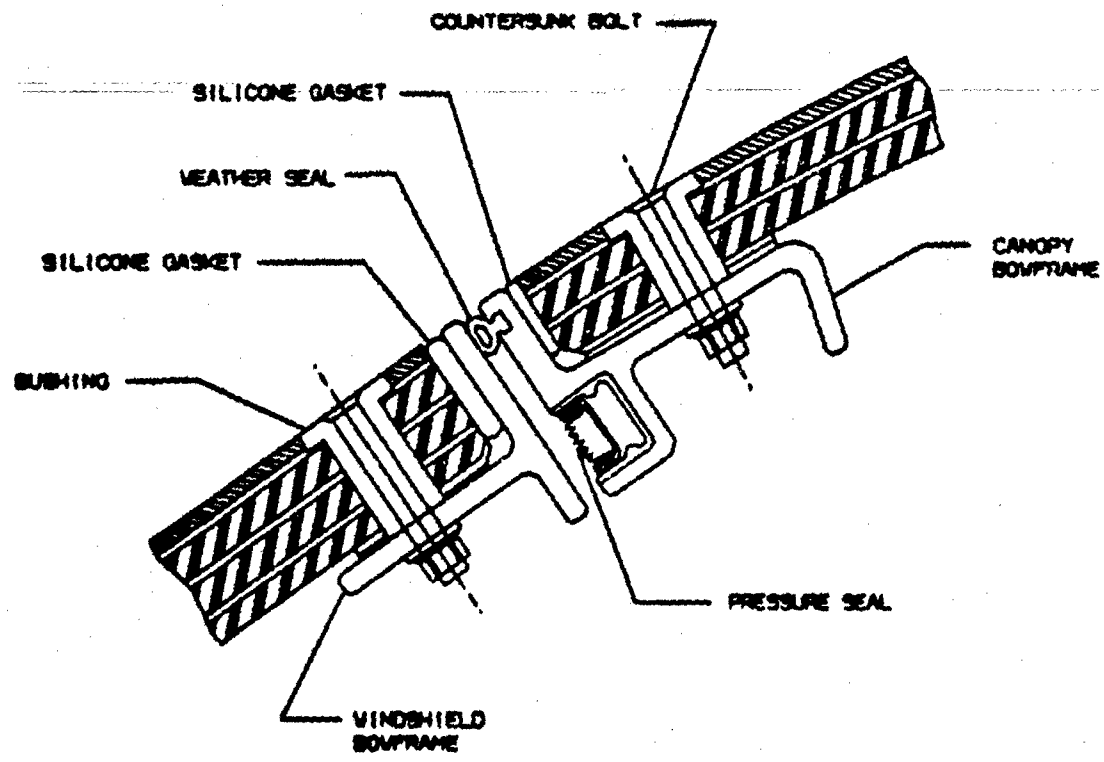


Figure 3.2-6 Interface Between Ground Attack Windshield and Canopy

The GA Transparency with its laminate summary are shown in Figure 3.2-7. Since the GA mission entails only modest elevated-temperature demands on the transparency, an all-plastic laminate was selected. In essence, this design was carried as a technology transition opportunity for insertion of emerging new polymer/casting systems. It should be noted that during this time period, coating systems were not available for plastic substrates that provided the required resistance to natural and combat hazards.

To accommodate the dual sill parting planes of the ground attack design, the canopy latching and unlatching motion follows the path of a large arc for a short distance. The instantaneous center of rotation of the canopy, the center of this arc, is geometrically defined by two lines projected normal to the sill planes. Figure 3.2-8 illustrates how the instantaneous center of rotation is defined. The effective motion at each of the hooks is approximately straight translation because of the large radius and the small arc angle. Each hook features a special profile and orientation to produce the desired rotational motion for the canopy.

The hooks protrude from the inside surface of the canopy frame to engage the latch rollers on the inboard flange of the longeron. The pilot can visually confirm latch engagement with ease. Hook fittings and latch lever fittings are incorporated in the canopy frame and fuselage longeron, respectively, to carry and distribute the latching loads.

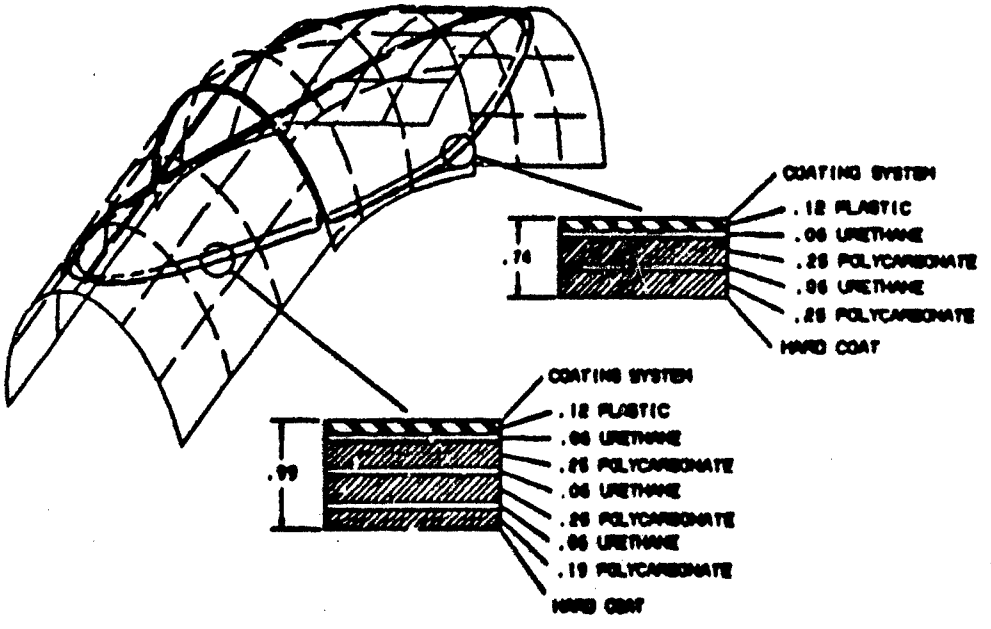


Figure 3.2-7 Transparency Geometry For Ground Attack Configuration

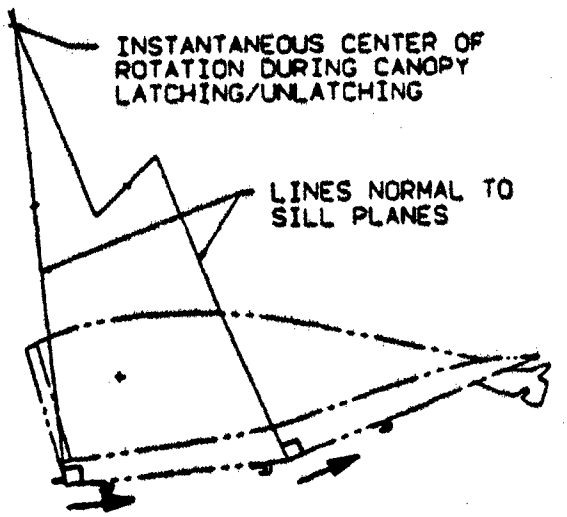


Figure 3.2-8 Instantaneous Center of Rotation, Ground Attack Canopy

For access to the cockpit, the canopy opens by pivoting about hinges at the aft ends of the frame. The canopy rotates approximately 34° to the normal, fully-opened position. The actuation system allows the canopy to be stopped and held at any position between fully closed and fully opened.

Figure 3.2-9 shows the Ground Attack Crew Station Geometry. The GA transparency has the largest area of clear vision for all MITS mission designs and is well suited for the GA role. The forward bowframe placement was driven to provide adequate clearance for emergency escape. The bowframe masks the transition from the single curvature windshield to the compound curvature canopy.

Air Superiority Design

In this section we will briefly summarize the primary features of the AS design. Figure 3.2-10 shows a side view of this design and its structural arrangement. The transparency system consists of three components that feature the same exterior ITO-coated glass-pressurized gap design employed by the MR configuration. Since birdstrike requirements are of lower priority for this mission than for the MR mission, the PC structural plies are thinner in this case (2 x 0.19 in. compared to 2 x 0.25 in. in the windshield area).

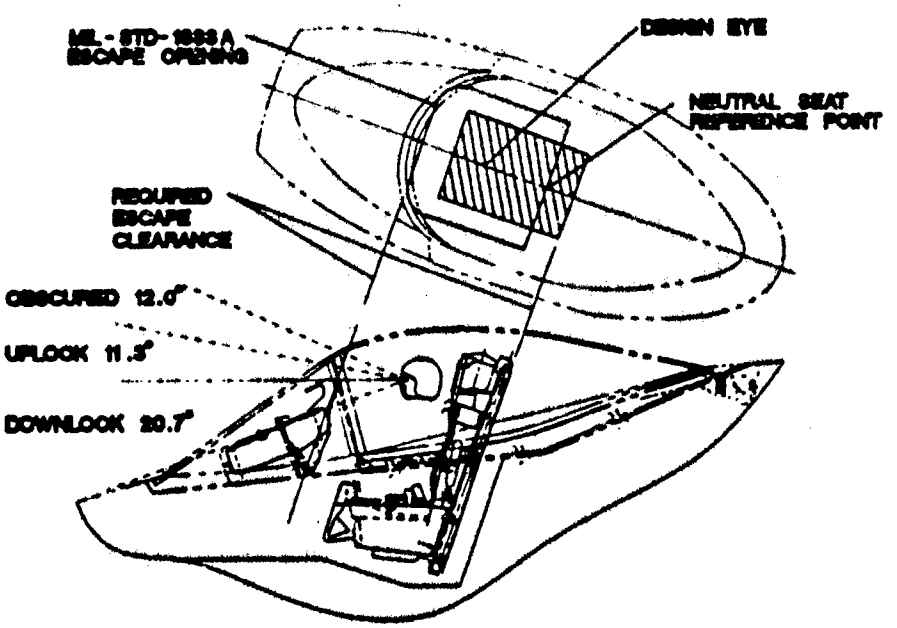


Figure 3.2-9 Ground Attack Crew Station Geometry

The windshield assembly features a forward-pivoting design for maintenance access. The canopy frame assembly, with an aft pivot for ingress/egress, and emergency escape, integrates the windshield/canopy, latch hardware, and emergency system components with the aircraft forebody. Dry seals are used for all interfaces between transparency and structure.

Additional information pertaining to the detail MR, GA, and AS transparency systems may be found in References 13, 14, 15, and 16.

An Integrated Support Plan and a System Safety Plan were written for the three MITS detail designs. These plans are described in detail in References 17, and 18, respectively.

3.3 Materials and Testing

The focus of Phase 2 materials activities was on ITO-coated chemically-tempered glass (Herculite II from PPG Industries) and on an 8240 high-temperature polymer. A high-temperature material was required for the MR and AS missions; the GA mission does not require high service-temperature capability. All missions require an exterior coating system to provide multifunctional resistance to natural and combat hazards.

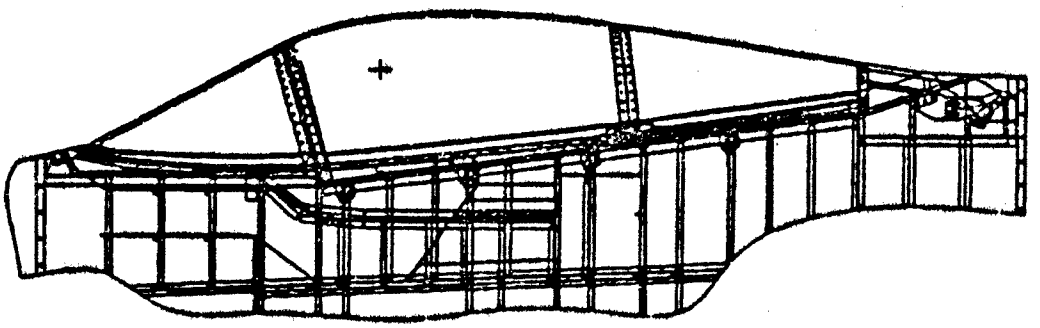


Figure 3.2-10 Summary Of Air Superiority Design Features

Table 3.3-1 summarizes a few of the pros and cons for glass versus plastic face-ply materials.

Table 3.3-1 Face-Ply Materials Tradeoffs

MATERIAL	PROS/CONS	P	C
Chem-Tempered Glass	• High-Temp Capability	✓	
	• Strong, Easily Coatable	✓	
	• Very Durable with ITO	✓	
	• Heavy, Impact Effects		✓
	• Hard to Form Compound		✓
Plastic	• Lighter Than Glass	✓	
	• Generally Easier To Form	✓	
	• Hard to Coat (ITO)		✓
	• Requires Complex Coating System		✓
	• Limited Service Temp.		✓

Glass provides an excellent substrate for conductive coatings and the bare ITO coating proved to be very durable, environmentally resistant, and well suited to our high performance requirements. This combination was therefore selected as the windshield outer ply for the MR and AS missions.

Transparent plastics with higher service temperature capability than PC and acrylic materials have not yet been demonstrated with environmentally resistant materials coatings at the scale and performance level required for MITS applications. Several materials systems are in various stages of development/test via either vendor, internal research & development, or other contract funding. S240 with an S417 coating system were selected for laboratory evaluation in this program. It is not yet optimized for all properties but demonstrated promise early in Phase 2 for its development and scale-up potential.

Four categories of tests were conducted at the University of Dayton Research Institute (UDRI), at PPG Industries, and at GD/FW during Phase 2 of the MITS Basic Program to aid in evaluation of the detailed designs:

- o Comparison of candidate face ply materials (e.g., rain erosion, stress craze, abrasion)
- o Determination of quantitative performance (e.g., ballistic fragment, reflectivity, transmittance/haze)
- o Discrimination between design details (e.g., hail, endurance)
- o Screening of advanced coatings for improved service life (reported in Section 5).

These tests and their results are described in the following paragraphs.

* MTS Flex Beam	* Hail Impact	* Ballistic
* Rain Erosion	* Endurance	* Salt Fog
* Air Cannon	* Stress Craze	* Thermal
* UV/VIS/NIR Transmission	* IR Reflection	* QUV
* Thermal Shock	* Tensile Loading	* Optics
* Salt Blast	* Photochromics	* Fungus
* Oscillating Sand Abrasion		

Conditioning of the samples was limited to QUV exposure. In many cases, tests were performed on samples with and without QUV simulated-weather conditioning. QUV exposure at GD/FW consisted of repeated cycles of 7 hours ultra-violet light at 60° C followed by 5 hours of condensing humidity at 45° C. The QUV cycle at UDRI was essentially the same except that temperature was held at a constant 120° F throughout the entire cycle. The samples were generally conditioned for four equivalent years using UVB-313 lamps and 168 hours per equivalent year. Figure 3.3-1 shows the initial and final transmission and haze values (before and after QUV) of the ITO/glass and S417/S240 samples.

Both the ITO/glass and S417/S240 face ply candidates were tested at GD/FW in oscillating sand (Bayer) abrasion and salt impingement. In the "Bayer" test, a sand "bath" is repeatedly shaken across the surface of the samples up to 600 times. Salt impingement tests were also run on the ITO/glass samples per ASTM F-1128 with slight modification to the maximum number of blasts. Test results are summarized in Figures 3.3-2 and 3.3-3.

Samples of the ITO/glass and fiberglass strap material were tested at UDRI for their resistance to fungi per ASTM G-21. Neither the glass nor the fiberglass samples supported any fungal growth. No significant change in transmission or haze was observed after exposure.

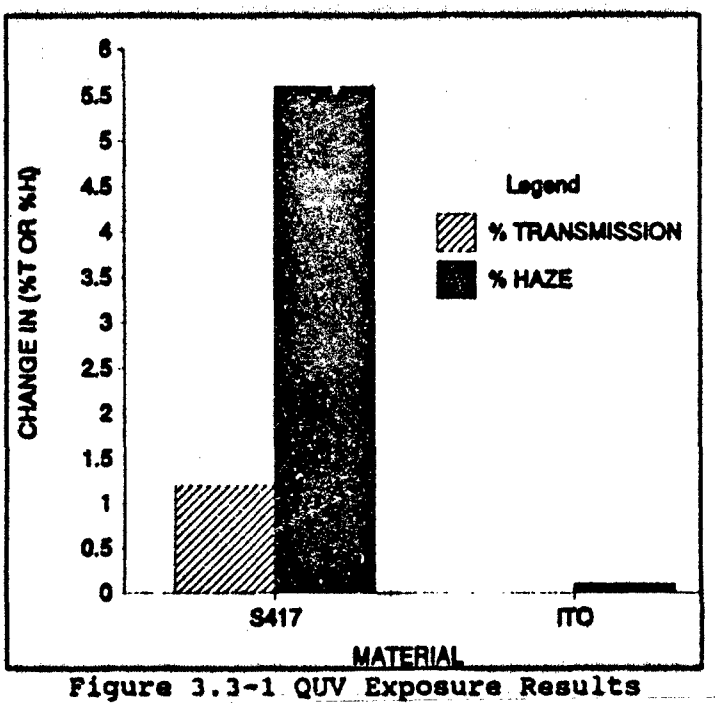


Figure 3.3-1 QUV Exposure Results

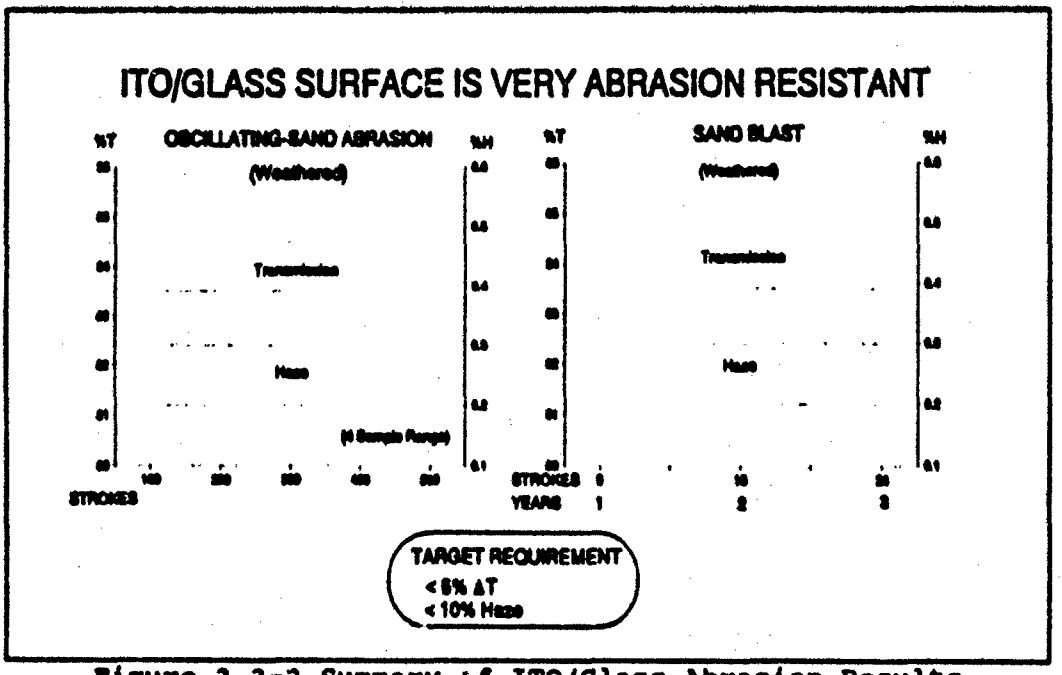


Figure 3.3-2 Summary of ITO/Glass Abrasion Results

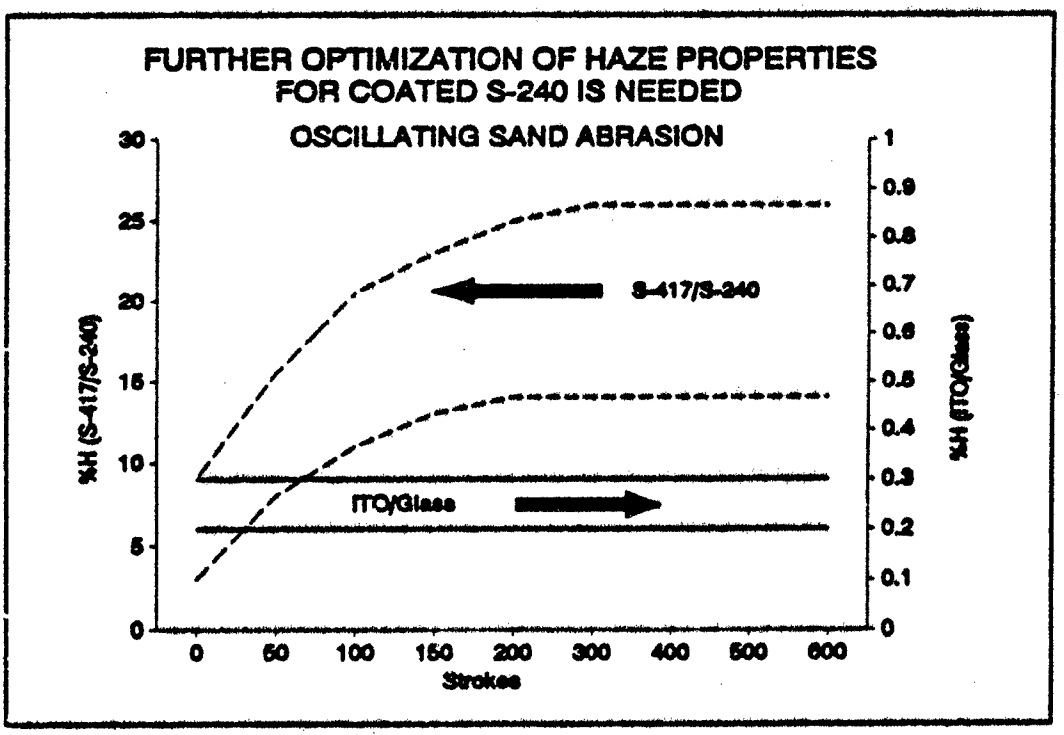


Figure 3.3-3 Summary of Oscillating Sand Haze Results

Thermal Shock tests were run at GD/FW on ITO/glass samples. The tests were run per MIL-STD-810D, Method 503.2, using a temperature range from -65° F to +300° F. No cracking of the glass or coating was observed.

Stress Craze tests were conducted at UDRI on both QUV-conditioned and baseline ITO/glass samples following the procedure outlined in AFWAL Report TR-85-3125. The test chemicals used were 50% sulfuric acid solution, isopropyl alcohol, toluene, ethylene glycol, JP4 jet fuel, and a 20% MEK/water solution. No crazing of the coating was observed with any of the chemicals up to 3,000 psi outer fiber stress level.

Various types of impact tests were conducted on a number of candidate laminate designs. Table 3.3-3 summarizes the impact response of comparable "gap" and "no-gap" glass-PC designs. The MITS flex-beam tests were conducted at 2,000 inches/minute crosshead displacement load rate (per AFWAL TR-85-3125). Ballistic tests used a chisel point fragment-simulating projectile. Threshold penetration velocities were used to rank the relative fragment resistance of the designs. Air-cannon impact tests used 2-inch steel ball bearings to measure the velocity and energy required to penetrate the laminates. Hail impact tests were conducted per ASTM F-320. The laminates are ranked based on the largest size hail not causing unacceptable damage.

Table 3.3-3 Summary of Impact Response

IMPACT RESPONSE IS SIMILAR FOR GAP / NO-GAP DESIGNS

TEST	DESIGN	
	GAP	NO-GAP
Ballistic Threshold Velocity for 30 cal., 44 gr, FSP	2780 ft/sec	2400 ft/sec
Air Cannon Penetration Energy for 2-Inch, 1.185-lb Ball Bearing	6471 ft-lbs (593 ft/sec)	5405 ft-lbs (542 ft/sec)
MTS Beam After QUV, 2000 in./min. Load Rate "Toughness" Energy	2263 in-lb	2503 in-lb
Hail (1000 ft./sec.)		
1/2" Ice Ball	No Damage	No Damage
1" Ice Ball	9,9,2 *	2,3,1
2" Ice Ball	1,1	.....

MULTIROLE DESIGN

- BALLISTIC FRAGMENT RESISTANCE > LEXGUARD
- NO SPALL
- GOOD DEBRIS/HAIL RESISITANCE

* 9 - Survived 9 Shot Requirement
2 - Failed on 2nd Shot
1 - Failed on 1st Shot
(Three Samples Each Design)

Table 3.3-3 indicated that the laminate designed with two plies of glass with an air gap is generally more impact resistant than the design with no air gap.

Samples of baseline and conditioned ITO/glass and S417/S240 were tested for rain erosion at 500 mph, 1 inch per hour rainfall, and 30 degree installation angle. The results of these tests are summarized in Table 3.3-4 for ITO/GLASS. The S417/S240 results are contained in References 13, 14, 15, and 16.

Infrared Reflectance measurements were made on the ITO/glass and S417/S240 surfaces for laser response and signature assessment. The reflectance curves are shown in Figure 3.3-4. The reflectance of the bare glass surface is also shown for comparison. Several different optical measurements were run on the PPG laminate configuration shown in Figure 3.3-5. Transmission measurements were made normal to the samples and at the installed position. Figure 3.3-5 is a summary of the various optical measurements that were performed. None of the samples had received any QUV conditioning.

Table 3.3-4 ITO/Glass Rain and Solvent Resistance

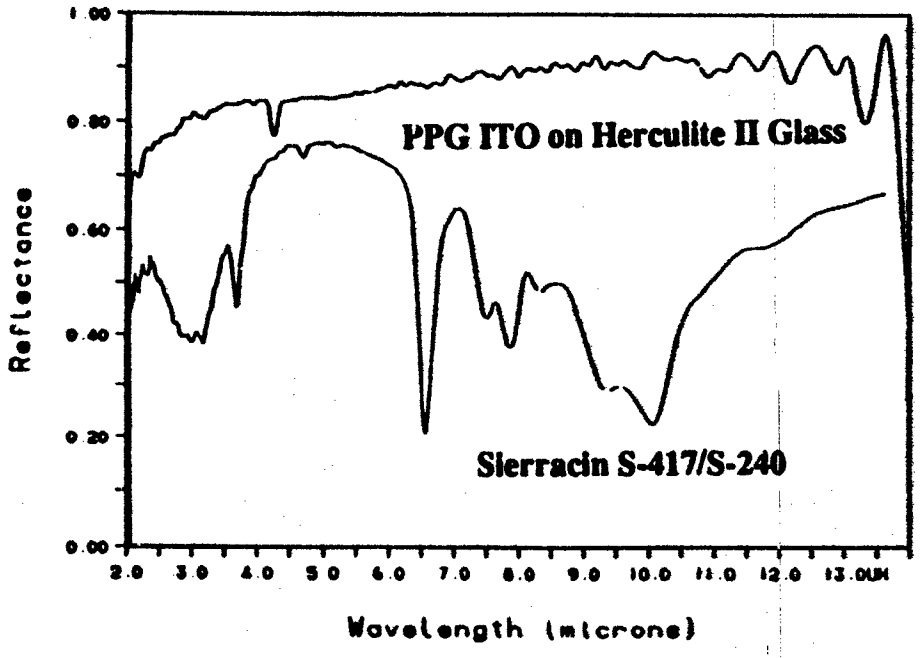
ITO/GLASS SURFACE IS RAIN AND SOLVENT RESISTANT				
TEST	RESULTS			COMMENTS
Stress Craze with: • Toluene • 50% Sulfuric Acid • Ethylene Glycol • Isopropyl Alcohol • JP4 • 20% MEK	NO CRAZING			• EXCEEDS MITS REQMNTS • EXTREMELY DURABLE SURFACE
Rain Erosion at 500 mph (After QUV)	Before	83% T	0.4 %H	
	After	83% T	1.6 %H	
	Coating Removal < 5%			
The Multirole Design Should Have Excellent Service Life (Based on Outer Surface Tests)				

The performance of several photochromic materials provided by Swedlow and PPG was investigated. Sample A was provided by PPG and Samples B and C were provided by Swedlow. Transmission measurements were made on the samples while in both the colored (darkened) and bleached (clear) states. These transmission values are shown in Table 3.3-5 with various combinations of transparent plies over the photochromic material. The tests were performed according to the procedure outlined in Reference 7. Details of this testing are covered in Reference 13.

Tensile Load tests were performed on samples of the ITO/glass laminate with bonded fiberglass straps. These straps are drilled and used in the designs as the structural mounting for the glass plies. They carry the pressure load from the glass to the frame. Interest was limited to the tensile strength of the glass and the bonding adhesive. Three of the samples failed due to the adhesive and two failed when the glass broke. The results of these tests are reported in References 13 and 14.

Aeroheating thermal tests were conducted at PPG Industries on 4-inch X 7-inch samples bolted together to simulate the bowframe area between the windshield and canopy sections of the MR and/or AS transparencies. These results are reported in Reference 14. The basic intent of these tests was two-fold: (1) to measure through-the-laminate temperatures for comparison to STAPAT predictions; and (2) to compare through-the-laminate temperatures for different outer bowframe materials (aluminum, graphite/epoxy, and fiberglass/epoxy) and different windshield and canopy laminates.

IR REFLECTANCE WAS MEASURED FOR CANDIDATE SURFACES



FOR EXAMPLE - -

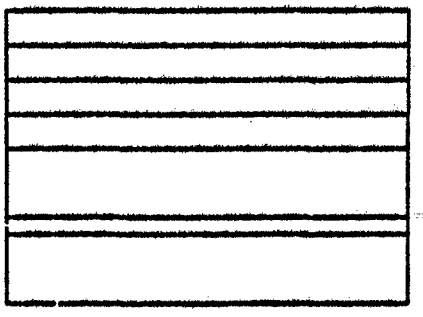
AT 10.6μm,

- ITO / GLASS IS ~90% REFLECTIVE
- S417 / S240 IS ~45% REFLECTIVE

Figure 3.3-4 IR Reference

OPTICAL MEASUREMENT SUMMARY			
PROPERTY	MEASURED VALUE	GOAL	COMMENT
% Transmission (Visible)	54%* at installed ∠	>65% at installed ∠	Shortfall
% Transmission (Visible)	70.5% Normal† 66.5% Normal*		
% Haze (Visible)	2.8% Normal †	<2% initially	Shortfall
% Transmission (Infrared) (700-900 nm)	50% Normal	>90%	Shortfall

Laminate Cross-Section



ITO/.0125" Glass
0.125" Silicone
0.125" Glass
0.125" Gap
0.25" Polycarbonate
0.06" Urethane
0.25" Polycarbonate

† Indicates measurements made according to ASTM D-1003
* Indicates measurements made according to AAMRL-TR-89-044

Figure 3.3-5 Optical Measurement Summary

The outer laminate surface was heated to approximately 365° F and held for several minutes. The most interesting results from these aeroheating tests were (1) laminate temperatures at the outermost PC ply near the bowframe bolts were less than predicted and well below the maximum temperature recommended for PC use. Also the temperatures inside the laminate were the lowest when the outer bowframe strap material was fiberglass (graphite/epoxy resulted in the next-lowest temperatures).

Laminate endurance tests were conducted at PPG in an effort to ascertain any subsequent performance differences in the PC structural layers for the gap and non-gap designs. Figure 3.3-6 diagrams the cross sections of the 29 inch square samples.

Table 3.3-5 Photochromic Dial-A-Tint Test Results

OVERLAYER	% TRANSMISSION					
	SAMPLE A		SAMPLE B		SAMPLE C	
	Bleached	Colored	Bleached	Colored	Bleached	Colored
None	91	57	90	31	66	12
0.125" Glass 0.06" Silicone 0.125" Glass	70	58	65	36	55	22
Polycarbonate	80	76	73	45	61	25
0.125" Glass 0.06" Silicone 0.125" Glass + Polycarbonate	62	60	57	40	49	24
ITO/0.125" Glass	74	57	68	38	57	17
(Response Time Typically 5-45 Seconds)						

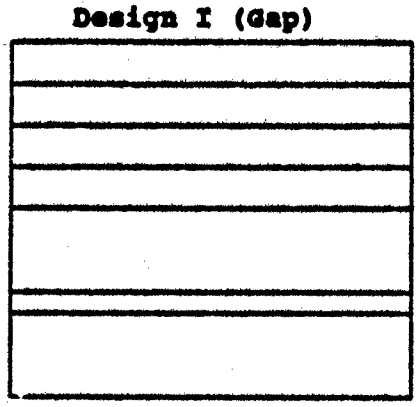


Figure 3.3-6 PPG Endurance Test Coupons

Figure 3.3-7 schematically describes the test set-up. One gap design and one non-gap design sample are mounted side by side in a temperature/pressure chamber that is held at 80° F and pressure cycled from 0-5 psi. This represented the cockpit conditions. The gap design sample was set up so that the gap was always pressurized to the "cockpit" conditions. The outer surface of the samples was cycled between -65 °F and +160 °F. The endurance test pressure/temperature cycle is shown in Figure 3.3-8.

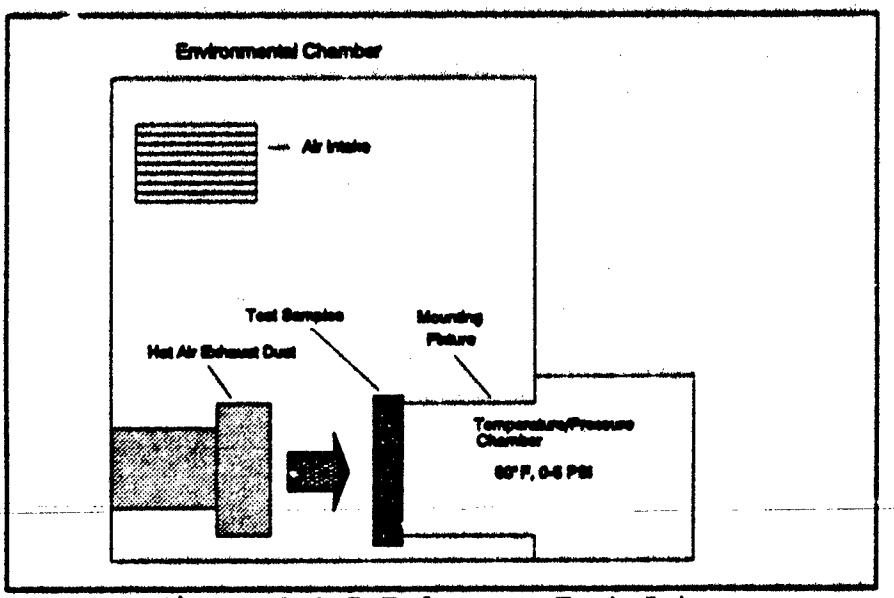


Figure 3.3-7 Endurance Test Setup

The samples were "fatigued" for 200 cycles near the end of MITS Phase 2. No visible differences were noted in the PC layers from the gap and non-gap designs. MTS-beam high-strain-rate-samples were cut from the PC plies of each sample and from an untested baseline sample. These samples (five of each type) were tested at UDRI shortly after the completion of Phase 2. Results from the flex beam tests are summarized in Figure 3.3-9.

The tests indicate only slight differences between the three types of samples. However, there does appear to be a trend with the baseline samples having the higher energy absorbing capability followed by the air gap design and then the silicone design.

Details of the results of the materials testing covered in this section are shown in References 13,14,15, and 16.

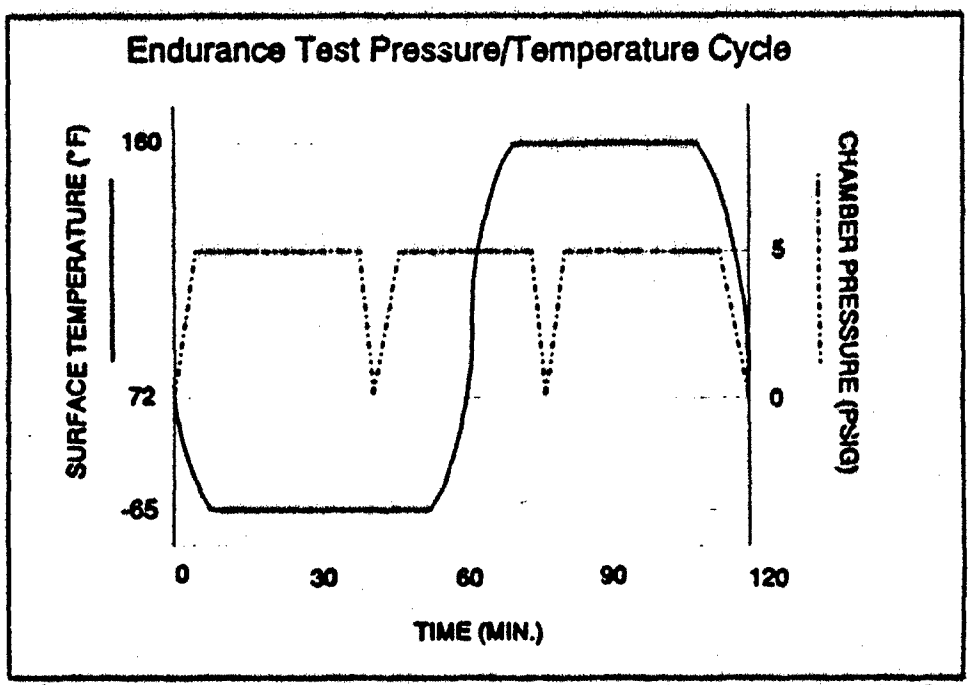


Figure 3.3-8 Endurance Test Pressure/Temperature Cycle

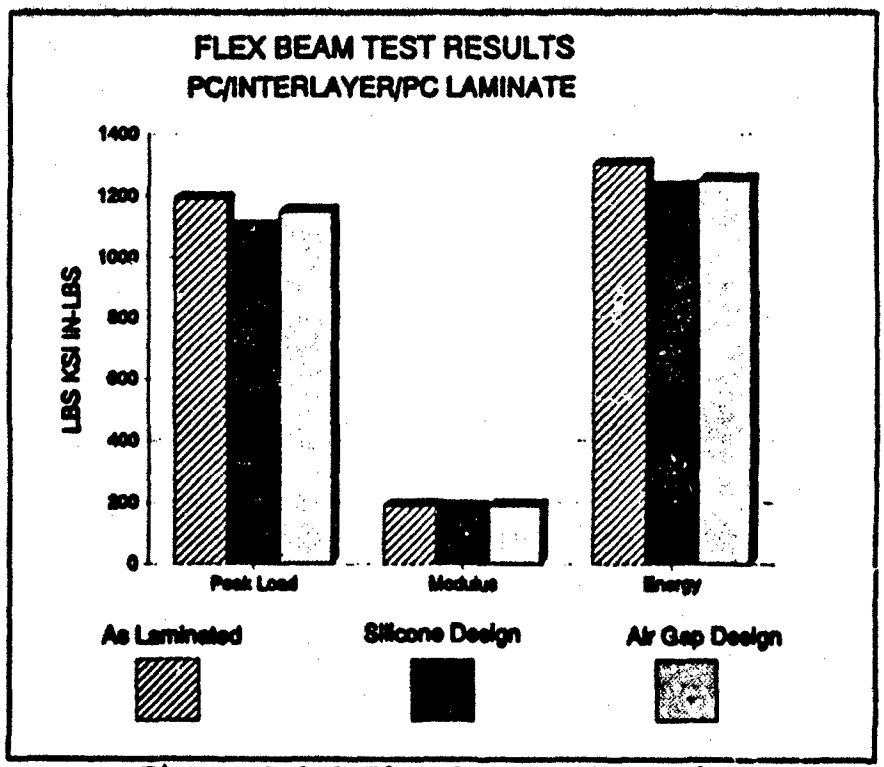


Figure 3.3-9 Flex Beam Test Results

3.4 Phase 3 Demonstrator and Testing

Midway through Phase 2, GD/FW and the MITS Air Force customer agreed that GD/FW should perform long lead tasks for the Phase 3 full scale testing effort. These tasks included expediting the choice of the Phase 3 Demonstrator, performing adaptive design as necessary and generating test plans.

The original plan for choosing the Phase 3 Demonstrator was to complete three detailed MITS Designs (MR, GA, AS), choose one of these three designs, build a full scale version and test it in Phase 3. The choice of the configuration was mutual between GD/FW and the Government: choose the one that addresses and satisfies the greatest number of MITS design requirement goals, which is the MR design. However, demonstrating the MR Configuration had become a much greater challenge since by this point in the program, not only was new tooling required for building the generic transparency system but additional new tooling was needed to build the generic forebody. This total tooling cost far exceeded the program resources at hand.

To avoid this excessive tooling cost, existing tooling was pursued for various canopy/forward fuselage combinations including the YF-22, YF-23, F-15, F-16, and F-18.

The final choice for the Phase 3 Demonstrator was to adapt the MITS MR configuration to an existing F-16 forward fuselage, since definite advantages were offered by the evolution of the MR forebody from a modified F-16 forebody(Section 2.5).

The Phase 3 demonstrator was intended to be a representative model of the MITS MR configuration to be used in the Phase 3 testing and evaluation program. Results from the testing were to be used to evaluate the capabilities of all three MITS designs to meet their required performance levels.

For the purpose of conducting the test program in a timely and cost efficient manner, the demonstrator was designed to be fabricated from existing F-16 hardware reworked into a MR configuration (see Figure 3.4-1).

The demonstrator will include three transparency sections: a forward windshield, a center canopy and an aft fairing as shown in Figure 3.4-2. A forward bowframe splice will be added to the canopy frame to separate the windshield and the canopy sections.

Windshield

The windshield section will use the same glass/gap laminate as that defined for the MR design (Figure 3.4-3) and will be contoured with a single curvature shape to provide the best optics and producibility possible with the glass plies. The glass outer ply will be covered with an ITO coating for good electrostatic drain and hazard resistance.

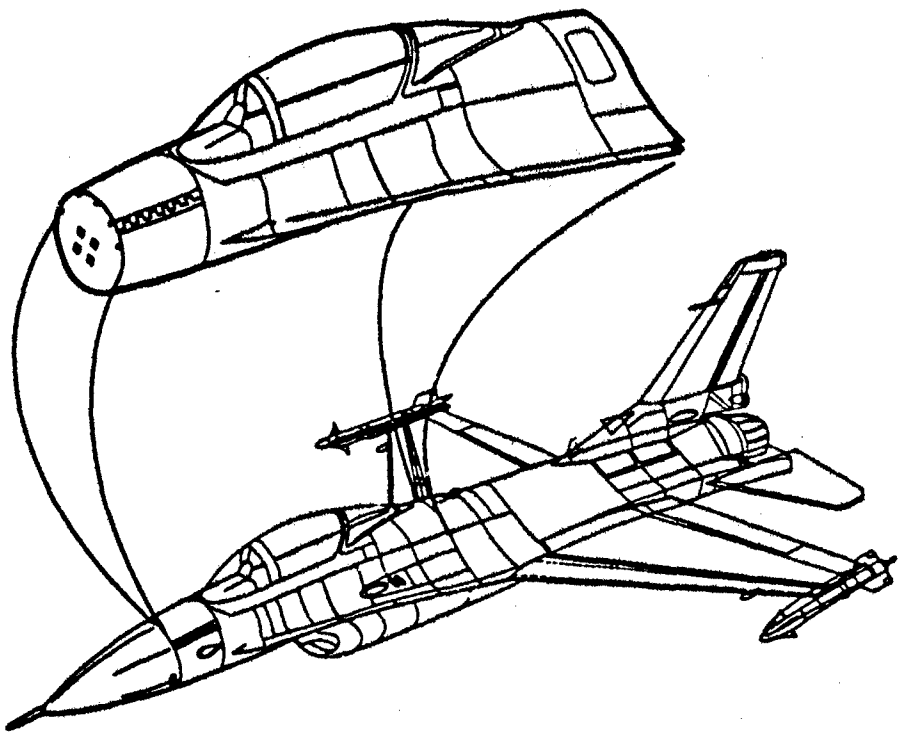


Figure 3.4-1 Commonality Between Multirole Baseline and F-16

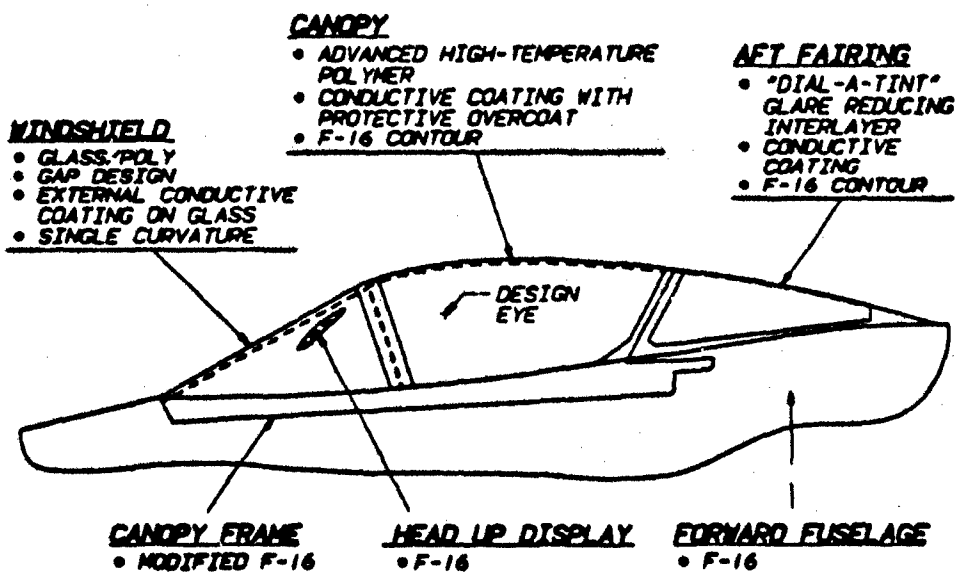


Figure 3.4-2 Demonstrator Configuration

The edge attachment design (Figure 3.4-4) will include a "dry seal" around the periphery to reduce removal and replacement times and a steel mesh grounding system to pass static charge from the ITO coating into the canopy frame.

Canopy

To minimize the amount of rework required on the canopy frame, the canopy section will not include any glass plies and will retain its original F-16 shape (Figure 3.4-5). The laminate will include a high temperature polymer face ply with a multi-functional coating applied to simulate the use of these types of materials on the GA design. The edge attachments will include a "dry seal" to enhance removal and replacement times.

Aft Fairing

The aft fairing section will be constructed with inner and outer plastic plies separated by an interlayer with a photochromic dye (Figure 3.4-6). This will demonstrate a variable tint capability to reduce glare and cockpit temperature. The aft fairing is fuselage-mounted and will retain its original F-16 shape.

The extent of the rework required on the demonstrator hardware to install the new transparency sections will be in the area of the forward windshield/bowframe integration. The new shape defined for the windshield will require the canopy frame attachment flanges to be modified since the simple curvature produces an inboard shift of the windshield surfaces (Figure 3.4-7). This modification can be accomplished with a minimal amount of new detail parts, including the bowframe and a few sections of new attachment flange (Figure 3.4-8). No modification to the forward fuselage section is anticipated beyond installing the aft fairing.

As noted in Figure 3.4-9, eight types of testing are planned for demonstrations during Phase 3. The test article/fixtures to be used range from a full transparency system mounted on an actual F-16 forward fuselage (birdstrike tests) to a full transparency system mounted on a "table-top" fixture (durability test) to coupons (laser tests). A thorough description of each of the planned tests is given in Reference 19.

4.0 REBASELINE OF PHASE 3

Upon completion of Phase 2, both the Air Force and GD/FW MITS Program Managers agreed that due to new knowledge gained that a rebaselining of Phase 3 would allow much better results and more efficient operation of the MITS Program. Therefore, the Government and GD/FW began the contractual process for rebaselining Phase 3.

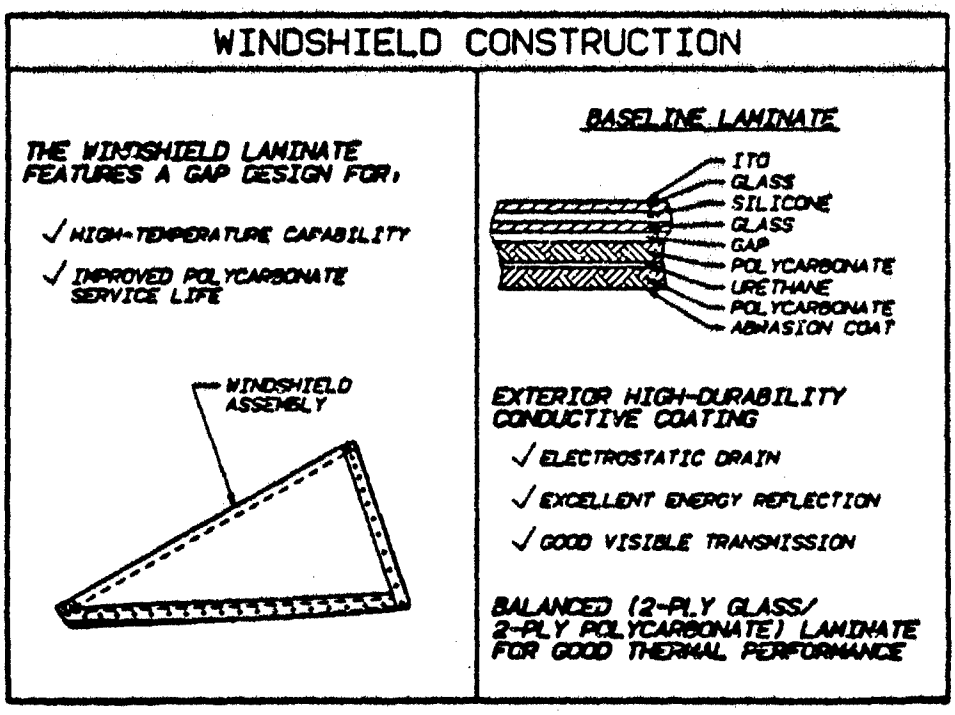


Figure 3.4-3 Demonstrator Windshield Configuration

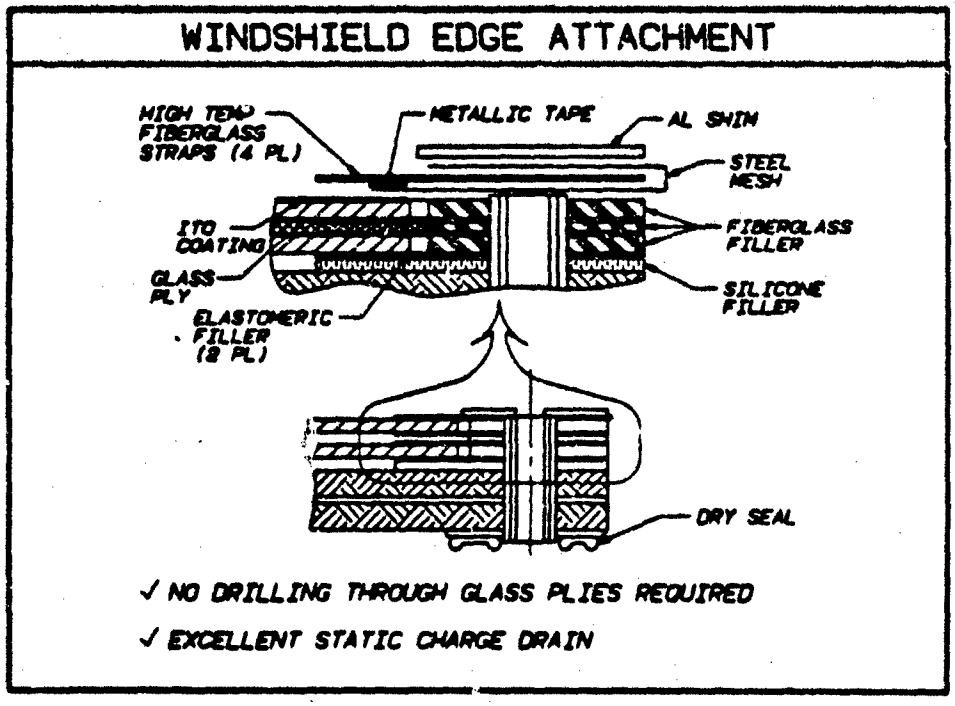


Figure 3.4-4 Windshield Edge Attachment

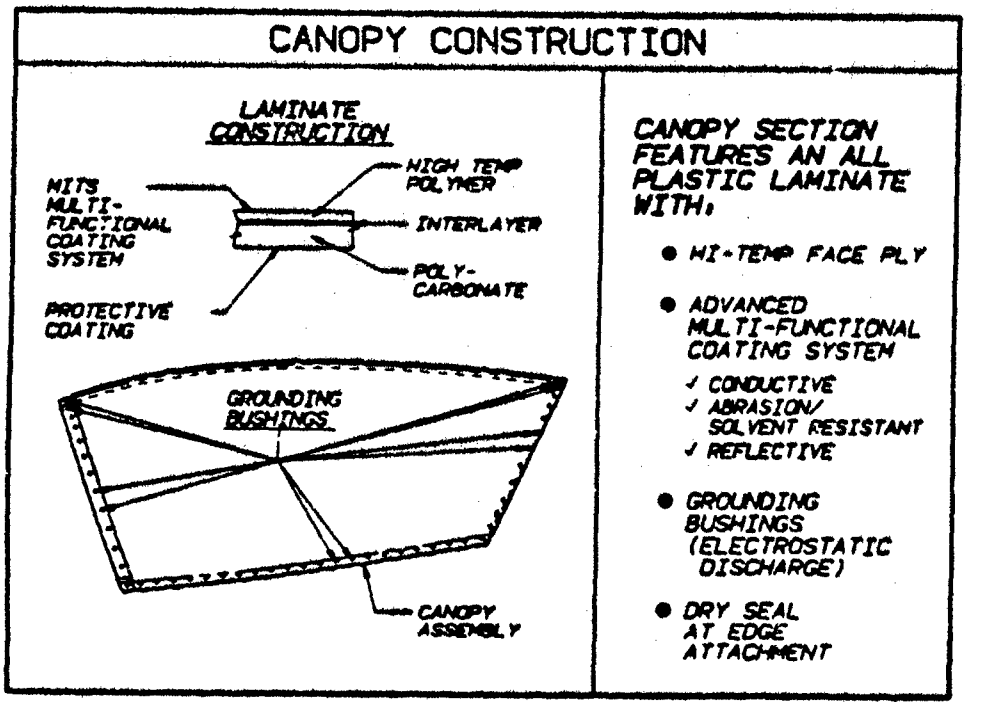


Figure 3.4-5 Demonstrator Canopy Construction

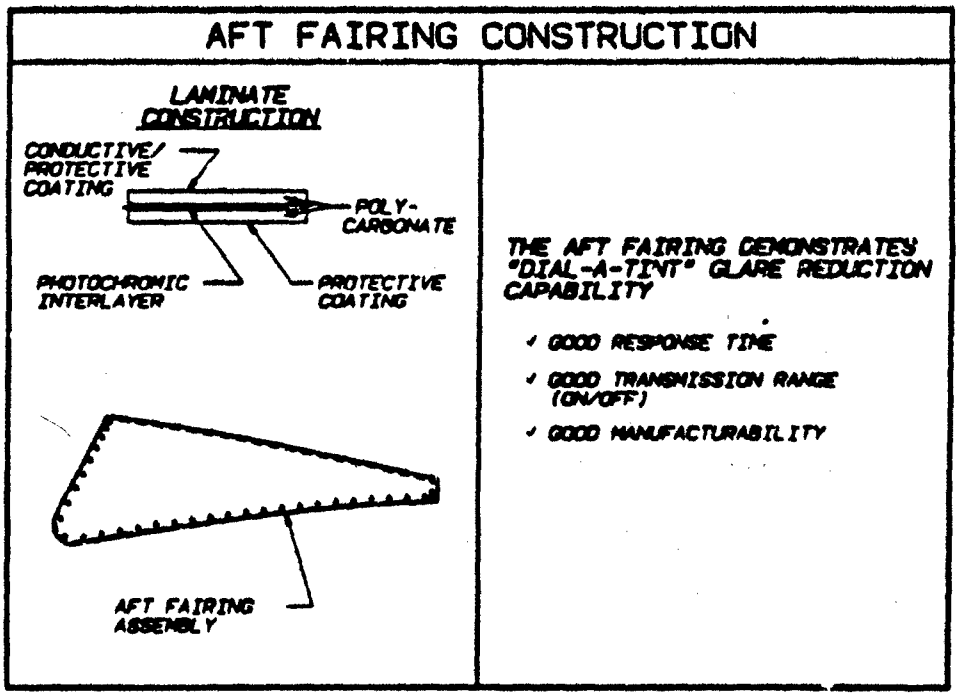


Figure 3.4-6 Aft Fairing Construction

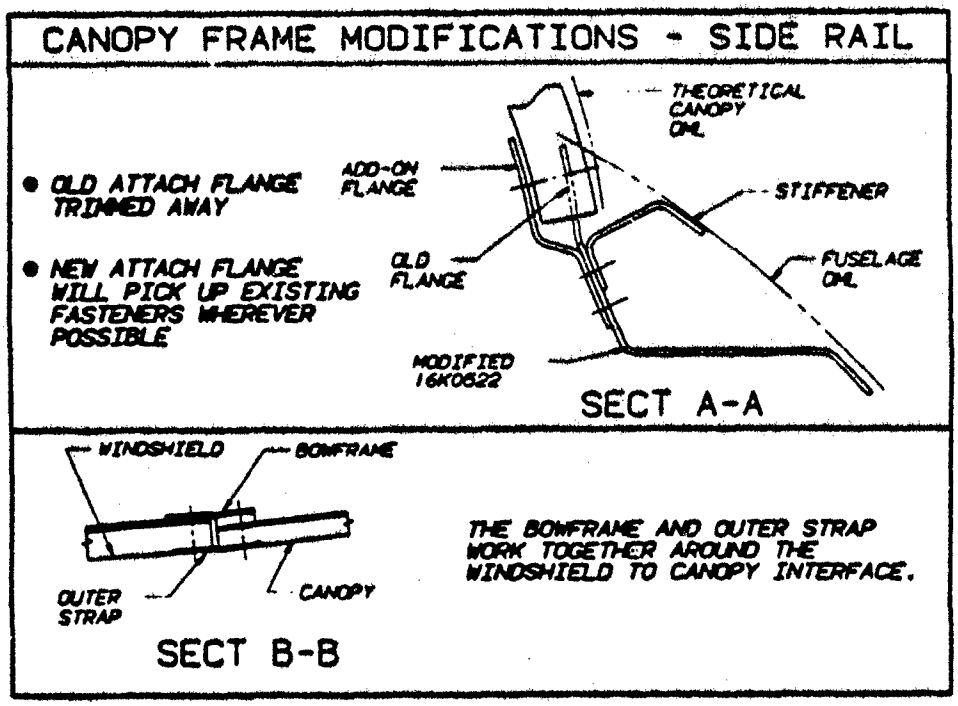


Figure 3.4-7 Modifications to Side Frame

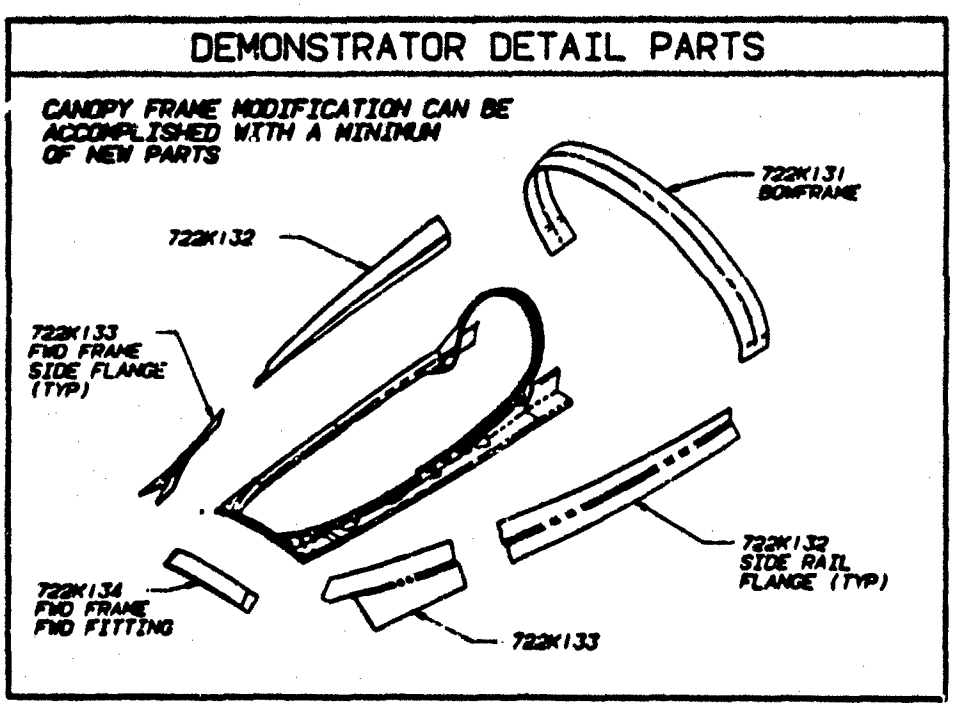


Figure 3.4-8 Demonstrator Detail Parts

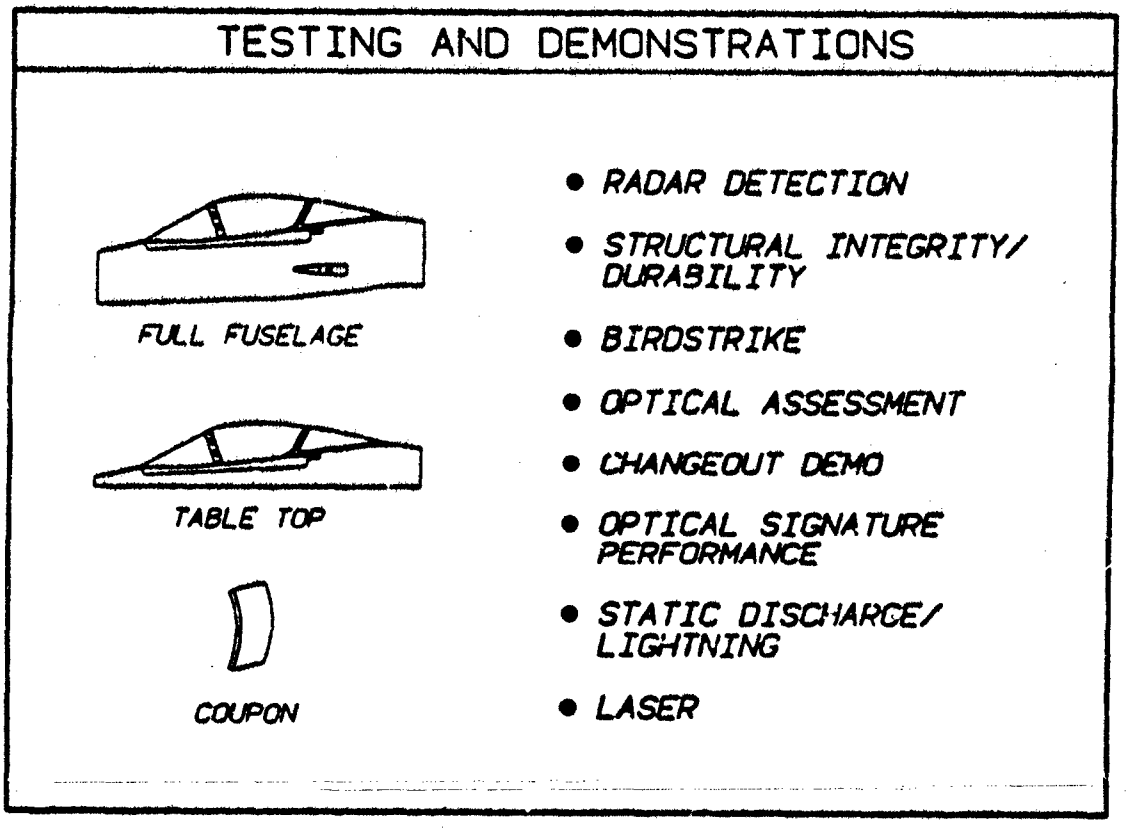


Figure 3.4-9 Phase 3 Testing Overview

As depicted in References 7, 20, 21, and 22, technical information transfer, testing and producibility and pre-planned product improvement were planned for Phase 3. Unfortunately, during this same time period, MITS Government funding sources underwent drastic reductions as a result of severe fiscal constraints. Consequently, the MITS Air Force Customer was compelled to request that GD/FW terminate the Basic part of the MITS Program for their convenience. That termination is being completed at present.

Fortunately, enough funding was salvaged to allow GD/FW (now LFWC) to continue on with an Advanced Canopy Coatings Add-On Project that had been negotiated during the early stages of Phase 2. The Advanced Canopy Coatings Project is described in Section 5.

5.0 ADVANCED CANOPY COATINGS PROJECT (ACCP)

5.1 ACCP - Phase 2

Within the overall scope of the MITS program, extra focus and resources were devoted to the goal of increased service life. A "project," known as the MITS Advanced Canopy Coatings Project (ACCP) was conducted with two primary objectives:

* Demonstrate a coating with a 4-year service life

* Develop a test methodology (lab tests) to validate that service life

While the goals of the total MITS effort typically required multifunctional exterior coatings systems, probably including a metallic or metal-oxide layer, the ACCP efforts had fewer requirements and did not necessarily include the need for an electrically conductive layer.

In general the following groundrules were adhered to during the coatings project: no new materials were to be developed; multiple sources were used for manufacture and test; service life was the single most important goal; MITS target requirement levels were used as ranking criteria; and near term demonstration on the F-16 transparency was planned. A bare (non-metallized) PC substrate was used as the baseline, but metallized versions were also included in the testing effort for possible future implementation.

The technical approach, detail testing and coatings test results for this phase of the ACCP are presented in a separate paper, entitled Advanced Canopy Coatings that appears elsewhere in this Air Force document.

5.2 ADVANCED CANOPY COATINGS PROJECT (ACCP) - PHASE 3

Phase 2 ACCP testing results indicated that four coatings looked very promising under "worse-case" conditions with excellent bare and metallized substrate combinations. These four coatings were Pilkington's 6831 EG (Enhanced Gold) and 6832, PPG's 5300 liner and Texstar's C659.

The original ACCP contract called for purchasing up to ten F-16 canopies, each, for applying two of the aforementioned "promising four" coatings. The coated F-16 canopies were to undergo optical, durability, birdstrike, and flight testing at government facilities. Then the testing methodology developed during Phase 2 was to be revised based on the full-scale test results.

At the time the MITS Air Force Customer requested that GD/FW rebaseline the basic part of the MITS Program (Section 4) the Government requested that Phase 3 of the ACCP be rebaselined, also.

The ACCP rebaseline process is almost complete and deviates from the original plan in two areas: (1) Production F-16 laminates have been replaced by two advanced laminates and (2) All four transparency manufacturers are participating rather than two as the original budget allowed for. The plan for the rebaselined Phase 3 of ACCP is shown in Figure 5.2-1. Texstar, PPG, Pilkington, and Sierracin are all now underway to provide the following for the Advanced Canopy Coatings Project: (1) Uncoated F-16 forward transparencies (Sierracin, Texstar) and (2) extended life coating

systems applied to these transparencies (Pilkington, PPG). Texstar will lay up a two-ply PC laminate and PPG will cast their 5300tm liner as the outer ply to this laminate. Texstar, with PPG's support, will then form this combination to the F-16 shape. Concurrently, Sierracin will lay up and form a three-ply laminate consisting of two outer PC plies and an acrylic inner ply to the F-16 shape. Pilkington will then flow coat the outer surface of this laminate with their 6831 Enhanced Gold coating system.

At present Texstar has "laid up" two units with the 2 ply laminate " in the flat" with acceptable optics and has shipped them to PPG. PPG has cast their 5300 liner to these two units and is in the process of shipping them back to Texstar for forming to the F-16 shape. Successful forming of these two units at Texstar will pave the way for batch processing of the remaining units.

Concurrently, Sierracin has formed two "3-ply" laminates and has delivered one of these units to Pilkington. Pilkington is preparing to apply their 6831EG coating to the Sierracin unit.

Present plans call for the MITS ACCP coated canopies to be available for Government testing beginning January 1994 and extending through July 1994. The ACCP Phase 3 is planned for completion in the fall of 1994.

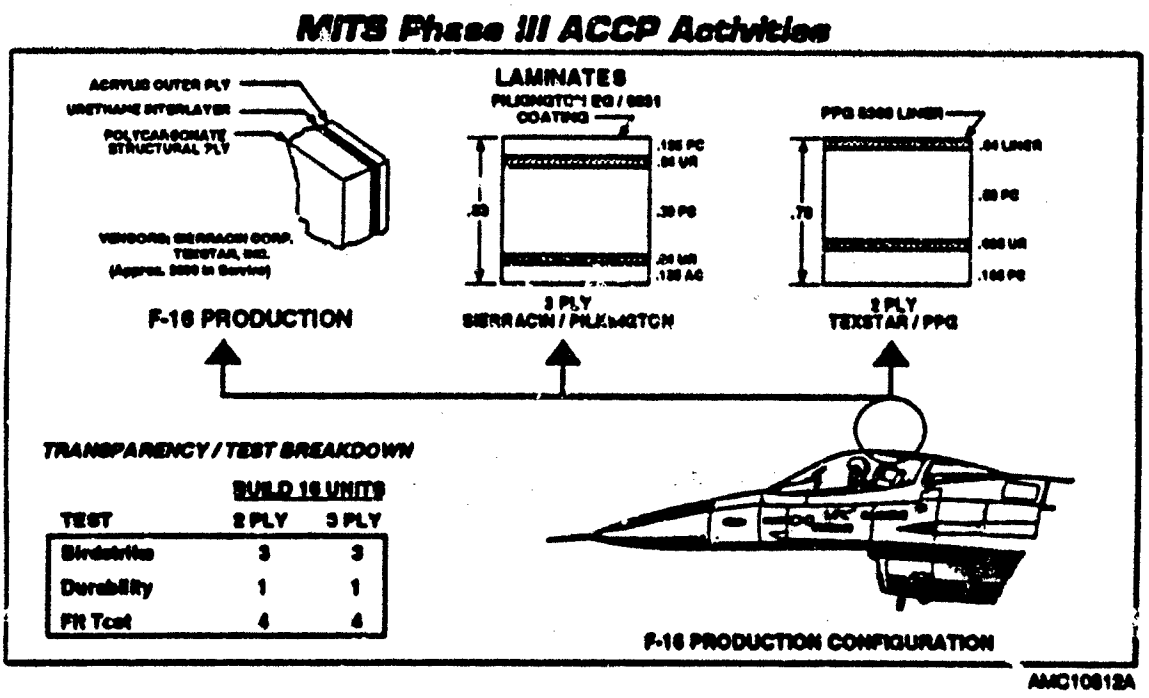


Figure 5.2-1 Rebaselined ACCP Phase 3 Activities

6.0 SUMMARY, PAYOFFS FOR THE MITS PROGRAM

The MITS Program was able to establish a number of design "firsts" for transparency systems.

Designed to over Eighty Requirements Goals

Approximately forty-five prioritized design drivers resulted from the three mission definitions used for this program. In addition to those prioritized and quantified target levels for a wide variety of performance features, an additional 40+ areas of concern were considered during the design process.

Established an Interactive Design Methodology

A design methodology was implemented to accomplish the required technology integration. This methodology featured iterative interaction between technical specialists and designers. The specialists analyzed particular requirements and recommended design features to satisfy those requirements. The designer then integrated these features into an overall design. The process featured tradeoffs and consensus decisions via a modified "quality function deployment" process and successfully accomplished several key design trades such as optics-RCS-aerodynamics.

Involved Transparency Manufacturers in Early Design Stages

The two MITS subcontractors, PPG and Sierracin/Sylmar, participated in all stages of the requirements analyses, tradeoff processes, and consensus design. At one stage of the design activities, the subcontractors regularly traveled to Fort Worth and met with the designers and specialists to drive in concert toward satisfactory design integrations.

These manufacturers cooperated with GD/FW and with each other to not just react to requirements but to act and recommend design features.

Incorporated a Novel Gap Design for the Transparency Laminate

In order to achieve the required high temperature capability and to extend the transparency service life, a pressurized "air gap" design was selected for the MR and AS configurations. This gap is to be pressurized with dry air/gas to the same pressure as the crew compartment. Consequently, the PC structural layers thus do not "flex" with pressure changes and potentially will have a significantly-improved service life.

The two ply glass outer laminate is thus separated from the two ply PC laminate, thereby reducing thermal mismatch problems.

Changeout of the outer ply is also facilitated by this design, and less haze is measured due to the reduced amount of silicone interlayers, for example.

Used MAGNA, STAPAT, and Ray Tracing as Design Tools

Computer-automated diagnostic and analysis tools were used to facilitate the technology integration process early in the design process. Although true Integrated Product Design/Concurrent Engineering was not achieved (due to the insufficient level of maturity of the analytical tools), substantial interaction was accomplished between analysis and design to define trade directions and quantify performance expectations.

Investigated High Temperature Polymers

Developmental high temperature candidate surface ply materials were evaluated and matured during the MITS efforts. The potential for use of these materials was assessed and a materials scaleup program was defined to alleviate this technology shortfall in high temperature plastic materials.

The GA mission configuration was designed to accommodate an all-plastic laminate in order to provide a technology insertion opportunity for developing multifunctional external coating systems. The Phase 3 demonstrator canopy section was also configured to allow substitution/insertion of the best available high temperature material and coating systems.

Used A Computerized Design Network

All surface contours were defined with a common CATIA 3-D data set, so that designers and analysts had a common set of models on which to work. This allowed aeroheating, aerodynamics, optics, and signature analyses to be accomplished with the identical contour being worked by the design team.

In addition to these "firsts" for transparency systems, the MITS Program has allowed for the first time an opportunity to concentrate on genetic transparency systems design at the prime contractor and subcontractor levels to identify, quantify , and prioritize all pertinent design requirements. MITS has allowed the pursuit of these requirements by applying the most advanced technologies and analysis techniques while utilizing the most advanced materials with maximum participation of transparency manufacturers.

Four MITS detailed designs with the following features:

(1) Forward Integral Bowsplice
(2) ITO Coated Glass Windshield
(3) Single-Acting Aft Pivot
(4) Static Discharge Reduction
(5) Non-Curing Dry Seals
(6) Multi-Functional Coatings
(7) Variable Solar Tint
(8) Supportability Enhancements

that encompass all pertinent mission profiles have been developed to the following levels:

(1) RCS Reduction
(2) Birdstrike Protection
(3) Aeroheating For Mach = 2.5
(4) Four-Year Service Life
    (Potential Air Force Savings: $250M)
(5) Four-Hour Change-Out/Two Technicians
(6) Laser Protection

Finally, MITS has allowed the latest multi-functional polymer surface coatings to be developed and to be demonstrated on flying F-16 production canopies and full scale testing to replace the outer acrylic ply by a protected PC ply - a significant improvement in F-16 service life, while featuring non-curing dry seals, electrostatic discharge prevention, and variable solar tint.

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F-15 ADVANCED TRANSPARENCY DEVELOPMENT: A TEAM APPROACH

1st Lt Guy Graening, WL/FIVR
Wright Laboratory

F-15 Advanced Transparency Development: A Team Approach

1st Lt Guy Graening, Wright Laboratory

ABSTRACT

The success of the F-15 Advanced Transparency Program can be attributed to a team approach that combined the goals of many organizations into a final product that offers a unique combination of improved performance and reduced cost. The changing mission of the F-15 aircraft combined with recent advancements in transparency design opened up an opportunity to develop an advanced transparency system for the F-15. The challenge involved bringing together individual organizations and creating a design that would meet the demands of the anticipated combat environment of the year 2000.

The team's work began with the Fighter Requirements, Safety, and Logistics offices of HQ ACC defining current areas of improvement and needs of future systems. The goals began materializing: 4 year service life, 4 hour change out, 500 knots birdstrike resistance. Warner Robins ALC needed to lower unit cost and could benefit greatly if the windshield was interchangeable between F-15A,B,C,D, and E models. Wright Laboratory drew upon UDRI, JATTIC, and the F-15 Systems Program Office to investigate the performance and logistical characteristics of the current F-15 transparency system as well as various candidate advanced systems. The team investigated birdstrike capability and risk, service life factors, and optical requirements.

The initial review of current and candidate designs resulted in a proposed advanced cross-section and frame. Sierracin Corporation manufactured two prototypes to help the team demonstrate the program goals. The advanced prototype underwent 500 knot birdstrike testing, optical evaluation, fit check, and flight evaluation with the involvement of McDonnell Douglas, AEDC, Armstrong Laboratory, and the 46th Test Wing.

The combined effort of all these organizations has resulted in the recent qualification and upcoming transition of the Advanced Windshield which features an interchangeable laminated polycarbonate windshield panel that is bolted to a reusable, quick change out frame. The Advanced Windshield will save nearly $20,000 per change out in the future and can be maintained with the current maintenance procedures. The F-15 Advanced Transparency Team will continue to apply the successful team approach to improve this new design with an abrasion resistant coating. In the future, a spall-resistant HUD and an Advanced Canopy could compliment the Team's Advanced Windshield.

Introduction

Twenty years after its introduction to the Air Force inventory, the F-15 fighter will have an advanced transparency system that incorporates many mature technologies. The timing could not be better. The role of the F-15 has changed dramatically, placing demands on system components originally designed for a different operational environment.

Today the F-15, especially the Strike Eagle model, flies low altitude missions which require increased birdstrike protection. In addition, the F-15 must respond to decreasing funding for maintenance and spares. These factors placed emphasis on lowering unit cost, extending the service life of the transparency, and reducing aircraft downtime for change out. The demand for increased performance at reduced cost challenged the key organizations in the F-15 community to develop an advanced transparency system.

Discussion

The challenge to develop an advanced transparency for the F-15 was risky and complicated for an individual organization to handle alone. Forming individual organizations into a team to solve a problem was not a new idea, but convincing them to be participating stakeholders from the beginning was a newer approach that ensured success. Several organizations had goals for the advanced transparency system that appeared to be in conflict with each other. Headquarters Air Combat Command needed a windshield that provided 500 knots of birdstrike protection for the aircrew and could be changed out in four hours. Warner Robins Air Logistics Center (WR-ALC) needed to lower unit cost and maintain a four year service life. The F-15 Systems Program Office wanted a windshield that was interchangeable between the A through D and E models.

Wright Laboratory (WL) also had an interest in the F-15 and could provide the spark that would ignite the effort and bring the individual organizations together into a team. WL was eager to find a customer for mature technologies such as laminated polycarbonate, computer modeling of the birdstrike event, and dry seals. A series of technology demonstrations on the F-15 would help WL transition products and would reduce the risk for the F-15 community's involvement in the program.

Wright Laboratory could not develop the prototype windshields for the technology demonstrations alone. WL worked with Sierracin Corporation through a research contract with the University Of Dayton Research Institute (UDRI) to design a cross-section similar to the A-7 and F-4 wrap-around prototypes. The cross-section was composed of two structural plies of

polycarbonate to provide the desired birdstrike resistance. Inner and outer plies of acrylic were added to maintain the current field polishing procedures (Figure 1). Two prototype panels were fabricated and bolted into F-15 A/B aluminum frames supplied by WR-ALC.

With two prototypes built, the technology demonstrations began. The first prototype underwent successful optical evaluation at Armstrong Laboratory according to the production specification. The prototype was then installed on aircraft F-15 101 by the 46 Test Wing, Eglin Air Force Base in September 1991. The flight evaluation would convince the F-15 community that laminated polycarbonate can withstand the fighter's demanding flight environment.

The second prototype underwent birdstrike testing at Arnold Engineering Development Center (AEDC), Arnold Air Force Base. The F-15 Systems Program Office and McDonnell Douglas Corporation provided test assets such as the forward fuselage, Head-Up Displays (HUD), and support structure. The windshield was impacted at the high quarter at 498 knots with a four pound bird. The panel contacted the wide field-of-view HUD combiner glass which had caused failure in the production acrylic windshield The polycarbonate windshield panel remained structurally intact, but the aft arch deflected 1.5 inches allowing bird to penetrate the cockpit. In addition, the production acrylic canopy and the HUD combiner shattered. The test proved that the F-15 could have 500 knots protection for the windshield with a properly designed aft arch. It also highlighted the need for an Advanced Canopy and a "Tough" HUD that would compliment the birdstrike characteristics of the Advanced Windshield.

The initial technology demonstrations helped convince WR-ALC to invest $1.8 million into designing a production version of the Advanced Windshield. The Joint Aircraft Transparency Technology Insertion Center assisted WR-ALC in writing the Statement of Work to ensure that the Advanced Windshield would meet performance and logistical requirements. Sierracin was awarded the development contract which included the responsibility to develop and manufacture the entire frame. Sierracin chose UDRI as a consultant to design the aft arch. Wright Laboratory would accomplish the qualification testing with the help of the 46th Test Wing and AEDC. At this point, the individual organizations had invested considerable assets in the F-15 Advanced Transparency Program and were committed to ensuring a team success.

The development contract began with improving the prototype design. Several arches were proposed that would decrease the 1.5 inch deflection experienced with the prototype in the aluminum frame. MAGNA and X3D, finite element codes specifically designed for analyzing bird impact, were used to evaluate each arch geometry and material combination. The team selected the cross-section shown in (Figure 2) which did not interfere with the

pilot line of sight and enabled the panel to be lifted out for quick change out. Nickel 718 alloy (Inconel) was chosen for the arch material because the analysis predicted it would limit arch deflection to 0.6 inches, providing the best chance of clearing the HUD and protecting the canopy (Figure 3). The desire to avoid HUD combiner glass spall and canopy fracture warranted the extra cost and machining time required for the nickel arch.

Along with the birdstrike performance goal, the team emphasized supportability requirements on an equal basis in the design of the Advanced Windshield. The design featured a dry seal applied to the panel at the factory that eliminated the long aircraft downtime for sealant curing. Bushings and nut plates were incorporated to achieve the quick change out and interchangeability goals.

With the preliminary design selected, the team moved on with fabrication of test units and 500 knot birdstrike qualification testing. The new assembly was impacted at 503 knots with a four pound bird resulting in a failure of the polycarbonate panel in tension along the bolt line. The nickel arch sustained no apparent damage, but the canopy and HUD were shattered. A post test analysis indicated that the arch began to deflect until the panel tore. The team needed to shift emphasis in the design to prevent the catastrophic failure mode.

The arch was redesigned so that a "safe" failure mode controlled the strength of the panel edge bolted to the arch. Originally, the team's design philosophy was to optimize the edge strength such that all failure modes (tensile failure of the panel, fastener shear failure, and shear tear-out) were equally likely. The number of fasteners in the arch was reduced so that fastener shear failure, the "safest" or least catastrophic mode, would occur first. If fasteners failed, energy would be dissipated before shear tear-out would occur. Tensile failure of the panel would occur last after two energy absorbing failure modes. To further reduce risk of catastrophic failure, the team revised the arch with a longer bottom flange to lower loads in the edge attachment.

The redesigned Advanced Windshield recently passed the birdstrike resistance performance goal. The redesigned nickel arch succeeded in supporting the laminated polycarbonate panel during a 506 knot birdstrike. In fact the arch limited deflection enough to protect the production canopy from damage. However, the arch could not limit deflection enough to prevent contact with the HUD as predicted in the analysis. The HUD combiner spalled and further emphasized the need for a Tough HUD.

Since the F-15 Advanced Transparency Team had such success at 506 knots, a second assembly was impacted at 544 knots. Again, the windshield passed with little structural damage and protected the production canopy. To solve the problem of combiner glass spall from the previous test, the Team applied a

urethane liner, 5300 by PPG Industries, to the HUD combiner. In theory, the liner would act as a safety net to contain glass spall. Although the liner deflected some debris, glass still spalled in the cockpit. A safer solution must be found that provides complete protection and does not interfere with optical and durability requirements of the HUD.

After meeting and exceeding the birdstrike resistance performance goal, the team pressed ahead with other milestones in the development contract. A fit check of the Advanced Windshield was conducted to verify sill, canopy, and external de-ice duct interfaces. A kit proof will follow to demonstrate quick change out, interchangeability, and Technical Order procedures. The Advanced Windshield will conclude the development contract with flight evaluation at operational environments such as Eglin, Luke, and Elmendorf Air Force Bases.

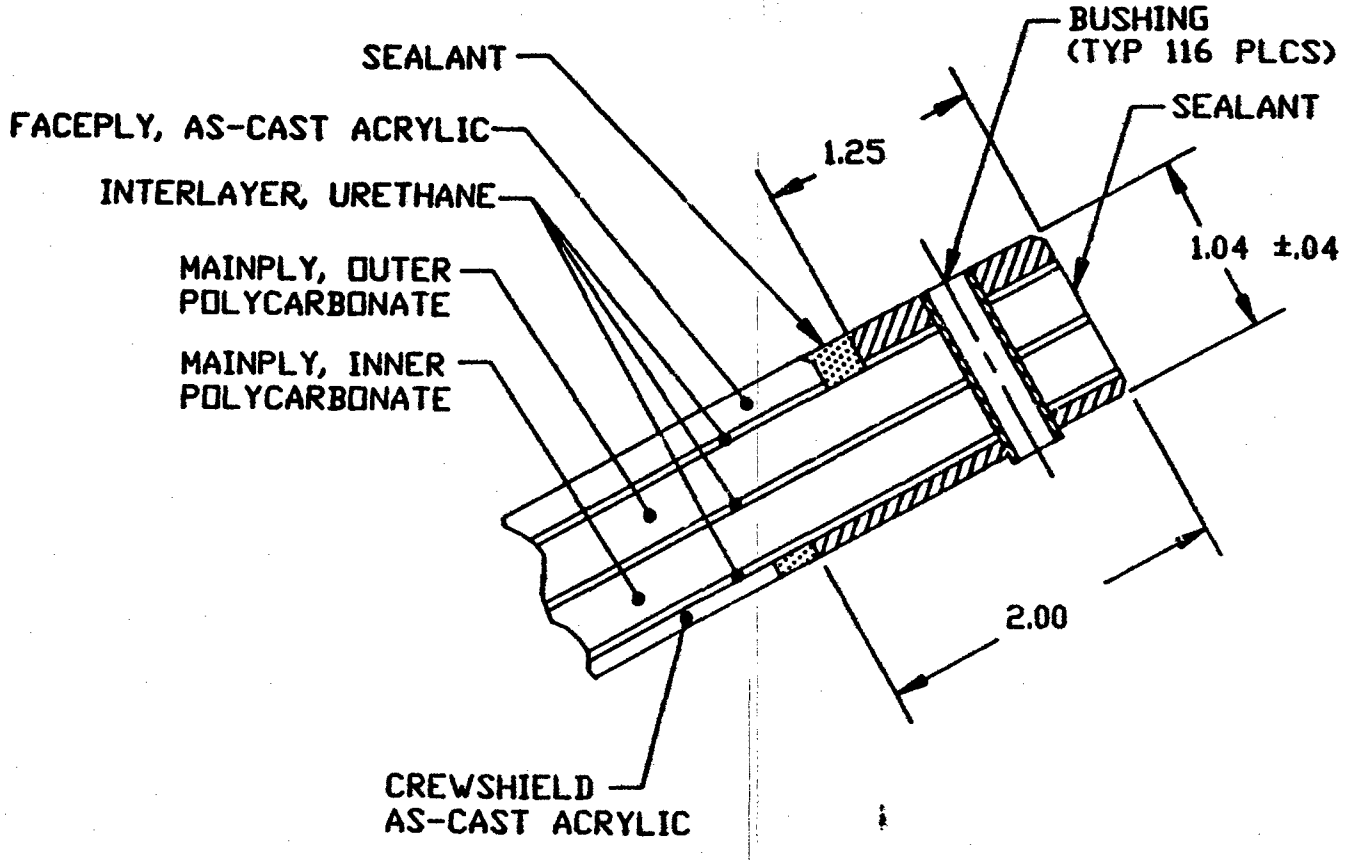
The F-15 Advanced Transparency Team's efforts have broadened beyond the development contract. The Team has focused on improving the durability of the Advanced Windshield from harsh environmental factors such as sand abrasion and rain erosion. Several proven durability coatings were applied to production parts for flight evaluation. If the coatings increase the service life of the production windshield, they will be incorporated into the design of the Advanced Windshield.

For the future, the Team will tackle the Tough HUD and Advanced Canopy projects to complete the F-15 Advanced Transparency system. The canopy presents a challenge in that achieving increased birdstrike resistance will affect the egress requirement for ejection through the canopy. Texstar Incorporated has joined the Team to fabricate prototype laminated polycarbonate canopies. Soon, the Team will demonstrate the maximum birdstrike protection possible without increasing system weight. The Advanced Canopy project will finish with an investigation of methods to balance aircrew protection from birdstrike and safe egress.

Summary

The F-15 Advanced Transparency Team will soon see their investment begin to payoff in terms of increased performance and reduced cost. The advanced windshield increases birdstrike protection to 544 knots for birds up to four pounds. The advanced windshield will be available as a preferred spare for $26 thousand including the frame and kit compared to the current price of $35 thousand. Follow on replacement panels will be $12 thousand and could easily incorporate a durability coating. Change out time will be five man-hours compared to the current time of 56 man-hours. Interchangeable panels will reduce the logistical cost of the aircraft and increase war-fighting capability.

In the future, the F-15 Advanced Transparency Team will concentrate on improving the Advanced Windshield and complementing it with a Tough HUD and Advanced Canopy. The performance of the Advanced Windshield can be improved with a durability coating and a high temperature outer ply. To provide complete mission compatible birdstrike protection to the aircrew, the HUD combiner and the canopy must be able to withstand the forces associated with a 544 knot bird impact. These future challenges are formidable, but the F-15 Advanced Transparency Team has developed an approach based upon investment, commitment, and cooperation that ensures success.



CROSS SECTION AFT ARCH

Figure 1

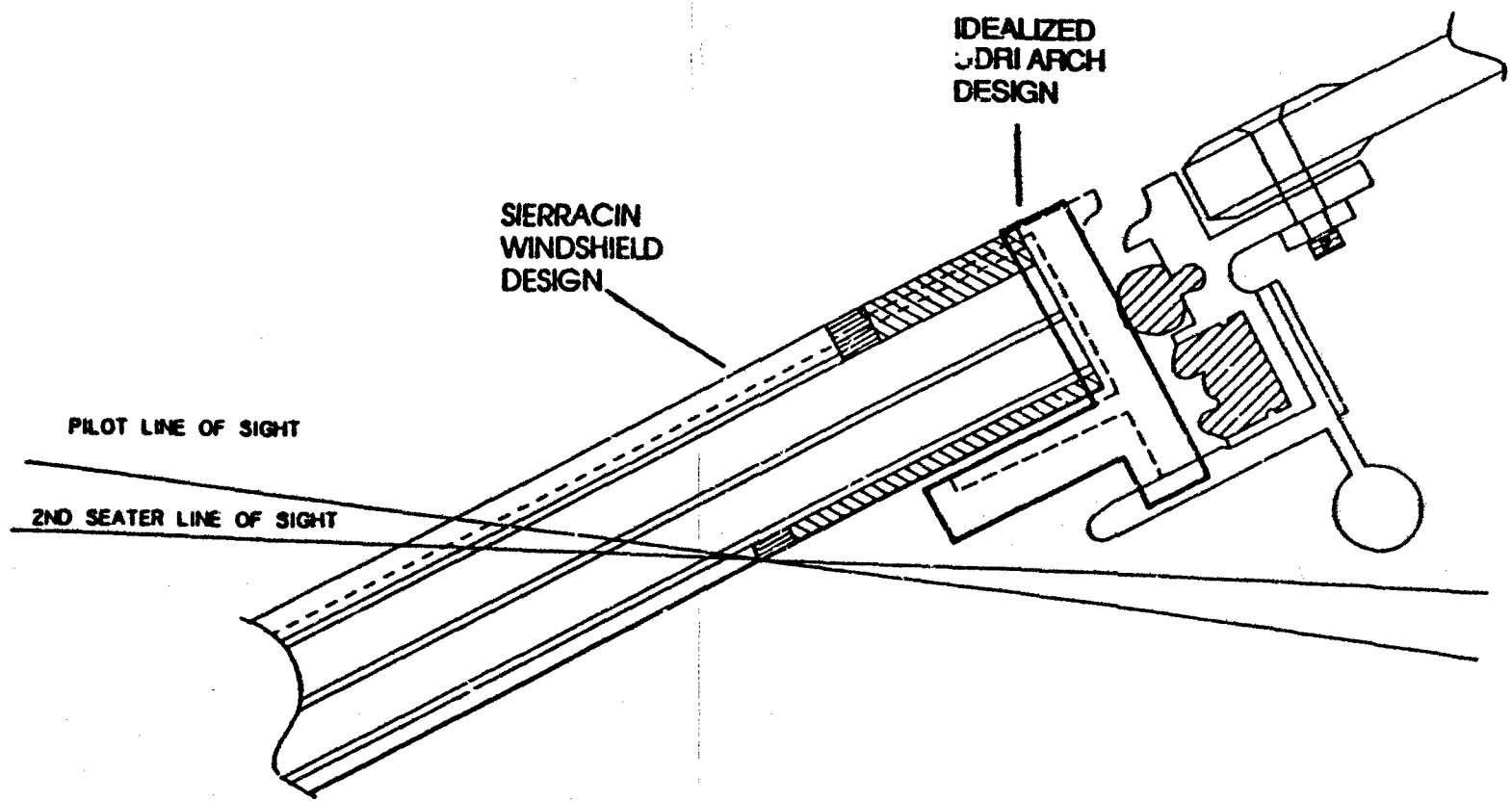
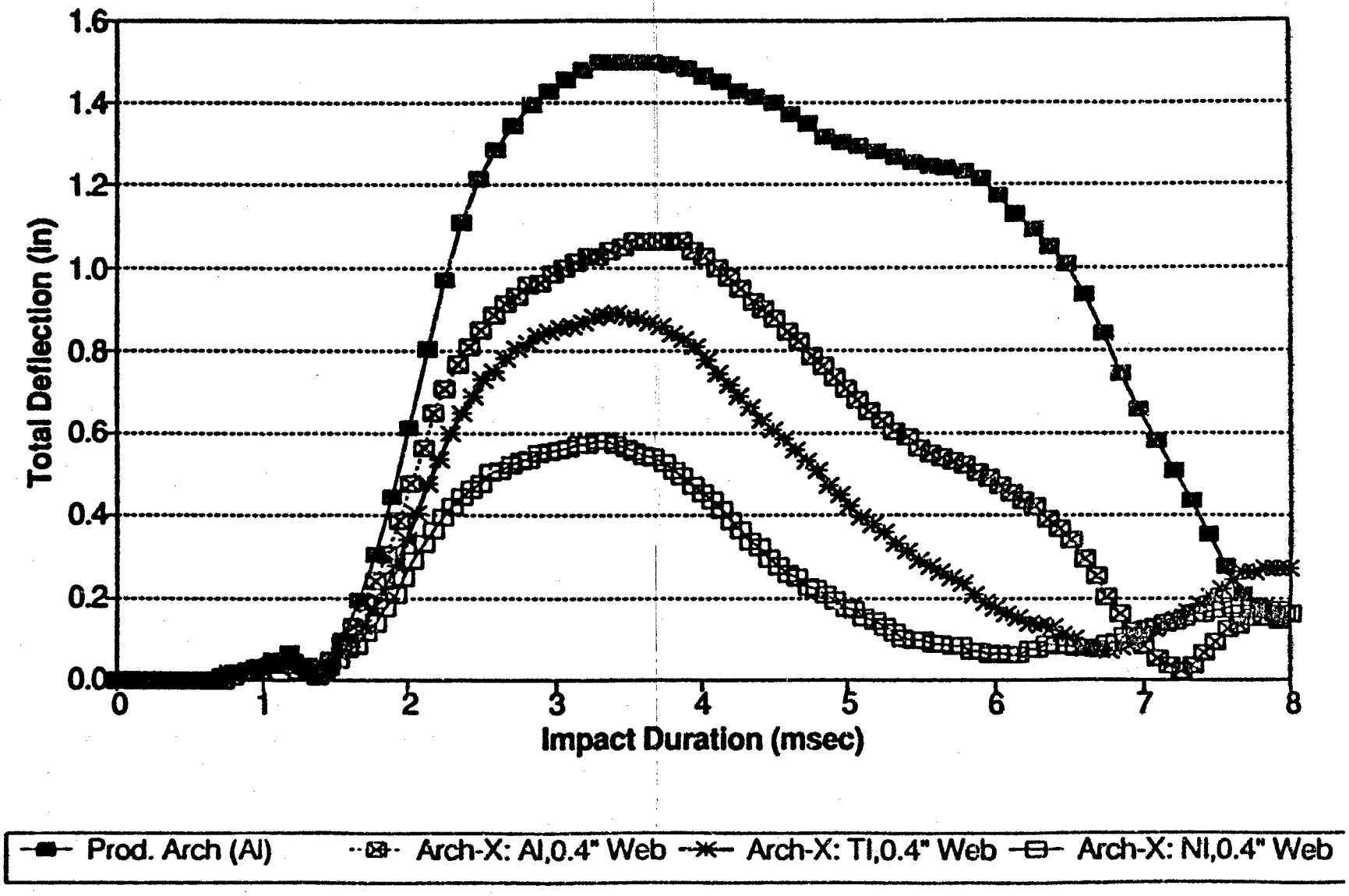


Figure 2. Arch Cross Section

Figure 3. Deflection Time History Comparison



T-38 BIRD IMPACT RESISTANT COMPOSITE WINDSHIELD FRAME DEVELOPMENT

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ABSTRACT

Due to increased high-speed flight training at low altitudes, T-38 aircraft flight crews need additional protection from windshield bird impacts. A windshield providing protection to 400 knots from a 4-lb bird impact has already been developed, but a compatible frame is needed to support the windshield during the bird-impact event while also improving flight crew visibility and frame durability. This paper covers the work to date in developing a composite windshield frame to meet flight crew needs. The frame developed for this program was the result of beam coupon and full-scale frame evaluations of numerous hybrid laminates of materials including, graphite, Kevlar, S-2 fiberglass, and stainless steel sheet. High strain rate testing of the hybrid laminated beam coupons was performed to identify mechanical properties for use in a finite element model used to perform an anaiysis of different configurations. Full-scale bird-impact testing was then performed on selected configurations. Testing showed the best design to be a S-2 fiberglass and stainless steel sheet laminate. Prototypes of the frame have been completed and are being installed on T-38 aircraft for flight evaluation by the United States Air Force and the National Aeronautics and Space Administration.

INTRODUCTION

Due to increasing low altitude high speed flight training, T-38 aircraft have incurred an increased number of windshield bird strikes and penetrations. As a result of this increased danger to the T-38 flight crews, the Air Education and Training Command (AETC) needed an improved windshield system to meet mission and flight safety needs. The Wright Laboratories Improved Windshield System Program Office (WL/FIVR) developed and flight tested an improved windshield system which met the flight safety requirements. The WL/FIVR windshield system consisted of a tapered hybrid composite reinforced magnesium frame and a PPG Industries developed laminated polycarbonate with PPG 5300 outboard liner windshield. The tapered hybrid composite

reinforcement developed by the University of Dayton Research Institute for the WL/FIVR reinforced the arch of the existing magnesium T-38 windshield frame. The hybrid composite consisted of a mix of Kevlar and S-2 fiberglass in an epoxy matrix; was tapered from the thickest point at the center of the arch to the thinnest at each side on the arch frame interface, and was bonded and bolted to the existing arch of the windshield frame. This system successfully meet AETC's requirement of withstanding a 4-lb bird impact at 400 knots (reference 6). However, flight testing of the system showed the hybrid composite reinforcement restricted the instructor pilots forward visibility. Also, corrosion of the aging cast magnesium frames forced the T-38 System Program Management Office (SA-ALC/LAS) to investigate replacing all T-38 magnesium frames.

The SA-ALC/LAS initiated an effort to develop a composite frame to replace the hybrid composite reinforced magnesium frame. The effort focused on developing a replacement frame that would work with the already developed improved windshield while improving pilot visibility. Full scale composite frames with constant cross-section arches of Kevlar in an epoxy matrix and Kevlar, S-2 fiberglass, and graphite in an epoxy matrix were developed and proved unsuccessful. A joint effort between the USAF Advanced Composites Program Office (ACPO), WL/FIVR, SA-ALC, and AETC was then initiated to develop a frame to meet the flight safety and mission needs.

This paper covers the work under this joint effort that successfully developed a bird-impact resistant composite windshield frame for the T-38 aircraft.

The active support of Mr Chris Szczepan, SA-ALC/LAS, is gratefully acknowledged. It is his never-ending efforts to provide the T-38 flight crews with improved flight safety that resulted in this program successfully developing a windshield system that increases the T-38's windshield bird-impact resistance.

DISCUSSION

The goals of this effort was to develop, prototype and test a composite windshield frame with an arch no thicker than 1.01 inches, require no modifications to the aircraft structure and would work with the already developed bird-impact resistant windshield to provide 400 knot 4-lb bird-impact resistance, minimize installation procedures, and improve damage tolerance and repairability compared to the existing cast magnesium frame. To accelerate the prototyping effort, the tooling developed under past T-38 composite frame efforts was used to manufacture the new composite frames (reference 4).

The approach used in developing a successful T-38 bird-impact resistant windshield frame was to first review past windshield arch designs, develop and test beam coupons representing arch cross-sections, perform a finite element analysis on cross-sections showing improvements over past designs, manufacture full-scale prototype frames of designs selected in the finite element

analysis, bird-impact test the selected frames, and then flight evaluate the frames that performed the best in the bird-impact testing.

Windshield Arch Design

To work efficiently with the windshield, the arch must be designed to yield just before the windshield is expected to fail. If the arch is too stiff, the transparency may tear out at the bolt holes or allow the bird to punch through. If the arch isn't stiff or strong enough the arch will deflect too much and allow bird debris to enter the cockpit and endanger the flight crew.

The development of the T-38 arch tapered hybrid composite reinforcement (reference 6) led to successful hybrid composite arches for the A-7 (reference 2) and F-4 (reference 3) aircraft. Both the A-7 and F-4 windshield systems using the hybrid composite tapered arches successfully withstood 4-lb bird impacts at speeds in excess of 480 knots. It initially appeared that these efforts provided enough information to quickly select the materials and manufacture T-38 prototype frames, however, both the A-7 and F-4 hybrid composite arches were over 1.5 inches thick and the T-38 arch couldn't be more than 1.01 inches thick. Thus, additional arch cross-sections of materials were evaluated to identify 1.0 inch thick beam coupons with mechanical properties that could meet the expected loading transmitted through the windshield to the frame.

To be successful, the arch needed an increased stiffness and strength over the two designs that failed to support the windshield in the earlier development effort. The new arch had to resist deflecting until the windshield had approached shear failure at the arch bolt holes. Once the windshield nears its maximum shear load capacity and the bird debris have passed over the canopy-arch interface, the arch needs to deflect and rotating to assist with absorbing and dissipating the load transferred from the windshield.

For a controlled deflection of the arch to occur, the arch had to be designed to yield first at the centerline. Past arch development efforts (reference 1) showed the stress in the structure away from the centerline should be less than 85% of the value at the arch centerline when the arch begins to deflect. The load the arch must dissipate was identified by first estimating the maximum shear load the windshield can withstand from a 8-inch bird-impact foot print:

$$P_w = T_u \, t \, w_b \qquad (1)$$

T_u = shear strength of windshield
t = thickness of windshield structural plies
w_b = width of bird footprint

For the T-38 windshield with a polycarbonate thickness of 0.6 inches and an 8 inch bird footprint, Eq 1 gives:

$$P_W = (5400 \text{ psi})\ (0.60 \text{ in})\ (8 \text{ in}) = 25{,}920 \text{ lb} \cong 26{,}000 \text{ lb}$$

Thus, the load required to yield the T-38 arch needed to be less than 26,000 lb.

BEAM COUPON SELECTION and TESTING

To identify improved arch cross-sections, a test program (reference 1) was conducted to characterize the mechanical properties of various thick hybrid composite materials to include known arch cross-sections of past failures. Twenty-five different laminate configurations were formed from combinations of Kevlar-49, S-2 glass, graphite, and 301 stainless steel in an epoxy matrix. Three coupons of each cross-section were loaded in bending using a four-point beam test. Test results included beam flexural strength, bending modulus, energy absorbed during loading, and failure mode of the laminate. The test results were evaluated to identify beam coupons with improved mechanical properties over the properties of the past failed cross-section coupons. The stacking sequence of the coupons, geometric and physical properties, and test results are identified in Appendix A.

Three beam coupons of each cross-section were fabricated into 1"x3"x14" laminates. These laminates were autoclave cured at 250 degrees F with 90 psi for two hours. Eight holes were drilled in each beam coupon to better simulate the actual arch cross-sections. In one set of 25 different laminated beam coupons 1/4 inch grade 5 bolts were used, 1/4 inch grade 8 bolts were used in another set of the 25 different samples, and 5/16 inch grade 8 bolts were used in the remaining set. The various bolts were used to identify changes in mechanical properties due to the different bolt strengths. Beam coupons with bolt placements are shown in Figure 1, with four point beam coupon test configuration and load fixture are illustrated in Figure 2. All tests were conducted at approximately 1000 in/min displacement rate to a total displacement of 2.5 inches.

The amount of deflection in the beam coupon needed to develop the similar stress from the maximum allowable full size arch deflection was determined to be 1.25 inches (reference 1). Table 5 in Appendix A shows the normalized energy absorbed for the coupons to deflect to 1.25 inches. Seven arch cross-sections showed increased energy absorption compared to the AFA beam coupons which were the best of the past failed arch cross-sections. Five of these cross-sections were selected for further evaluation.

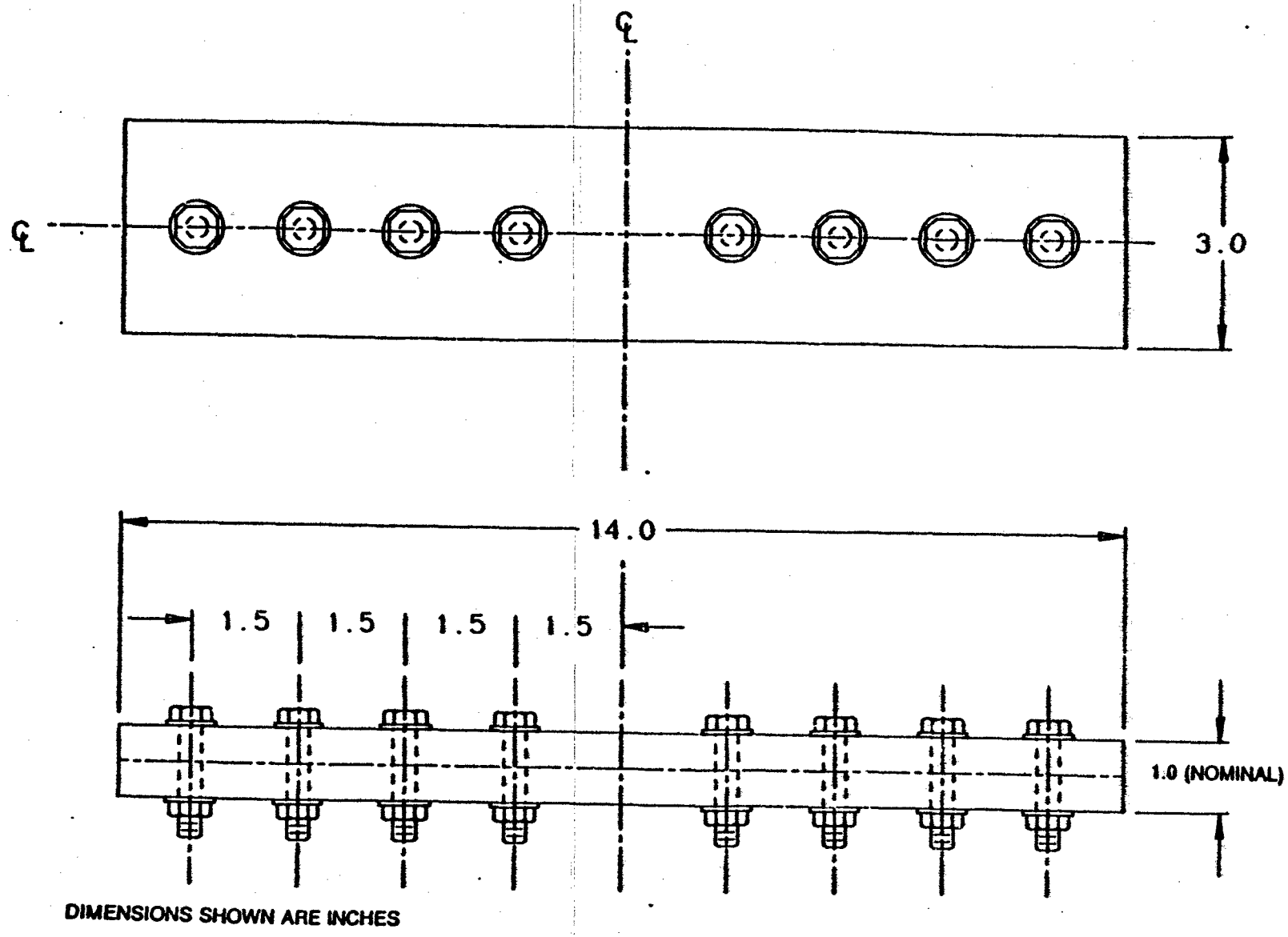


Figure 1. Bolt Locations in Hybrid Composite Specimens.

4 POINT BENDING TEST

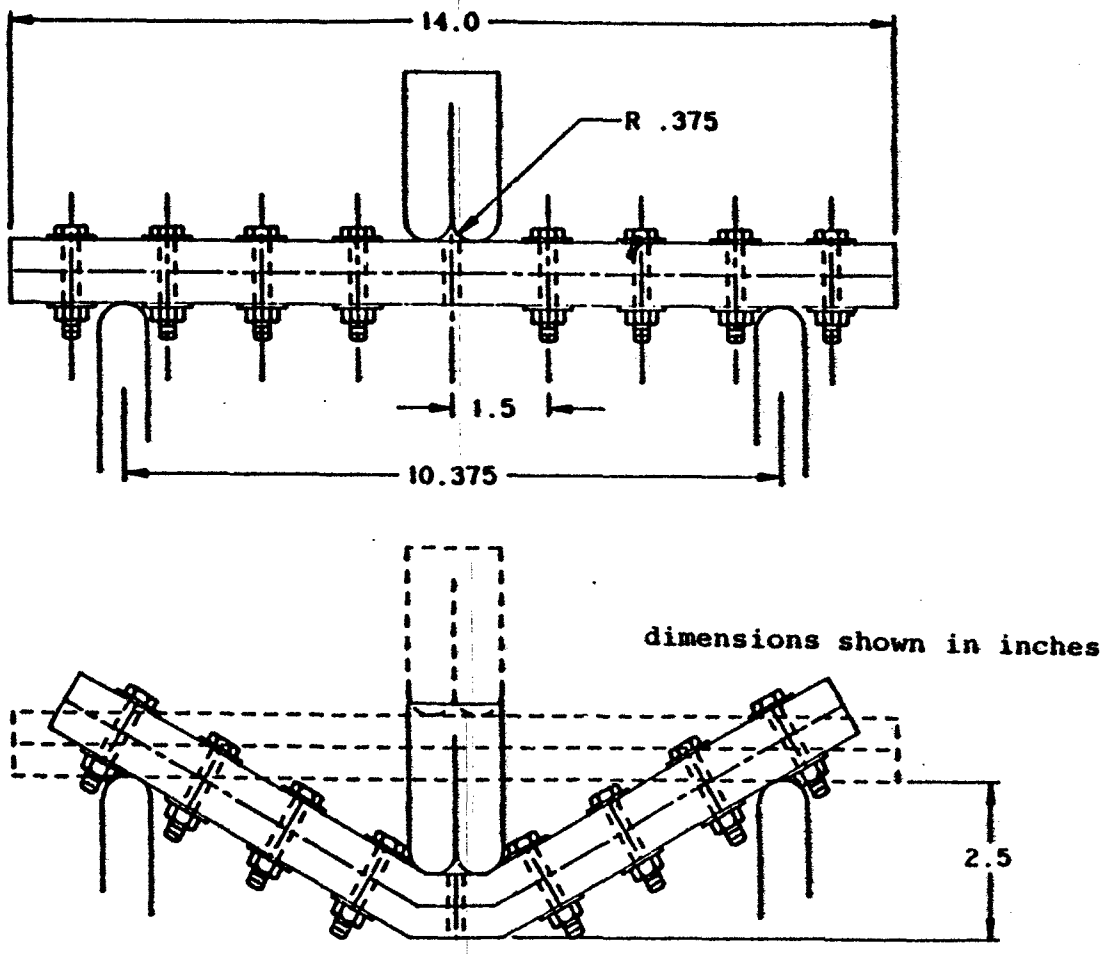


Fig. 2 Four Point Bend Test Configuration

FINITE ELEMENT ANALYSIS of SELECTED LAMINATES

A nonlinear, static finite element model of a simulated T-38 arch was used for the analysis. The analysis was performed by the University of Dayton Research Institute, UDRI, under contract from WL/FIVR. The model allowed the effects of different properties of the five laminates and tapering the arch to be evaluated based on a variety of criteria. Analysis criteria included: arch strength, energy absorption, centerline deflection and rotation, failure location, and reactions at the sill attachment. The results of the finite element analysis of the prototype composite arch were compared with baseline cases developed from arch systems which had been previously tested.

The finite element method was used for the modeling and analysis of the aft arch system. The computer program MAGNA was used because of its nonlinear analysis capabilities and familiarity to UDRI. A nonlinear, static model of the structure was used for all of the analyses.

Figure 3 depicts the finite element model of the T-38 aft arch system. Directions for loads, displacements and rotations used in the remainder of this paper refer to the coordinate system shown in this figure. Due to symmetry of the arch, only one-half of the arch was modeled. The model consisted of three-node curved beam elements.

Boundary conditions were set by centerline symmetry and sill interface requirements. Only v-displacements and θ_X rotations were permitted at the centerline. To model the θ_Z rotational stiffness of the structure at the sill, a spring was included one element away from the sill attachment. The spring had a stiffness of 2000 lb/in and connected nodes 37 and 40 as shown in Figure 3.

Loads consisted of equal-magnitude vertical nodal loads applied at the 6 nodes closest to the centerline which correspond with the 4 inch half width of the bird footprint. For all runs except baseline 4 (where plasticity was included) the loads were applied incrementally until the highest axial stress in the arch exceeded the ultimate stress for the laminated material.

Five finite element cases of past T-38 bird-impact tested frames were run for baseline results and 22 cases were run to study the 5 hybrid composites selected in the beam-coupon testing. Baseline case 1 was the original magnesium arch. Baseline cases 2 and 4 were a 4130 steel tube reinforced magnesium arch. Baseline cases 3 and 5 evaluated the composite reinforced magnesium arch. Appendix B gives the centerline section properties of the various baseline and prototype arches. As shown in reference 1, the prototype arch cases analyzed the effects of arch taper on each of the 5 hybrid composite materials. Figure 4 shows the regions of taper used in the analysis.

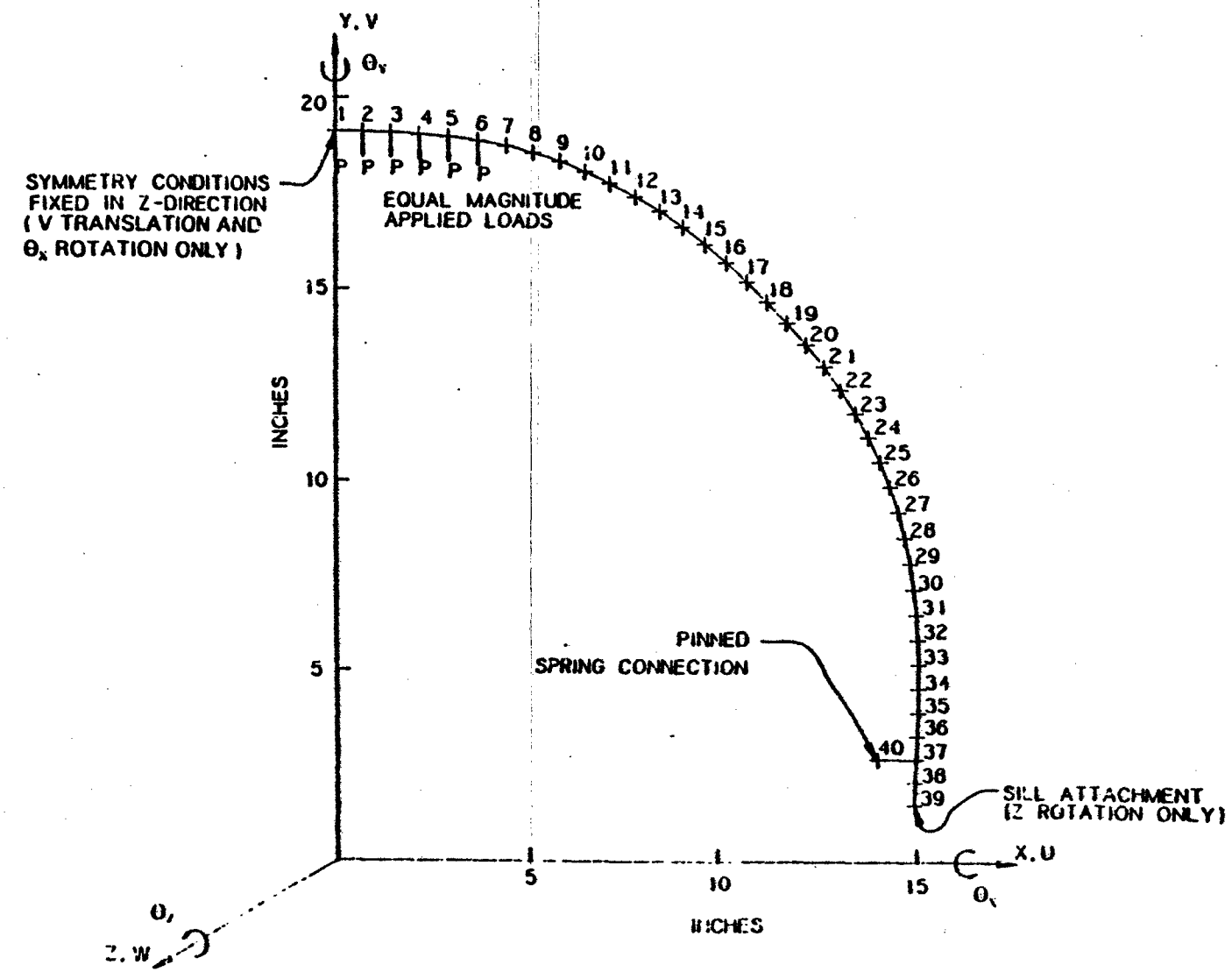


Figure 3. Beam Element Model of T-38 Windshield Aft Arch.

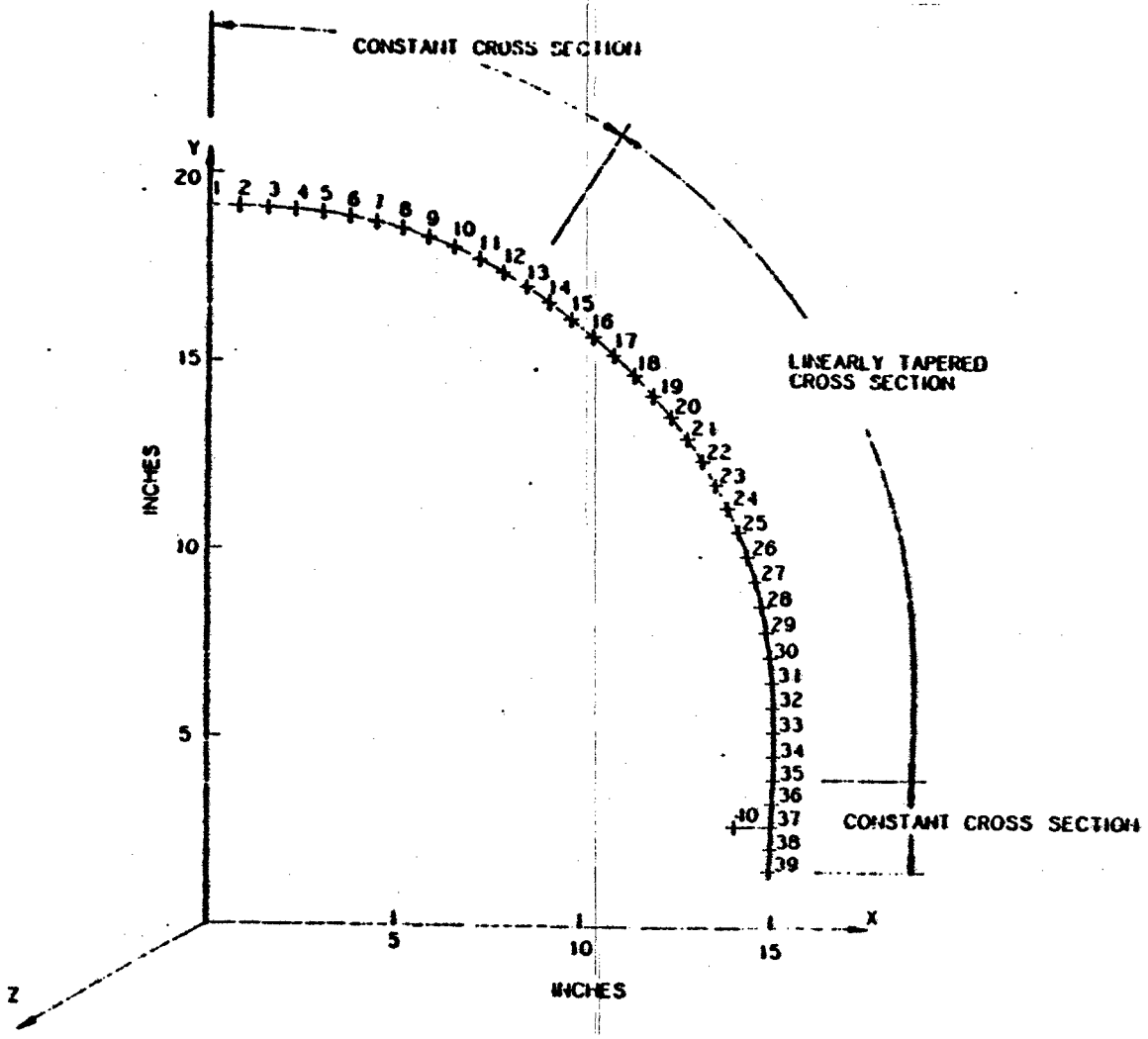


Figure 4. Region of Linear Taper for Baseline Cases 3 and 5 and Prototype Arch.

EVALUATION RESULTS

Twenty-seven cases were analyzed with the finite element method. All cases except baseline case 4 were run until the axial stress at the most highly stressed location on the arch reached the allowable stress of the material. Baseline case 4 was run until a 1-inch deflection was achieved at the centerline. The values of the analysis for the prototype arches were compared to the analysis values in baseline 4 and 5.

Values at peak load are summarized in Table 1. This table shows the forces acting on the windshield and fuselage at the load required to yield the arch. The peak load was used to determine the maximum load which the windshield had to carry for each case. These values are lower than the 26,000 lbs which was the maximum expected shear strength of the transparency.

The best measure for predicting the performance of an arch during bird-impact is the total energy required to deflect the arch. Since an arch deflection of 3 inches was expected, the strain energy to deflect the arch by this amount was chosen for use in comparing the different cases. As shown in Table 1, only materials PF and TF had an absorbed energy at 3 inches of deflection which exceeded that of the composite reinforcement (baseline case 5).

In order to ensure the arch yields initially at the centerline, it is desirable to limit the stress at all other places around the arch below some fraction of the centerline stress. Since the composite reinforcement (baseline case 5) showed an acceptable yielding pattern in bird-strike tests, the stress distribution for this case, shown in Figure 5, was used as a baseline for the prototype arches. Comparison of Figure 6 with Figure 5 shows the stresses at the aft outer location for all five tapers of material PF were within the stress envelope of baseline case 5. Comparison of Figure 7 with Figure 5 shows the stresses at the forward inner location for material PF cases which taper to less than 0.875 inch at the sill exceed the stress distribution of baseline case 5. Thus, large amounts of taper could cause the arch to fail away from the centerline.

SELECTION of ARCH DESIGNS

Based on the finite element analysis, prototype arches from materials TF and PF were recommended for full-scale evaluation. The analysis showed arches of materials TF and PF failed non-catastrophically, and absorbed a large amount of energy after first ply failure. To reduce the loads reacted by the fuselage, the arches were recommended to be tapered to 0.875 inch at the sill. It was assumed that the arch made of material TF would behave in a similar manner to the steel-tube reinforced magnesium arch. The increased mass of material TF relative to the steel-tube reinforcement was expected to reduce the amount of deflection relative to the steel reinforcement. While material PF was thinner than the composite reinforced magnesium arch the analysis suggested that material PF would perform in a similar manner as the composite reinforcement.

TABLE 1
RESULTS SUMMARY FOR T38 ARCH DESIGN CASES

		-------- EXTRAPOLATED ULTIMATE VALUE ---------								
					--- SILL REACTIONS** ---					
		PEAK LOAD*	DISPL.	ROT.	Fx	Fy	Fz	Mz	ABSORBED ENERGY AT YIELD #	ABSORBED ENERGY AT 3.0" ##
		(LB)	(IN)	(DEG)	(LB)	(LB)	(LB)	(IN-LB)	(IN-LB)	(IN-LB)
BASELINE:										
1 - MAGNESIUM ARCH		1400	0.29	0.49	-272	700	180	52	195	--
2 - STEEL REINFORCEMENT (load to yield)		6320	0.23	0.07	-1443	3160	23	62	729	16489
3 - COMPOSITE REINFORCMENT (loaded at fwd. edge of Mg arch)		11440	1.43	-0.15	-2488	5720	984	440	8600	21486
4 - STEEL REINFORCEMENT (elas-plas to 1 in. deflect)		11810	1.00	0.18	-3074	5905	43	141	7230	28486
5 - COMPOSITE REINFORCMENT (loaded at fwd. edge of composite)		11350	1.63	2.03	-2037	5675	1523	476	9660	20824
PROPOSED:										
MATERIAL	TAPER***									
PF	NONE	15440	1.71	4.83	-3282	7720	1837	361	13760	21729
	0.875	14170	1.82	4.17	-2681	7085	1481	453	13990	20673
	0.750	12930	1.98	3.76	-2193	6465	1223	626	13830	19111
	0.625	11370	2.20	3.08	-1455	5685	810	904	13580	17215
	0.500	9850	2.50	1.84	-700	4925	612	1249	13720	15684
KF	NONE	11280	1.14	2.56	-2638	5640	1284	246	6520	20153
	0.875	10350	1.23	2.24	-2098	5175	1125	321	6500	18411
	0.750	9430	1.31	1.96	-1677	4715	837	398	6400	15979
	0.625	8310	1.46	1.60	-1174	4155	595	608	6380	14700
	0.500	7300	1.64	0.91	-667	3650	453	805	6390	12841

(continued)

TABLE 1 (cont.)

MATERIAL	TAPER***	EXTRAPOLATED ULTIMATE VALUE — PEAK LOAD* (LB)	DISPL. (IN)	ROT. (DEG)	SILL REACTIONS** Fx (LB)	Fy (LB)	Fz (LB)	Mz (IN-LB)	ABSORBED ENERGY AT YIELD # (IN-LB)	ABSORBED ENERGY AT 3.0" ## (IN-LB)
VF	NONE	9440	1.14	2.03	-2006	4720	1093	247	5430	19129
	0.875	8660	1.23	1.75	-1692	4330	947	322	5440	17401
	0.750	7930	1.31	1.45	-1371	3965	708	393	5310	15773
	0.625	6950	1.45	1.07	-927	3475	486	587	5260	13656
	0.500	6060	1.66	0.44	-360	3030	385	901	5370	11705
AFA	NONE	8910	0.97	2.22	-1926	4455	993	208	4330	15723
	0.875	8180	1.14	1.94	-1823	4090	956	269	4320	13899
	0.750	7500	1.11	1.70	-1339	3750	650	329	4230	13158
	0.625	6600	1.23	1.43	-939	3300	458	486	4140	11501
	0.500	5400	1.33	0.83	-550	2700	330	563	3910	9590
TF	NONE	13840	1.20	2.69	-3014	6920	1574	257	8430	27594
	0.750	11560	1.39	2.05	-2111	5780	1045	425	8330	22651

* Load at initial yield.
** See Figure 4 for reference coordinates.
*** Values indicate thickness of tapered cross-section at sill.
# Energy absorbed at first ply failure.
Energy absorbed after 3-inch displacement.

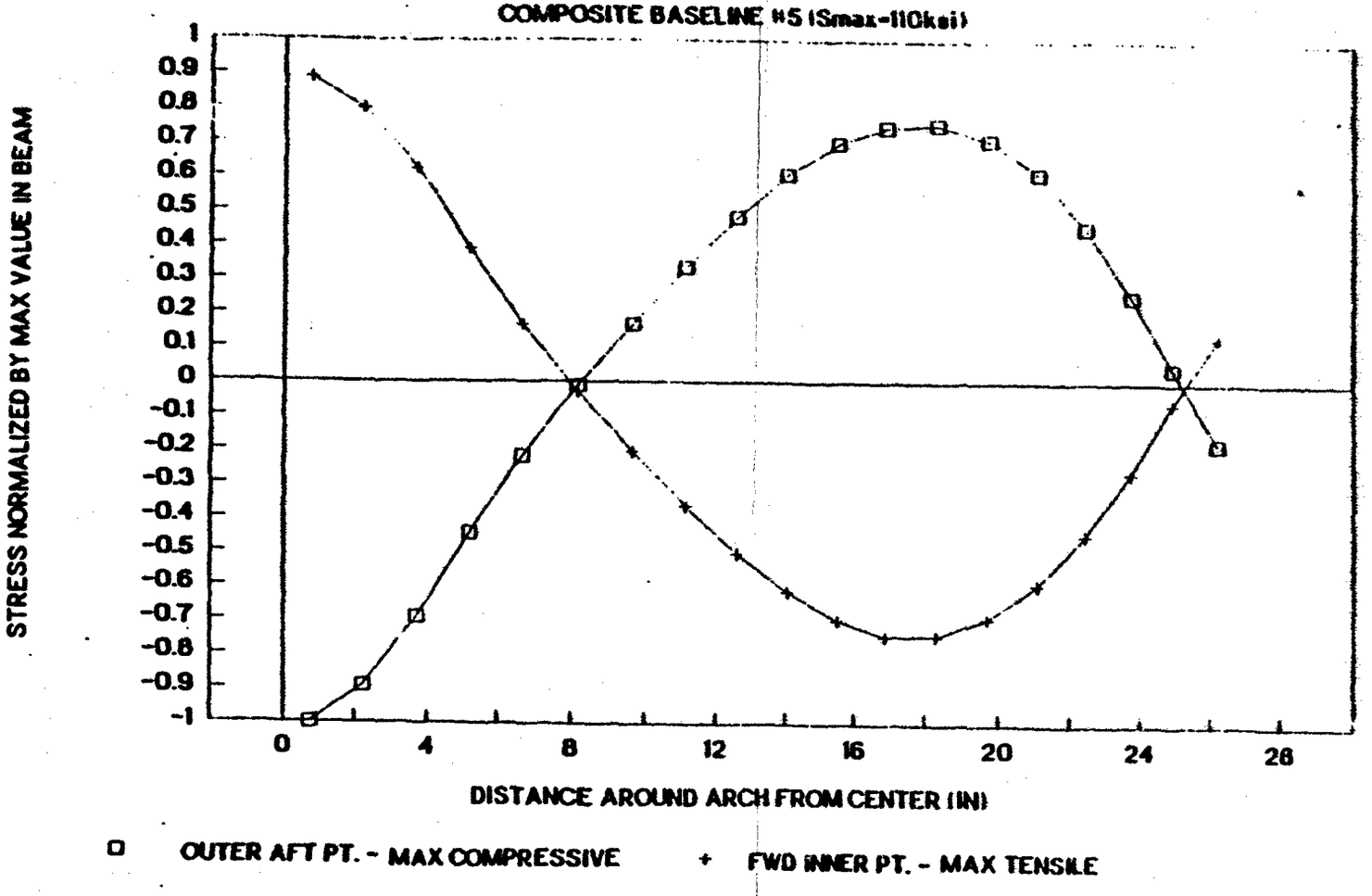


Figure 5. Axial Stress Distribution at Onset of Yield for T-38 Arch Composite Reinforcement. Stresses are at the outer aft corner (max. compression) and inner forward corner (max. tensile) on the reinforcement.

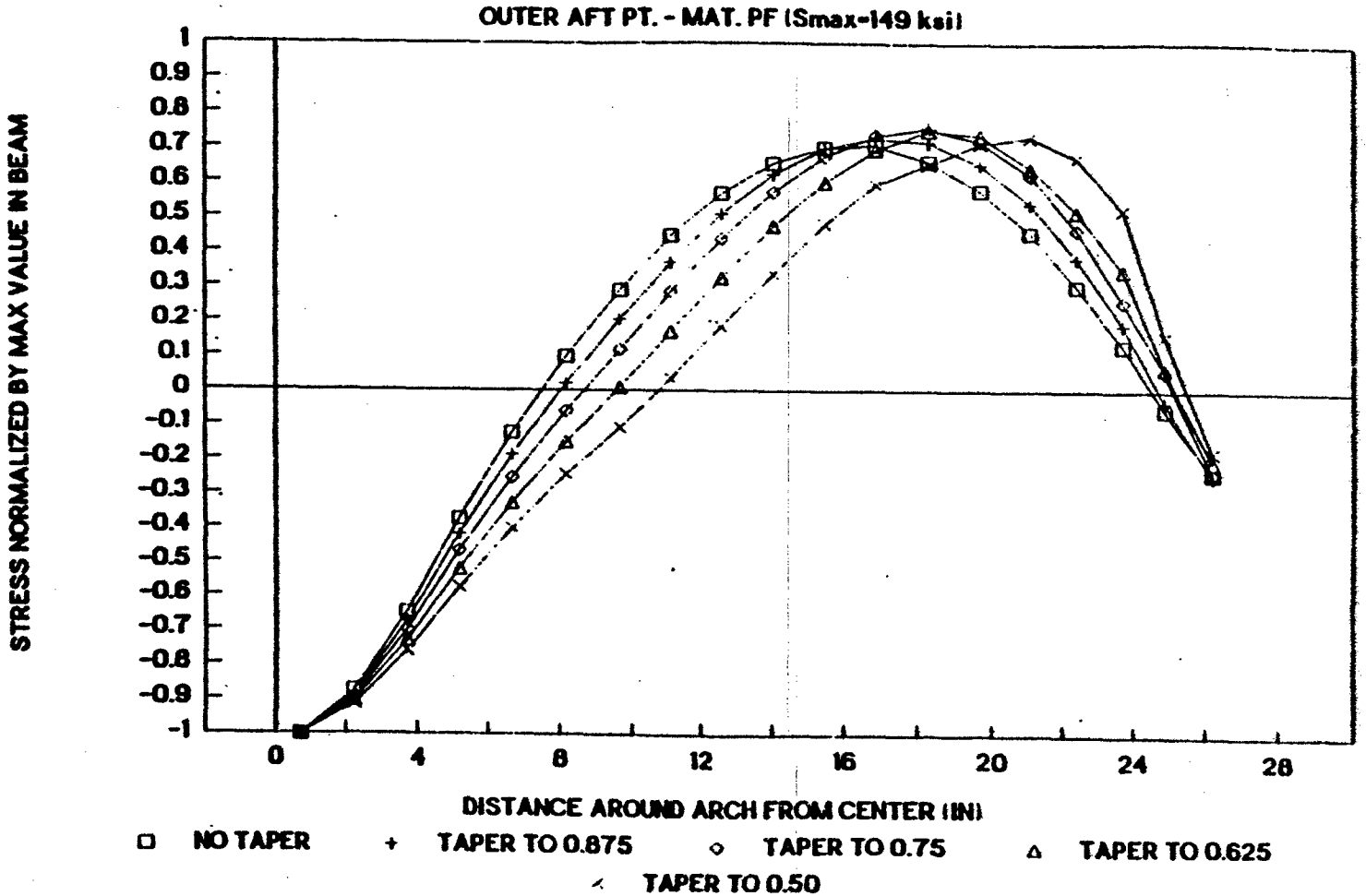


Figure 6. Axial Stress Distribution at Onset of Yield for Material PF Prototype Arches. Stresses are at the outermost aft corner of the arch.

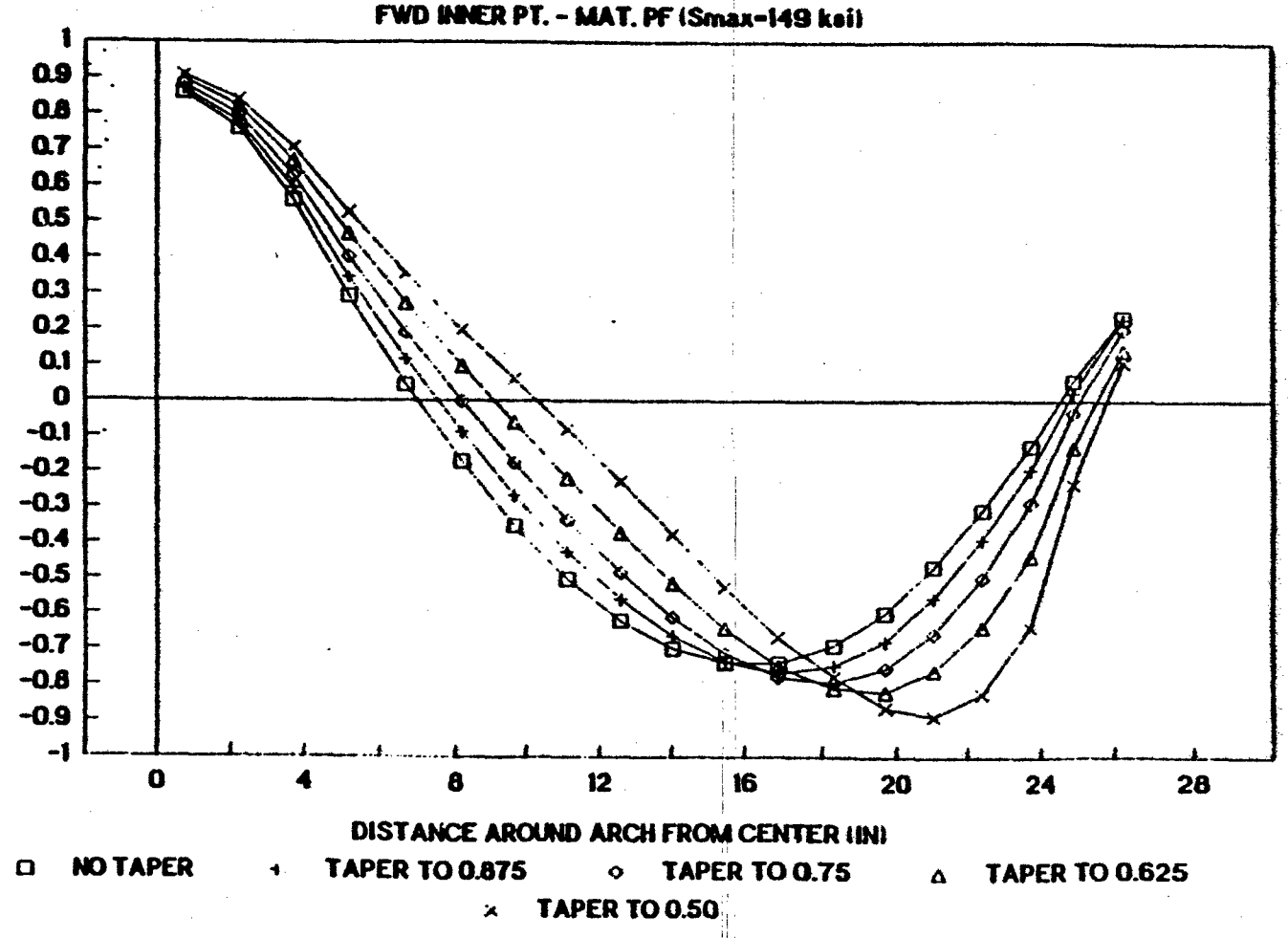


Figure 7. Axial Stress Distribution at Onset of Yield for Material PF Prototype Arches. Stresses are at the innermost forward corner of the arch.

One of the goals of this program was to maximize the use of already developed tooling. The tooling provided was designed for basically a constant cross-section arch. Thus, further analysis (reference 1) was performed to identify a basically constant-geometrically-shaped arch which had a tapering stiffness. The analysis indicated that replacing some of the outer 0° plies with 90° plies would reduce the stiffness by the required amount, but would also cause a reduction in strength. Only bird-impact testing would show how the reduction in strength would affect the arch performance.

WINDSHIELD FRAME MANUFACTURING

From the finite element analysis, modified arch designs of PF (named PFL) and TF (named PFH) were manufactured at the Sacramento Air Logistics Center, McClellan AFB, California (SM-ALC). As noted earlier, the designs were modified to taper the stiffness of the arch without geometrically tapering the arch. The designs were also modified to allow for reduction in the number of materials required for manufacturing. As a result, all unidirectional material was replaced with fabric material and all Kevlar material was replaced with S-2 fabric material. The frames were manufactured in the same manner as the earlier tested composite frames (reference 4). The SM-ALC modified the existing tooling provided and the designs of the metal hardware for the frame to correct for fit problems, ease the manufacturing process, and allow for easier field repair when needed.

BIRD IMPACT TESTING

Bird-impact testing of frame designs PFL and PFH were conducted at PPG Industries, Huntsville, Alabama (reference 5). Transparencies of the already successfully 400-knot bird-impacted and flight tested PPG 5300 liner/ polycarbonate design were installed on the frames. The windshield system was then installed on a T-38 forward fuselage section for bird-impact testing. New acrylic panels were placed in the student-pilot canopy frames used and repaired in past bird-impact testing. Bird-impact location was on the windshield centerline at 9-inches forward of the aft arch edge. The windshield installation angle of 27.5 degrees was fixed to represent the aircraft in a straight-and-level attitude. Camera triangulation data was used to develop arch and windshield deflection data. Testing was performed with 4-lb birds at 402 knots.

The windshield frame with the PFH arch design performed the best with maximum deflection of the windshield system at approximately 3.5 inches (Figure 9) and no bird debris in cockpit. The PFL arch windshield frame design performed in a similar manner as the composite reinforced arch magnesium frame with maximum windshield deflection being approximately 4.0 inches (Figure 8) and allowed some bird debris in the cockpit. The student pilot canopies in both test incurred damage. Testing film footage showed the canopy damage occurred when the arch rebounded from the deflection and not from any bird debris impacting the canopy. The canopy frame used in this testing

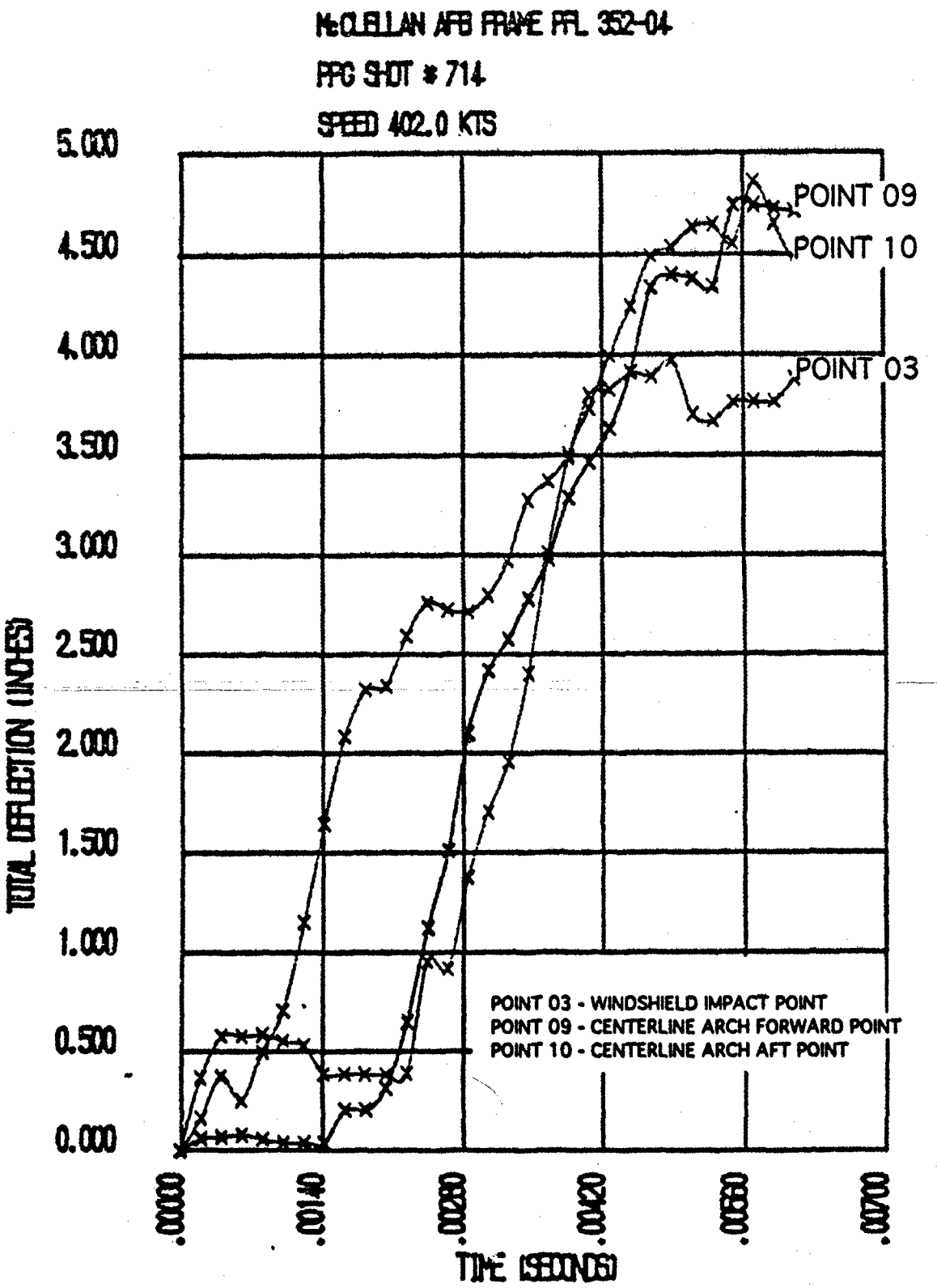


Figure 8. FRAME PFL BIRD IMPACT RESULTS
DEFLECTION vs TIME

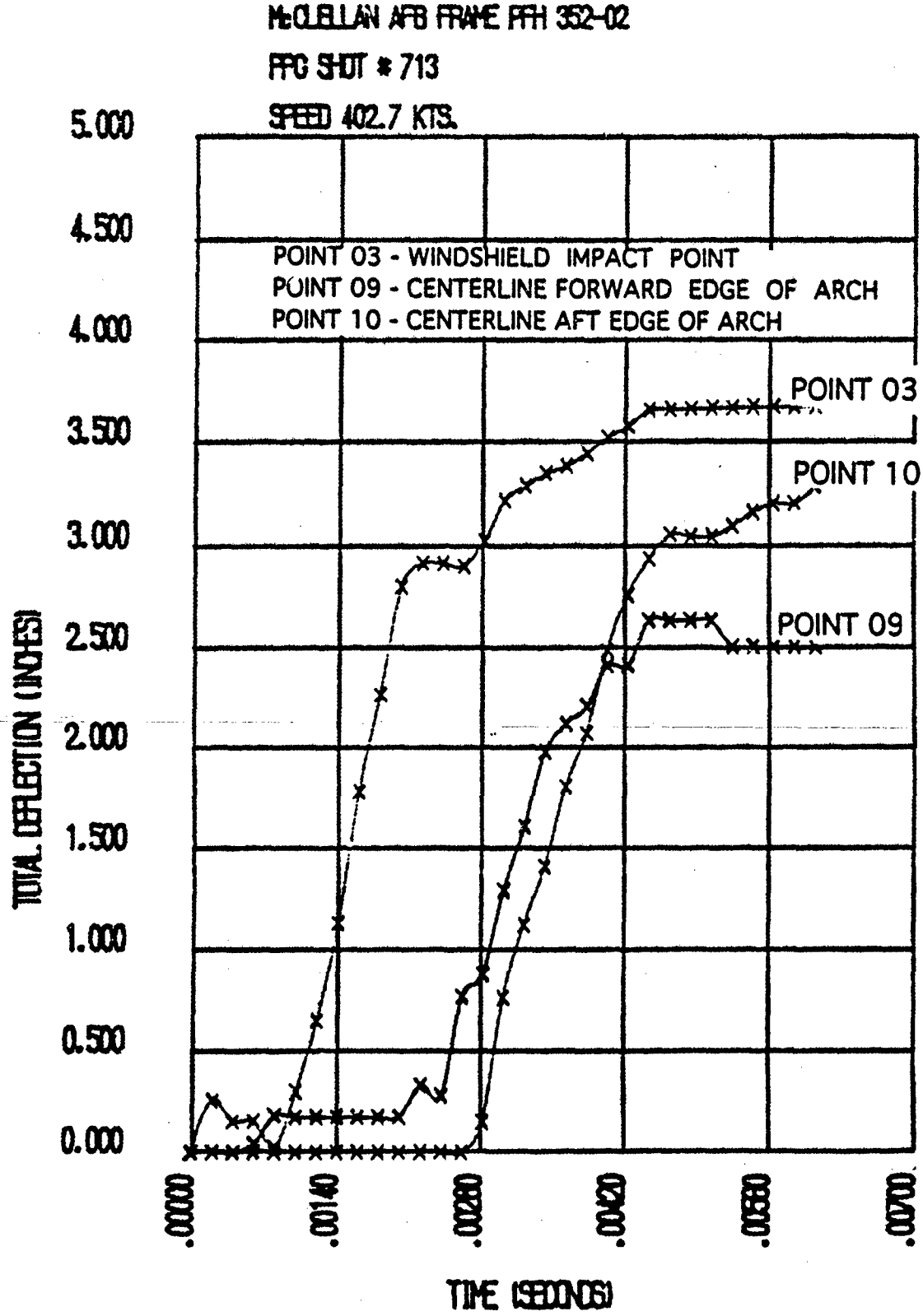


Figure 9. FRAME PFH BIRD IMPACT RESULTS
DEFLECTION vs TIME

was one that had received extensive damage from past bird-impact testing and was bolted and welded together to retain the shape needed to hold the canopy transparency. Thus, the canopy failures were attributed to the weakened canopy frame and not the windshield arch design. Only testing with flight quality canopy frames would show how the canopy would perform when the windshield incurs a similar type of bird-impact.

STATE of PROTOTYPE FRAME PROGRAM

As a result of the bird-impact testing, the composite windshield frame design with the PFH arch was chosen for flight evaluation. Two frames have been manufactured by SM-ALC and delivered to PPG Industries for windshield installation. After windshield installation, one windshield system will be provided to Randolph AFB, Texas and the other frame will be provided to NASA in Houston, Texas for flight evaluations. Upon completion of flight evaluation, six additional windshield systems will perform a long-term field evaluation on variousT-38 aircraft at operational bases. The AETC and SA-ALC are expected to make a production decision during the long-term field evaluation. Once a production decision is made, the SM-ALC will complete the tooling, design and manufacturing data needed to reproduce the composite frames. This data will be provided to SA-ALC for follow-on spare procurements.

SUMMARY

A T-38 aircraft composite windshield frame with an arch consisting of laminated S-2 glass fabric and stainless steel sheets in an epoxy matrix has been successfully developed. With the PPG Industries 5300 liner/polycarbonate windshield the frame meets the design goals of withstanding the impact of a 4-lb bird at 400 knots. The frame also meets the design goals improved damage tolerance and increased repairability through use of composite materials, and requires no changes to existing fuselage structure or installation procedures of existing cast magnesium windshield frames.

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APPENDIX A

T-38 BEAM-COUPON PHYSICAL AND MECHANICAL PROPERTY DATA

ii. Beam-Coupon Stacking Sequence (Table 1)

iv. Composite Material Specifications (Table 2)

v. Beam-Coupon Geometric and Physical Properties (Table 3)

vii. Beam-Coupon Test Data Summary (Table 4)

ix. Beam-Coupon Hybrid Composite Materials Summary (Table 5)

x. Figure 1. Beam-Coupon Post-Test

TABLE 1
STACKING SEQUENCE FOR COMPOSITE BEAMS

Cross-Section Designation*	Lay-Up **
AF	$[(0_k)_7[(\pm45_k)_2(0_k)_7(\pm45_k)_4]_2\ (\pm45_k)_2(0_k)_7(\pm45_k)_4(0_k)_8]_s(\pm45_{kw})_8(\pm45_k)$
AO	$[[(0_k)_8(\pm45_k)(0_k)_8(\pm45_k)_3]_7(0_k)_7]_s(\pm45_{kw})_8(\pm45_k)$
AFA	$[[(0_c)_7/(\pm45_k)]_2[(0_k)_7/(\pm45_k)]_3[0_{Sg})_4/(\pm45_{Sg})]_3]_s\ (\pm45_{kw})_2(0_{kw})_2 (\pm45_{kw})_2(\pm45_k)$
BO	$[(\pm45_k)_8(0_k)_{31}(0_{Sg})_{42}]_s(\pm45_{kw})_8(\pm45_k)$
CO	$[(\pm45_k)_8(0_k)_{11}(0_{Sg})_2(0_k)_8(0_{Sg})_8(0_k)_8(0_{Sg})_7(0_k)_4(0_{Sg})_{12}(0_k)_2(0_{Sg})_{16}]_s (\pm45_{kw})_8(\pm45_k)$
DO	$[(\pm45_k)_8[(0_{Sg})_3(0_k)_2]_{14}(0_k)_3]_s(\pm45_{kw})_8(\pm45_k)$
EO	$[(0_k)_{11}(\pm45_k)(0_{Sg})_2(0_k)_8(\pm45_k)(0_{Sg})_8(0_k)_8(\pm45_k)(0_{Sg})_7(0_k)_4(\pm45_k)(0_{Sg})_{12} (0_k)_2(\pm45_k)_2(0_{Sg})_{16}]_s(\pm45_{kw})_8(\pm45_k)$
EF	$[(0_k)_7(\pm45_k)(0_{Sg})(0_k)_8(\pm45_k)(0_{Sg})_2(0_k)_4(\pm45_k)(0_{Sg})_3(0_k)_3(\pm45_k)(0_{Sg})_8(0_k)_2 (\pm45_k)(0_{Sg})_8]_s\ (\pm45_{kw})_4(\pm45_k)$
FO	$[[(0_k)_{11}(0_{Sg})_2(0_k)_8(0_{Sg})_8(0_k)_8(0_{Sg})_7(0_k)_4(0_{Sg})_{12}(0_k)_2][(\pm45_k)_8(0_{Sg})_{16}]]_s (\pm45_{kw})_8(\pm45_k)$
GO	$[[(0_k)_{11}(0_{Sg})_2(0_k)_8(0_{Sg})_8(0_k)_8][(\pm45_{kw})_8(0_{Sg})_7(0_k)_4(0_{Sg})_{12}(0_k)_2(0_{Sg})_{16}]]_s (\pm45_{kw})_8(\pm45_k)$
HO	$[(\pm45_k)_2[0_k/0_{Sg}]_{28}(\pm45_{Sg})(0_{Sg})_{28}(\pm45_{Sg})]_s\ (\pm45_{kw})_8(\pm45_k)$
IO	$[(\pm45_k)_2[0_{St}/(0_k)_8]_7(0_{Sg})_7(\pm45_{Sg})]_s\ (\pm45_{kw})_8(\pm45_k)$
JO	$[(\pm45_k)_2[(0_{St})/(0_k)_{13}]_4(\pm45_k)_2]_s\ (\pm45_{kw})_8(\pm45_k)$
KF	$[[(0_k)_2/(0_{St})]_2(0_k)_{13}(\pm45_k)(0_{Sg})_4(0_k)_7(\pm45_k)(0_{Sg})_8(\pm45_k)_2]_s\ (\pm45_{kw})_8(\pm45_k)$
LF	$[[(0_k)_2/(0_{St})]_4(0_k)_7(\pm45_k)(0_{Sg})_8(0_k)_8(\pm45_{Sg})(0_{Sg})_8(\pm45_{Sg})]_s\ (\pm45_{kw})_8(\pm45_k)$

(A) 11

TABLE 1 (continued)

MF	$[[(0_k)_2/(0_{St})]_7(\pm45_k)(0_{Sg})_4(0_k)_4(\pm45_{Sg})(0_{Sg})_7(\pm45_{Sg})]_s\ (\pm45_{kw})_e(\pm45_k)$
NF	$[[(0_{St})/(0_k)_4]_4(\pm45_k)(0_{Sg})_4(0_k)_8(\pm45_{Sg})(0_{Sg})_7(\pm45_{Sg})]_s\ (\pm45_{kw})_e(\pm45_k)$
OF	$[[(0_{St})/(0_k)_4]_4(\pm45_k)(0_k)_{11}(\pm45_k)(0_k)_8(\pm45_k)]_s\ (\pm45_{kw})_e(\pm45_k)$
PF	$[[(0_{St})/(0_{Sg})_3]_4(\pm45_{Sg})(0_{Sg})_{10}(\pm45_{Sg})(0_{Sg})_8(\pm45_{Sg})]_s\ (\pm45_{Sgw})_e(\pm45_{Sg})$
RF	$[(0_{Sta})(0_k)_{11}(\pm45_k)(0_{Sg})_6(0_k)_8(\pm45_{Sg})(0_{Sg})_{11}(\pm45_{Sg})]_s\ (\pm45_{kw})_e(\pm45_k)$
SF	$[(0_{Sta})(0_k)_8(0_{Sta})(0_{Sg})_8(0_k)_8(\pm45_{Sg})(0_{Sg})_8(\pm45_{Sg})]_s\ (\pm45_{kw})_e(\pm45_k)$
TF	$[(0_{St})_{38}(\pm45_{kw})_e(\pm45_k)]_T$
VF	$[(0_k)_{20}(\pm45_{Sg})(0_{Sg})_{19}(\pm45_{Sg})]_s\ (\pm45_{kw})_e(\pm45_k)$
WF	$[[[(0_k)/(0_{Sg})]_4(\pm45_k)]_4(0_{Sg})_3(\pm45_k)]_s\ (\pm45_{kw})_e(\pm45_k)$

*Second letter of cross-section identification designates material supplier, where:

O = Ferro
F = Fiberite

KEY:

k	Kevlar 49
kw	Kevlar wrap ***
Sg	S-2 glass
Sgw	S-2 glass wrap***
c	Graphite
St	0.015" 301-1/2 hard stainless steel
Sta	0.063" 301-1/4 hard stainless steel
s	Symmetric
T	Total

** [Core structural plies] (outer wrap plies) (single ply to splice seam in final wrap ply)

*** Wrap plies ≡ outer plies that encase inner structural core plies.

(A) iii

TABLE 2
COMPOSITE MATERIALS SPECIFICATIONS

Material:	Tensile Strength [ksi]		Modulus [Msi]		Poisson's Ratio*	thickness [in]
	long.	trans.	long.	trans.	LT	
Ferro (CE-321R epoxy matrix):						
S-2 Glass 6781-style fabric	81.98	64.07	4.68	3.58	0.154	0.0087
S-2 Glass tape	150.8	-	5.76	-	0.337	0.0046
Kevlar-49 285-style fabric	63.84	51.51	4.09	3.82	0.092	0.0078
Kevlar-49 tape	151.8	-	9.70	-	0.371	0.0063
Fiberite (7714A epoxy matrix):						
S-2 Glass fabric	72.79	66.93	3.85	3.84	0.128	0.0076
S-2 Glass tape	161.78	-	5.92	-	0.333	0.0123
Kevlar-49 fabric	66.47	59.18	4.06	3.73	0.094	0.0102
Kevlar-49 tape	164.60	-	11.62	-	0.385	0.0096

Material:	Yield Strength [ksi]	Tensile Strength [ksi]	Percent Elongation	thickness [in]
Aerospace Alloys:				
301-1/4 hard stainless steel	81.2	138.6	42	0.0620
301-1/2 hard stainless steel	117.6	175.0	29	0.0150

* Specimen loaded longitudinally with strain measured longitudinally and transversely.

(A) 1v

TABLE 3
GEOMETRIC AND PHYSICAL PROPERTIES OF COMPOSITE SPECIMENS

SPECIMEN ID	BOLT DESIGN	DENSITY LBS/CU IN	WIDTH IN	THICKNESS IN	AREA SQ IN	MOMENT OF INERTIA IN^4
AO-1	1/4" GD5	0.0501	2.9815	1.0215	3.0456	0.2648
AO-2	5/16" GD8	0.0495	2.9670	1.0350	3.0708	0.2741
AO-3	1/4" GD8	0.0499	2.9750	1.0250	3.0494	0.2670
AF-1	1/4" GD5	0.0490	2.8200	1.0200	2.8764	0.2494
AF-2	1/4" GD5	0.0492	2.8150	1.0220	2.8769	0.2504
AF-3	1/4" GD8	0.0492	2.8470	1.0230	2.9125	0.2540
AFA-1	1/4" GD5	0.0550	2.9635	1.0930	3.2391	0.3225
AFA-2	5/16" GD8	0.0539	2.9760	1.1490	3.4194	0.3762
AFA-3	1/4" GD8	0.0539	2.9530	1.1750	3.4698	0.3992
BO-1	1/4" GD5	0.0585	3.1280	1.0150	3.1749	0.2726
BO-2	5/16" GD8	0.0568	3.0300	1.0550	3.1966	0.2965
BO-3	1/4" GD8	0.0542	3.2150	1.0630	3.4175	0.3218
CO-1	1/4" GD5	0.0568	2.9715	1.0590	3.1468	0.2941
CO-2	5/16" GD8	0.0578	2.9495	1.0175	3.0011	0.2589
CO-3	1/4" GD8	0.0569	2.9550	0.9975	2.9476	0.2444
DO-1	1/4" GD5	0.0580	3.0075	1.0305	3.0992	0.2743
DO-2	5/16" GD8	0.0557	2.9160	1.1135	3.2470	0.3355
DO-3	1/4" GD8	0.0567	2.9230	1.1300	3.2738	0.3422
DO-1A	1/4" GD5	0.0569	2.9560	1.0220	3.0210	0.2630
DO-2A	5/16" GD8	0.0570	2.9600	1.0230	3.0281	0.2641
DO-3A	1/4" GD8	0.0572	2.9370	1.0100	2.9664	0.2522
EO-1	1/4" GD5	0.0564	2.9600	1.0410	3.0814	0.2783
EO-2	5/16" GD8	0.0570	2.9215	1.0250	2.9945	0.2622
EO-3	1/4" GD8	0.0569	2.9470	1.0200	3.0059	0.2606
EF-1	1/4" GD5	0.0556	2.8130	1.0180	2.8636	0.2473
EF-2	5/16" GD8	0.0558	2.8120	1.0200	2.8682	0.2487
EF-3	1/4" GD8	0.0560	2.8260	1.0125	2.8613	0.2444
FO-1	1/4" GD5	0.0513	2.9905	1.1750	3.5138	0.4043
FO-2	5/16" GD8	0.0578	2.9120	1.0215	2.9746	0.2587
FO-3	1/4" GD8	0.0579	2.9535	0.9825	2.9018	0.2334
GO-1	1/4" GD5	0.0569	2.9425	1.0465	3.0793	0.2810
GO-2	5/16" GD8	0.0566	2.9490	1.0560	3.1141	0.2894
GO-3	1/4" GD8	0.0564	2.9450	1.0500	3.0923	0.2841
HO-1	1/4" GD5	0.0589	2.9500	1.0460	3.0857	0.2813
HO-2	5/16" GD8	0.0574	2.9670	1.0860	3.2222	0.3167
HO-3	1/4" GD8	0.0587	2.9400	1.0645	3.1296	0.2955
IO-1	5/16" GD8	0.0969	2.9470	0.9870	2.9087	0.2361
IO-2	1/4" GD5	0.0963	2.9670	0.9870	2.9284	0.2377
IO-3	1/4" GD8	0.0969	2.9450	0.9900	2.9156	0.2381

(A) v

TABLE 3 (continued)

SPECIMEN ID	BOLT DESIGN	DENSITY LBS/CU IN	WIDTH IN	THICKNESS IN	AREA SQ IN	MOMENT OF INERTIA IN^4
JO-1	1/4" GD5	0.0769	2.9500	0.9835	2.9013	0.2339
JO-2	5/16" GD8	0.0765	2.9370	0.9950	2.9223	0.2411
JO-3	1/4" GD8	0.0762	2.9460	0.9960	2.9342	0.2426
KF-1	1/4" GD5	0.0666	2.8420	1.0120	2.8761	0.2455
KF-2	5/16" GD8	0.0667	2.8215	1.0220	2.8836	0.2510
KF-3	1/4" GD8	0.0663	2.8415	1.0270	2.9182	0.2565
LF-1	1/4" GD5	0.0803	2.8250	1.0260	2.8985	0.2543
LF-2	5/16" GD8	0.0804	2.8195	1.0260	2.8928	0.2538
LF-3	1/4" GD8	0.0816	2.8195	1.0250	2.8900	0.2530
MF-1	1/4" GD5	0.0989	2.9270	0.9870	2.8889	0.2345
MF-2	5/16" GD8	0.0946	2.9415	1.0355	3.0459	0.2722
MF-3	1/4" GD8	0.0985	2.9460	1.0355	3.0506	0.2726
NF-1	1/4" GD5	0.0804	2.8160	1.0160	2.8611	0.2461
NF-2	5/16" GD8	0.0796	2.8265	1.0225	2.8901	0.2518
NF-3	1/4" GD8	0.0800	2.8315	1.0185	2.8839	0.2493
OF-1	1/4" GD5	0.0747	2.9390	0.9830	2.8890	0.2326
OF-2	5/16" GD8	0.0720	2.9200	1.0210	2.9813	0.2590
OF-3	1/4" GD8	0.0734	2.9640	0.9930	2.9433	0.2418
PF-1	1/4" GD5	0.0903	2.8210	1.0165	2.8675	0.2469
PF-2	5/16" GD8	0.0903	2.8275	1.0125	2.8628	0.2446
PF-3	1/4" GD8	0.0899	2.8310	1.0185	2.8834	0.2493
RF-1	1/4" GD5	0.0862	2.8145	0.9880	2.7807	0.2262
RF-2	5/16" GD8	0.0865	2.9505	0.9385	2.7690	0.2032
RF-3	1/4" GD8	0.0824	2.9450	0.9875	2.9082	0.2363
SF-1	1/4" GD5	0.1082	2.8195	1.0265	2.8942	0.2541
SF-2	5/16" GD8	0.1093	2.8265	1.0180	2.8774	0.2485
SF-3	1/4" GD8	0.1080	2.8345	1.0205	2.8926	0.2510
TF-1	1/4" GD5	0.1678	2.8250	1.0330	2.9182	0.2595
TF-2	5/16" GD8	0.1665	2.8460	1.0330	2.9399	0.2614
TF-3	1/4" GD8	0.1739	2.8260	1.0310	2.9136	0.2581
VF-1	1/4" GD5	0.0570	2.8145	1.0290	2.8961	0.2555
VF-2	5/16" GD8	0.0541	2.8205	1.0825	3.0532	0.2981
VF-3	1/4" GD8	0.0569	2.8125	1.0260	2.8856	0.2531
WF-1	1/4" GD5	0.0561	2.8125	1.0270	2.8884	0.2539
WF-2	5/16" GD8	0.0547	2.8090	1.0275	2.8862	0.2539
WF-3	1/4" GD8	0.0559	2.8230	1.0330	2.9162	0.2593

TABLE 4
COMPOSITE BEAM TEST DATA SUMMARY

SPECIMEN ID	PEAK LOAD LBS	BENDING MODULUS MSI	ULTIMATE STRESS PSI	TOTAL ENERGY IN-LBS	TOTAL NORM ENERGY LBS/IN	NORMALIZED ENERGY TO 1.25" DISP
AO-1	16959	4.469	78700	21191	5294	4052
AO-2	13596	4.001	61760	21040	5257	4251
AO-3	16479	4.325	76120	17404	4351	4056
AF-1	16432	4.788	80860	23301	6068	4304
AF-2	16907	4.726	83020	18174	4737	4192
AF-3	17583	4.887	85200	16295	4211	4464
AFA-1	20934	6.624	85370	27397	6754	4612
AFA-2	20190	5.371	74192			5564
AFA-3	21984	5.784	77950	28909	7003	4642
BO-1	20935	3.918	93790	26765	6460	4934
BO-2	18271	4.673	78220	25193	6167	4964
BO-3	19871	3.543	78970	24161	5648	4373
CO-1	21374	4.122	92600	24904	6179	4593
CO-2	19863	4.738	93910	24478	6170	4198
CO-3	19791	4.428	97180	23723	6002	4518
DO-1 *	21174	3.894	95720	24418	6047	3999
DO-2 *	15719	3.829	62770	26261	6517	4058
DO-3 *	16388	3.796	64530	27353	6766	4228
DO-1A	20626	4.062	96450	24642	6195	4153
DO-2A	19611	3.826	91400	26300	6603	4760
DO-3A	21603	4.006	104100	23374	5922	4323
EO-1	21805	5.290	98140	25386	6345	4270
EO-2	19993	5.174	94040	27020	6847	4655
EO-3	23445	5.039	110400	26344	6641	4631
EF-1	21686	5.252	107400	24188	6314	4872
EF-2	21703	5.339	107100	24048	6276	5018
EF-3	21170	5.550	105500	22488	5859	4658
FO-1	13335	3.998	46630	23619	5670	3608
FO-2	10544	5.301	50100	18412	4681	3470
FO-3	12376	4.142	62670	21132	5369	3774
GO-1	22253	4.930	99700	26844	6730	4863
GO-2	18432	4.887	80920	28756	7180	4749
GO-3	20706	4.826	92070	25680	6428	4657
HO-1	25083	4.771	112200	27348	6844	4690
HO-2	20534	4.466	84720	28056	6922	4429
HO-3	25406	4.981	110100	27068	6759	4749
IO-1	17529	7.701	88150	31140	7916	4654
IO-2	16048	7.285	80160	24438	6181	4269
IO-3	19313	7.030	96600	24330	6183	4744

* Specimen exothermed to in excess of 350°F.

TABLE 4 (continued)

SPECIMEN ID	PEAK LOAD LBS	BENDING MODULUS MSI	ULTIMATE STRESS PSI	TOTAL ENERGY IN-LBS	TOTAL NORM ENERGY LBS/IN	NORMALIZED ENERGY TO 1.25" DISP
JO-1	18532	6.356	93764	26092	6633	4499
JO-2	18823	5.986	93460	29941	7615	4821
JO-3	20001	6.426	98810	27075	6868	4956
KF-1	20523	7.056	101800	28586	7417	5557
KF-2	19912	6.095	97550	33896	8819	5669
KF-3	22149	6.527	106700	31347	8103	5823
LF-1	18322	7.606	88950	24354	6324	4770
LF-2	20825	6.711	101300	27510	7154	5433
LF-3	22980	7.301	112000	26338	6851	5337
MF-1	18711	8.738	94740	20893	5338	4929
MF-2	20011	7.070	91600	30271	7612	5637
MF-3	25008	7.836	114300	23660	5942	5082
NF-1	20011	7.917	99390	22537	5881	4625
NF-2	20960	6.419	102400	33831	8790	5203
NF-3	17651	7.063	86760	25171	6538	4924
OF-1	17109	6.824	86980	20011	5102	4057
OF-2	18303	6.018	86810	32028	8127	4508
OF-3	14705	7.638	72640	24863	6283	4283
PF-1	29981	6.430	148500	27672	7211	6107
PF-2	30497	7.277	151900	38521	10032	7119
PF-3	30043	6.601	147700	32440	8427	6392
RF-1	22512	7.274	118300	25612	6736	5486
RF-2	21150	7.527	117500	32871	8452	5775
RF-3	19181	6.718	96430	30347	7717	4965
SF-1	13135	8.973	63830	25804	6709	4340
SF-2	13500	8.015	65539	34647	9012	5236
SF-3	13206	8.279	64590	21531	5585	4327
TF-1	26476	7.675	126800	39478	10233	7681
TF-2	26630	7.508	126600	46092	11882	8105
TF-3	25862	8.053	124300	41645	10797	7784
VF-1	16839	5.422	81580	26128	6798	5179
VF-2	17490	4.841	76400	26187	6709	5296
VF-3	19623	5.815	95690	25669	6687	5258
WF-1	22149	5.116	107800	25670	6686	4682
WF-2	20081	4.774	97760	26480	6902	4790
WF-3	18941	4.898	90780	23338	6052	4136

TABLE 5
HYBRID COMPOSITE MATERIALS SUMMARY

SPECIMEN ID	DENSITY LBS/CU IN	PEAK LOAD LBS	BENDING MODULUS MSI	ULTIMATE STRESS PSI	TOTAL ENERGY IN-LBS	TOTAL NORMALIZED ENERGY LBS/IN	NORMALIZED ENERGY TO 1.25" DISP LBS/IN
TF	0.1694	26323	7.745	125,900	42405	10971	7857
PF	0.0902	30174	6.769	149,367	32878	8557	6539
KF	0.0665	20861	6.559	102,017	31276	8113	5683
RF	0.0850	20948	7.173	110,743	29610	7635	5409
VF	0.0560	17984	5.359	84,557	25995	6732	5244
MF	0.0973	21244	7.882	100,213	24941	6297	5216
LF	0.0808	20709	7.206	100,750	26067	6776	5180
AFA	0.0543	21036	5.926	79,137	23153*	6879*	4939
NF	0.0800	19541	7.133	96,183	27180	7070	4917
EF	0.0558	21520	5.381	106,667	23575	6149	4849
JO	0.0765	19119	6.256	95,345	27703	7039	4759
BO	0.0565	19692	4.045	83,660	25373	6092	4757
GO	0.0566	20464	4.881	90,897	27093	6779	4756
SF	0.1085	13280	8.422	64,653	27327	7102	4634
HO	0.0583	23674	4.739	102,340	27491	6842	4623
IO	0.0967	17630	7.339	88,303	26636	6760	4556
WF	0.0556	20391	4.929	98,780	25163	6547	4536
EO	0.0567	21748	5.168	100,860	26250	6611	4518
CO	0.0572	20343	4.429	94,563	24368	6117	4436
DOA	0.0570	20613	3.964	97,317	24772	6240	4412
AF	0.0491	16974	4.800	83,027	19257	5005	4320
OF	0.0734	16706	6.827	82,143	25634	6504	4283
AO	0.0499	15678	4.265	72,193	19878	4967	4120
DO	0.0568	17761	3.839	74,340	26011	6443	4095
FO	0.0557	12085	4.480	53,133	21054	5240	3617

Results are average of three tests except (*) which are average of two tests.

(A) 1x

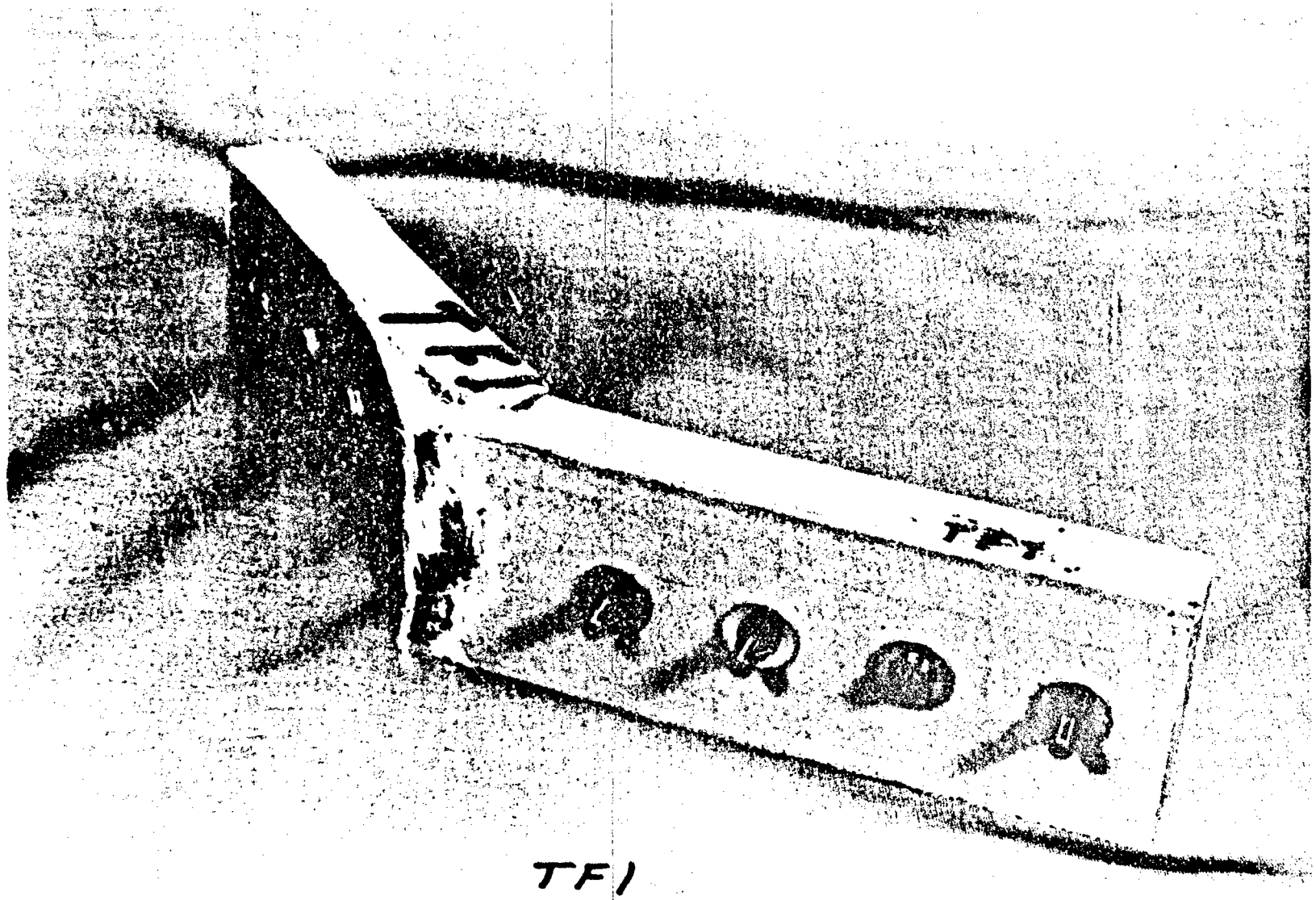


Figure 1. Beam-Coupon Post Test

APPENDIX B

T-38 BASELINE AND PROTOTYPE ARCHES SECTION PROPERTIES

ii. Section Properties for Magnesium Arch (Figure 1) (Baseline Case 1)

iii. Section Properties for Steel Tube Reinforcement (Figure 2) (Baseline Cases 2 and 4)

iv. Centerline Section Properties for Composite Reinforcement (Figure 3) (Baseline Cases 3 and 5)

v. Sill Section Properties for Composite Reinforcement (Figure 4) (Baseline Cases 3 and 5)

vi. Prototype Arches Material Properties (Table 1)

vii. Centerline Section Properties for Prototype Composite Arch (Figure 5)

viii. Sill Section Properties for Tapered Prototype Arches (Figure 6)

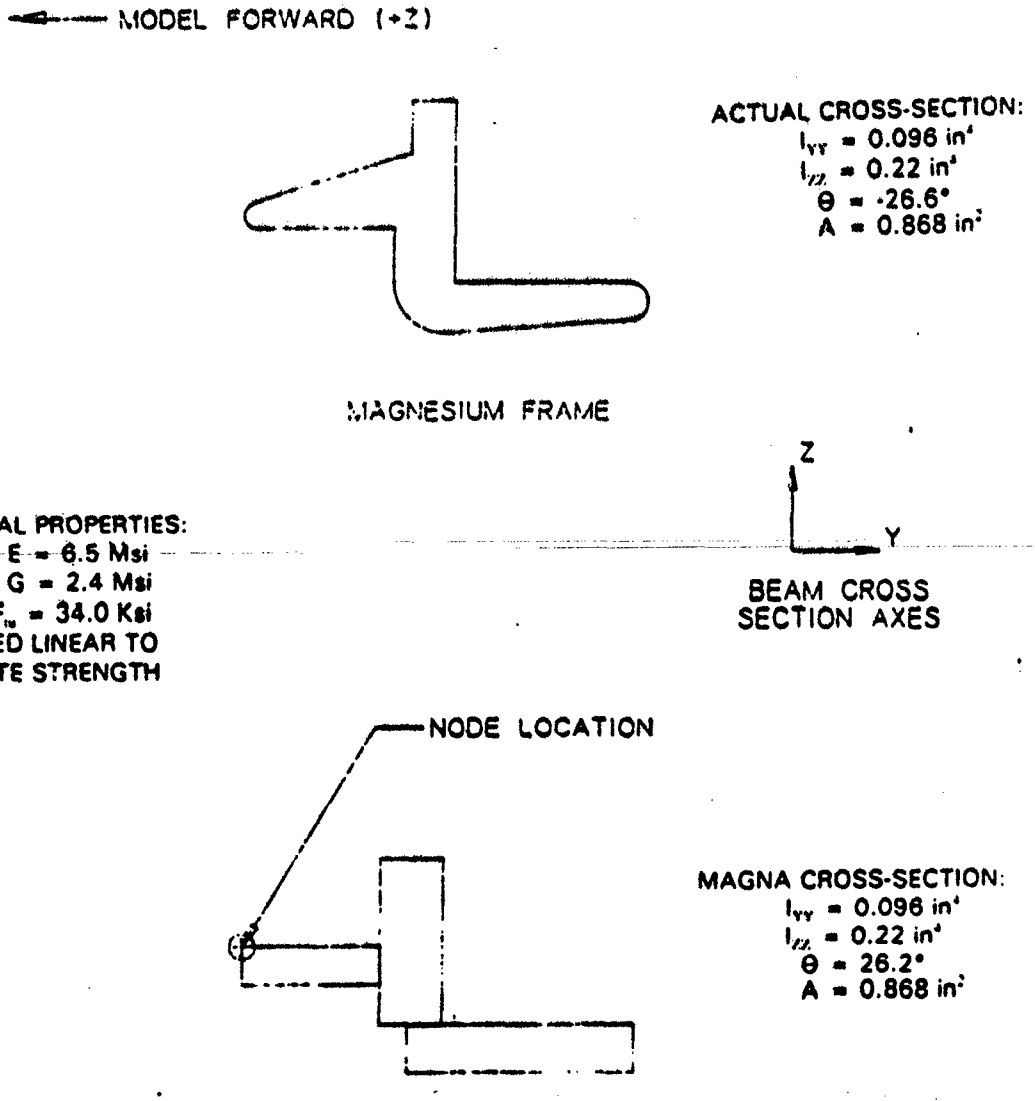


Figure 1. Section Properties for Magnesium Arch (Baseline Case 1).

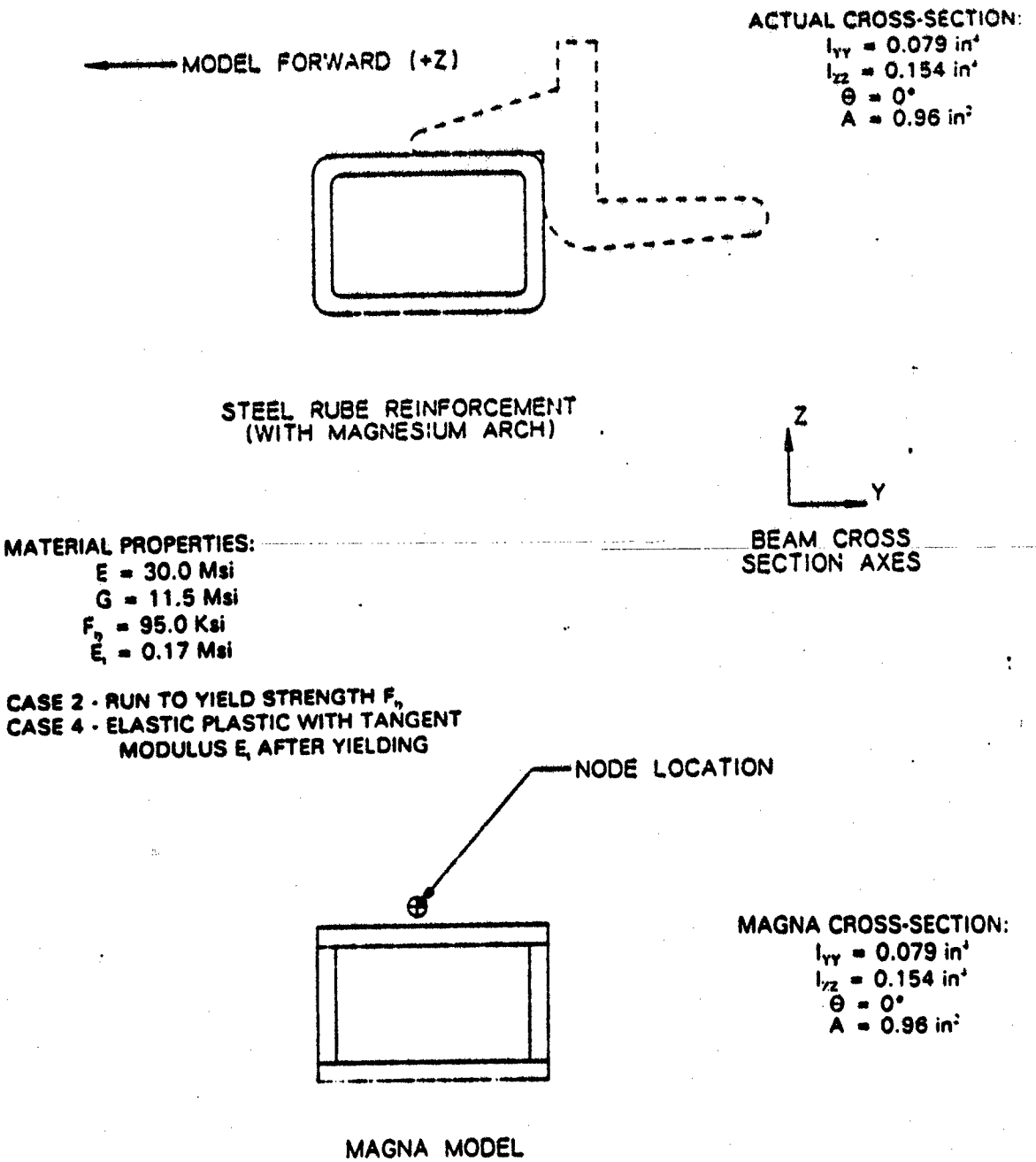
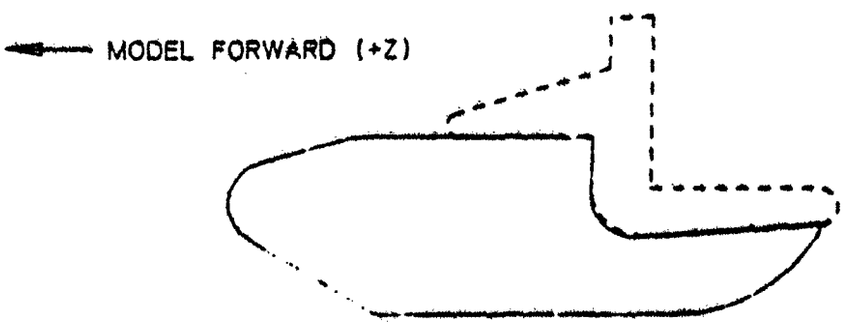


Figure 2. Section Properties for Steel Tube Reinforcement (Baseline Cases 2 and 4).

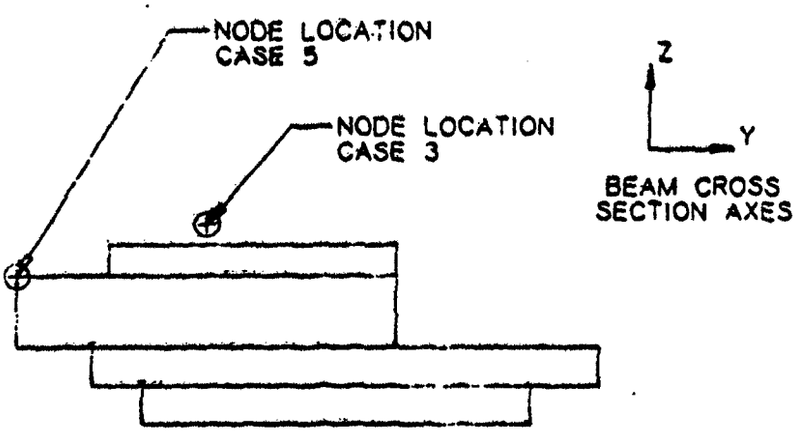
ACTUAL CROSS-SECTION (CENTERLINE):
I_{YY} = 0.201 in⁴
I_{ZZ} = 1.569 in⁴
θ = -8.2°
A = 2.43 in²



MATERIAL PROPERTIES:
E = 8.0 Msi
G = 0.878 Msi
F_{tu} = 110.0 Ksi
E_t = 0.17 Msi

ASSUMED LINEAR TO ULTIMATE STRENGTH

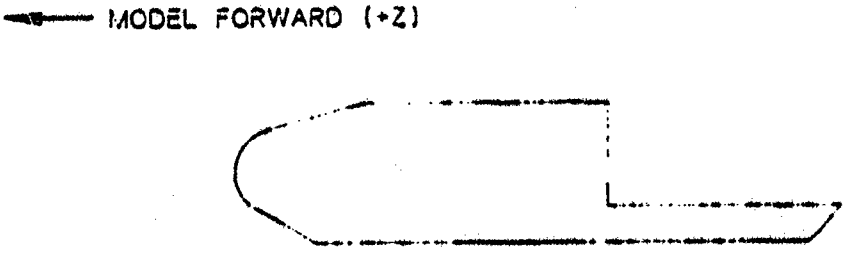
COMPOSITE REINFORCEMENT
(WITH MAGNESIUM ARCH)



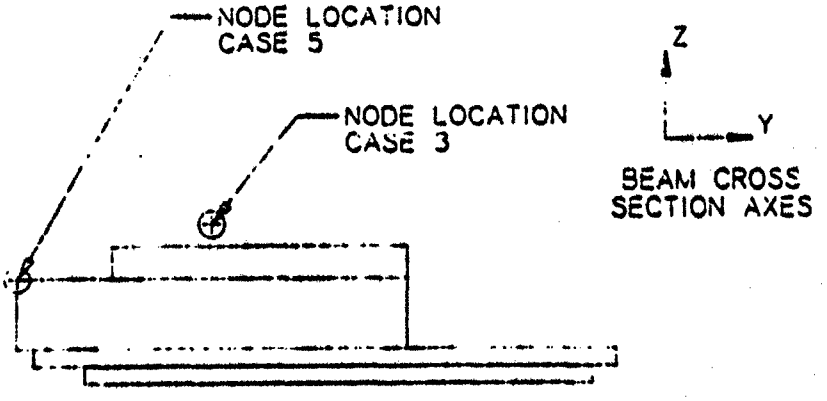
MAGNA MODEL

MAGNA CROSS-SECTION (CENTERLINE):
I_{YY} = 0.201 in⁴
I_{ZZ} = 0.154 in⁴
θ = -8.1°
A = 2.43 in²

Figure 3. Centerline Section Properties for Composite Reinforcement (Baseline Cases 3 and 5).



COMPOSITE REINFORCMENT



MAGNA MODEL

Figure 4. Cross-section at Sill for Composite Reinforcement (Baseline Cases 3 and 5).

(B) v

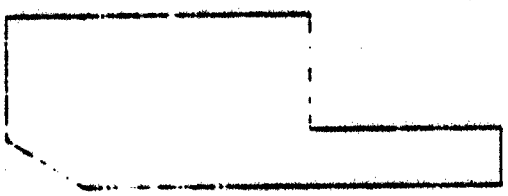
TABLE 1

PROPERTIES OF MATERIALS FOR PROTOTYPE ARCH

MATERIAL	F_{tu} (Ksi)	E (Msi)	G (Msi)
PF	149.4	6.769	0.49
KF	102.0	6.559	0.55
VF	84.6	5.359	0.66
AFA	79.1	5.926	0.46
TF	125.9	7.745	0.66

ACTUAL CROSS-SECTION (CENTERLINE):
I_{yy} = 0.191 in^4
I_{zz} = 1.06 in^4
θ = -11.0°
A = 2.11 in^2

MODEL FORWARD (+Z)



PROPOSED COMPOSITE

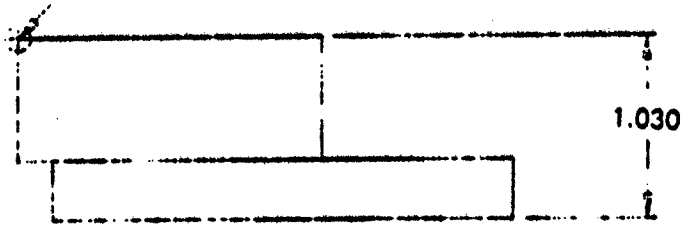
MATERIAL PROPERTIES ARE LISTED IN TABLE 1

MATERIALS ARE ASSUMED LINEAR TO ULTIMATE STRENGTH



BEAM CROSS SECTION AXES

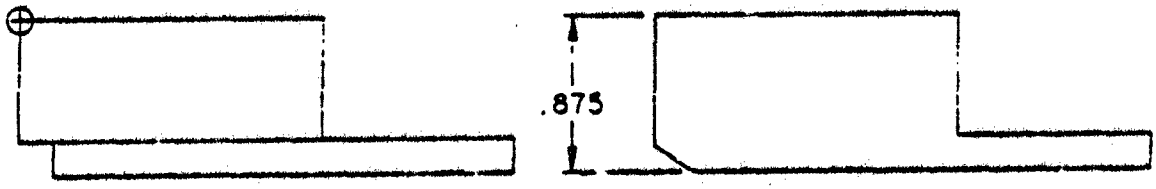
NODE LOCATION



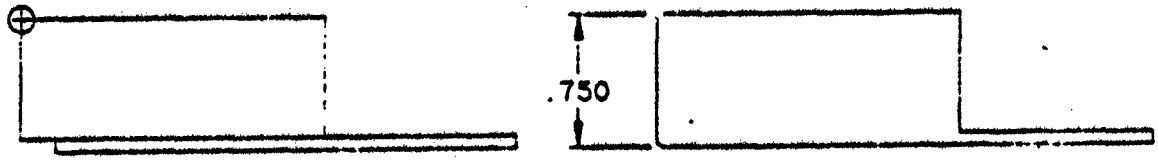
MAGNA MODEL

MAGNA CROSS-SECTION:
I_{yy} = 0.193 in^4
I_{zz} = 1.05 in^4
θ = -11.1°
A = 2.10 in^2

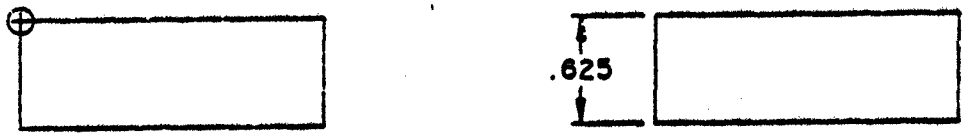
Figurer 5. Centerline Section Properties for Prototype T-38 Composite Arch.



FINAL TAPER WIDTH = 0.875 INCH



FINAL TAPER WIDTH = 0.750 INCH



FINAL TAPER WIDTH = 0.625 INCH



FINAL TAPER WIDTH = 0.500 INCH

MAGNA MODELS

PROPOSED CROSS SECTIONS
AT SILL ATTACHMENT

Figure 6. Cross-Sections for Tapered T-38 Prototype Arch (Cross-Sections are at sill).

SESSION III

UNDERSTANDING CURRENT MATERIALS - PART A

Chairman: B. S. West
University of Dayton

Co-Chairman: E. Kochis
Boeing

Coordinator: M. R. Unroe
Materials Directorate
Wright Laboratory

INTERLAYER DESIGN FOR GLASS/PLASTIC TRANSPARENCIES

Thomas G. Rukavina
PPG Industries Inc.

Interlayer Design for Glass/Plastic Transparencies

by
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Pittsburgh,PA 15238

Abstract

Unbalanced glass/plastic composites fabricated with thin chemically tempered glass and polycarbonate plastic have many performance advantages over all-plastic or all-glass transparencies. The durability of glass combined with the lightweight and high impact resistance of polycarbonate can produce a lighter weight, longer-lived, high strength composite. These advantages, though well known, can be offset by disadvantages, which are related to the not too simple task of marrying together two dissimilar materials.

Polymeric interlayers that are able to mitigate shear stress during fabrication, as well as in the field during temperature fluctuations, can improve the long-term performance of glass/plastic transparencies. This paper presents the results of a systematic study of a polyurethane interlayer structure-property relationships, including the behavior of these polymers in glass/polycarbonate beam samples. The effects of urethane content and molecular weight on glass transition temperature, tensile strength, elongation, shear modulus, and glass/plastic beam center- deflections are discussed.

I Introduction

Thin chemically-tempered glass laminated with an interlayer to polycarbonate, at first glance, seems to be the ideal combination of materials to a transparency designer. Glass adds stiffness to the composite, has excellent abrasion resistance, blocks short wavelength ultraviolet light, and weathers better than any synthetic plastic. Polycarbonate contributes lightweight and outstanding impact resistance. However, a problem that fabricators commonly confront is that after lamination, following cooling from a temperature which allows good adhesion of the interlayer to the glass and plastic surfaces, the composite transparency warps and does not match the intended contour. This problem is due to the large mismatch in coefficients of expansion of glass and polycarbonate which causes a shear stress to develop in the composite. The stress is transmitted to the interlayer from the shrinking plastic, as the composite temperature cools to room temperature. When the shear stress is not mitigated successfully at the interlayer, the composite warps, putting the glass in tension.

There are several potential solutions to the contour problem at room temperature. One is to lower the lamination temperature, thus reducing the expansion of the plastic. This method is practical only if the interlayer can be laminated at low temperatures. Also the activation energy for adhesion must be low or adhesion problems may occur in the field. Another way of circumventing the problem is to insulate the plastic during lamination. A third solution is to use an interlayer that is cast-in-place and cured at room temperature.. Another solution is to design a sheet interlayer that laminates with good adhesion to the substrates, preferably above the maximum use temperature of the transparency, and that has a low shear modulus over a wide temperature range. This is the topic of this paper.

The low shear modulus is necessary to reduce stress induced via temperature extremes in flight. High shear modulus can result in

delamination, stress-crazing of the polycarbonate structural ply, and glass breakage. Depending on the final application, the interlayer can be tailored to be low - temperature melting and low modulus over a wide temperature range, low-temperature melting and high modulus at room temperature and below, or higher melting and low modulus over a wide temperature range. The interlayers prepared for this study can all be laminated at 150°F or higher.

Polyurethane chemistry is very diverse, and allows the tailoring of materials with vastly different properties, from hard plastics with high glass transition temperatures, to resilient elastomers with low glass transition temperatures. Thermoplastic, segmented polyurethane elastomers are linear block copolymers that are prepared by reacting a diisocyanate with a polyol and a short chain diol chain extender. The elastomers are usually microphase-separated to various degrees into what are termed "soft" and "hard" segments. Excellent performance properties over wide temperature ranges can be attributed to this segregation. The soft segments consist of a high molecular weight polyol and a diisocyanate, though ideally they would be composed only of polyol segments. The ideal is seldom the case. The hard segments consist of a short chain diol and a diisocyanate. There is a thermodynamic driving force for the phase separation, which is the difference in polarity of the two phases. The polarity difference depends on the soft and hard segment solubility parameters. The completeness or incompleteness of the phase separation is dependent on not only thermodynamic compatibility, but the viscosity of the polymer, and the mobility of the hard segment.[1]

Previous studies have indicated that lower melting ranges and lower modulus elastomers result from the use of short chain diols that have an odd number of carbon atoms, though the length is also important.[2] This type of diol is referred to as non-reinforcing, whereas if the diol has an even number of carbon atoms it is termed a reinforcing diol. Branched diols are also non-reinforcing. We have measured the melting points of pure hard segments composed of short chain diols of different lengths and dicyclohexylmethane-4,4'-diisocyanate. The confirming results are shown in Fig.1. If one

chooses a straight chain aliphatic diisocyanate with an odd number of carbon atoms one will observe a similar effect. See Fig.2. [3]

Another way of lowering the melting point while maintaining strong mechanical properties at room temperature include using mixed diols. This can broaden and lower the melting point. Also, one can use a low molecular weight polyol with a small amount of a reinforcing diol, or a larger amount of a non-reinforcing diol. The resulting asymmetry in the hard segment domains is responsible for the lower modulus and melting points. This is a consequence of weaker hard domain interactions.

In this study we use a polytetramethylene oxide(PTMO) polyol (Mn=1000), 1,5 pentanediol, and dicyclohexylmethane-4,4'-diisocyanate. The polyether was chosen for its good hydrolytic stability, low glass transition temperature, and its low tendency to crystallize at low urethane contents. Other short chain diols that were tested included diethylene glycol, butanediol, and thiodiethanol. Poor oxidative stability was observed in formulations prepared with diethylene glycol. Also, melting points were too low. This can be attributed to the compatibility and mixing of the soft and hard segments, which effectively increased the homogeneity of the polymer. The thiodiethanol imparted color and odor to the polymers. The butanediol polymers had higher melting points and yielded higher modulus polymers than those based on pentanediol.

II. Experimental Section

Materials.

The PTMO was obtained from Dupont and is referred to as Teracol® 1000 (Mn = 1000). The pentanediol was obtained from Aldrich Chemical Co. and has a formula weight of 104 g/mole. The dicyclohexylmethane-4,4',- diisocyanate is referred to as Hylene® W and was obtained from Dupont. Butylstannoic acid (BSA) was used as the catalyst in 200, 400, and 600 ppm concentrations. BSA was obtained from Pfaltz & Bauer Chemical Co. The catalyst was analyzed for any iron impurities before using. By analysis the iron content of BSA is < .01ppm. By The stoichiometries were calculated for 10%, 12%, and 14% urethane content (Wu), using the true hydroxyl

numbers (OH number, plus acid number, plus % water). The polyurethanes were prepared at stoichiometry with no excess diisocyanate added, 3% excess diisocyanate, and 6% diisocyanate, in order to vary the molecular weight. The 10, 12, and 14% Wu samples had calculated weight % hard segment contents (Wh) of 3, 12, and 20% respectively. The linear polyurethanes were prepared by a one-step bulk polymerization, cast between 14" x 14" Teflon-coated stainless steel plates, and cured at 200°F for 24 hours into .125" thick sheets. None of the samples were post- annealed.

Glass/Polycarbonate Beam Samples

40" x 6" glass/polycarbonate laminates were fabricated using .120" thick chemically strengthened glass, .250" thick polycarbonate, and .125" thick interlayer. The beams were laminated in an air autoclave at 210°F and 200psi pressure.

Experimental Design.

A Box-Behnken quadratic response surface experiment was designed using RS-Discover experimental design software. A total of 15 experiments were run. In the designed experiment the urethane content was varied from 10 to 14% and the weight average molecular weight varied from 77,000 to 260,000 Mw (based on polystyrene standard.) The molecular weight was controlled by varying the NCO/OH ratio. The degree of polymerization, $Xn = (1+r)/(1-r)$, where r is the ratio of the reactants. The cure of the polymers was monitored by ATR-FTIR by measuring the relative absorbance = $\log (I_o/I_{NCO})/\log (I_o/I_{CH})$, where I_o is the incident radiation and I is the transmitted radiation. The disappearance of the isocyanate band at 2265 cm^{-1} was monitored. The butylstannoic acid(BSA) concentration was also varied in order to vary the degree of polymerization, though was high enough at all concentrations that complete polymerization was attained in all samples. It was found that BSA did react somewhat, as the molecular weight decreased slightly with increasing BSA concentration.

Instrumentation.

Molecular weights were measured on a Waters Gel Permeation Chromatograph using a Waters 600E pump, a 401 RI detector, and a Linear Ultrastyragel column.

Tensile strength, % elongation, and Young's modulus were measured on an Instron machine. Crosshead speeds were 20 in./min., at a 3 in. gauge length.

Hardness measurements were made by hand with a Shore A durometer instrument.

Glass transition temperatures were measured on a Torsional Braid Analyzer (Plastics Analysis Instruments, Inc., Princeton, N.J.) Fiberglass braids were impregnated with the monomer mixtures and the polymers were cured on the braid. The fiberglass braid showed flat response over the temperature ranges used in the experiments.. Relative rigidity was measured as F^2 which is proportional to $1/P^2$ where F= frequency of the oscillation of the sample and P is the period in seconds. F^2 is proportional to the in-phase elastic portion of the shear modulus (G'). The log decrement [$\Delta = \ln (A_i/A_{i+1})$], where A is the amplitude of the generated wave, was measured as the log of the ratio of the amplitudes of successive oscillations, and is directly proportional to the ratio of the out-of-phase or viscous portion of the shear modulus (G") to G' [$\Delta = \pi G''/G' = \pi \tan \partial$], where ∂ is the phase difference between the stress and the deformation.

Differential Scanning Calorimetry was done on a Dupont 910 DSC. Heating rates were 10°C/min. under a nitrogen atmosphere. Sample weights varied from 16.5 to 80mg.

Melting points were measured on a Dupont Thermomechanical Analyzer 943 using a 1 gram sample load and a heating rate of 5°C/min.

III. Results and Discussions

Instron Results.

Fig.3 shows the contour plot of tensile strength as a function of average molecular weight and urethane content. The tensile strength increases with increasing urethane content and molecular weight. At low urethane content, below 12-12.5%, there is a stronger dependence on molecular weight than at high urethane content. One

can see that the shape of the contour changes as the urethane content is increased, i.e., the slope becomes steeper approximately above 160,000 MW, and urethane contents above 12 - 12.5%.

Fig. 4 shows the contour plot of % elongation as a function of urethane content and increasing molecular weight. The % elongation decreases with increasing urethane content and with molecular weight. The shape of the contour stays constant, but contour spacing increases from low to high urethane content and molecular weight. This means that the rate of decrease in % elongation slows. There are two phenomena responsible for this. The chain entanglement density is increasing with molecular weight, making it more difficult to unravel the coiled polymer chains. The elastic modulus is increasing due to an increased resistance to uncoiling. Above a particular molecular weight maximum, the chains will break before they can unravel further. Secondly, the hard segment content increases as the urethane content increases. If the hard segments are viewed as reversible crosslinks or fillers that reinforce the material,[4] and thereby increase strength, then the hard segment domains also resist uncoiling or elongation, regardless of the molecular weight.

Hardness Measurements.

Fig. 5 shows the contour plot of Shore A durometer hardness as a function of urethane content and molecular weight. The hardness increases as the urethane content increases and increases with molecular weight, but again the dependence on molecular weight is stronger at low urethane contents. From the contour spacings it is seen that the hardness increases faster generally from 10-13% urethane content than from 13-14%. This behavior is consistent with the elastic modulus behavior.

The development of the morphology can be followed with time by simply measuring the hardness as a function of time (Fig. 6). The plot shows that it takes approximately two weeks for a polyurethane with Wu = 14.0% to approach a constant Shore A durometer hardness of 80. At first one might guess that these low modulus materials should develop their final properties much faster than two weeks.. This may be the case for polymers made with butanediol, being that

the hard segments are more symmetrical, and there is a stronger hard domain interaction. The thermodynamic driving force for segregation is smaller for pentanediol and the time to segregate is long.[5] It may be that the mutual exclusion of hard and soft segments is rapid, but the development of order in the hard domains is slow.[6]

An interesting practical implication is that the shear modulus of the material stays low for a period of time that is longer than the time for polycarbonate, in a glass/plastic laminate, to cool to room temperature after autoclaving. Hence, shear stress can be absorbed by the interlayer as the polycarbonate cools and contracts. After the plastic cools to room temperature and the modulus of the interlayer continues to increase, it should have very little effect on the contour of the laminate.

Differential Scanning Calorimetry.

The DSC plot of a 13.5% urethane content polymer (Fig. 7) shows a hard segment melting transition(Tm,h) at around 79°C. If one runs the same sample within one hour of the first run the hard segment transition does not reappear. Again, there is a time dependency for domain organization as was seen in the durometer data of Fig. 6. The temperature of the first-run transition, Tm,h, did not differ for the 12% and 14% polymers. What did change was the magnitude of the enthalpy of the transition. No hard segment melting transition was observed for the 10% urethane content samples, which had a hard segment content of only 3.3%. Fig. 8 shows the melting enthalpy or enthalpy of fusion plotted as a function of weight % hard segment. The enthalpy increases with increasing hard segment content above 3.3%.

Generally one observes a Tm,h increase with increasing hard segment content, but usually at much higher hard segment contents than we have prepared, and with reinforcing hard segments.[1] This was not seen using pentanediol at relatively low hard segment contents. The amount of mixing of the hard and soft segments can, however, be determined by how high the soft segment Tg,s increases above the pure soft segment oligomer Tg,o. Thus, the greater (Tg,s -Tg,o), the greater the mixing of the two phases, as the hard segments restrict the mobility of the soft segments. For the 14% urethane content polymers Tg,s = -53°C and Tg,o = -82°C. Thus (Tg,s

-Tg,o) = 29°C. The soft segment Tg is 29°C above that of the pure oligomer polyol, indicating phase mixing. The anchoring of soft segment ends in the hard domains has been shown to raise the *Tg,s* by 4°C.[5] Phase boundary hydrogen bonding can also increase the *Tg,s*.

Polyether soft segments in most cases will yield more complete phase separation than polyester soft segments.[9] This can be explained by comparing solubility parameters. The solubility parameter for the polyether is lower than that of the ester, and much lower than that of the short chain diol. The molecular weight differences of the polyols also determine the degree of phase separation.[1] A 2000 molecular weight polyether polyol-based polyurethane will phase separate more completely than a polyurethane made with a 1000 molecular weight polyol. This can be seen by the *Tg,s* differences. The *Tg,s* for the 2000 molecular weight soft segment will be closer to the *Tg,o* (oligomer) than the 1000 molecular weight soft segment Tg,s. This can be interpreted to mean that fewer hard segments exist in the 2000 molecular weight soft segment phase. One should also note that at the same urethane content the hard segment content will be higher using a 2000 MW soft segment than a 1000 MW soft segment (see Fig. 9).

Torsional Braid Analysis.

A Torsional Braid Analyzer (TBA) is a freely oscillating torsional pendulum that utilizes a fiberglass braid. The braid is impregnated with a polymer, plasticizer, or virtually anything that will stay trapped within it. Even powder coatings have been tested successfully using TBA. The braid allows a polymer to be tested at temperatures higher than the maximum temperature at which the polymer can support a weight without deforming. The instrument torques the sample about 4° and releases it to oscillate at its own resonant frequency, which is usually below 1Hz. The material properties are calculated as shown in Fig. 10.

Our polymers were cured on a braid in the TBA apparatus. See Fig. 11 for the results from a typical isothermal experiment. Fig. 12 shows the log decrement vs. temperature plot for three different urethane contents from this study: 10, 12, and 14%. The Tg's from the maximum in the log decrement plot are -44°C, -29°C, and -5°C,,

respectively. The magnitude of the log decrement plot decreases with increasing urethane content indicating phase mixing. The purity of the soft segment can also be gauged by the steepness of the slope of the low temperature side of the log decrement plot. The slope decreases as urethane content is increased indicating again phase mixing. In Fig. 13, which is a contour plot of Tg as a function of molecular weight and urethane content, we can see that the Tg is a linear function of urethane content between 10 and 14%. It is not very dependent on molecular weight between 77,000 and 260,000. Though Tg usually goes up with increasing molecular weight because of a decrease in free volume, and fewer chain ends, the dependence varies for different polymers and molecular weight ranges.

A plot of F^2 vs. temperature is shown in Fig. 14. The Tg's of the three different urethane content polymers can be seen on this plot as the inflection points in the plots. The width of the Tg region or the steepness of the slope of the Tg region is different for each polymer. The width increases with higher urethane contents, which confirms a greater heterogeneity or phase separation for the higher hard segment polymer. Notice also that the rubbery plateau extends to lower temperatures the lower the urethane content.

Glass/Plastic Laminated Beams.

Flat 40" x 6" polycarbonate/glass beam samples were laminated in an air autoclave at 210°F and the center deflections of the the samples were measured when the laminate cooled to room temperature. Measurements were made on a flat table with the glass surface oriented up. See Fig. 15 for a schematic of the laminate. Fig. 16 is a step surface plot of beam center deflection as a function of urethane content and molecular weight. At low urethane contents the center deflections increase very slowly with increasing molecular weight, but at high urethane contents the deflections jump up rapidly. At low molecular weights the center deflections increase slowly with urethane content, but more rapidly with urethane content at high molecular weights. Thus, precise control of the molecular weight is critical for use of this type of interlayer in glass/plastic laminates. The shear modulus and the resulting changes in the ability to take up shear are very sensitive to molecular weight changes, particularly at urethane contents above 12%.

Fig. 17 shows a plot of center deflection vs. temperature for a 12.5% urethane content polymer with a Mw = 150,000. It shows similar data for three 13.5% urethane content polymers with molecular weights equal to 86,000, 210,000, and 305,000. The beams made with the 13.5% urethane, 86,000 molecular weight interlayer (Tg = -14°C) and the 12.5% urethane,150,000 molecular weight interlayer (Tg= -21°C) have virtually identical behavior over a wide temperature range. This is so even though the glass transition temperature is lower for the 12.5% urethane content polymer. These beams were completely flat at room temperature. Shown for comparison in the figure are data for a polyester polyurethane and a low temperature melting interlayer that is commercially available for use in glass/plastic laminates. Neither is at all flat at room temperature. Although two experimental beams were comparatively flat at room temperature, warpage at low temperatures resulted in center deflections of approximately 1.25" at -20°F for these lower modulus polymers. Deflections extended beyond this for the higher modulus polymers. The effect of wide differences in molecular weight, at constant urethane content, on the deflection of the beams is also reflected in the plot.

Concluding Remarks and Future Direction.

The proper design and control of polyurethane interlayer properties can result in reduced stress and improved contour in glass/plastic laminates at room temperature and above. That has been demonstrated here. Furthur improvements in interlayers should focus on a widening of the service temperature range at low temperatures. Though flattening the modulus - temperature behavior is important, ideally, the material should be able to take up shear stress over a wide range of temperatures. Silicones, for example, show very little modulus change with temperature, but cannot shear easily because they are crosslinked. A commercially available heat-vulcanized silicone sheet material was also tested in our lab in a simple beam construction. It showed a center deflection of 0.6" at room temperature when laminated at 210°F. Work in our lab with polydimethylsiloxane-polyethyleneoxide organofunctional siloxane-based polyurethanes has shown that transparent polymers are possible with glass transition temperatures as low as -50°C (-58°F) (See Fig. 18) and flatter modulus - temperature behavior than the polytetramethyene oxide glycols. Cooper, *et. al.*, reported on

a series of polydimethylsiloxane-polyurethane elastomers based on MDI, 1,4 butanediol, and an organofunctional polydimethylsiloxane that showed glass transition temperatures as low as -124°C with a very flat modulus over a very wide temperature range. See Fig. 19.[10] These types of hybrid polymers may greatly enhance the performance of glass/plastic transparencies over a wider temperature range than is possible with a polyether polyurethane.

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3) Saunders and Frisch, *Polyurethanes: Chemistry and Technology,* Part 1, 1962.

4) Cooper, S., et. al., *Polymer,* 1985, Vol. 26, January, 55-69.

5) Pascault, et. al., *J. of Polymer Sci.,* Vol. 21, 415-423(1983).

6) Kwei, T. K., *J. of Applied Polymer Science,* Vol. 27, 2891-2899(1982).

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8) Roller, M., *J. Coatings Technology,* Vol. 54, No. 691, 33-40, August, 1982.

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10) Cooper, S. L., et. al., *J. of Polymer Sci., Polymer Physics Edition,* Vol. 23, 2319-2338 (1985).

Figure 1

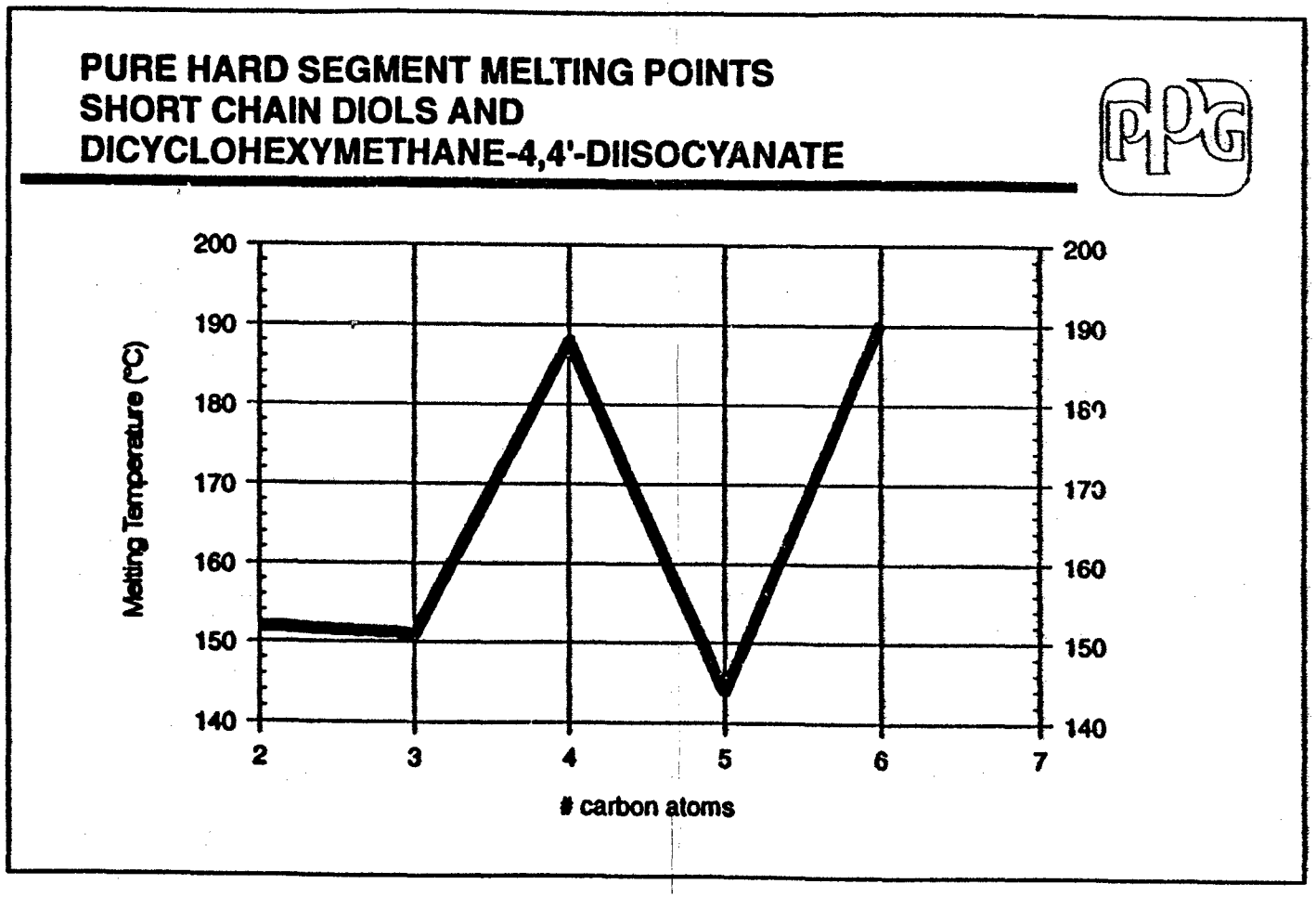

PURE HARD SEGMENT MELTING POINTS
SHORT CHAIN DIOLS AND
DICYCLOHEXYMETHANE-4,4'-DIISOCYANATE
PPG
Melting Temperature (°C)
200
190
180
170
160
150
140
2
3
4
5
6
7
carbon atoms

Figure 2

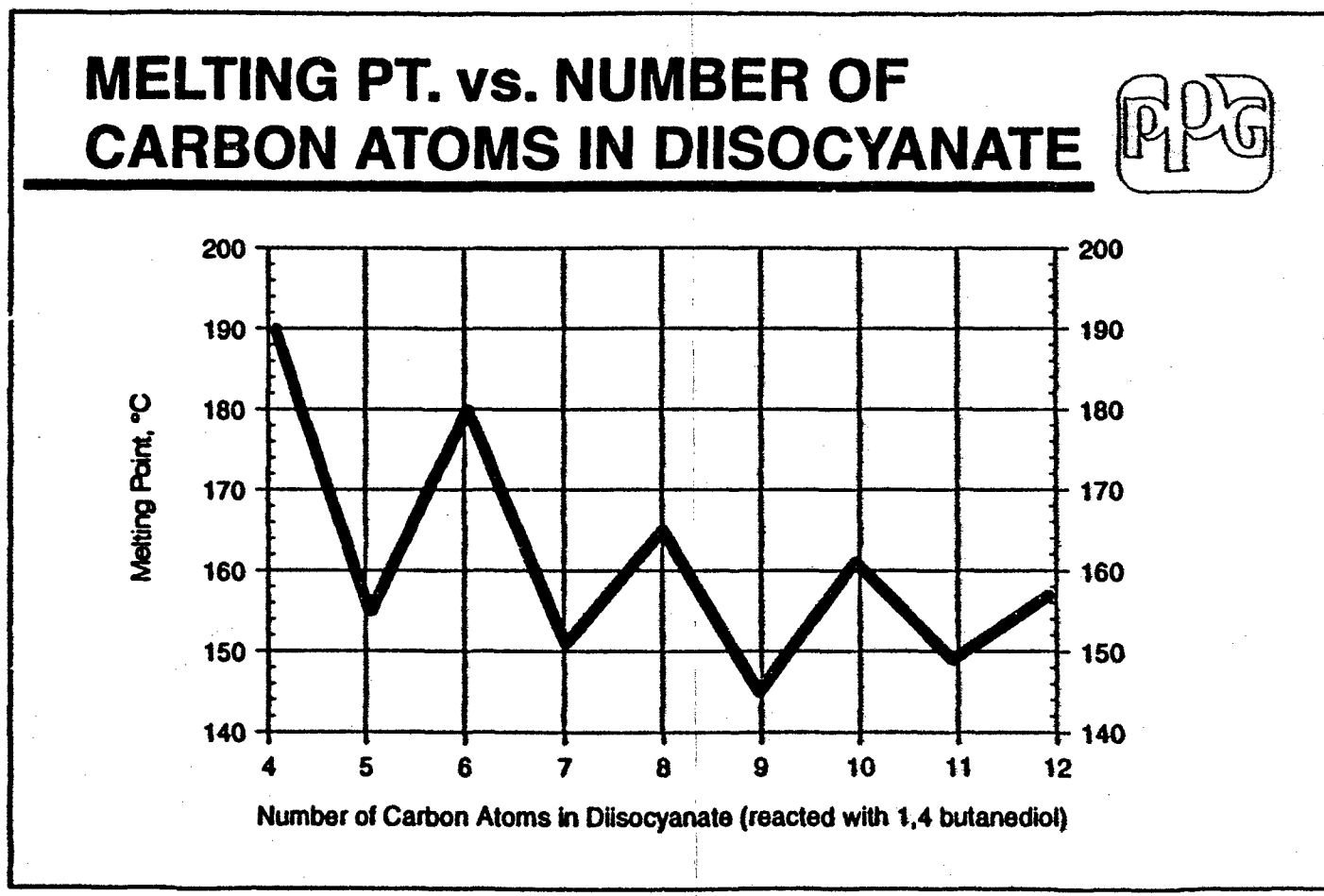

MELTING PT. vs. NUMBER OF CARBON ATOMS IN DIISOCYANATE
PPG
Melting Point, °C
200
190
180
170
160
150
140
4
5
6
7
8
9
10
11
12
Number of Carbon Atoms in Diisocyanate (reacted with 1,4 butanediol)

Figure 3

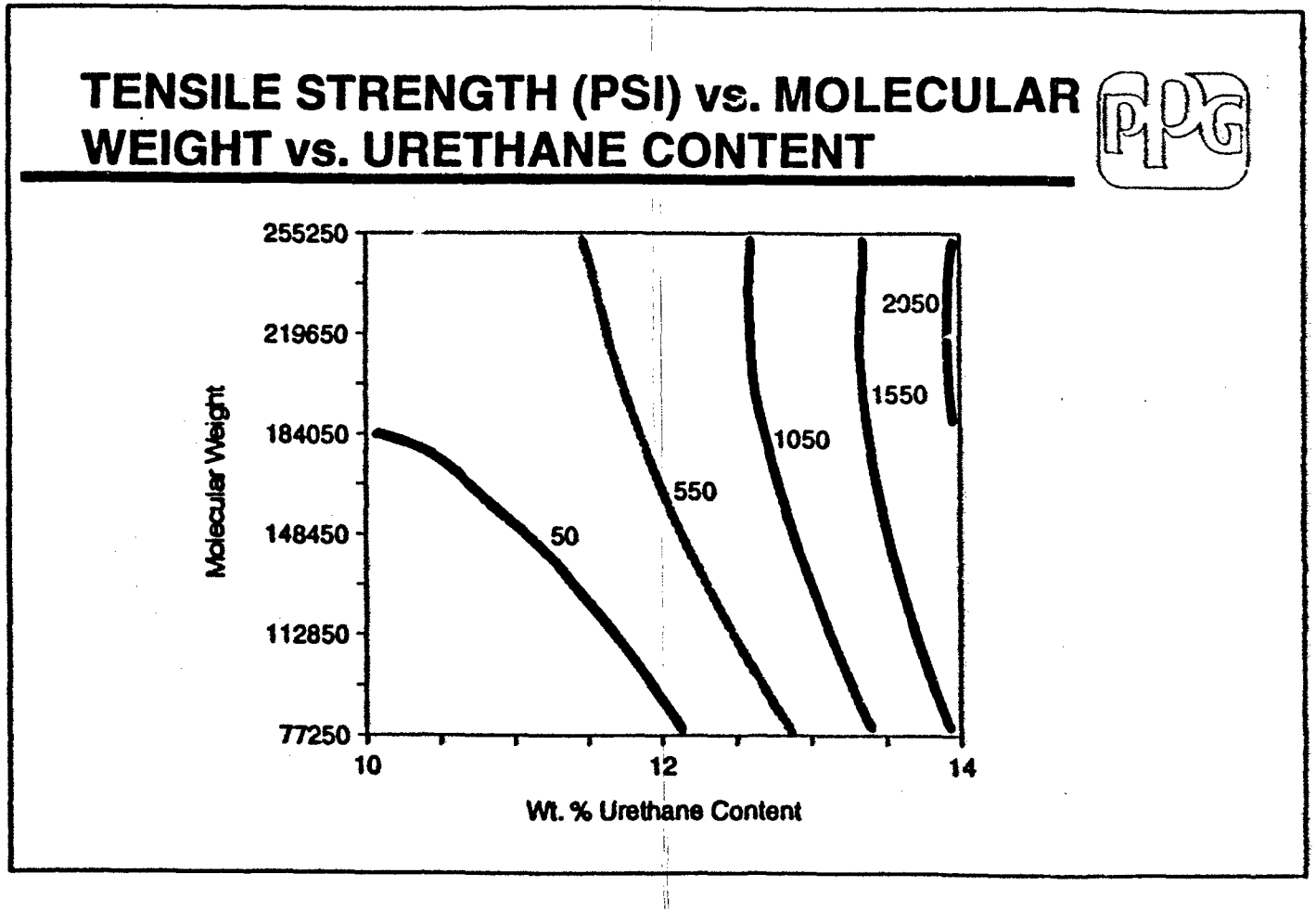

TENSILE STRENGTH (PSI) vs. MOLECULAR WEIGHT vs. URETHANE CONTENT
PPG
Molecular Weight
255250
219650
184050
148450
112850
77250
10
12
14
Wt. % Urethane Content
50
550
1050
1550
2050

Figure 4

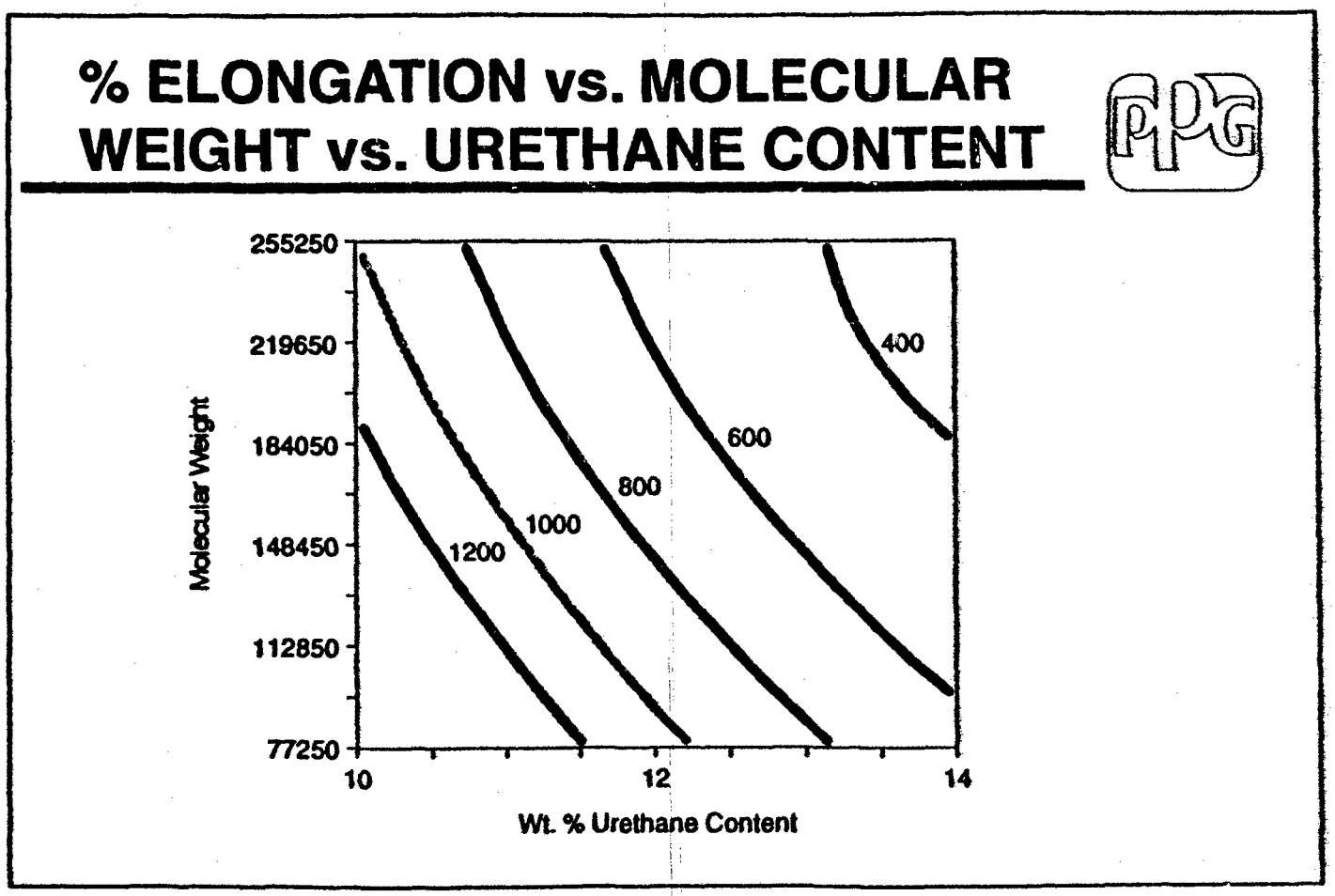

% ELONGATION vs. MOLECULAR WEIGHT vs. URETHANE CONTENT
PPG
255250
219650
184050
148450
112850
77250
Molecular Weight
400
600
800
1000
1200
10
12
14
Wt. % Urethane Content

Figure 5

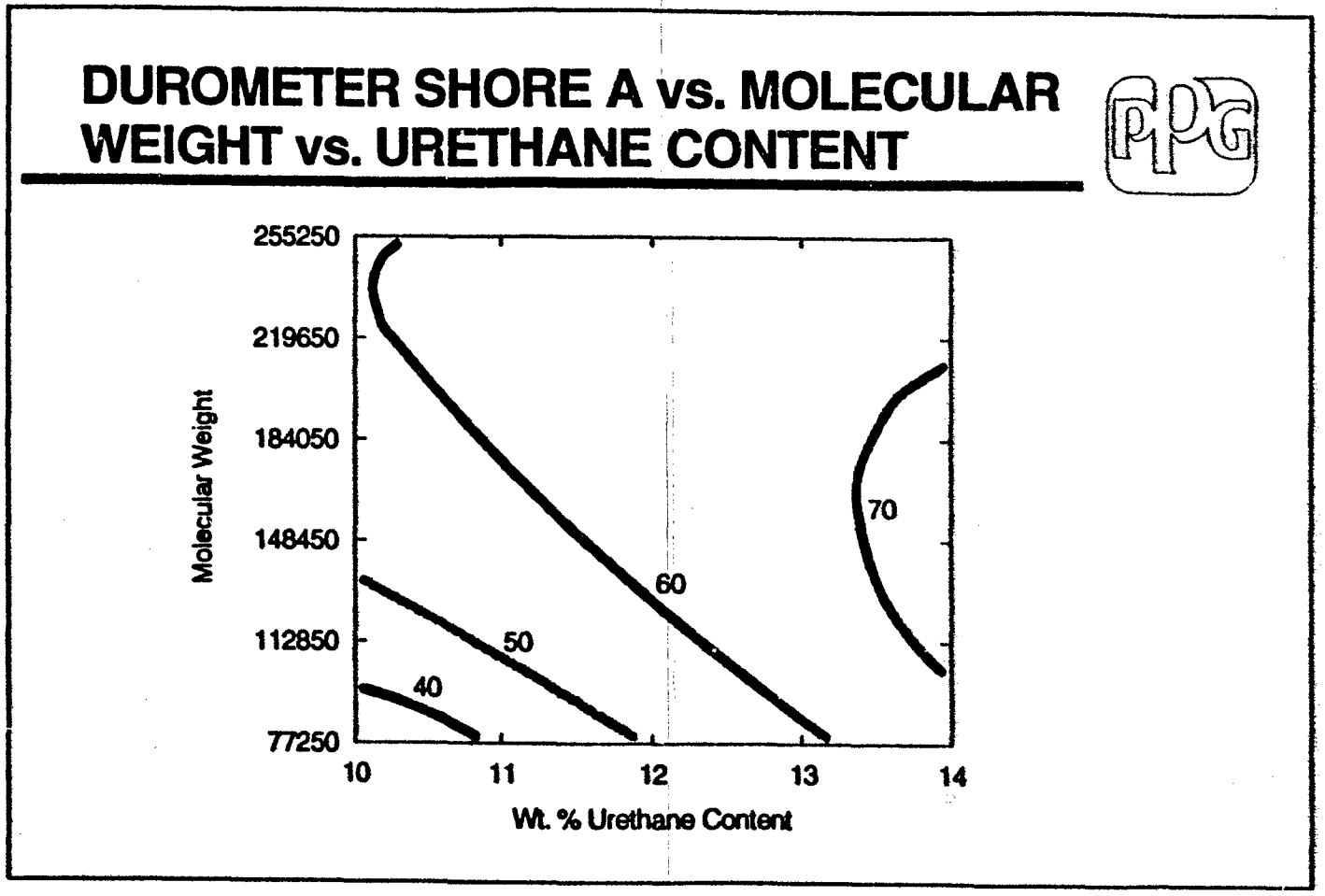

DUROMETER SHORE A vs. MOLECULAR WEIGHT vs. URETHANE CONTENT
PPG
Molecular Weight
255250
219650
184050
148450
112850
77250
10
11
12
13
14
Wt. % Urethane Content
40
50
60
70

Figure 6

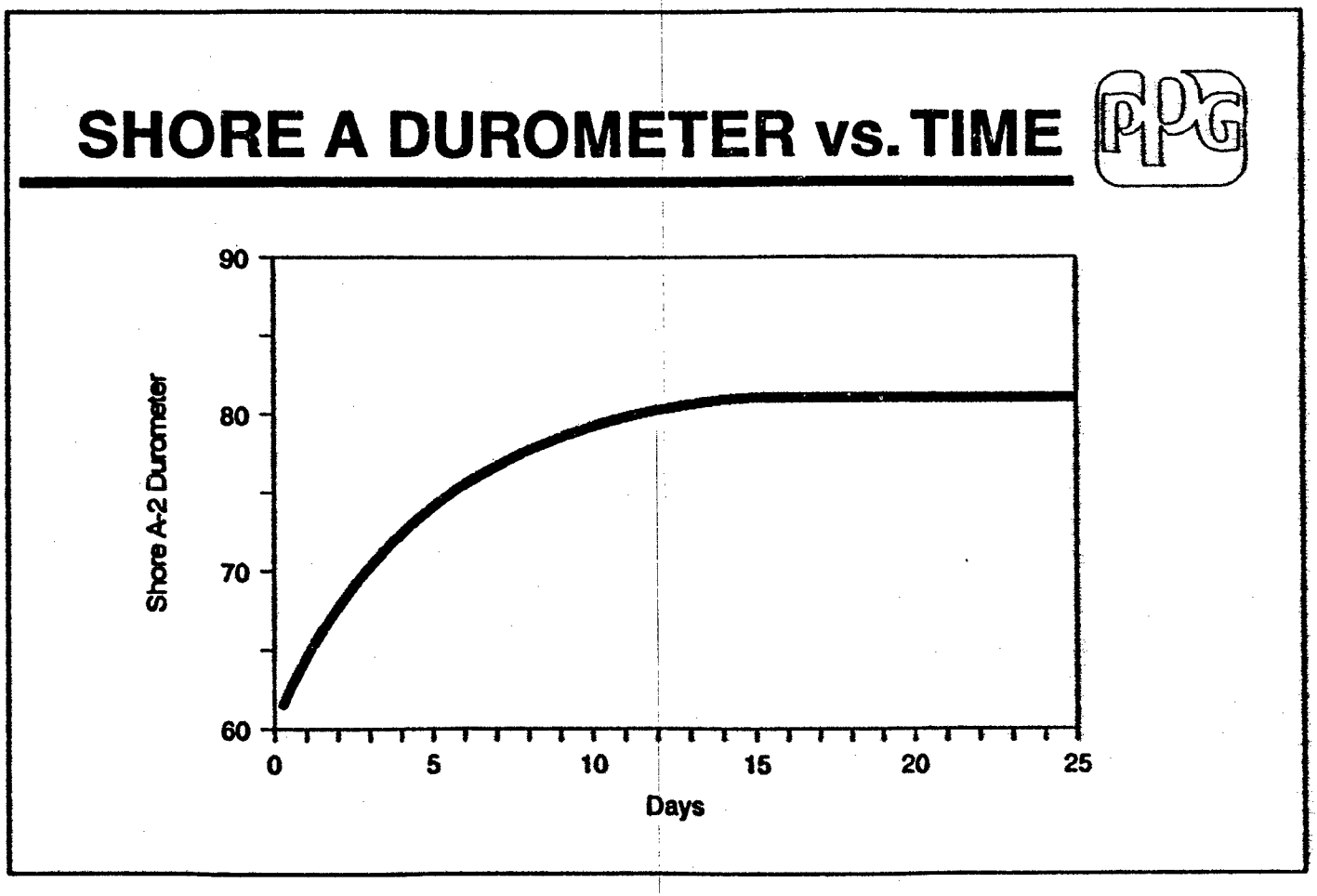

SHORE A DUROMETER vs. TIME
PPG
Shore A-2 Durometer
90
80
70
60
0
5
10
15
20
25
Days

Figure 7

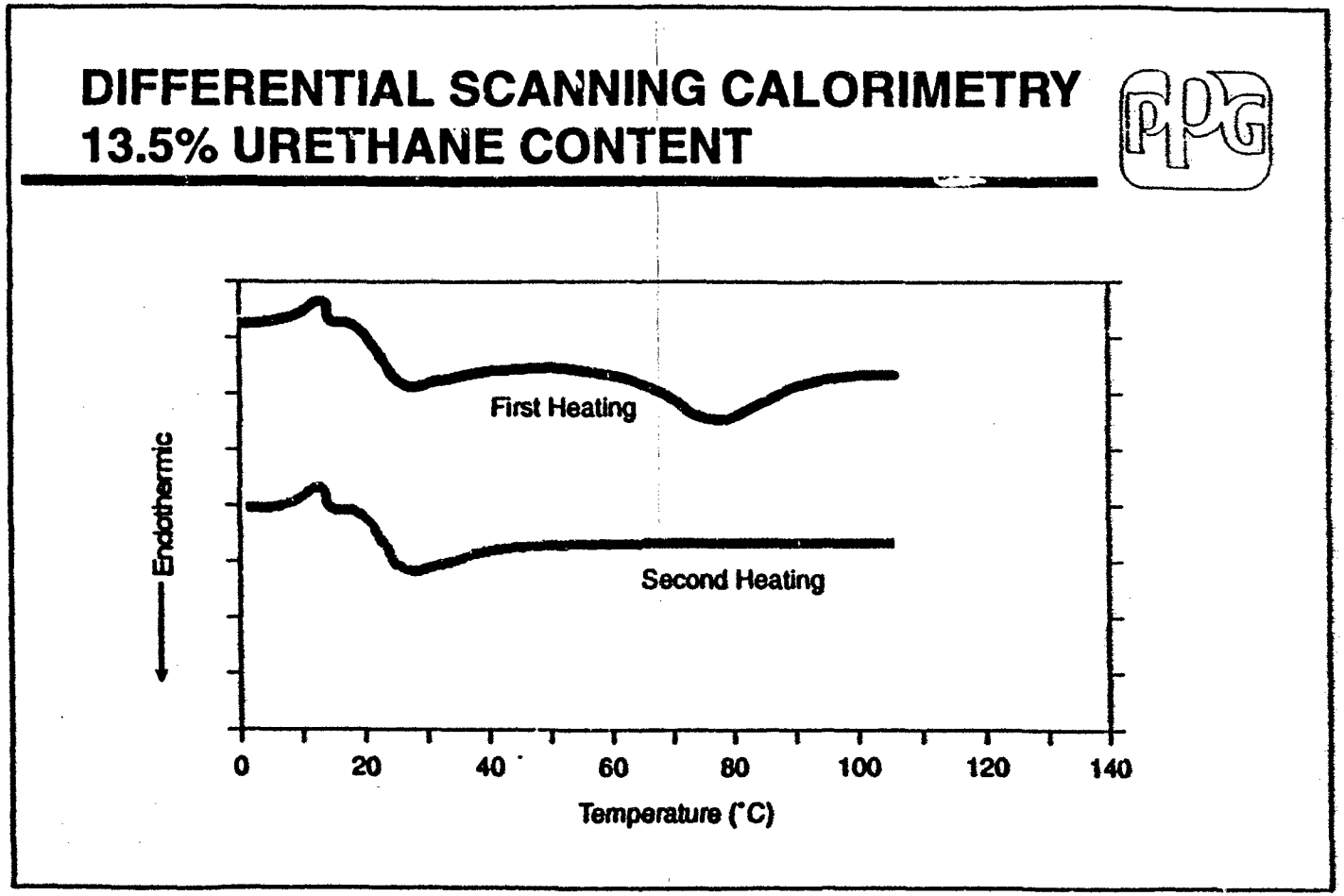

DIFFERENTIAL SCANNING CALORIMETRY
13.5% URETHANE CONTENT
PPG
First Heating
Second Heating
Endothermic
0
20
40
60
80
100
120
140
Temperature (°C)

Figure 8

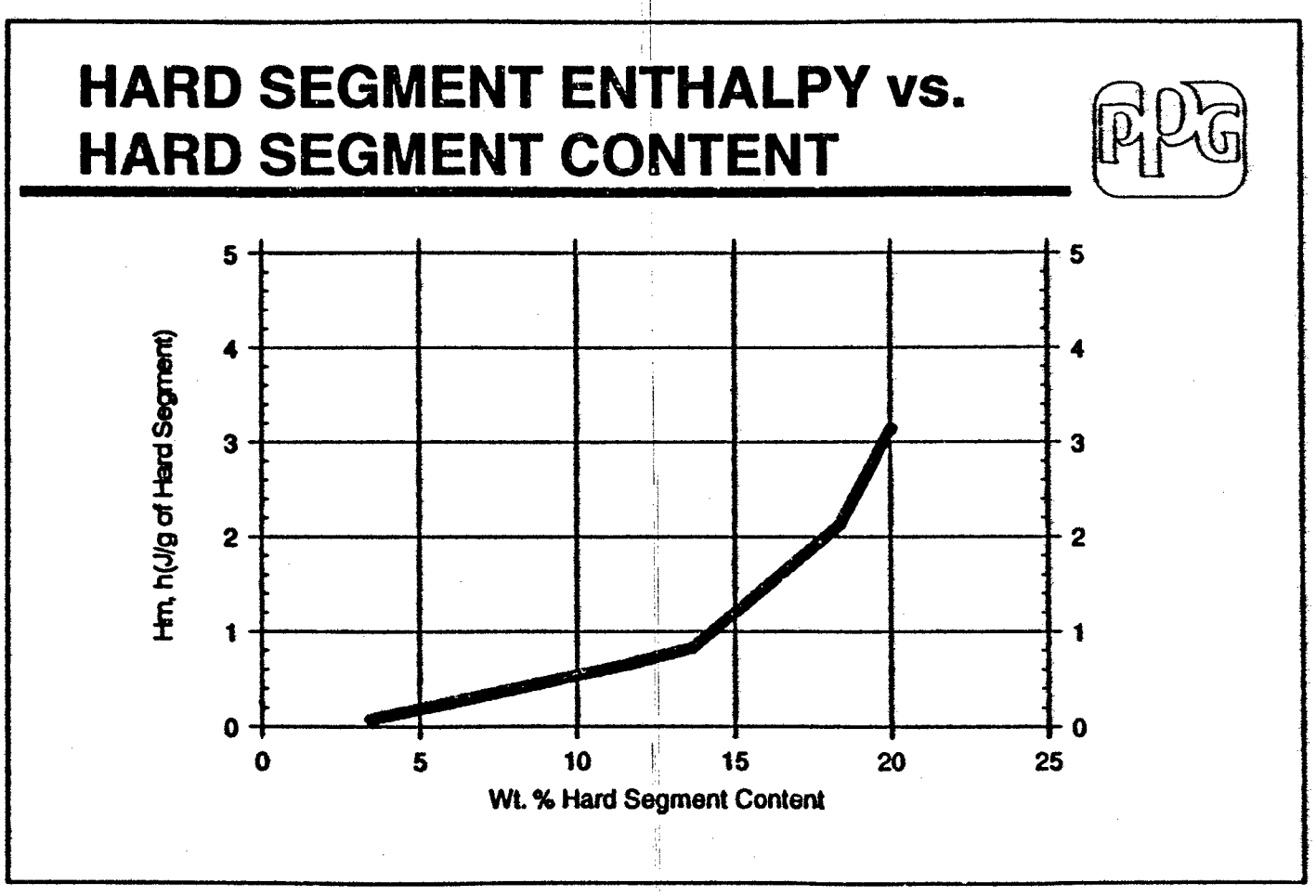

HARD SEGMENT ENTHALPY vs.
HARD SEGMENT CONTENT
PPG
Hm, h(J/g of Hard Segment)
0
1
2
3
4
5
0
5
10
15
20
25
Wt. % Hard Segment Content

Figure 9

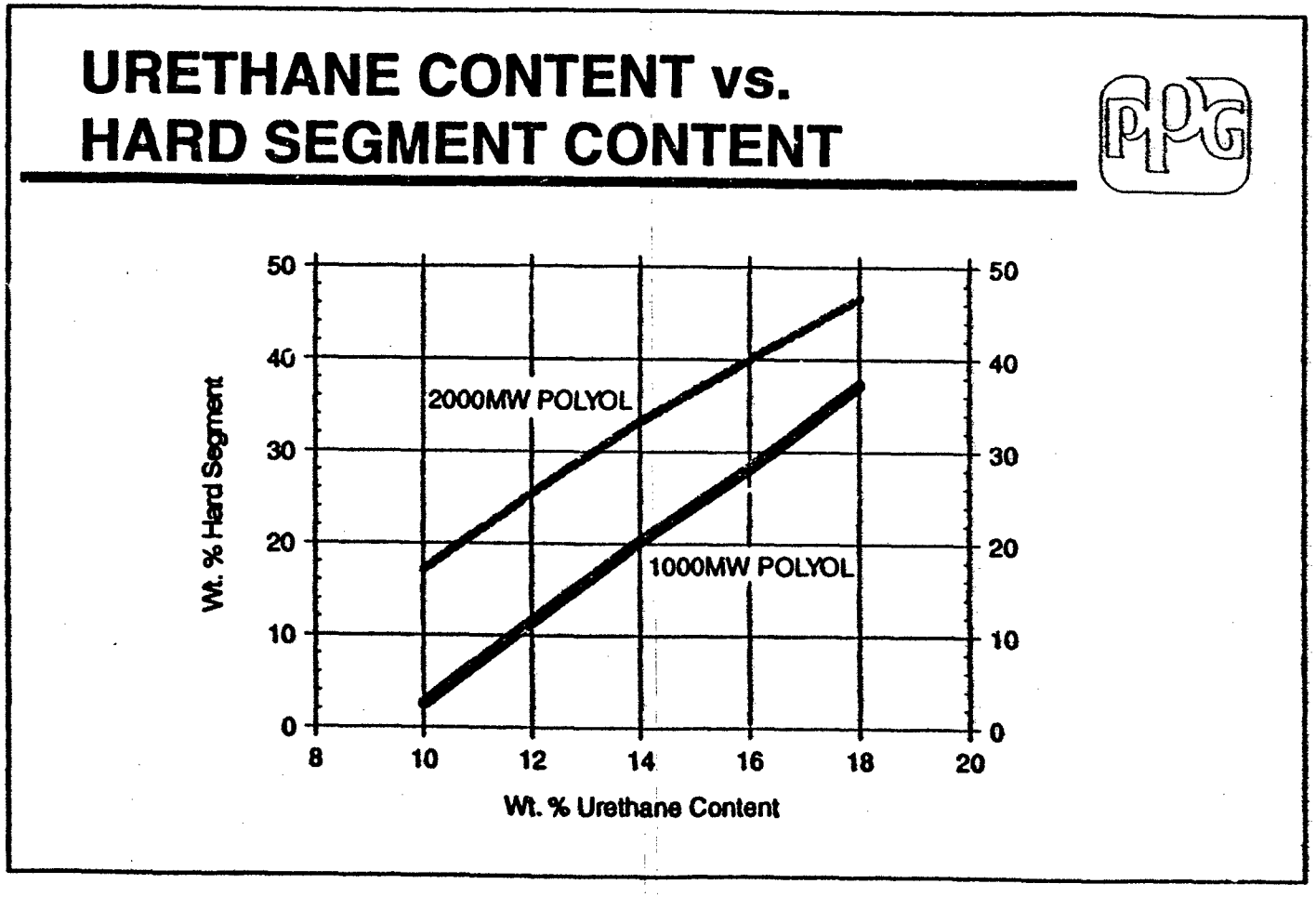

URETHANE CONTENT vs.
HARD SEGMENT CONTENT
PPG
Wt. % Hard Segment
50
40
30
20
10
0
2000MW POLYOL
1000MW POLYOL
8
10
12
14
16
18
20
Wt. % Urethane Content

Figure 10

MATHEMATICAL RELATIONSHIPS

SHEAR MODULUS = $G' = KI(2\pi/P)^2 [\ 1 + (\Delta/2\pi)^2\ ]$

$G' \cong 4\pi kI(1/P)^2$

LOSS MODULUS = $G'' = 4\pi KI\propto/P$

LOG DECREMENT = $\Delta = \pi G''/G' = \propto P = \pi TAN\delta$

Figure 11

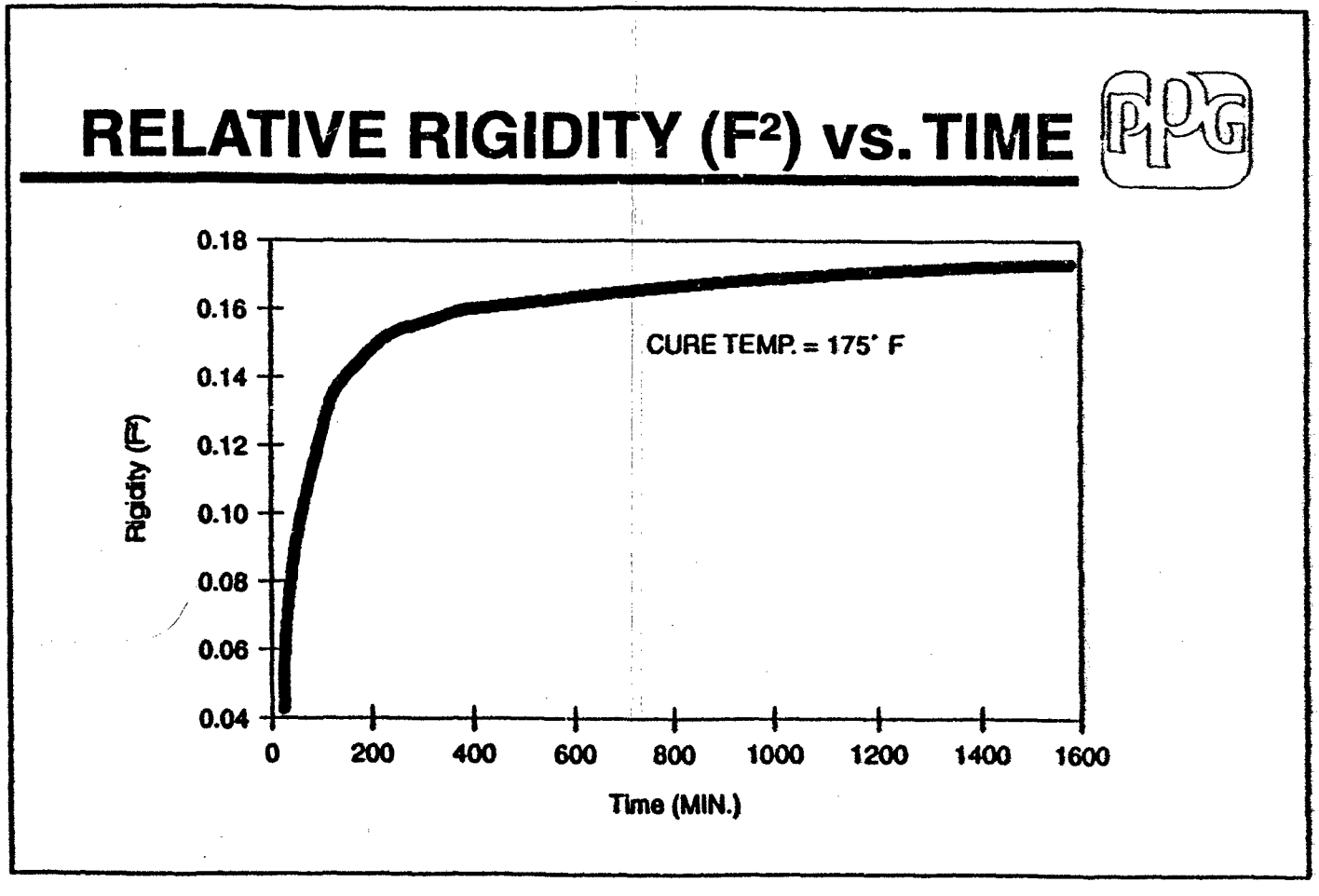

RELATIVE RIGIDITY (F²) vs. TIME
PPG
CURE TEMP. = 175° F
Rigidity (F²)
0.18
0.16
0.14
0.12
0.10
0.08
0.06
0.04
0
200
400
600
800
1000
1200
1400
1600
Time (MIN.)

Figure 12

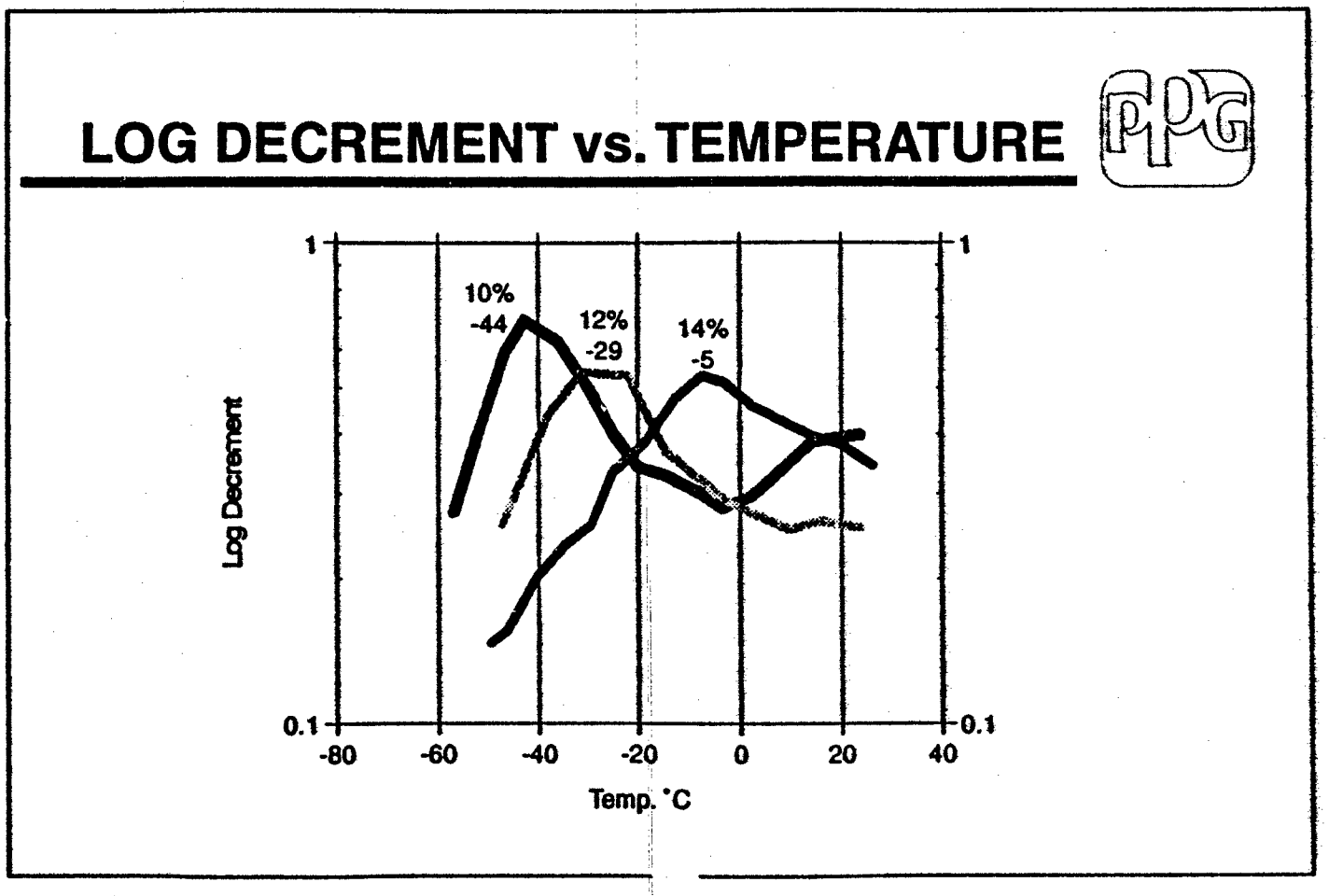

LOG DECREMENT vs. TEMPERATURE
PPG
1
1
10%
-44
12%
-29
14%
-5
Log Decrement
0.1
0.1
-80
-60
-40
-20
0
20
40
Temp. °C

Figure 13

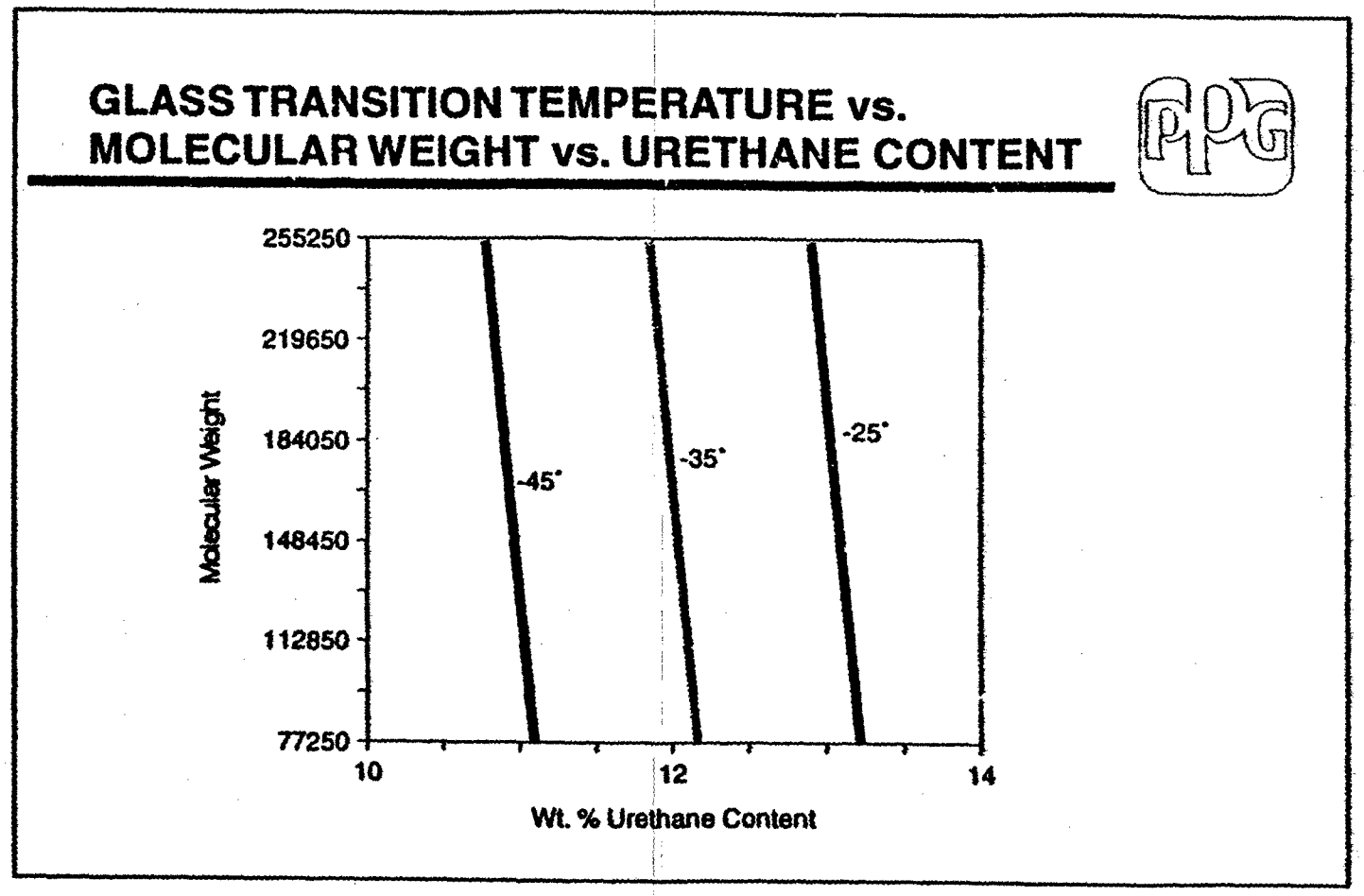

GLASS TRANSITION TEMPERATURE vs.
MOLECULAR WEIGHT vs. URETHANE CONTENT
PPG
255250
219650
184050
148450
112850
77250
Molecular Weight
-45°
-35°
-25°
10
12
14
Wt. % Urethane Content

Figure 14

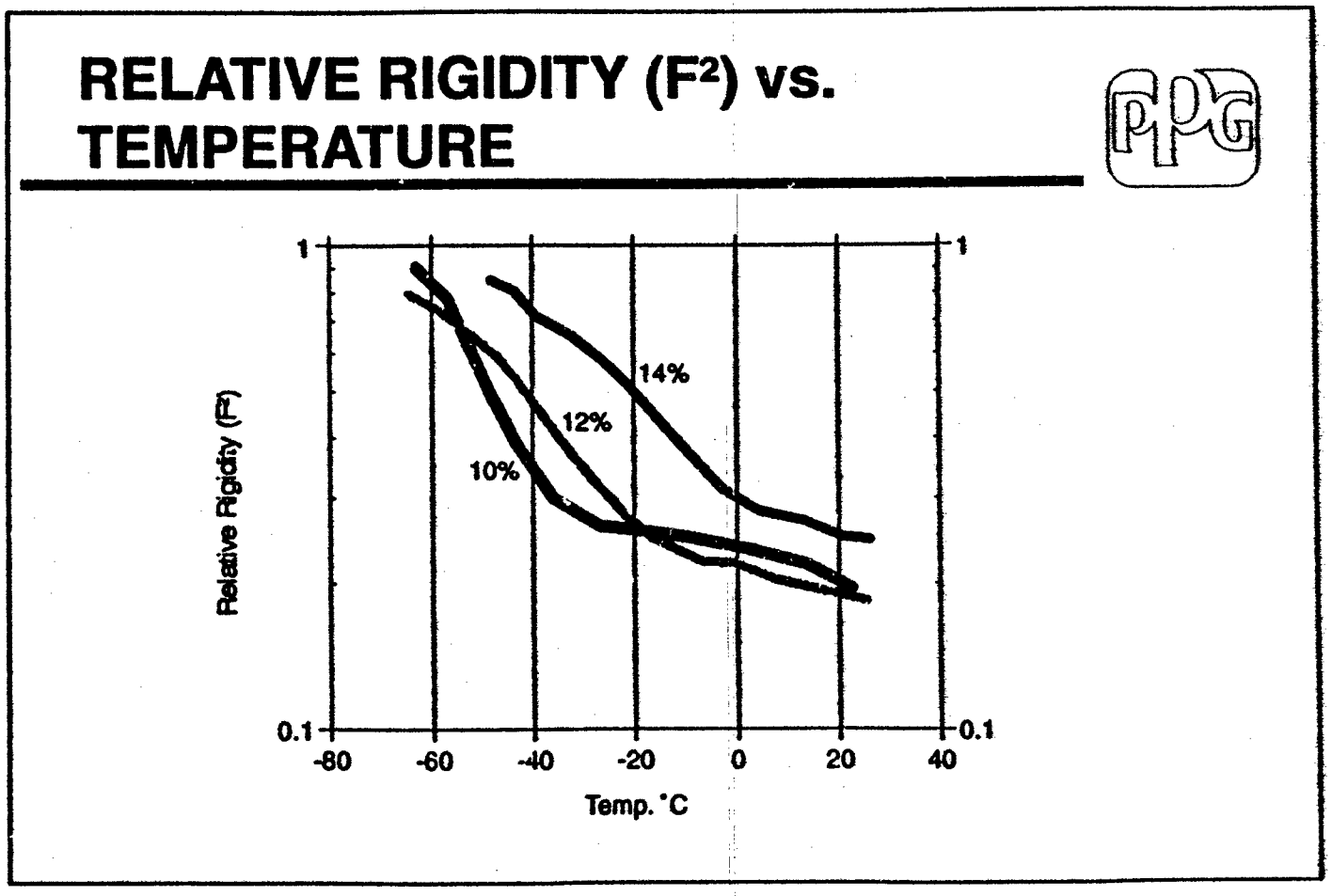

RELATIVE RIGIDITY (F²) vs. TEMPERATURE
PPG
Relative Rigidity (F²)
1
0.1
14%
12%
10%
-80
-60
-40
-20
0
20
40
Temp. °C

Figure 15

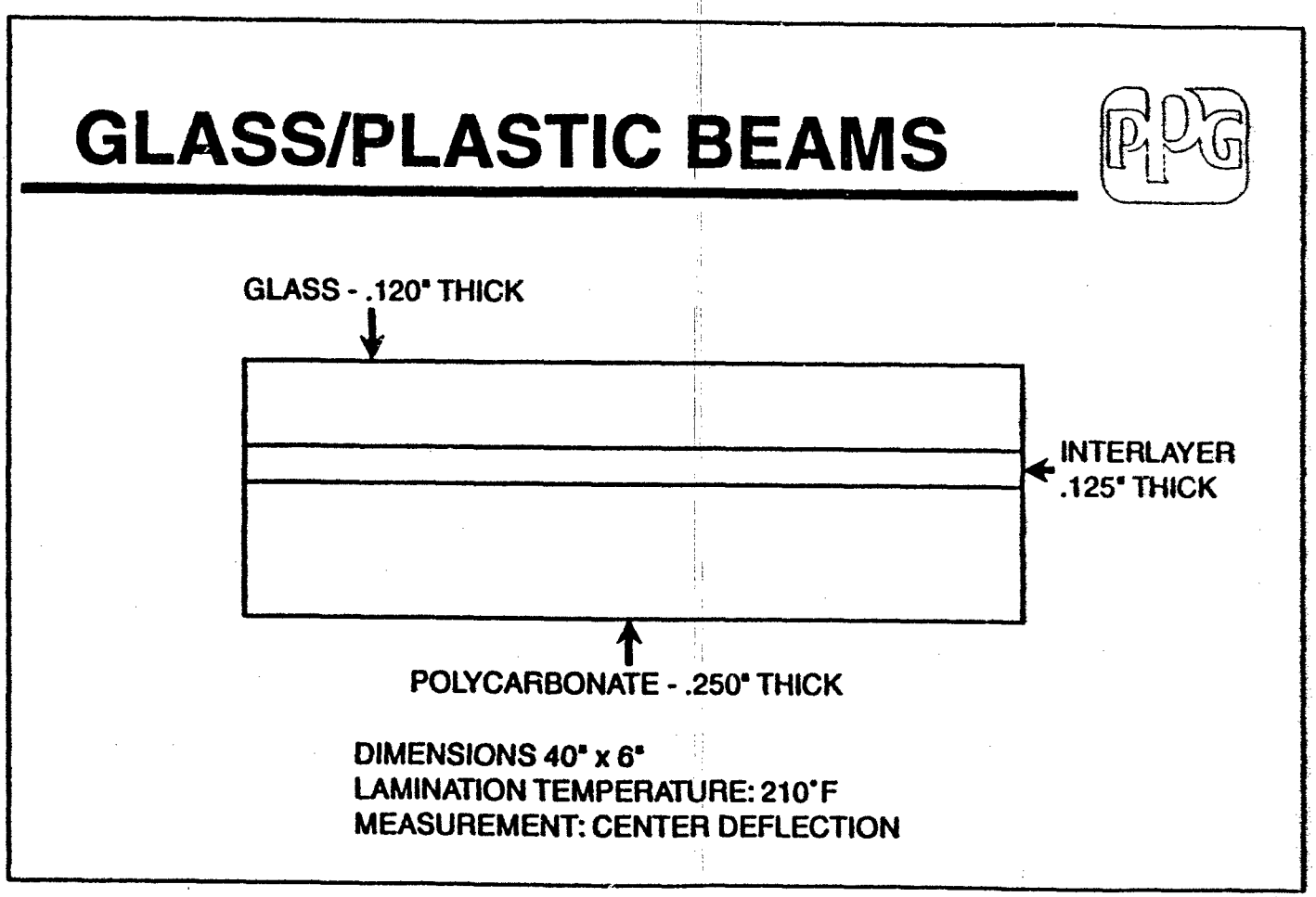

GLASS/PLASTIC BEAMS
GLASS - .120" THICK
INTERLAYER
.125" THICK
POLYCARBONATE - .250" THICK
DIMENSIONS 40" x 6"
LAMINATION TEMPERATURE: 210°F
MEASUREMENT: CENTER DEFLECTION

Figure 16

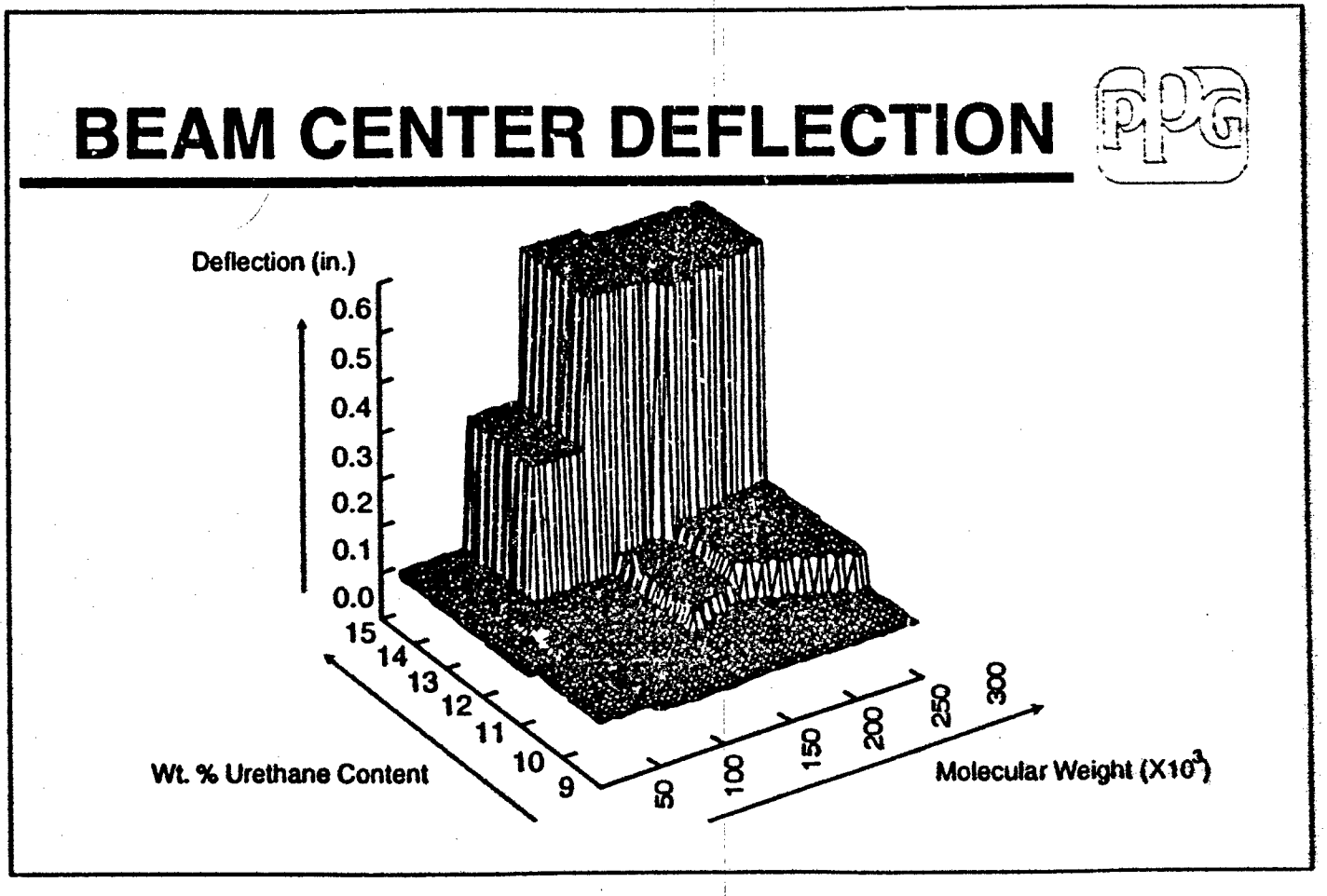

BEAM CENTER DEFLECTION
Deflection (in.)
0.6
0.5
0.4
0.3
0.2
0.1
0.0
15
14
13
12
11
10
9
Wt. % Urethane Content
50
100
150
200
250
300
Molecular Weight (X10³)

Figure 17

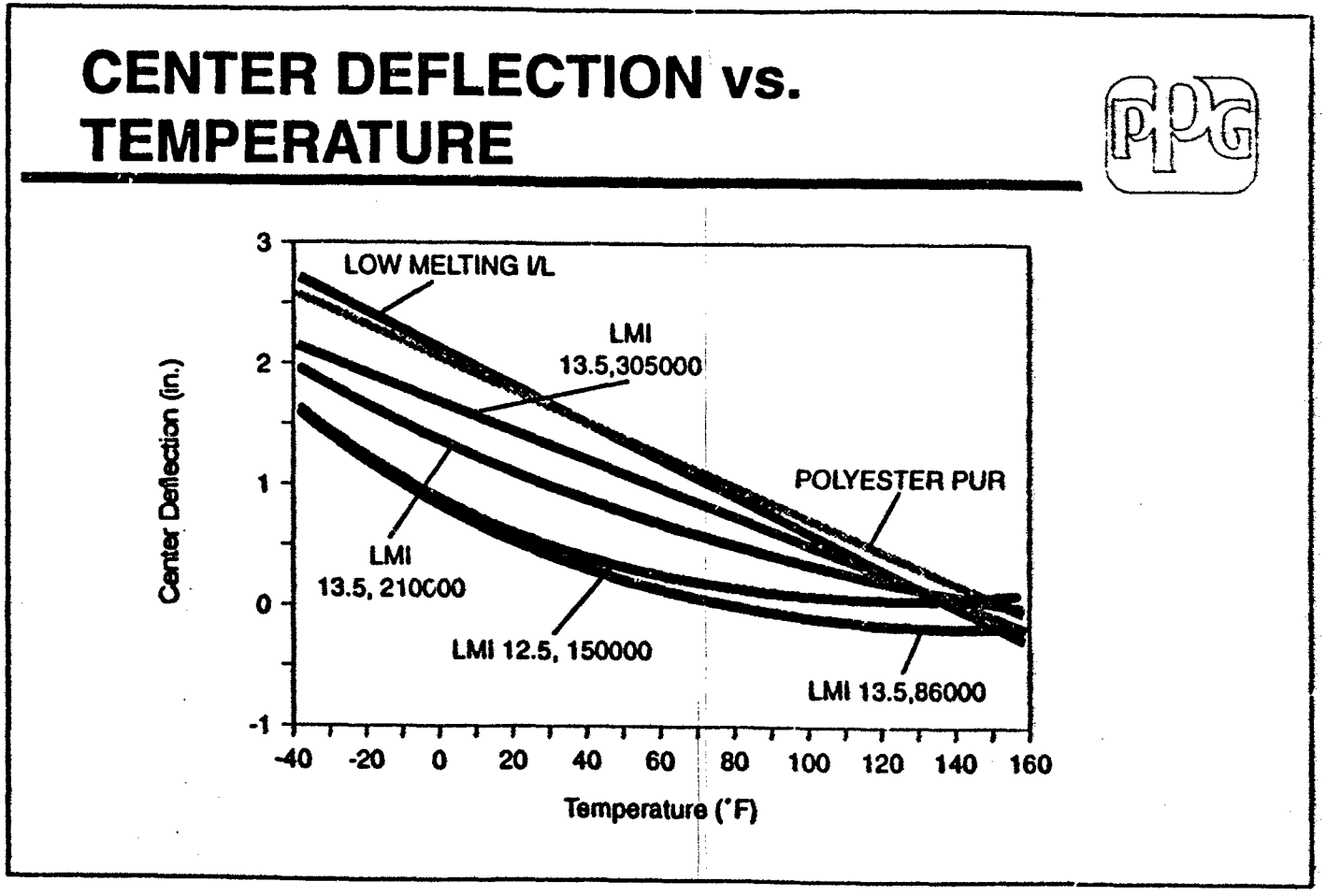

CENTER DEFLECTION vs. TEMPERATURE
PPG
LOW MELTING I/L
LMI 13.5,305000
POLYESTER PUR
LMI 13.5, 210000
LMI 12.5, 150000
LMI 13.5,86000
Center Deflection (in.)
3
2
1
0
-1
-40
-20
0
20
40
60
80
100
120
140
160
Temperature (°F)

Figure 18

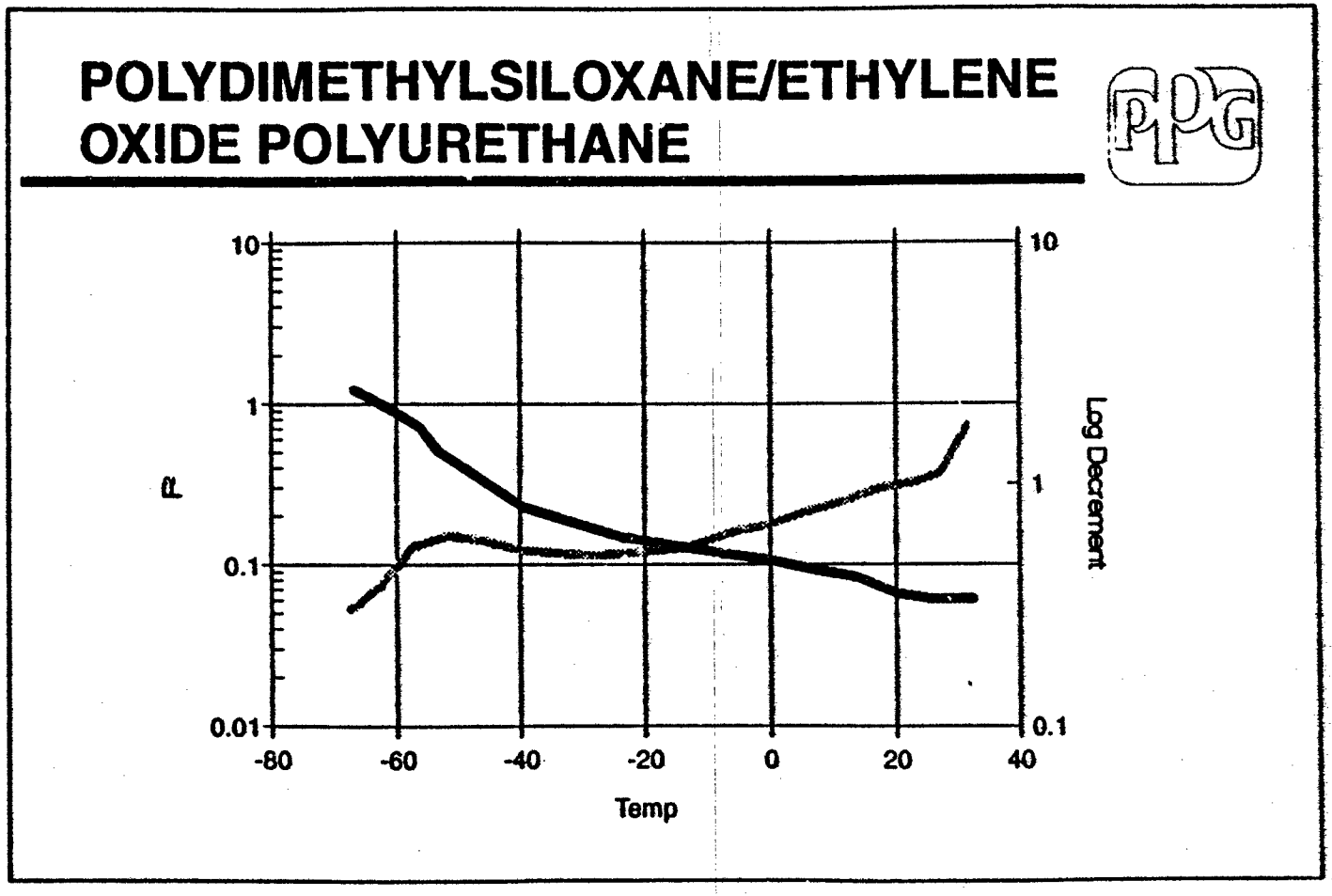
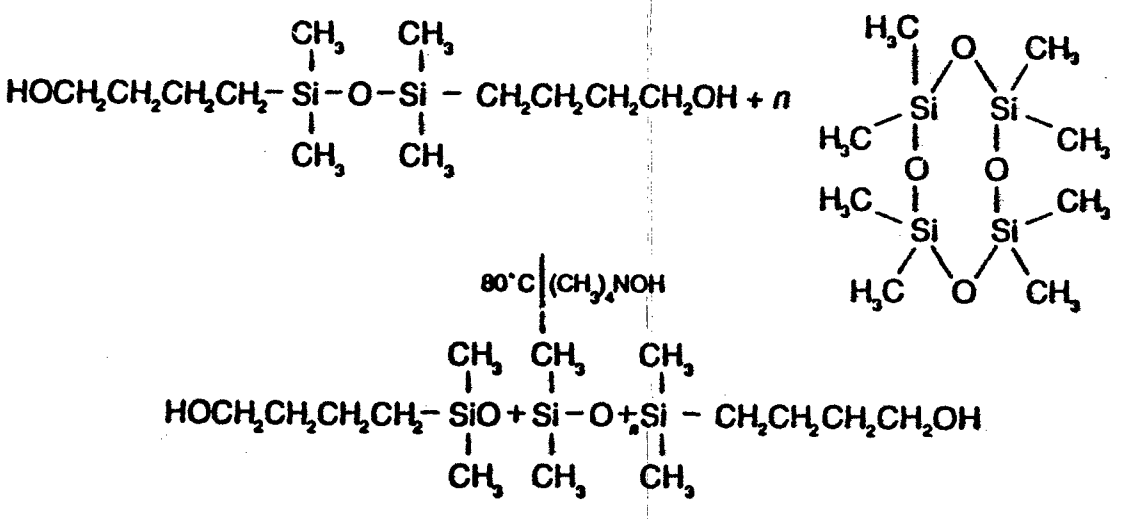

POLYDIMETHYLSILOXANE/ETHYLENE OXIDE POLYURETHANE
PPG
10
1
0.1
0.01
10
1
0.1
-80
-60
-40
-20
0
20
40
Temp
Log Decrement

Figure 19

ORGANOFUNCTIONAL POLYDIMETHYLSILOXANE SYNTHESIS



YU, COOPER, J. POLYM. SCI., 23, 2319-2338 (1985)

USE OF PHOTOCHROMIC MATERIALS IN AIRCRAFT TRANSPARENCIES

J. A. Andrechuk
M. V. Moncur
Pilkington Aerospace



PILKINGTON
AEROSPACE

USE OF PHOTOCHROMIC MATERIALS IN AIRCRAFT TRANSPARENCIES

J. A. Andrechak & M.V. Moncur

Pilkington Aerospace

ABSTRACT

New photochromic dyes have become available over the past several years through Pilkington research efforts targeted at ophthalmic, automotive, architectural and aerospace applications. Pilkington photochromic dyes incorporated into suitable polymer hosts show improved fatigue life and are deeper coloring than materials available previously. Significant dye activation occurs even in designs where the photochromic material is screened by a plastic or glass outer ply.

Pilkington photochromic dyes are readily incorporated into polyurethane materials used in aircraft transparencies and can be subjected to standard fabrication processes without degradation. Full-scale transparencies with photochromic polyurethane interlayers have been fabricated using conventional processing methods. Transparency systems incorporating such photochromics are being investigated for cockpit solar shading and in flash protection systems.

This paper describes properties of Pilkington photochromic dyes incorporated into transparent polymer hosts. Data are presented on light transmission characteristics of darkened and bleached states, darkening and fading rates, fatigue life, polymer host property effects and temperature effects. Processing studies and limitations are discussed and prior performance test programs related to aircraft transparency applications are described.

INTRODUCTION

Photochromism is defined as a reversible transformation between two states of a single chemical compound which is induced in at least one direction by electromagnetic radiation and in which the two states of the compound have different absorption spectra. The reverse conversion may be induced thermally or by radiation of a different wavelength (Ref. 1).

Photochromic compounds of interest for transparency applications are those with high visible transmittance in one state ("colorless" or "bleached" form) which is converted to a state with an absorption band in the visible range ("colored" or "darkened" form) when exposed to UV radiation. Several classes of organic photochromic materials meeting these criteria have been identified.

The Pilkington Group have a long-standing interest in organic photochromics for possible use in architectural, automotive, ophthalmic and aircraft applications. Pilkington researchers have synthesized and evaluated many of the dye types reported in the literature, including spiropyrans, chromenes, fulgides and spironaphthoxazines. Recent research efforts have primarily focused on spiroindolinonaphthoxazines (SINOs, Figure 1a) because of their good photochemical stability, range of available colors in the activated state and low background color in the bleached state. Chromene dyes (Figure 1b) have also been investigated in detail.

SINOs and chromenes are activated by radiation at UV/blue wavelengths, which induces a ring opening reaction (Figure 1). The open isomers have strong absorption bands in the visible spectrum. The reverse ring closing reaction is thermally induced.

MOLECULAR ENGINEERING

A synthesis program at Pilkington Group Research has yielded many new photochromic dyes, including over fifty new SINOs. Dye structure/property relationships have been established, resulting in improved ability to tailor dye structures to enhance performance features. Practical synthetic methods for dye manufacture have also been developed.

One significant Pilkington development was identification of a class of SINO dyes with enhanced photochromic performance: 6'-amino SINOs. Typical SINOs are activated by UV radiation and darken to a blue color, with peak absorbance at about 600 nm. Amino substitution at the 6' position shifts the activation band of the closed, colorless form to longer wavelength, resulting in increased activation in sunlight with UV filtering (i.e., through glass or a transparent plastic). 6'-Amino substitution shifts peak absorbance of the open, colored form to slightly shorter wavelength. Open 6'-amino SINO isomers also have increased absorbance, so dyes are deeper coloring than typical SINOs (Figure 2).

SINO and chromene dye structures can be modified to shift the closed state peak absorbance to longer or shorter wavelength, which significantly alters darkened state color. The rate of the thermal reverse (bleaching) reaction can also be modified by altering the dye structure. The practical utility of such dye structure modifications is illustrated in the following sections.

INCORPORATION INTO POLYMER HOSTS

Photochromic dyes can potentially be combined with transparent polymers by all standard processing techniques used to incorporate additives. With thermoplastic polymers, the dye may be incorporated by extrusion/melt blending to produce photochromic resin granules, which are processed by standard methods, including molding, extrusion and coating. Dyes can also be combined with thermoplastics directly during sheet extrusion, injection molding or coating processes.

With thermoset polymers, the dye is preferably dissolved in a prepolymer or monomer mix which is processed by conventional casting, coating, potting or bonding methods. Photochromic dyes have also been imbibed into polymer articles by use of a solvent carrier.

Pilkington studies have shown that polyurethanes are excellent hosts for SINO and chromene photochromics. Dyes can be incorporated into polyurethanes by casting, coating, extrusion or injection molding without degradation. Photochromic activity and stability in polyurethane hosts are excellent. Dye fatigue life can be enhanced and darkening and fading rates can be manipulated by tailoring polyurethane host compositions. Properties of photochromics in polyurethane hosts are described in the following sections.

SINOs and other photochromic dye types have also been successfully incorporated into cast acrylate plastic materials used for ophthalmics. These materials also show excellent photochromic activity and stability. Attempts to incorporate SINOs into polycarbonate have been less successful, as dyes suffer significant degradation under polycarbonate extrusion and injection molding conditions.

PHOTOCHROMIC SYSTEMS FOR SOLAR SHADING

Photochromic Activity

6'-Amino SINOs in polyurethanes display the full darkening range needed for solar shading at concentrations of less than 0.2 wt %. Under simulated standard Air Mass 2 (AM2) solar exposure conditions, a typical 6'-amino SINO (compound code CG1) in a polyurethane interlayer host fully darkens in 1 to 2 minutes (Figure 3). Absorbance or optical density (OD) levels of 1.0 to 1.5 are achieved with dye concentration of 0.05 wt % , corresponding to visible integrated light transmittance of <10%. Full fading is achieved in 2 to several minutes depending on host properties (Figure 4).

Solar darkening and fading characteristics of two other 6'-amino SINOs (PW8 and IB8) in a polyurethane host are presented in Figures 5 and 6. These data illustrate effects of dye structural modifications on photochromic activity and kinetics. IB8 incorporates a minor modification to the PW8 structure, which reduces the rate of the thermal bleaching reaction. This structural change has little effect on the darkening rate (Figure 5) but significantly slows fading (Figure 6).

The slower fading dye, IB8, also shows greater darkening after extended solar exposure. At a dye concentration of 0.04 wt %, darkened integrated visible transmittance of the IB8 sample is 2.9% compared to 7.6% for PW8. Since open and closed forms are in equilibrium under these conditions, reducing the back reaction rate shifts equilibrium towards the open state, resulting in a greater concentration of the colored form and hence higher absorbance.

Changes to the host polymer which alter the bleaching reaction rate have a similar effect on photochromic activity. For example, increasing the modulus of the host polymer would be expected to slow the bleaching reaction rate and thus increase darkening under equilibrium solar exposure conditions.

Effect of Filters

Conventional SINOs show little photochromic activity in sunlight when filtered by glass or transparent plastics. The activation band of 6'-amino SINOs is shifted towards the visible relative to SINOs, so they exhibit strong photochromism in filtered sunlight.

Figures 7 illustrates effects of UV filtering on activity of 6'-amino dyes (PW8 and IB8) incorporated at 0.04 wt % into a polyurethane. Filters are materials used in aircraft transparencies: glass, polycarbonate, Acrivue 350S stretched acrylic (per MIL-P-25690) and Acrivue 590 (a polyurethane plastic). Data show that darkening under simulated solar conditions is reduced by all filters but dyes are still very active and low equilibrium transmission levels are reached. Dye concentrations can be adjusted to readily compensate for the attenuating effect of a filter.

Temperature Effects

Temperature has a very significant effect on photochromic activity of SINO and chromene dyes under solar exposure conditions. The thermal bleaching reaction rate increases as temperature increases, resulting in reduced equilibrium darkening (Figure 8). At some temperature limit, which varies depending on the dye structure and the host, the bleaching reaction is so fast that photochromic activity is lost. Most SINOs in polyurethanes show little activity above 50°C.

Dye and host modifications that reduce the rate of the bleaching reaction will increase the upper temperature limit for photochromic activity. However, these changes also slow the fade rate. Such systems may bleach too slowly at low temperatures to be of practical utility for solar control. This strong temperature dependence is an inherent limitation of this type of photochromic dye. Use for solar control is limited to applications where the temperature can be controlled within about a 30 - 40°C range.

Neutral Colored Photochromic Systems

SINOs described in previous sections are blue to blue/violet in the darkened state. Pilkington dye design and synthesis efforts have also yielded red, green and yellow photochromics. Level absorbance over the full visible range can be achieved by using mixtures of three or four dyes (Figure 9). This results in a colorless/gray/black transition during darkening, which is desirable for some solar shading applications.

Balanced, neutral coloring mixed dye systems in polyurethane hosts have been developed by Pilkington and are undergoing performance testing. Unfiltered darkening and fading transmission spectra of one mixed system are shown in Figures 10 and 11.

Mixed dye systems must be adjusted for specific exposure conditions and filters. Each dye in the mixture has a different activation spectrum, so changes in the activating light can alter color during darkening. Compensation for different activation conditions and filtering normally requires only adjustment of individual dye concentrations.

Figure 12 presents photochromic activity data for a dye mixture in polyurethane which is neutral when exposed to unfiltered solar radiation but not neutral when activated through a filter. The filter used in this example was a commercial aircraft passenger window assembly, consisting of two stretched acrylic panes and a polycarbonate scratch shield.

Fatigue Life

Organic photochromic compounds gradually lose photochromic range on exposure to activating radiation and other environmental factors, such as oxygen, water and elevated temperatures. Degradation is due to chemical side reactions which convert the photochromic compound into an inactive derivative. The observed result is development of background color in the bleached state and lower absorbance in the darkened state.

SINOs as a class exhibit very good photochromic fatigue resistance. The host polymer also has a strong influence on system durability. Polyurethane compositions which enhance photochromic fatigue resistance have been developed and are being evaluated as interlayers in transparent laminates. Excellent stability has been demonstrated in accelerated fatigue testing.

Figures 13, 14 and 15 present accelerated fatigue testing data for two glass laminates with photochromic polyurethane interlayers. One laminate contains a blue 6'-amino SINO and the other a neutral dye mixture. Performance of both systems is essentially unchanged after 1000 hours of QUV-340 exposure. The blue laminate has also been exposed outdoors for 12 months in south Florida with no change in photochromic properties.

Free-standing polyurethane films and bi-layer designs are expected to have lower fatigue resistance than laminates because the photochromic material is not as well protected from the environment. Such designs have not been as fully characterized as laminates.

PHOTOCHROMIC FLASH PROTECTION SYSTEMS

The fundamental photochromic process responsible for generation of the colored isomer of SINO and similar photochromics is very fast, occurring in less than one nanosecond in solution (Ref. 2). The switching rate of these compounds in a polymer host is slower but still very fast with proper design of the host. Polyurethanes with 0.2 wt % IB8 6'-amino SINO exposed to a 0.84 J/cm^2 xenon flash lamp pulse for 500 µsec reach maximum absorbance of > 1.4 in 200 µsec. Lyes are strongly activated by flash exposure through glass or plastic filters (Ref. 3).

Ability to darken rapidly under flash conditions allows these photochromics to be used in passive nuclear flash protection systems. The limited temperature range for darkening with solar exposure does not apply for flash exposure. The darkening reaction rate is very much higher than the fading rate unless the ambient temperature is very high. Since equilibrium is not reached under flash conditions, darkening occurs over a broad temperature range, well above the upper temperature limit for solar darkening.

Slow fading (or unwanted solar darkening) of a passive flash protection system can be a problem at low temperatures. This can be overcome by dye and host modifications to increase the thermal bleaching rate or by controlling the temperature above the solar darkening limit of the system.

Pilkington Optronics in the UK are evaluating photochromic dye technology described in this paper for visor applications (Ref. 4). Performance targets for their program have been achieved.

Pilkington Aerospace participated in two programs involving development of integral nuclear flash protection systems incorporated into aircraft transparencies. Systems employ photochromic dyes in polyurethane interlayers coupled with other technology. Flash testing completed to date has yielded positive results and has highlighted design improvements which will further enhance performance.

During one program, Pilkington Aerospace demonstrated ability to manufacture full-scale transparencies with a thermoset polyurethane interlayer containing a photochromic dye. Processing methods and transparency structural properties were not affected by addition of the dye component.

CONCLUSIONS

1. Recent Pilkington advancements in photochromic technology have yielded materials with high photochromic activity, a wide range of darkened colors including neutral (gray/black), good fatigue life and compatibility with transparent polymers, particularly polyurethanes.

2. Photochromic polymer systems with practical utility for aircraft transparency solar shading and flash protection applications are now available.

3. Photochromics incorporated into polyurethane interlayer materials in laminated transparency constructions have demonstrated excellent performance in testing programs for both transparency solar shading and flash protection applications.

4. Incorporation of photochromics into polyurethane coatings/liners and other polymeric materials may also be feasible but has not been fully demonstrated.

ACKNOWLEDGMENT

Most developments summarized in this paper are accomplishments of Pilkington Group Research personnel (Pilkington Technology Centre, Lancashire, UK): P. Holmes, W. Maltman, S. Marsden, M. Ormsby, M. Rickwood, A. Staunton, J. Williams and D. Wood.

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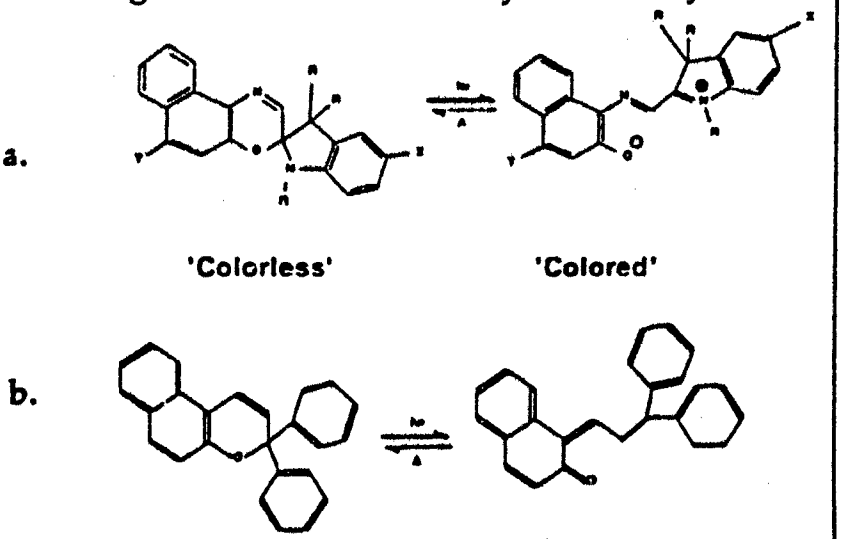
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Figure 1. Photochromic Dye Chemistry



a . SINO (Y = NR_2: 6'-amino SINO) B. Chromene

Figure 2. Absorption Spectra of SINO Photochromics

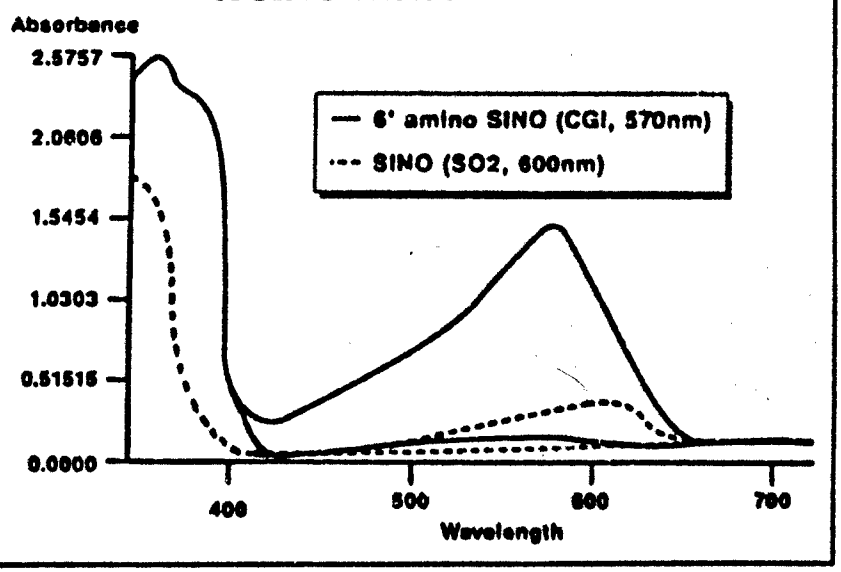


Figure 3. Darkening Transmission Spectra of a 6′-Amino SINO in Polyurethane

Simulated AM2 Solar Exposure, 21°C

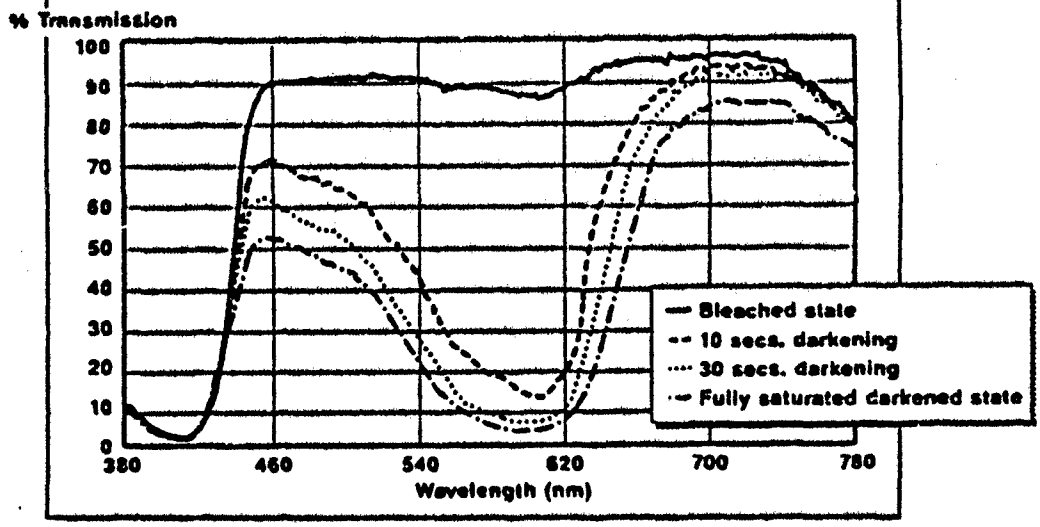


Figure 4. Fading Transmission Spectra of a 6′-Amino SINO in Polyurethane

Simulated AM2 Solar Exposure, 21°C

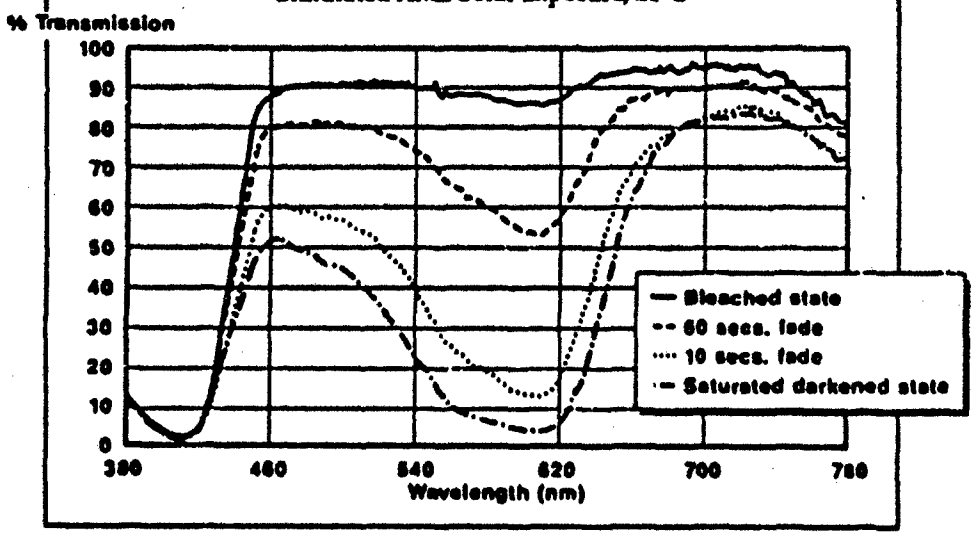
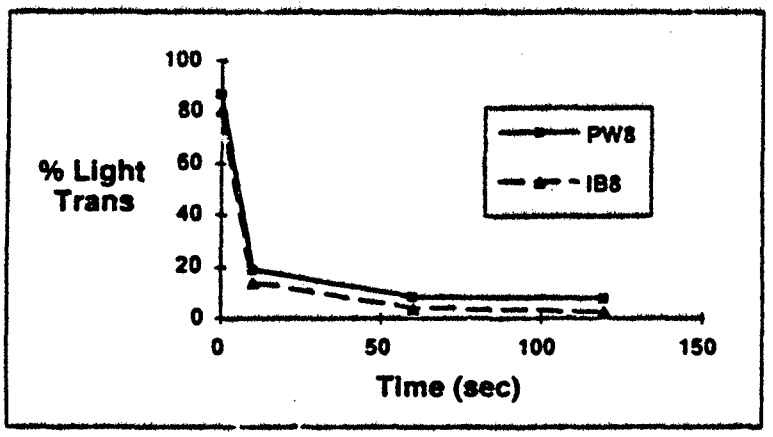


Figure 5 - Integrated Visible Transmission (Darkening) of PW8 and IB8 6'-Amino SINOs in Polyurethane

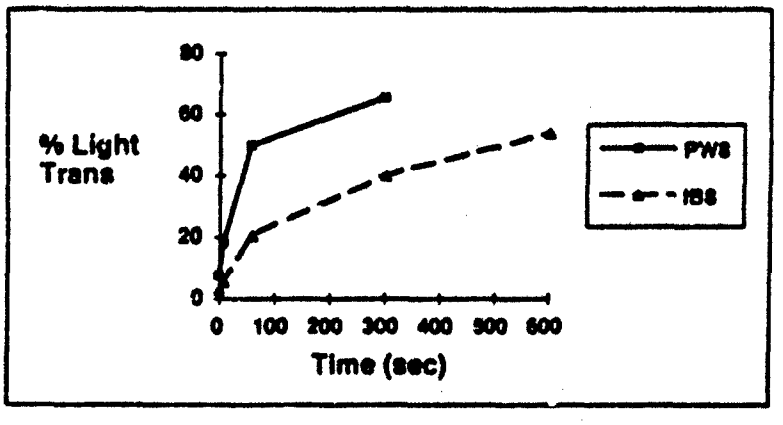
Simulated AM2 Solar Exposure, 21°C



0.04% Dye in Polyurethane, No Filter

Figure 6 - Integrated Visible Transmission (Fading) of PW8 and IB8 6'-Amino SINOs in Polyurethane

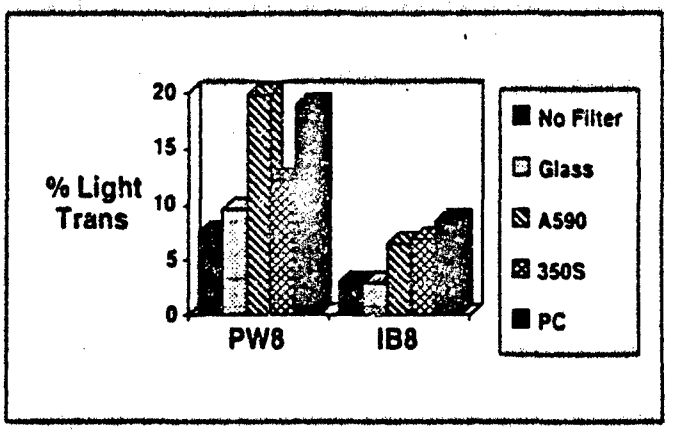
Simulated AM2 Solar Exposure , 21 °C



0.04% Dye in Polyurethane, No Filter

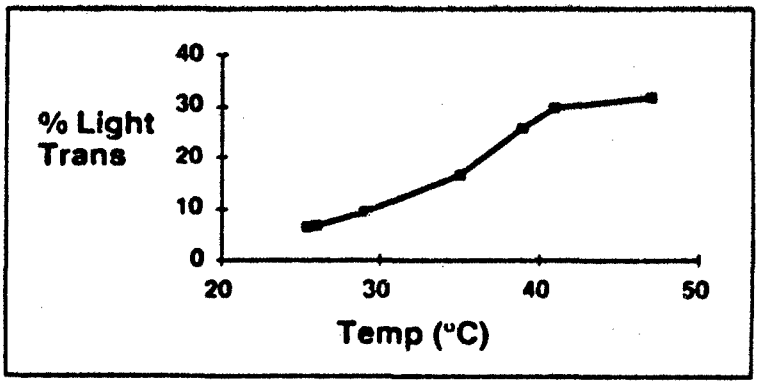
Figure 7 - Effect of Filters on Equilibrium Darkening of 6'-Amino SINOs in Polyurethane

Equilibrium - Simulated AM2 Solar Exposure, 21 °C



0.04% Dye in Polyurethane

Figure 8 - Effect of Temperature on Equilibrium Darkening of SINOs in Polyurethane



- Glass / Polyurethane / Glass Laminate
- 1 Min Exposure, Simulated AM2 Solar Exposure
- 0.10% IB8 / 0.07% PW8 in Polyurethane

Figure 9. Absorption Spectra of a Photochromic Mixture for a Neutral Coloring System

Simulated AM2 Solar Exposure, 21°C

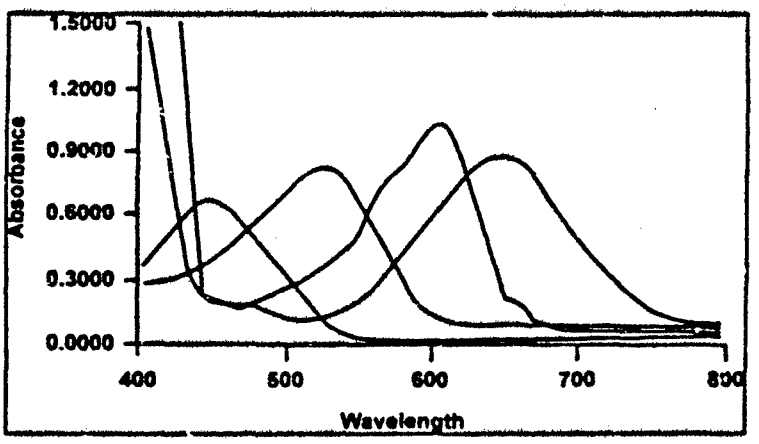
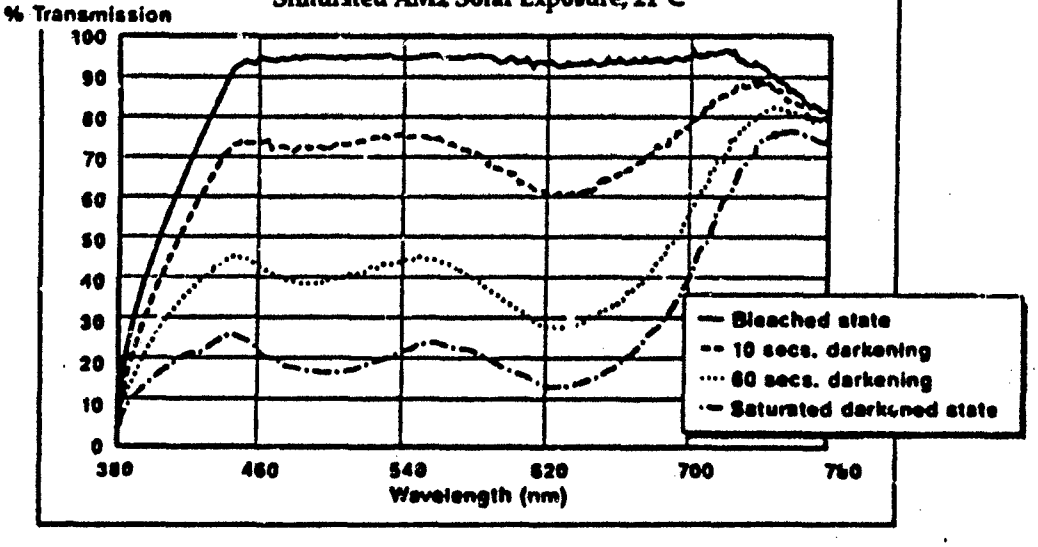
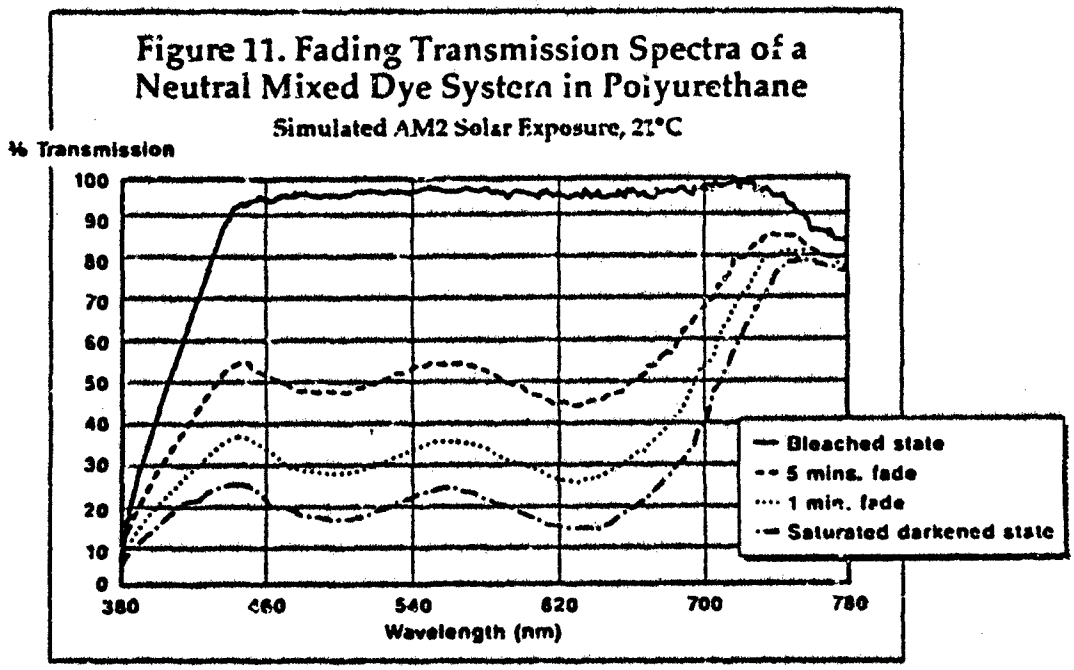


Figure 10. Darkening Transmission Spectra of a Neutral Mixed Dye System in Polyurethane

Simulated AM2 Solar Exposure, 21°C




Figure 11. Fading Transmission Spectra of a Neutral Mixed Dye System in Polyurethane
Simulated AM2 Solar Exposure, 21°C
% Transmission
100
90
80
70
60
50
40
30
20
10
0
380
460
540
620
700
780
Wavelength (nm)
Bleached state
5 mins. fade
1 min. fade
Saturated darkened state

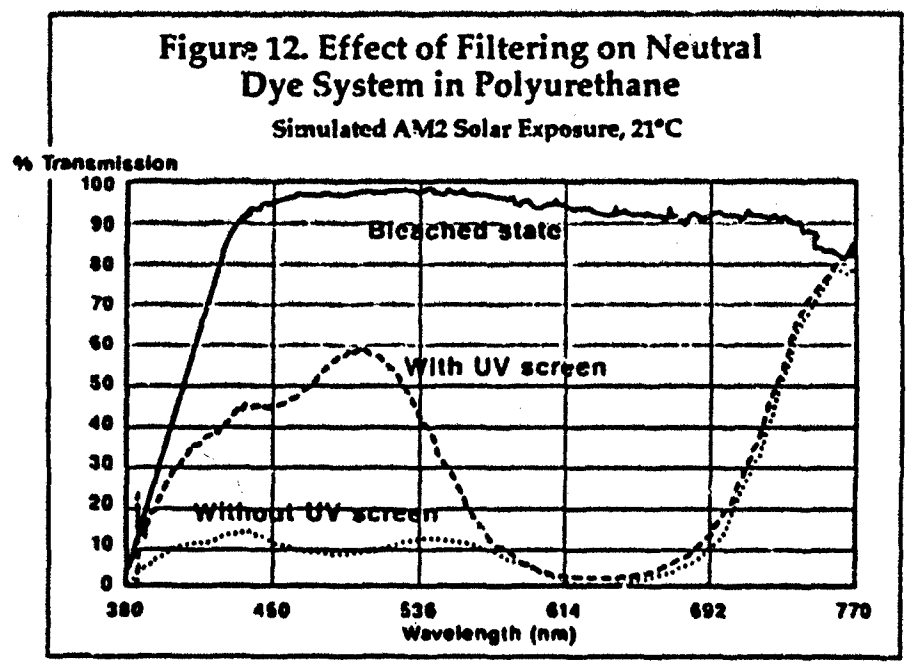
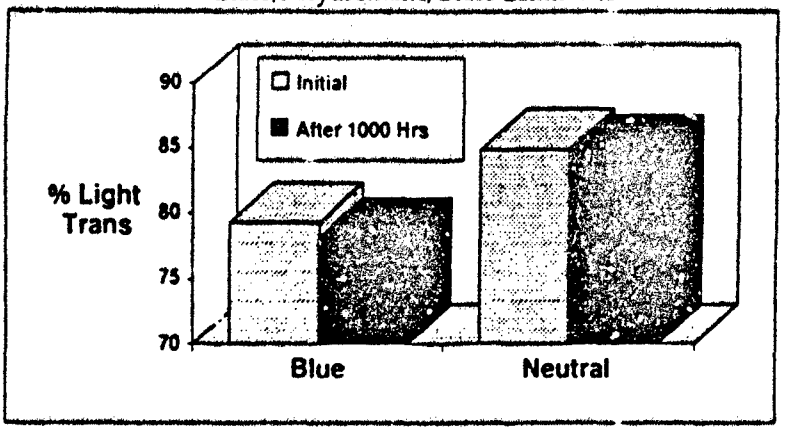

Figure 12. Effect of Filtering on Neutral Dye System in Polyurethane
Simulated AM2 Solar Exposure, 21°C
% Transmission
100
90
80
70
60
50
40
30
20
10
0
380
460
536
614
692
770
Wavelength (nm)
Bleached state
With UV screen
Without UV screen

Figure 13 - Effect of QUV-340 Exposure on Bleached State Transmission of a 6'-Amino SINO in Polyurethane

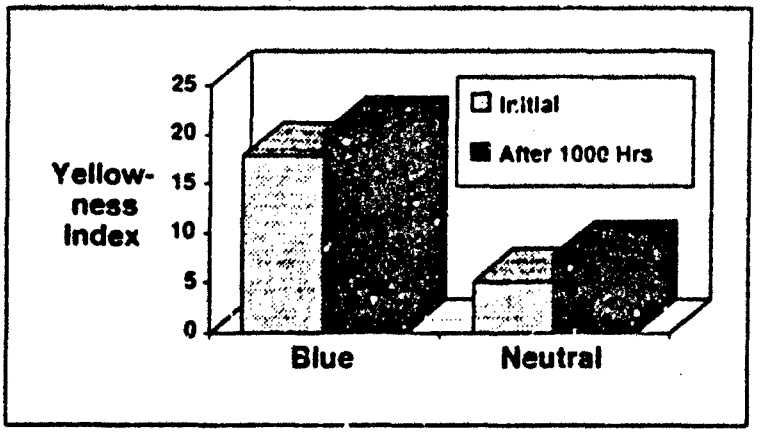
Glass/Polyurethane/Glass Laminates



Total Dye Concentration: 0.2%

Figure 14 - Effect of QUV-340 Exposure on Bleached State Yellowness Index of a 6'-Amino SINO in Polyurethane

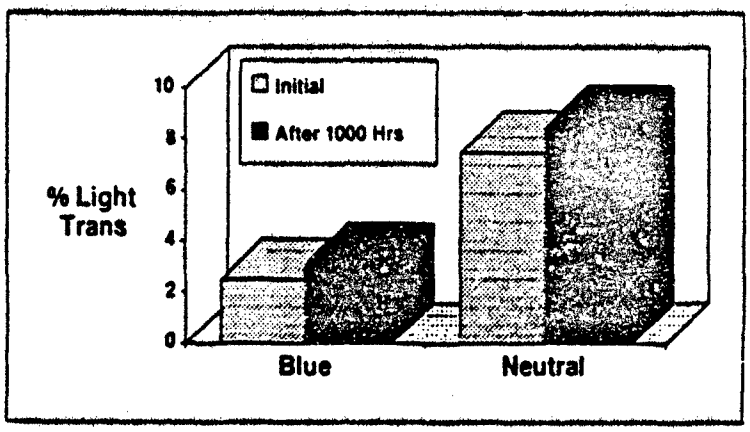
Glass/Polyurethane/Glass Laminates



Total Dye Concentration: 0.2%

Figure 15 - Effect of QUV-340 Exposure on Darkened State Transmission of a 6'-Amino SINO in Polyurethane

Glass/Polyurethane/Glass Laminates



Equilibrium, Simulated AM2 Solar Exposure, 0.2% Total Dye

HIGH TEMPERATURE URETHANE (S-240)

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High Temperature Urethane (S-240)

Presented by Khushroo H. Lakdawala at the
Conference on Aerospace Transparent
Materials and Enclosures

16th Conference

Sponsored by
Flight Dynamics Directorate and
Materials Directorate of the
United States Air Force
Wright Laboratory

San Diego, California
9-13 August 1993

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and
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HIGH TEMPERATURE URETHANE (S-240)

by

Khushroo H. Lakdawala
John Raffo

Sierracin/Sylmar Corporation

ABSTRACT

S-240 is a Sierracin/Sylmar Corporation proprietary formulation based on polyurethane chemistry, developed to improve upon the limitations of existing transparency materials. Analytical techniques to evaluate the properties and reactivity of the monomer of S-240 are illustrated. Polyurethanes are conventionally used, as interlayers, in laminates of various cross sections for aircraft windshields. In a vast majority of windshield designs for high-speed aircraft the outer ply is comprised of acrylic, and the structural ply is polycarbonate. Both these materials have field service limitations, ie acrylic tends to crack and craze readily, whereas polycarbonate is susceptible to hazing, poor abrasion resistance, and is readily attacked by solvents. S-240, a high temperature urethane, is capable of being used as an outer ply and circumvents the common drawbacks associated with acrylic faced (outer) plies. Laboratory tests indicate that S-240 has superior high temperature, impact, abrasion, and solvent/craze properties and is presently available for full scale application testing.

Introduction

The major materials currently being used, as an exterior ply, for aircraft transparencies are methyl methacrylate based polymers and/or glass. Certain inherent limitations exist with either class of materials. In a prior study, premature failure of F-16 windshields was attributed to crazing or cracking of the acrylic face ply in a vast majority of cases. Based on these observations, Sierracin has developed a material capable of enhanced performance and increased service life. Development of an ideal material can only be achieved by an iterative process where the formulation changes are directly correlated with field service performance. The tremendous versatility in formulating a urethane molecule makes it an ideal candidate for these purposes.

Polyurethanes are a broad class of materials having a common urethane (--NH(C=O)-O--) chemical linkage. The urethane groups are predominantly formed by the reaction of an isocyanate with a hydroxyl containing compound (Equation 1). The great variations that are possible in the type and functionality of the hydroxyl and isocyanate raw material components make polyurethanes a uniquely versatile class of polymers. The possibility of custom tailoring the chemical structure to suit a particular application's requirements has opened a range of new uses for polyurethanes. In addition to urethane functional groups, polyurethanes could also contain amide, ether, ester, urea, amine, etc type functional groups. To date, however, no material is available that can withstand the harsh environmental, chemical and mechanical rigors experienced by the outer ply of a sophisticated aircraft and still provide the desired service life expected by Operational Commands. S-240 addresses the drawbacks associated with a conventional methacrylate-based face ply material. Additionally, the high temperature capabilities of S-240 are a significant asset of this material.

Characteristics of Segmented Polyurethanes

The structure of typical segmented polyurethanes consists of alternating blocks of flexible chains of low glass transition temperature (Tg), amorphous or low melting components (soft segments), and a relatively rigid block component (hard segments) that has a crystalline melting point well above room temperature. The varying polarity and chemical nature of both the blocks tends to separate the polymer into two phases designated "soft" and "hard". The hard blocks tend to associate into domains because of chemical structure and hydrogen bonding. The extent of phase separation has a significant effect on the final properties of the polyurethane.

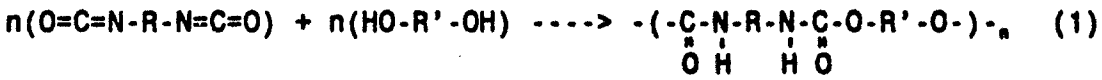
Polyurethanes are typically categorized based on the ratios and types of the above components. Soft segment concentration (SSC) is defined as the ratio of the mass of polyol chain without the terminal hydroxyl groups to the total mass of the polymer, usually expressed as a percentage. The difference, making the total 100%, is the hard segment concentration (HSC). A one phase polyurethane can also be obtained by reacting the soft segment polyol with isocyanates,

without the use of a chain extender to obtain one dimensional polyurethanes with significantly reduced properties such as tensile strength, modulus, etc.

The structure, concentration and organization of the hard segment blocks have a dominant influence on the physical and mechanical properties of the polymer. Characterizing the property changes that occur with formulation changes is important in order to develop a polymer for specific end-use applications. Analytical techniques like differential scanning calorimetry (DSC), thermomechanical analysis (TMA), Fourier transform infrared spectroscopy (FTIR) and dynamic mechanical analysis (DMA) are ideal tools for polymer characterization in relation to useful and desired practical properties.

Chemistry and Morphology of Polyurethanes

Polyurethanes are polymers which have the characteristic linkage -NH-(C=O)-O-. These are formed when a diol undergoes step-growth polymerization with a diisocyanate. Segmented polyurethanes are obtained by two basic procedures: the prepolymer technique and the 'one shot' method. In the former case, the polymer is synthesized by 'capping' or terminating the reactive groups on the polyol with an excess of difunctional isocyanate in specific equivalent amounts required to obtain the requisite soft or hard segment concentration. This "pseudo" prepolymer macrodiisocyanate is subsequently used in chain-extending with a low molecular weight diol or glycol (Figure 1). The soft and hard segments are each more nearly uniform in length. In the 'one shot' case, the diisocyanate, polyol and glycol react directly to yield a more polydispersed polymer (Figure 2) where the length of the hard and soft segment blocks is more random. Typical isocyanate reactions are as follows:



$$n(O{=}C{=}N\text{-}R\text{-}N{=}C{=}O) + n(HO\text{-}R'\text{-}OH) \longrightarrow -\left(-\underset{\underset{O}{\|}}{C}\text{-}\underset{\underset{H}{|}}{N}\text{-}R\text{-}\underset{\underset{H}{|}}{N}\text{-}\underset{\underset{O}{\|}}{C}\text{-}O\text{-}R'\text{-}O-\right)-_n \quad (1)$$

Urethane Formation



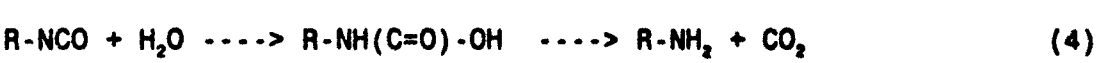
$$R\text{-}NCO + HO\text{-}R' \longrightarrow R\text{-}NH(C{=}O)O\text{-}R' \quad (2)$$

Urea Formation



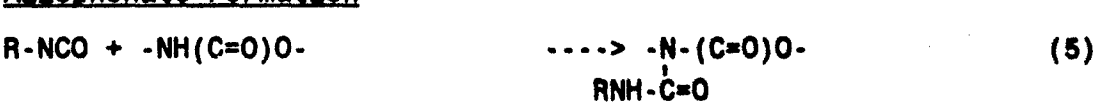
$$R\text{-}NCO + H_2N\text{-}R' \longrightarrow R\text{-}NH(C{=}O)NH\text{-}R' \quad (3)$$

Reaction with Water



$$R\text{-}NCO + H_2O \longrightarrow R\text{-}NH(C{=}O)\text{-}OH \longrightarrow R\text{-}NH_2 + CO_2 \quad (4)$$

Allophonate Formation



$$R\text{-}NCO + -NH(C{=}O)O\text{-} \longrightarrow -\underset{\underset{RNH\text{-}\overset{\overset{O}{\|}}{C}}{|}}{N}\text{-}(C{=}O)O- \quad (5)$$

Biuret Formation

$$R\text{-}NCO + \text{-}NH(C{=}O)NH\text{-} \longrightarrow \text{-}N(RNH\text{-}C{=}O)\text{-}(C{=}O)\text{-}NH\text{-} \quad (6)$$

An ideal segmented polyurethane can be prepared only if equations (3) through (6) are suppressed. The polymerization kinetics of polyurethane formation is not exactly of second order. The overall kinetics is of third order at very low concentrations (-OH) but of second order at higher concentrations. The structure of a segmented polyurethane is highly heterogenous and depends strongly on the reaction conditions as well as the reactivity of the individual components, choice of catalyst, and the presence of impurities. Stoichiometric adjustments are necessary to obtain a system with optimal molecular weight. The structural formulae of the industrially important diisocyanates are given in Figure 3. A synergistic combination of some of these isocyanates has proven to provide beneficial optical and physical properties which has enabled us to develop the S-240 class of polymers comprised of a linear compact molecule with significant intermolecular interactions and good phase separation.

Segmented polyurethanes possess a two phase morphology wherein the hard and soft segments are, to varying degrees, incompatible with each other (Figure 4). Despite this incompatibility, however, a significant amount of mixing between hard and soft segment blocks exists. Hydrogen bonding between hard and soft blocks or between individual hard blocks can beneficially affect the physical properties of the polymer (Figure 5). In the latter case a three-dimensional molecular domain structure can exist to produce polymers with the temperature capabilities similar to, crosslinked or thermosetting systems.

The appearance of spherulites in the polymer may be due to the building of hard blocks in a transverse direction to the molecular axis (Figure 4). The morphology of segmented polyurethanes is also dependent on the chemical constitution and thermal history of the polymer. The performance of polyurethanes at elevated temperatures is dependant on the structure of the hard blocks and their ability to remain associated at these temperatures. DSC analysis indicates a transition temperature for the soft blocks at greater than -30°C (-22°F), an endotherm corresponding to the dissociation of hydrogen bonding in the soft segments at less than 80°C (176°F), an endotherm due to break-up of the urethane hydrogen bonds at less than 150°C (302°F), and, lastly, transitions above 160°C (320°F) which are attributed to the thermal dissociation of the hard block aggregates.

Properties of S-240

Although, as previously mentioned, the chemical structure and physical properties of polyurethanes can vary widely with minor changes in formulation, all data presented in this study has been taken from material of the same Sierracin/Sylmar proprietary formulation to assure continuity.

Tensile Properties

The tensile strength was measured using a United Calibration test machine with a 1000 lb load cell. The specimens were cut as per ASTM D 1708. The average values from 5 specimens has been reported. The samples were mounted in mechanical grips with serrated faces. The tensile properties are summarized in Table 1. The exact variation in the hard segment content affects the modulus, tensile strength and ultimate elongation significantly. This effect is illustrated in the S-240 formulations 'A' through 'D'.

Thermal Analysis

The softening characteristics were determined by thermomechanical analysis (TMA) using a Perkin-Elmer Thermomechanical Analyzer, TMS-2 with a Perkin-Elmer System 4 thermal analysis microprocessor controller. The samples were heated in a helium atmosphere under a 10u psi load. The heating rate was 10°C (18°F) per minute. The samples were run from 30°C (86°F) to the softening point of the sample. The TMA trace in Figure 6 displays the effect of the chain extender (diol) on the softening point of the urethane. The softening point of S-240 was 193°C (379°F).

The curing characteristics of the polymer were evaluated using a Perkin-Elmer DSC-4 (differential scanning calorimeter). The instrument was purged with nitrogen. A premixed sample weighing 8 to 15 mg was encapsulated in a high pressure aluminum pan and crimped before placing into a DSC sample chamber. The test was run from 40°C (104°F) to 220°C (428°F) at 10°C (18°F) per minute. The DSC studies gave significant insight into optimization of catalyst level and curing conditions. The DSC thermograms depicted in Figure 7 indicate the effect of catalysts on the curing characteristic of S-240 system. The curing kinetics were interpreted from dynamic experiments. The amount of catalyst decreased the peak curing temperature from 173°C (343°F) to 113°C (235°F).

Dynamic Mechanical Analysis

The dynamic mechanical properties were evaluated using a Rheometrics Dynamic Analyzer (RDA-700). The samples were run in a rectangular torsion mode. A liquid nitrogen controller was used to attain appropriate temperature. The test measurements were taken at 10°C (18°F) intervals with a 3 minute equilibration duration. Figure 8 depicts a typical DMA trace of S-240. The polymer system manifests significant phase separation as indicated by the tan delta trace. In general, use of DMA analysis has allowed tremendous insight into the formulation aspects of S-240, and has cut development time considerably.

The extent of phase separation attained for a combination of isomers at specific levels of hard segment was extremely critical. Figure 9 demonstrates the variation in modulus for such specific combinations. Lastly, the DMA traces of the two most widely used transparency materials, acrylic and polycarbonate, are compared with S-240 (Figure 10). The modulus of S-240, from room temperature to

the glass transition temperature, was lower than either of the materials. However, the enhanced dissipation capability offsets the effect of lower modulus as the tough characteristics of S-240 enhance the resistance to crazing and cracking as compared to acrylic.

Abrasion Resistance

The abrasion characteristics of S-240 were evaluated as per ASTM F-735, using a Bayer abrader. The delta haze values, after 600 strokes, for a series of S-240 formulations are given in Table 2. The apparent increase in haze after abrasion is directly related to the increased stiffness in the formulation.

Impact Testing

The impact performance of thin sections of S-240 sheet was evaluated using a Gardner impact tester, with a half inch hemispherical indenter. Table 3 illustrates the impact characteristics of a series of S-240 formulations. Impact performance is significantly superior to acrylic of similar thickness and is similar to polycarbonate, the most impact resistant material used in aircraft transparency application.

Craze Resistance and Chemical Exposure

Tables 4 and 5 summarize the ability of S-240 to withstand various solvents. The craze testing was performed on a 15 x 1 x 0.07 inch thick beam sample according to Mil-P-8184E. The poor craze resistance of acrylic at 71°C (160°F) is in sharp contrast to the superior craze resistance of S-240. The chemical exposure studies were done using a 15 minute liquid saturated patch test. Also, S-240 performs satisfactorily when exposed to various, simulated chemical warfare agents.

High Temperature Capabilities

Extensive testing was done to evaluate the performance of S-240 in comparison to other high temperature materials readily available in the market. The various materials tested are listed in Table 6. A comparative study exposed the materials to 177°C (350°F), 204°C (400°F), and 246°C (475°F) for one hour.

Degradation is enhanced at temperatures above the glass transition temperature (Tg). Consequently, modulus was measured, using DMA, at fixed temperatures and varying frequencies (frequency sweep) before and after the thermal exposure. Decrease in modulus at a specific frequency indicates the extent of degradation. Table 7 summarizes the change in modulus after temperature exposure for the different materials. S-240 was the only material that survived 246°C (475°F) exposure testing.

Process Characteristics of S-240

Cast sheets of S-240 were successfully laminated to polycarbonate with a urethane interlayer (configuration similar to an F-16). The

cast laminate was subsequently vacuum formed to stretch the outer S-240 ply over 28 percent. Adhesion of S-240 to the urethane was excellent. No delamination or other adverse effects occurred and no abnormalities were observed after the forming cycle.

Conclusion

Sierracin S-240 is a potential faceply material which, when laminated to a suitable structural ply, is expected to improve the service life of advanced transparencies significantly. S-240 eliminates the drawbacks associated with conventional, external acrylic based cross sections. The overall advantages of an S-240 system are summarized in Table 8.

Sierracin is currently under contract to scale-up the S-240 technology to full size operational transparencies.

Acknowledgement

The authors wish to recognize their co-workers Don Ahern, David Han, Connie Maglalang and Haimo Zhang for their participation in developing various aspects of S-240.

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FORMULATION	MODULUS 10 % (PSI)	STRENGTH (PSI)	ELONG. (%)
S-240 A	7300	2222	520
S-240 B	16000	3400	320
S-240 C	19672	4557	240
S-240 D	30706	4483	185
S-240 E	27810	7982	50

Table 1 TENSILE PROPERTIES of S-240 Formulations

MATERIAL	DELTA HAZE (%)
ACRYLIC	35
POLYCARBONATE	53
S-235 LINER	0.3
S-240 A	8
S-240 B	25
S-240 C	16
S-240 D	36

Table 2 Abrasion Resistance of S-240
Bayer Abrader, 600 cycles

MATERIAL	FAILURE THRESHOLD (IN.-LB)
ACRYLIC	3
POLYCARBONATE	>160
STRETCHED ACRYLIC	8
S-240 A	120
S-240 B	>160
S-240 C	>160
S-240 D	15

Table 3 IMPACT TESTING
Gardner Impact Tester. Sample thickness 0.07"

MATERIAL	75 F (2500 PSI)	160 F (2500 PSI)
ACRYLIC	NO CRAZE	CRAZE (5 MIN.)
S-240	NO CRAZE	NO CRAZE

Table 4 CRAZE TEST 50 / 50 IPA / H2O

CHEMICAL MATERIAL	THF	TOLUENE	IPA	CONC. HCl	CONC. H3PO4	CONC. HF
ACRYLIC	DISSOLVES	DISSOLVES	OK	OK	SLIGHT HAZE	HAZY
S-240 A	SWELL	SWELLS	OK	HAZY	HAZY	HAZY
S-240 B	OK	OK	OK	OK (YLW)	HAZY	OK
S-240 C	OK	OK	OK	OK (YLW)	HAZY	OK
S-240 D	OK	OK	OK	OK (YLW)	HAZY	OK

Table 5 Chemical Resistance of S-240 Formulations

DUREL	--	AROMATIC POLYESTER
AEC	--	COPOLYMER of an AROMATIC ESTER and CARBONATE
PC 9350	--	POLYCARBONATE ISOMER
GAC 590	--	THERMOSET POLYURETHANE
S -240	--	THERMOPLASTIC POLYURETHANE

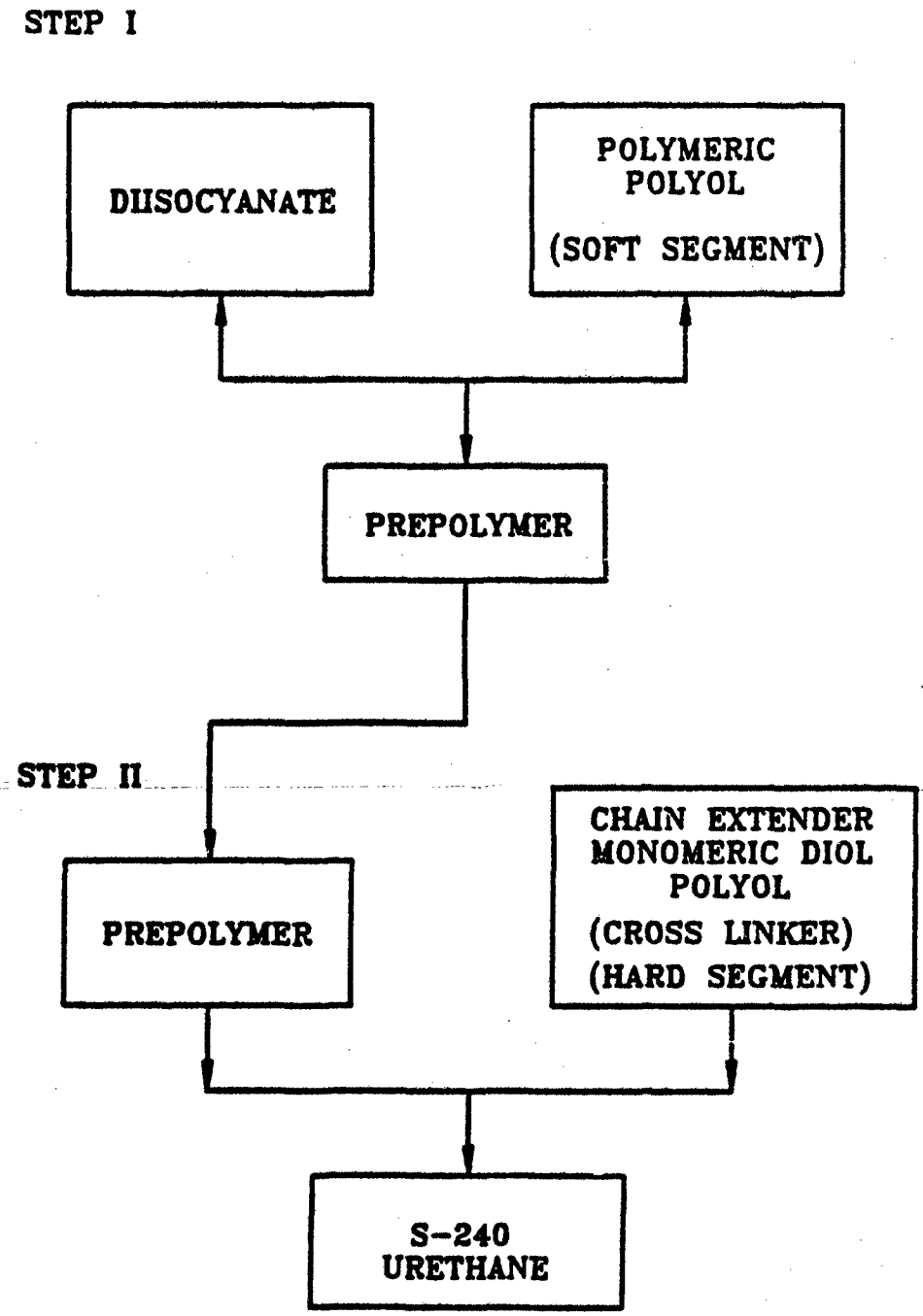
Table 6 List of High Temperature Materials Evaluated

MATERIAL	AS RECD (G'X EXP 7)	MOD. G' 350F 1 hr. (% CHNG)	MOD. G' 400F 1 hr. (% CHNG)	MOD. G' 475F 1 hr. (% CHNG)
AEC	2.84	2.79 (1.4)	2.26 (20.1)	FAILED
GAC 590	12.13	-	10.31 (15.8)	FAILED
DUREL	2.63	2.53 (3.8)	FAILED	FAILED
S-240	10.79	10.33 (4.2)	8.13 (24.6)	7.17 (33.5)

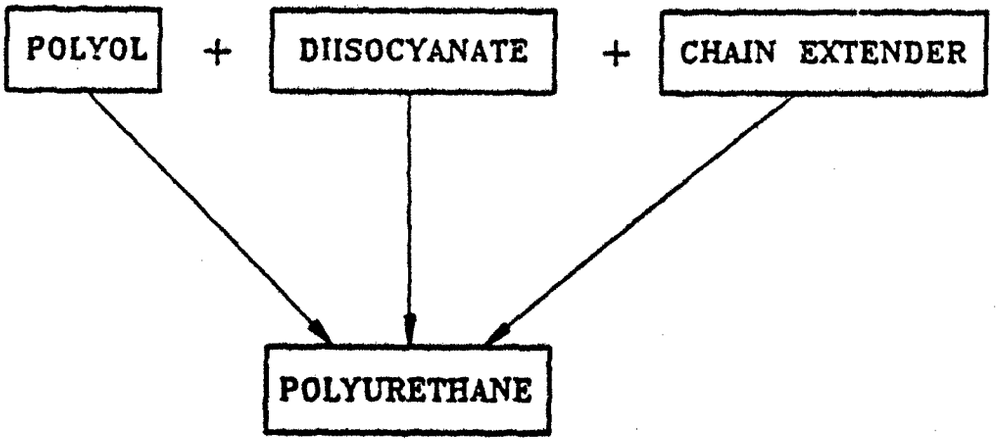
Table 7 Effect of High Temperature Exposure on Shear Modulus DMA FREQUENCY SWEEP @ 220C (428F)

MATERIAL	UV	ABRASION	CRAZE	IMPACT	MAX. TEMP. (F)
ACRYLIC	GOOD	FAIR	POOR	POOR	250
PC	POOR	POOR	POOR	EXCL	320
DUREL	FAIR	POOR	POOR	FAIR	380
AEC	FAIR	POOR	POOR	GOOD	410
PC-9350	FAIR	POOR	POOR	GOOD	410
GAC-590	--	GOOD	GOOD	POOR	435
S-240	GOOD	GOOD	GOOD	EXCL	475

Table 8 MATERIAL PROPERTIES SUMMARY



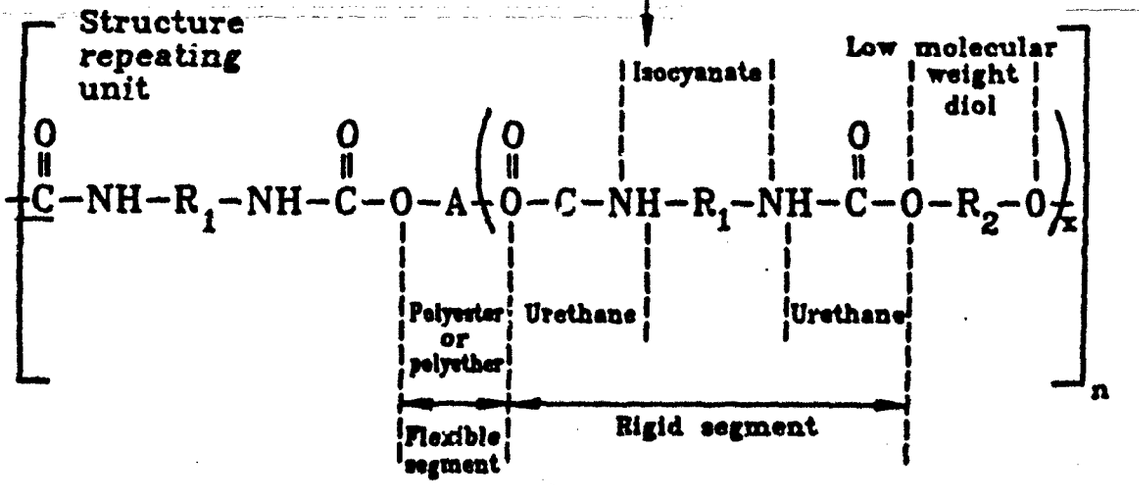
URETHANE MANUFACTURE – PREPOLYMER ROUTE
Fig. 1



Diol chain extension

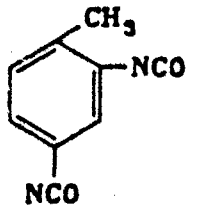
$$HO-A-OH + OCN-R_1-OCN + HO-R_2-OH$$

Linear polyester or polyether diol — Diisocyanate — Low molecular weight diol

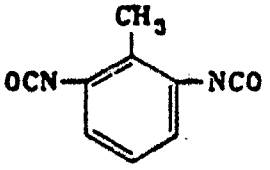


URETHANE MANUFACTURE - ONE SHOT PROCESS

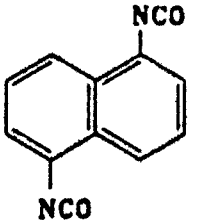
Fig. 2



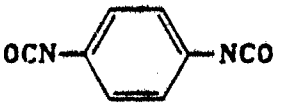
2,4-Toluene diisocyanate



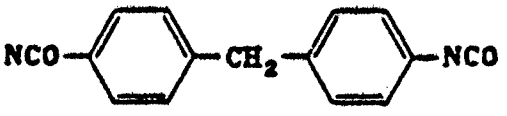
2,6-Toluene diisocyanate (TDI)



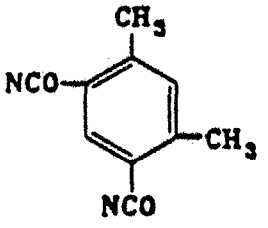
1,5-Naphthalene diisocyanate (NDI)



para-Phenylene diisocyanate (PPDI)



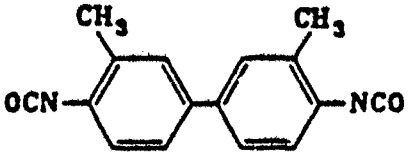
4,4'-Diphenylmethane diisocyanate (MDI)



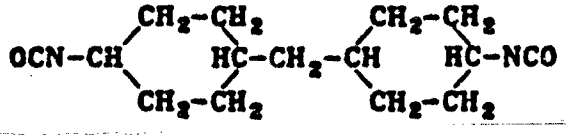
4,6'-Xylylene diisocyanate (XDI)

$OCN(CH_2)_6NCO$

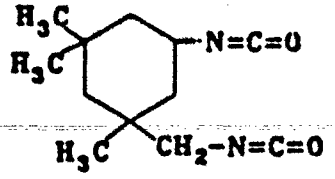
1,6-Hexamethylene diisocyanate (HDI)



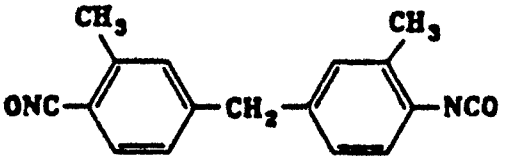
3,3'-Tolidene 4,4'-diisocyanate.
(3,3'-Dimethyldiphenyl 4,4'-diisocyanate)(TODI)



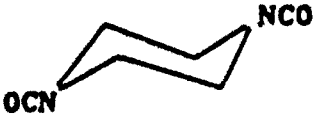
4,4'-Dicyclohexylmethane diisocyanate
(H_{12}MDI)



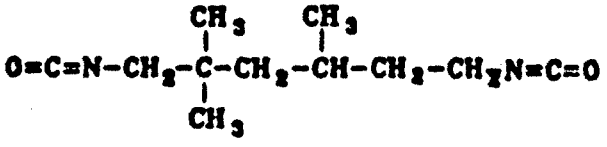
3-Isocyanatomethyl-
3,5,5-trimethylcyclohexyl isocyanate.
(Isophorone diisocyanate)(IPDI)



3,3'-Dimethyl-diphenylmethane
4,4'-diisocyanate



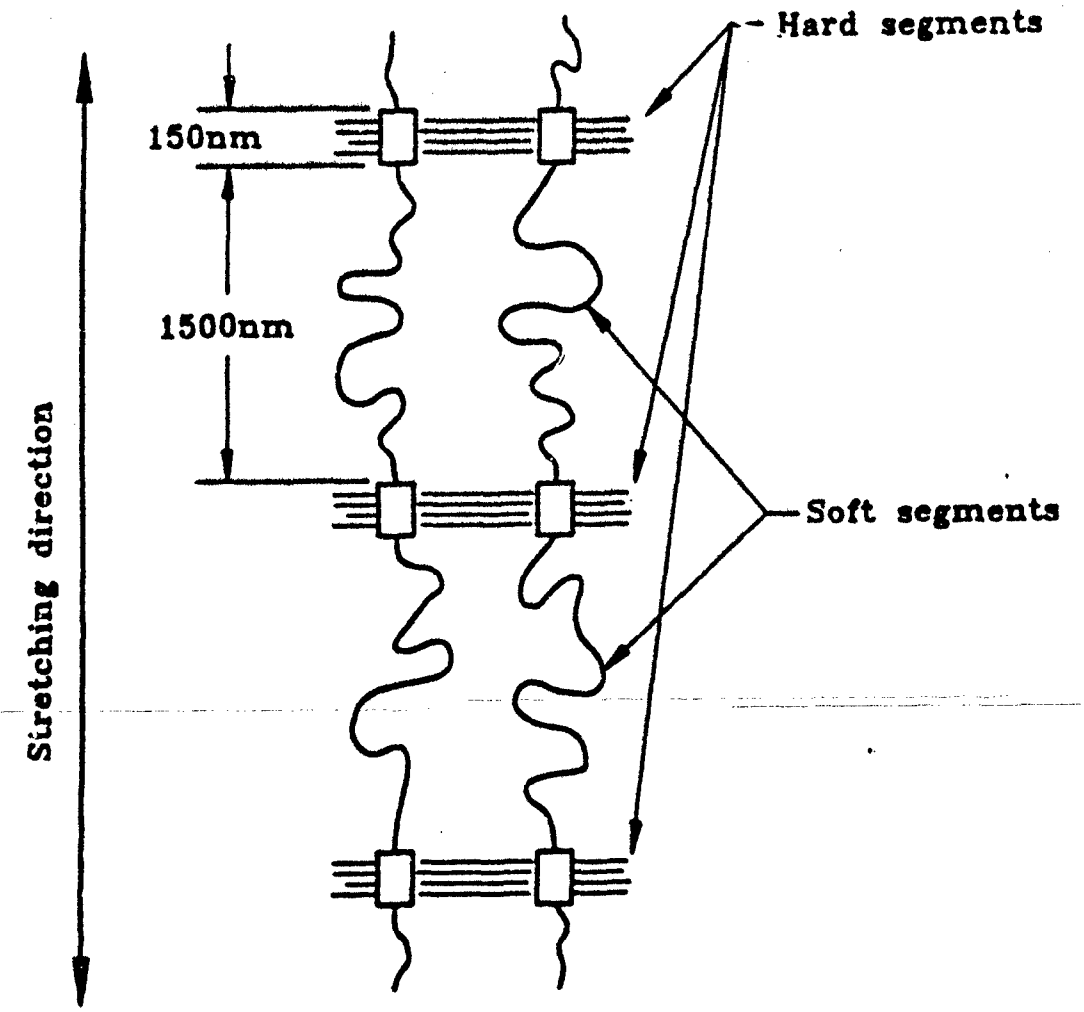
Cyclohexyl diisocyanate
(CHDI)



2,2,4-(2,4,4-) Trimethylhexamethylene diisocyanate.
(2,2,4-Trimethyl-1,6-hexamethylene diisocyanate)(TMDI)

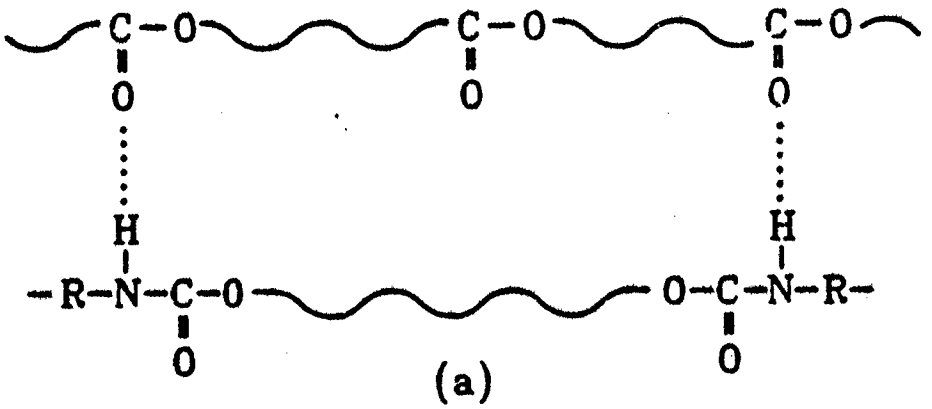
STRUCTURAL FORMULAE OF DIISOCYATES

Fig. 3

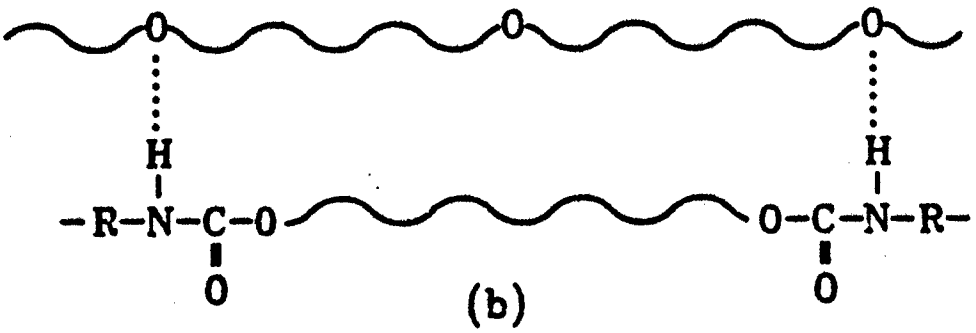


SCHEMATIC REPRESENTATION OF SEGMENTED POLYURETHANES

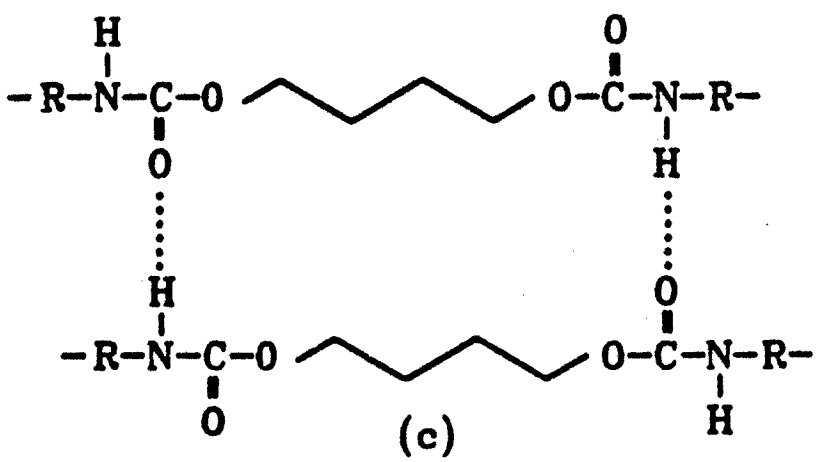
Fig. 4



(a)



(b)



(c)

MANIFESTATION OF HYDROGEN BONDING IN POLYURETHANES
(a) ester-urthane (b) ether-urethane (c) urethane-urethane
Fig. 5

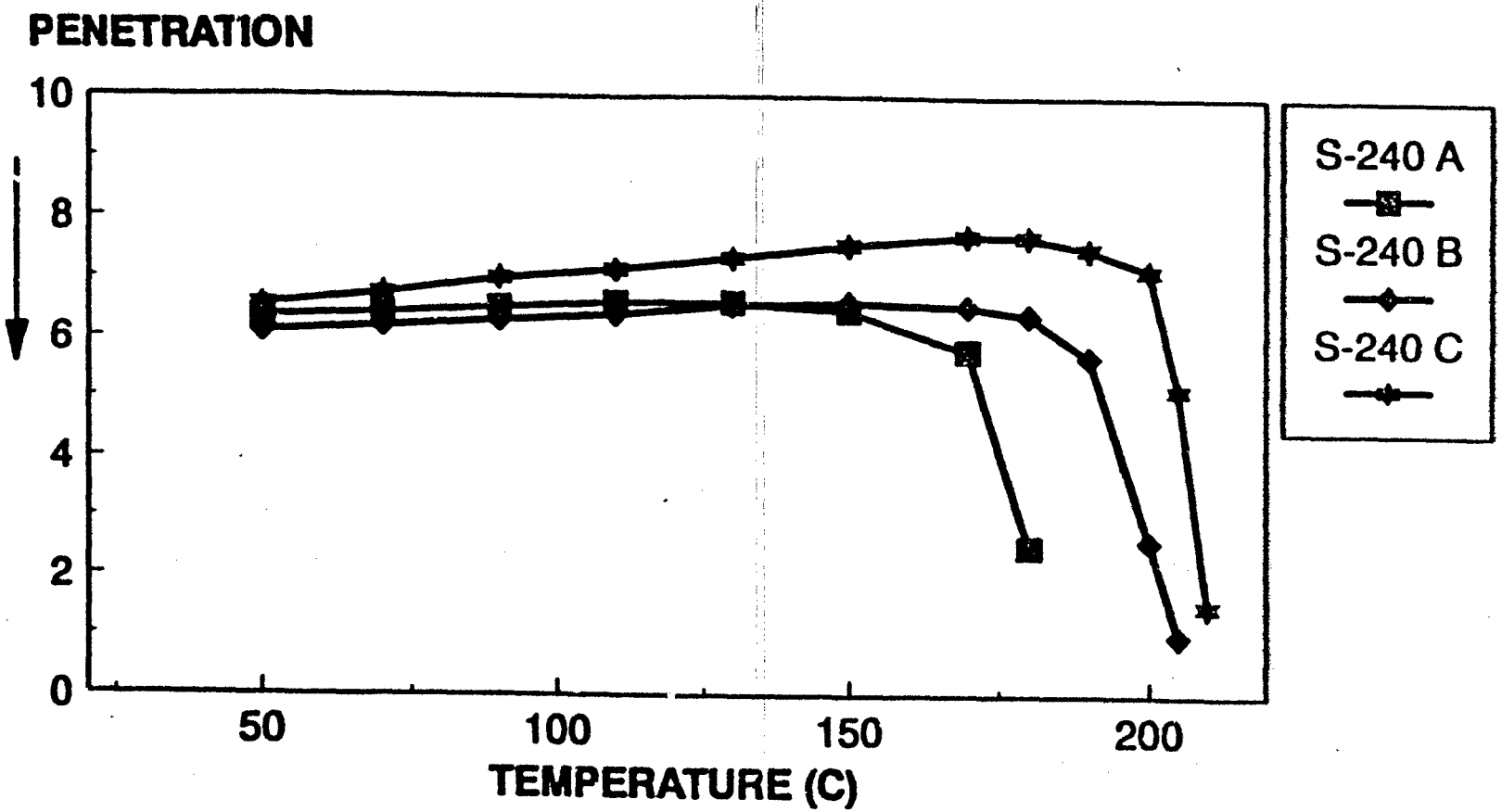


Fig. 6 TMA TRACE for S-240 Formulations

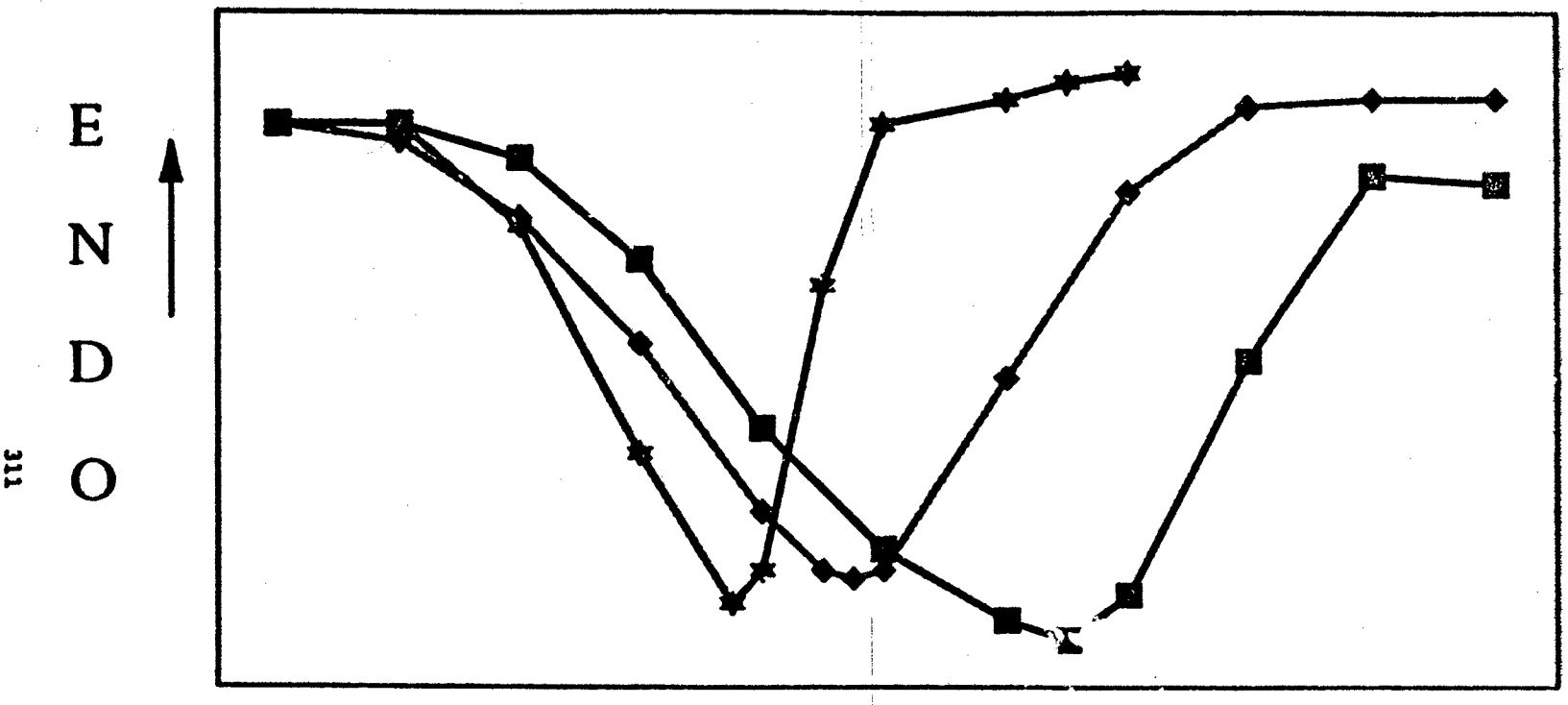
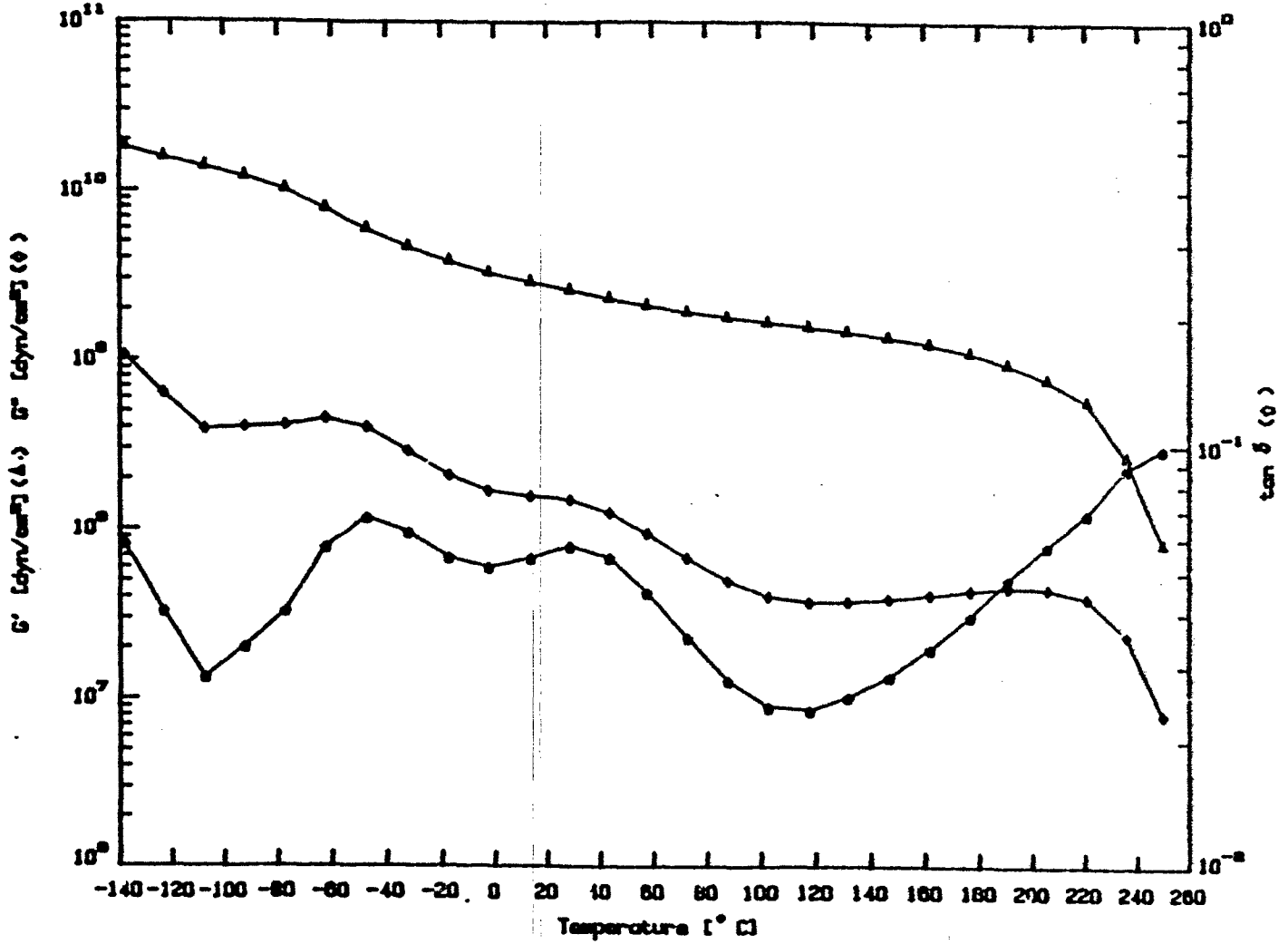


Fig. 7 DSC THERMOGRAMS for S-240



DYNAMIC MECHANICAL PROPERTIES OF S-240

Fig. 8

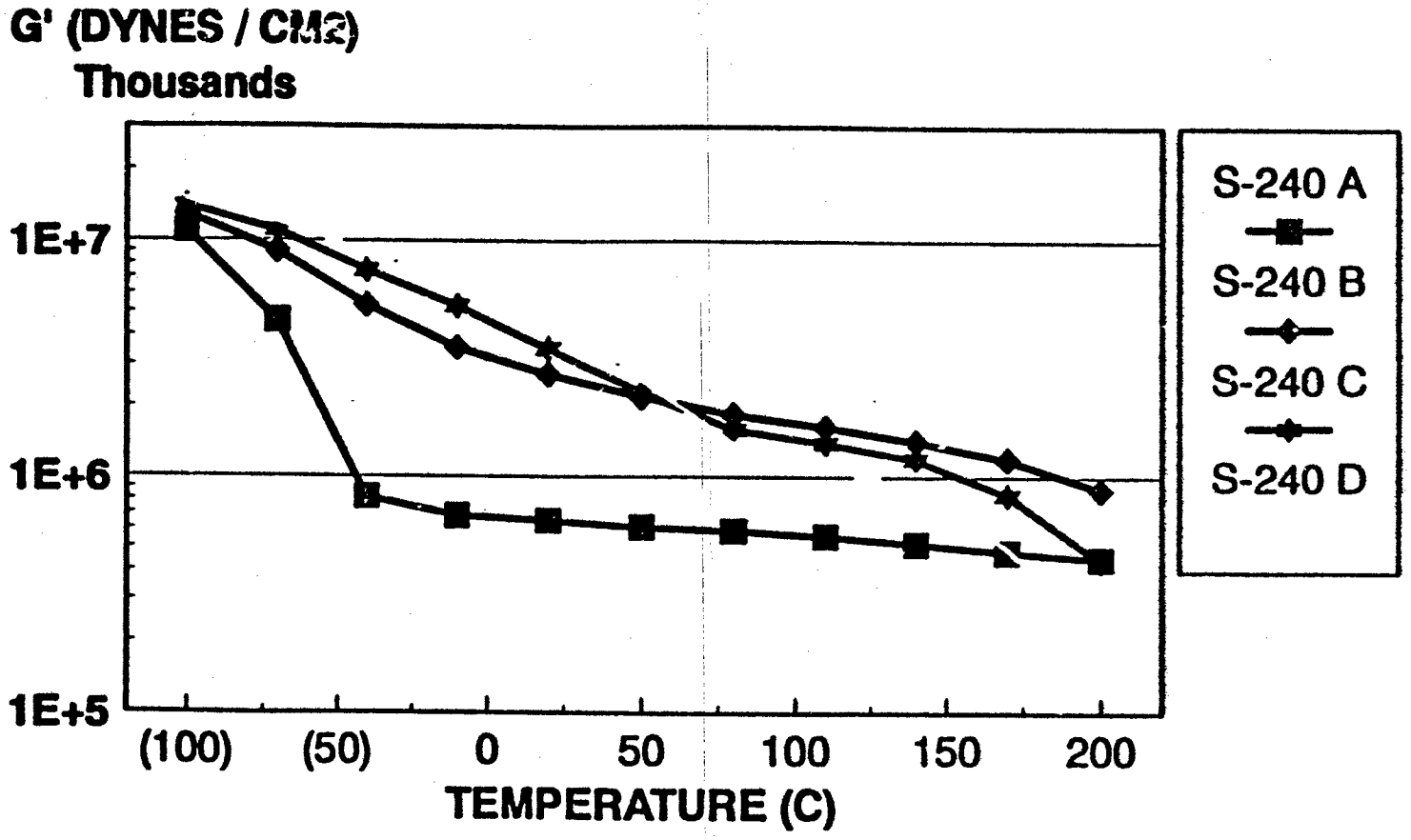


Fig. 9 DMA TRACE for S-240 Formulations

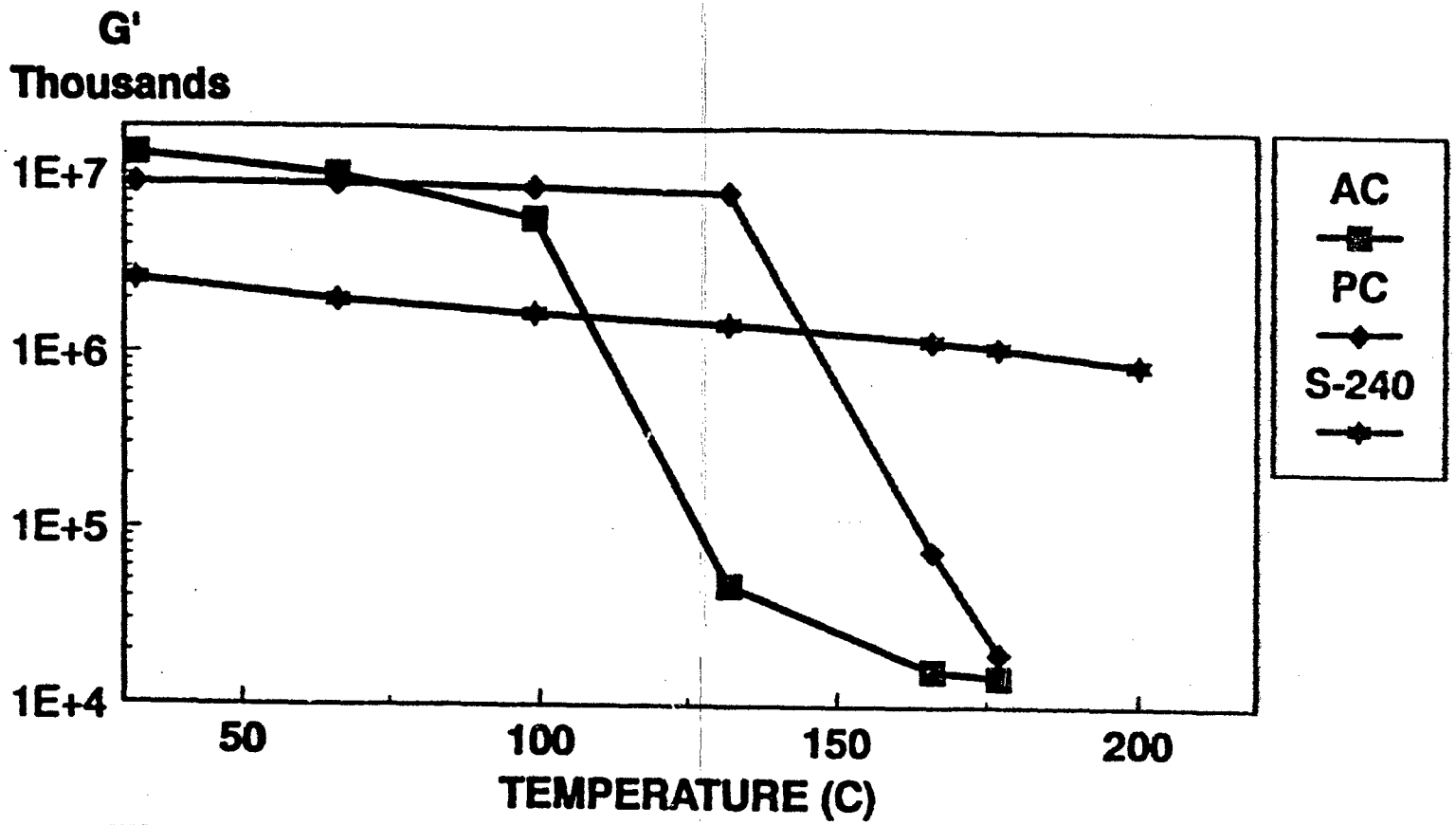


Fig. 10 COMPARATIVE DMA TRACE acrylic (AC), polycarbonate (PC) and S-240

SESSION III

UNDERSTANDING CURRENT MATERIALS - PART B

Chairman: E. Kochis
Boeing

Co-Chairman: B. S. West
University of Dayton

Coordinator: M. R. Unroe
Materials Directorate
Wright Laboratory

SOLID PARTICLE IMPACT EFFECTS ON AIRCRAFT TRANSPARENCIES

Robert G. Oeding
PDA Engineering

SOLID PARTICLE IMPACT EFFECTS ON AIRCRAFT TRANSPARENCIES

Robert G. Oeding
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2975 Redhill Avenue
Costa Mesa, CA 92626

ABSTRACT

Both military and commercial aircraft may be forced to operate in environments which contain significant loadings of dust, sand and debris. Distributions of small solid particulates can extend to flight altitudes and remain airborne for significant periods. Detection and avoidance of such particle environments by operational aircraft is not feasible with current instrumentation. Thus, the response of aircraft transparencies to solid particle impacts is an important issue.

Under government sponsorship, PDA Engineering has developed and currently operates a test facility to simulate the effects of solid particle impacts on critical aircraft surface materials. With this facility, the key impact parameters (i.e. particle size, speed, impact angle and total mass loading) can be controlled independently. Data from the facility are used to assess the vulnerability of existing aircraft surfaces/components, evaluate erosion resistant materials/protective techniques and to provide an experimental database for the development of predictive response models. Aircraft windscreens, canopies and sensor windows are especially vulnerable to particle environments. The response of aircraft transparencies to particle impacts will be presented in terms of the physical damage mechanisms and optical degradation.

Particle impact damage extends beyond the laboratory. The hazard to aircraft operations posed by airborne particulates has been well established through a number of commercial aircraft encounters with volcano-generated ash clouds. A Boeing 747-400 co-pilot front windscreen which flew into a Mt. Redoubt ash cloud was obtained and evaluated. The physical and optical degradation characteristics of this transparency which occurred in flight are presented.

INTRODUCTION

The hazards to aircraft operations posed by airborne sand/dust environments have been well established in recent years through commercial aircraft encounters with volcano-generated ash clouds and military aircraft experiences in Desert Shield/Storm operations. These experiences have shown that a number of aircraft surfaces and components are vulnerable to solid particle impact damage. Specifically, volcanic ash cloud encounters have resulted in the loss of engines (for a significant period of time), severe hazing of forward windscreens and landing light lens, plugging of pitot-static probes and removal of paint on leading edges and forward-facing surfaces.

Aircraft transparencies including windscreens, canopies and sensor windows are particularly vulnerable to solid particle impact damage. Although windscreen hazing is not as serious as engine loss, damage to transparencies tends to occur at much lower particle loadings than do engine related effects. Thus, for particle environments less severe than volcanic ash clouds, for example, transparency degradation will likely be a primary damage mode. For military aircraft operations, aircrew visibility loss and/or sensor performance degradation can have a significant impact on mission performance and crew safety. In addition, aircraft transparencies are expensive components and the reduced service life associated with operations in sand/dust environments has important cost implications.

The interaction of solid airborne particles with aircraft systems is a complex one and involves several issues including: (1) the characteristics of the ambient particle environment, (2) the interaction of the particles with the aircraft flowfield, (3) the actual response of aircraft surfaces to the particle impacts and (4) the effects of particle-induced damage on aircraft system performance and component service life. It is important to recognize that the damage produced by irregularly shaped solid particles is significantly different than that due to waterdrop impacts. Although an extensive database exists for rain erosion effects on materials of interest, these data are generally not of value in assessing solid particle impact damage. The physics of the impact process and materials response are quite different for waterdrop and sand/dust particle impacts.

Although a better understanding of ambient particle environments and the perturbation of those environments by the aircraft flowfield is important, the primary need is an understanding of the material / component response so that materials related vulnerability issues can be addressed. To characterize the impact response of critical aircraft materials, a valid experimental simulation of particle impact effects was required to assess vulnerability of existing aircraft surfaces and components, evaluate erosion resistant materials, assess mitigation techniques and provide an experimental database for the development of predictive response models.

When the issue of aircraft dust erosion was first addressed in the early 1980s, no suitable facility existed for characterizing dust erosion effects on aircraft surfaces. Single impact test capabilities did exist for investigating the physics of the impact process; however, this type of approach could not simulate realistic particle loadings over multi-sample arrays. Although a fairly extensive solid particle erosion database existed at that time, it was for materials and impact conditions that were not relevant to aircraft or missile flight through particle environments. These erosion data were generated for low velocity impacts (< 100 m/s) of alumina, silicon carbide or quartz particles of which only quartz is indicative of natural environments. The standard erosion facilities such as particle laden arc heaters, combustion tunnels, and rocket sled ranges were not suitable to simulate the subsonic velocity regime, had high uncertainties on particle size and impacted mass, limited (short) test times, and were complex and expensive facilities to operate.

The capability to conduct coupon scale tests in a well characterized and controlled particle environments was established at PDA Engineering's laboratory facilities through DOD support in 1983. This facility was later expanded and has been in operation since 1986 under the sponsorship of the Defense Nuclear Agency (DNA). Test programs have been conducted for a number of government agencies and industry users. A variety of materials including transparencies, paints, coatings and composite materials have been evaluated.

This paper describes the experimental technique and associated facility developed for simulating solid particle impact effects on various aircraft surfaces including transparencies. The response of typical transparency materials to particle impacts as well as the effects of the key impact parameters are examined. Also, as an example of erosion effects on actual flight hardware, the characterization of a Boeing 747 front windscreen which encountered a Mt. Redoubt ash cloud in December 1989 is presented in terms of both physical damage and optical degradation.

DUST EROSION FACILITY

The general requirement for simulating the response of surface materials to particle impacts is to effectively match the key physical parameters associated with the impact phenomenology. These parameters include particle type/size, mass concentration, impact velocity, impact angle, and air flow. Thus, simulation of aircraft dust erosion effects requires a facility which can produce steady, well controlled, low-concentration particle flow environments. Since aircraft operation is predominantly at subsonic cruise speeds, a high velocity test capability is not a primary requirement. However, accurate control of impacted particle mass and velocity are required over a broad range of velocities in order to determine the erosion characteristics of critical materials.

In order to provide a suitable test facility for addressing aircraft erosion issues, PDA Engineering, developed what is currently known as the DNA Dust Erosion Test Facility. This facility was designed to meet a wide range of flight environments and provides precise control over the key physical parameters in a laboratory test environment. The facility is located at PDA Engineering's laboratory facilities in Santa Ana, California. Specific simulation requirements included particle sizes from <38 to 250 μm, particle mass fluxes as low as 0.5 $mg/cm^2/min$, run times from a couple minutes to over an hour, particle velocities over a broad range of subsonic Mach numbers (20 - 300 m/s) and adjustable impact angles (15 - 90 deg with respect to the specimen surface).

Simulation Approach

In order to accomplish simulation requirements, dust particles are accelerated in a small diameter (approximately 0.25-inch) high speed gas jet and caused to impinge on a test specimen as illustrated in Figure 1. Since the diameter of the dust jet is smaller than the test specimen area, the specimen holder and jet are articulated so that the test specimen is moved through the dust jet in a uniform manner. This articulation provides a uniform particle loading (dust mass intercepted per unit surface area) over an approximately square area of 310 cm^2 (i.e. 6.9-inch square).

The transport gas stream is provided by either compressed air or pure nitrogen (GN2) with regulators and pressure transducers to measure and control the pressure at the nozzle inlet. Dust particles are metered into the transport gas stream from a pressurized screw feeder system (Accu-Rate Series 300). With this system both screw type, diameter and RPM can be controlled to provide a wide range of flow rates with particle sizes from a few micron to millimeters. Particle flows are uniform and stable. The dust mass flow rate is established by prior calibration for each particle size range or special blend. The total particle mass delivered to the specimen is then determined from the measure exposure time and the calibrated particle feed rate.

Dust velocity is determined as a function of the nozzle inlet pressure and the particle size by pre and post-test calibration. A laser doppler based velocimeter (LDV) is used to rapidly obtain accurate velocity distributions for large particle sample sizes. Thus, for a given test with a specified particle size, a specific test velocity is selected from this velocity versus pressure calibration. Once the pressure condition is established, particle velocity is verified via both pre and post-test measurements using the LDV system. Particle impact uniformity over the test area is determined by exposing a 6 x 6-inch glass sheet at the test conditions of interest. Haze measurements are then conducted at 25 locations over the glass sample with the standard deviation the haze values providing a measure of exposure uniformity.

Particle size, velocity and impact angle can be controlled independently. This provides an excellent capability to parametrically evaluate the response of critical aircraft materials to solid particle impact effects. Materials from such components as leading edges, windscreens, radomes, paints and any special coatings can be evaluated in a well controlled laboratory environment under realistic particle impact conditions. Details of this facility are described in Reference 1. A photograph of the primary test hardware is presented in Figure 2 which shows the gas flow control panel, an earlier (fluidized bed type) particle feed system, the test chamber, switching box for diagnostic instrumentation and a strip chart recorder. Separate areas for specimen preparation, physical examination/measurement and optical evaluation are also part of the laboratory facility but are not shown.

Test Particles

The primary concerns in selecting a test particles for simulating erosion response are (1) that the particle composition and shape representative of real environments and (2) that the material is available in a broad range of sizes. Although a variety of particle types and sizes can be used test facility, most of the testing conducted on transparencies to date has utilized a crushed silica material ranging in size from <38 μm to 177 μm. The material, referred to as foundry sand, is 96% silica with a trace of iron (providing a brown color). The material is purchased in bulk and specific particle size ranges are obtained by sieving. Ten screen sizes are available and particle cuts between most of the adjacent screen sizes are maintained in stock. The three different sizes of the crushed silica material are shown in the photomicrographs in Figure 3. The particle shape and composition is quite similar to desert sand samples from the middle east.

Test Specimen Configuration

Due to the facility design and test approach, flat test specimens are the standard configuration; however, specimens with moderate curvature can be accommodated with some loss in exposure uniformity. Test fixtures exist to handle up to 6 x 6-in square (flat) specimens. For materials screening tests, an array of 9 specimens of 2 x 2-inches (see Figure 4) or 16 specimens of 1-inch diameter can be accommodated in existing test fixtures. The 1-inch diameter sample array is used primarily for sensor window materials. The use of sample array for screening tests reduces the number of exposures required and provides a direct comparison between materials. Also, the specimens can be exposed incrementally with physical characterization and/or performance measurements conducted after each exposure. Thus, material response (e.g. mass loss, haze, IR transmissivity, etc.) can be determined as a function of the particle loading (gm/cm^2) for specific test conditions (i.e. particle size, velocity and impact angle).

Relationship to Flight Environment

The dust erosion test facility differs from the real flight environment in that a stationary test specimen is impacted by a moving dust field. Whereas the key parameters in the flight environment are the static particle cloud concentration (mass of particles per unit volume) and the aircraft velocity, in the dust erosion facility the key parameters are the particle mass loading (mass of particles per unit surface area) and particle velocity. For the range particle flow rates and velocities that can be achieved in the facility, the key parameter for correlating material damage is the particle mass loading since it represents the total particle mass impacting a unit area of the test surface. The relationship between the particle mass loading in the test facility, the impact velocity, and the particle cloud concentration in the flight environment is as follows:

$$\frac{\Delta m}{A_S} = 0.006\, C_C\, U_p\, \Delta t\, \sin\alpha \qquad (1)$$

where

Δm = particle mass impacting reference surface area (A_S),
C_C = ambient cloud concentration (gm/m^3)
U_p = impact velocity (m/sec)
A_S = surface reference area (= 310 cm^2)
α = impact angle (normal impact = 90°)
Δt = exposure time (minutes)

In the test facility, the particle mass flow rate ($\Delta m/\Delta t$) is controlled by adjusting the screw feeder system. This involves adjusting either the speed of the screw and/or the screw diameter. For the current facility configuration, stable particle flows can be achieved for particle flow rates between approximately 0.2 and 20 gm/min.

For specific test conditions, the corresponding equivalent cloud concentration range that can be simulated in the dust erosion facility is determined from Equation (1). Solving for the cloud concentration, C_C, yields:

$$C_C = 166.7 \frac{\Delta m}{U_p\, A_S\, \Delta t\, \sin\alpha} \qquad (2)$$

For example, with normal particle impact at a velocity of 250 m/s (820 ft/sec), equivalent cloud concentrations between 0.00043 and 0.0430 gm/m^3 can be simulated in the dust erosion facility corresponding to the maximum particle flow rate (20 gm/min) and the minimum flow rate (0.2 gm/min).

TRANSPARENCY RESPONSE

Aircraft transparencies are particularly vulnerable to solid particle impact damage. These include cockpit transparencies, windscreens and canopies, which are required for aircrew vision as well as sensor windows such as those used on FLIR systems. Strategic, C^3 and commercial aircraft generally have large composite windscreens which utilize a glass outer ply. These glass plies are strengthened by chemical or thermal tempering thus providing a hard surface for resisting minor abrasion damage. Tactical aircraft canopies, on the other hand, generally have complex shapes which require formable plastic materials such as acrylic and polycarbonate. These materials are softer than glass and more easily pitted and scratched. Sensor windows utilize glasses and polycrystalline materials with special coatings in order to achieve high transmittance in selective wave bands. Both substrate materials and coats are susceptible to particle impact damage.

The response of the transparency surface to particle impacts is material dependent. Glasses and polycrystals are brittle materials which can easily fracture under particle impact loads while plastics are soft and deform when impacted by a solid particles. Since the transparency is a critical component in an optical imaging system, even minor surface damage may seriously degrade the optical performance of the system. Optical degradation occurs well before any significant material mass loss occurs. Impact induced surface craters act as scattering centers for incident light. This scattered light appears as a veiling luminance which reduces the image contrast and impairs aircrew visibility and optical detector performance. This effect is generally referred to as hazing and is measured for visible transparencies in the laboratory with a Hazemeter in accordance with Federal Test Standard No. 406, Method 3022 or ASTM Method D 1003 (Reference 2). For sensor windows with transmission bands in the mid-wave (MWIR) and long-wave infrared (LWIR), spectral transmission measurements provide a simple and sensitive approach for assessing particle impact damage. Also, on sensor windows particle impacts remove or damage anti-reflection and/or other special coatings which may be required to achieve system performance goals.

Damage Modes

Erosion tests have been conducted on samples of primary windscreen and canopy materials, coatings designed for erosion protection and a variety of sensor window materials. Specific transparency materials that have been tested are summarized in Table 1. The most extensive database has been developed for glasses and acrylics which include variations in velocity, impact angle, particle size and dust loading. The primary optical performance measurements obtained for the exposed transparency specimens are luminous transmittance and haze. Haze is quite sensitive to surface damage and provides a convenient and objective measure of optical degradation.

Photomicrographs of typical surface craters produced by impacts on chem-tempered glass, aircraft grade polycarbonate and cast acrylic are shown in Figure 5. For brittle materials such as glasses and IR transmitters, the damage mode is localized fracturing at the particle impact site. Lateral cracks are formed below the surface which may extend well beyond the point of impact. Although damaged material may remain in place, the fracture pattern acts as a light scattering surface. The occurrence and fracture pattern is dependent on impact parameters. The effect of impact angle on haze level for glass and acrylic specimens is presented in Figure 6 for a specific particle environment and impact velocity. For brittle material such as glass, the maximum haze level occurs at the normal impact angle. Plastic materials are more ductile. Particles impacting normal to the surface deform the surface and imbed while particles impacting at shallow angles tend to "plow" material from the surface forming elongated craters as shown in the photomicrographs. Thus, with plastics maximum surface damage occurs at shallow angles of the order of 30° from the surface. Since particle impacts on installed cockpit transparencies generally occur at angles less than 30°, plastic canopies are generally more susceptible to hazing than comparable glass transparencies.

The relative performance of the primary cockpit transparency materials at shallow impact angles typical of aerodynamically deflected particle impact conditions on windscreens and canopies is presented in Figure 7. Performance is presented in terms of measured haze as a function of the particle loading. Maximum acceptable haze values for production cockpit transparencies are generally less than 4%. Significant visual degradation may occur at haze levels greater than approximately 15% depending on ambient lighting conditions. For shallow impact angles, material rankings in order of increasing haze (i.e. surface damage) are polycarbonate, acrylic, glass and a good elastomeric protective coating (over a primary substrate). For the conditions listed in Figure 7, polycarbonate reaches haze values of 15% at dust loads of the order of 0.005 g/cm^2 while a chem-strengthened glass can withstand a dust load of 0.024 g/cm^2 before incurring the same degree of haze. The erosion resistance of the primary cockpit transparencies can be further enhanced through the use of a tough elastomeric coating such as a high tear-strength polyurethane on the external surface of the transparency. The better elastomeric coatings offer significant protection from both hazing (as shown in Figure 7) and damage to substrate surface coatings.

Damage characteristics of sensor windows are similar to those of glass windscreens. The particle impact, if sufficiently energetic, causes localized pitting which contributes to surface scattering of the infrared radiation and reduces the quality of the signal reaching the sensor. Thus, the optical performance of infrared-transmitting windows and domes may be significantly reduced. Even without pitting the substrate material, damage to anti-reflection and abrasion-resistant surface coatings can result in significant degradation in optical performance. Since the optical performance of infrared (IR) sensor windows is

based on IR spectral transmittance measurement , it is difficult to compare sensor window optical performance with that of windscreen and canopy. Based on general surface damage characteristics, the baseline LWIR transmitting materials such as zinc sulfide, zinc selenide and germanium appear to be more vulnerable to surface damage than the cockpit transparencies. Based on limited data, MWIR transmittance materials such as magnesium fluoride and spinel, on the other hand, appear to be significantly more resistant to particle impact damage than the LWIR materials. The vulnerability of sensor window is also enhances, compared to windscreens and canopies, by the fact that they are usually mounted nearly normal to the particle flow. This is important since impact angle effects for brittle materials result in maximum damage near the normal impact angle.

Test Parameter Effects

The effects of key impact parameters on haze level have been evaluated parametrically for unstrengthened soda lime glass and should be representative of other brittle materials. The most sensitive parameter is particle velocity. Effects of particle velocity on haze are shown in Figures 8 and 9 for specific dust loads and impact angles, respectively. Haze varies approximately as the square of the velocity for these conditions. Impact angle is also a sensitive parameter for brittle materials at shallow impact conditions. As shown in Figure 10, haze increases with impact angle for glass until a maximum is reached at normal impact conditions. Particle mass loading, which is essentially a measure of the number of particles impacting the surface, is a less sensitive parameter than velocity or angle. Haze varies linearly with dust load (as seen in Figure 7) until the surface becomes obscured to the point that particles begin to impact previously damaged areas. However, this non-linearly effect usually only occurs at very high haze levels.

For IR transmitting materials, the loss in average spectral transmittance over a specific IR band is used as a sensitive and convenient measure of window degradation. As an example, Figure 11 presents, for a typical LWIR window material, measured transmittance loss as a function of particle loading for several impact velocities. A semi-log format is used in Figure 11 in order to display the three order-of-magnitude span in particle loading. As the figure indicates, for a 200 m/s impact velocity, significant optical damage occurs at particle loadings less than 0.01 gm/cm^2. These damage levels in terms of threshold particle loading (the particle loading where measurable damage occurs) are similar to those of acrylic and polycarbonate shown in Figure 7. In both cases, only very low particle loadings are needed to product significant optical degradation. The strong influence of particle velocity on the threshold particle loading level is also apparent for the IR window material. For a specific level of optical degradation, the associated particle loading varies as approximately the square of impact velocity.

EXAMPLE OF INFLIGHT DAMAGE

A number of aircraft encounters with volcanic ash clouds over the past several years have demonstrated the vulnerability of critical aircraft components and systems to severe particle environments. As related to transparencies, an examination of windscreen damage from volcanic ash clouds not only characterizes the damage level but also can provide insights into the particle and flow characteristics over that particular region of the aircraft.

One of the more serious incidents involved a Model 747-400 commercial airliner making a descent into Anchorage Alaska at 20:25 on 15 December 1989, the day after an eruption of near-by Mount Redoubt. The flight crew and air traffic controllers were unaware of the ash cloud hazard. The aircraft entered the ash cloud at 25,000 feet and experienced flameouts of all four engines. The flight crew was able to restart two of the engines at 17,000 feet, after several attempts. The remaining engines were finally restarted after several more attempts at 13,000 feet and the aircraft was able to make a manual approach and landing at Anchorage.

Damage to the aircraft was extensive including such items as engines, contamination to various systems, erosion of protruding components/parts, pitot-static system failure and hazing of forward transparencies. A more detailed description of the incident and damage to the aircraft can be found in References 3 and 4. The forward facing transparencies suffered significant damage due to ash particle impacts during Mt. Redoubt encounter. A detailed examination of the co-pilot's front windscreen has provided important information that may be useful in characterizing both the ambient ash cloud and the aerodynamic flow of the ash over the forward aircraft surfaces. The front windscreen was a glass-polycarbonate laminate having a thermally tempered outer glass ply which was exposed to the ash flow. Examination of the windscreen involved two different approaches: (a) microscopic examination of the windscreen outer surface and (b) a variety measurements to characterize erosion effects on windscreen optics.

Windscreen measurements consisted of gridboard photographs of the complete part, luminous haze, contrast and inferred particle flow direction taken at a variety of gridded points across the windscreen surface. Results showed that the damage pattern varied significantly over the windscreen surface. This is illustrated in Figure 12, a photograph of the windscreen in front of an optical grid board. An area adjacent to the centerline of the aircraft and covering approximately one-third of the windscreen was totally obscured. Except for a small area along the the lower edge shadowed by the deicing nozzle, none of the original glass surface remained undamaged in this area and visual resolution was not possible regardless of the target. Damage decreases from the centerline to the outboard edge. As seen in the photograph, a region adjacent to the outer edge remained essentially undamaged.

In order to characterize erosion damage over the surface of the windscreen, optical measurements were made a fixed grid locations. Figure 13 shows a photograph of the co-pilots windscreen at the installed attitude. Haze, contrast and surface damage measurements were made at each of the circular grid points shown in the photograph. A contour plot of the haze distribution based on these measurements is presented in Figure 14 and shows the smooth decrease in haze from the centerline to the outboard edge.

Surface damage was characterized by two types of craters: (1) thin elongated craters resembling scratches and (2) standard brittle fracture patterns. Figure 15 presents photomicrographs of the surface in both the highly damaged area near the aircraft centerline and at a location near the outer edge of the windscreen. These two types of surface damage can be clearly seen in the "outboard" photomicrograph. The surface "scratches" appeared to be caused by particles closely following the airflow and impacting at very shallow angles. They were not due to cleaning or other forms of surface abrasion.

The local direction of these "scratches" was measured and is shown as lines drawn through each grid point in the photograph in Figure 13. A contour plot of these inferred flow angles is shown in Figure 16. Although the flow angle contours are not as smooth as the haze data, they do indicate the increasing angle (as measured from the centerline) with distance from the centerline edge of the windscreen. The brittle craters appear to be the result of larger particles impacting at steeper angles. The overall pattern of damage appears to be consistent with particle/flowfield interaction effects. Severe damage occurred along the centerline (more normal impact angles) and decreased as the windscreen curved toward the outboard edge of the aircraft (shallower impact angles).

Data obtained from an examination of inflight transparency damage such as the 474 windscreen may prove useful in developing a better understanding of both the particle flow characteristics over the forward portion of the aircraft as well as the ash cloud particle size/concentration ranges. Comparisons of the inferred surface flow data obtained from the examination of the 747 windscreen with computational fluid dynamic calculations may useful in validating computational methods for predicting particle flow and impact conditions.

In order to infer integrated ash particle loadings or particle size ranges, laboratory induced damage on samples of the outer ply material with ash particles would be required. Preliminary tests on tempered glass specimens at very shallow impact angles using volcanic ash particles indicate that the damage modes seen on the 747 windscreen can be duplicated in the laboratory. Although not a simple task, the correlation of laboratory and windscreen damage in terms of crater size and density may provide insight into ash cloud characteristics.

SUMMARY

Airborne sand and dust environments represent a serious hazard to both commercial and military aircraft. Encounters with dense particle environments such as volcanic ash clouds can lead to serious aircraft damage and possible loss, if prompt action is not taken by the air crew. Dense particle environments may also be encountered by military aircraft during low altitude penetration missions or tactical battlefield support. Examination of a front windscreen from a commercial 747-400 airliner which encountered a volcanic ash cloud from Mt. Redoubt in Alaska illustrates the severe damage produced by dense particle environments.

Natural desert or sand environments such as those experienced in the middle east during Desert Shield/Storm are usually less severe than the volcanic ash clouds and are primarily a threat to aircraft transparencies. Windscreens, canopies and sensor windows are particularly vulnerable to particle impacts which may affect mission performance and crew safety. However, the primary effect of these low density environments is to significantly reduce transparency service life which increases aircraft down time and maintenance costs.

The overall particle impact phenomenology is complex and involves several issues such as environments, aircraft flowfield interactions, materials response and damage assessment. Since protecting or hardening exposed materials is the most direct approach to reducing damage and improving service life, the evaluation of material response to particle impact environments is a critical area. Solid particle impact phenomenology is significantly different than that of waterdrops and the extensive rain erosion database that has been developed for aircraft materials and components cannot be readily use to predict materials performance in solid particle environments. A solid particle effects database similar to that for rain erosion is needed to better characterize and improve the hardness of aircraft transparencies. To meet this need, a laboratory scale facility has been established for evaluating particle impact effects on critical aircraft surfaces and components. This facility is located at PDA Engineering laboratories in Santa Ana, California and has been in operation under DNA sponsorship since 1986.

The primary damage mode associated with particle impacts on transparencies is optical degradation due to light scattering from surface pits and craters. This can occur at very low particle loadings well before any significant surface mass loss occurs. Surface damage is a strong function of impact velocity and angle. Impact angle effects may also be material dependent as is seen with brittle and ductile materials. For shallow impact angles associated with forward windscreen and canopy surfaces, plastic materials experience significantly higher damage levels than glass surfaces. At near normal impact angles associated with sensor windows, for example, the opposite is true with glass or brittle IR transmitting materials experiencing the greater degradation.

Test results indicated that the use of an elastomeric liner (such as high tear-strength polyurethane) as a protective external coating can significantly reduce surface damage and hazing. This approach may be of particular value for plastic windshields and forward canopy areas on tactical aircraft which must operate in desert environments. For sensor windows, mitigation efforts have been directed primarily towards the use of hard external coatings to protect fragile AR coatings and window substrates. Specifically, some hard carbon or diamond-like coatings have shown improved impact damage resistance under certain conditions.

In addition to improving material hardness, methods to enhance the deflection of particles by the aircraft flowfield have received some limited attention. Flow deflectors or similar devices would deflect the surface air flow (and particles) from critical areas such as windscreen areas used for target acquisition, refueling, landing and etc. The primarily difficulties associated with this approach are the requirements for sophisticated two-phase computational fluid dynamic (CFD) analyses and the potential for disrupting the aircraft aerodynamics and affecting overall aircraft performance.

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3. Brantley, S.T.,"The Eruption of Redoubt Volcano, Alaska, December 14,1989 - August 31, 1990", U.S. Geological Survey Circular 1061, 1990.

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Table 1. List Transparency Materials Evaluated in Dust Environments

APPLICATION	TYPE	MATERIAL	VARIATIONS
WINDSCREENS & CANOPIES	GLASS	Soda Lime	Strengthened Unstrengthened
	PLASTIC	Acrylic Polycarbonate	Stretched Cast A/C Grade Commercial
SENSOR WINDOWS	MWIR	Magnesium Fluoride ALON Spinel Calcium Aluminate	Various Coatings " " "
	LWIR	Zinc Sulfide Zinc Selenide Germanium Silicon Diamond	Various Coatings " " " "

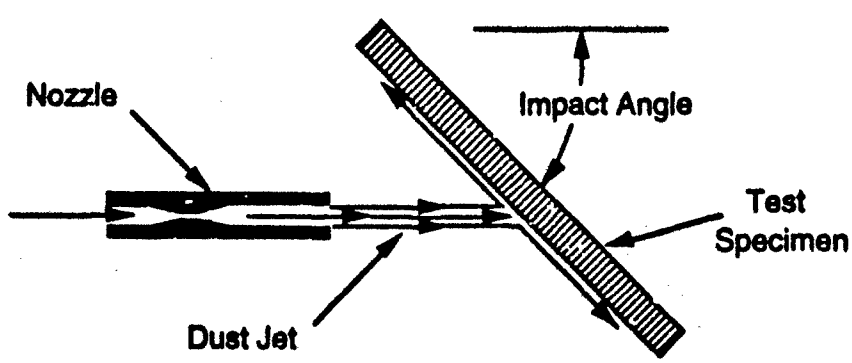


Figure 1. Test Configuration

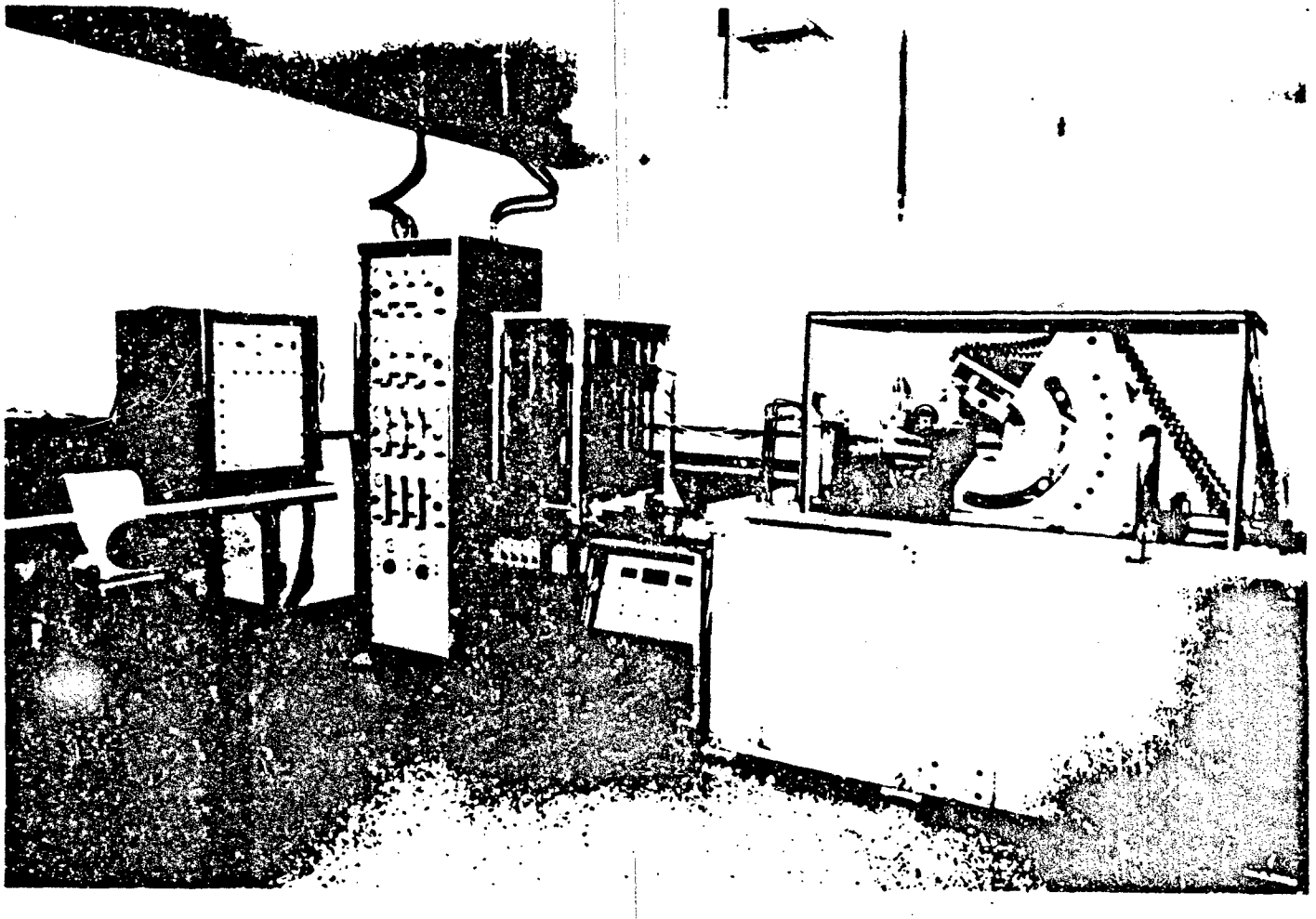


Figure 2. DNA Dust Erosion Test Facility

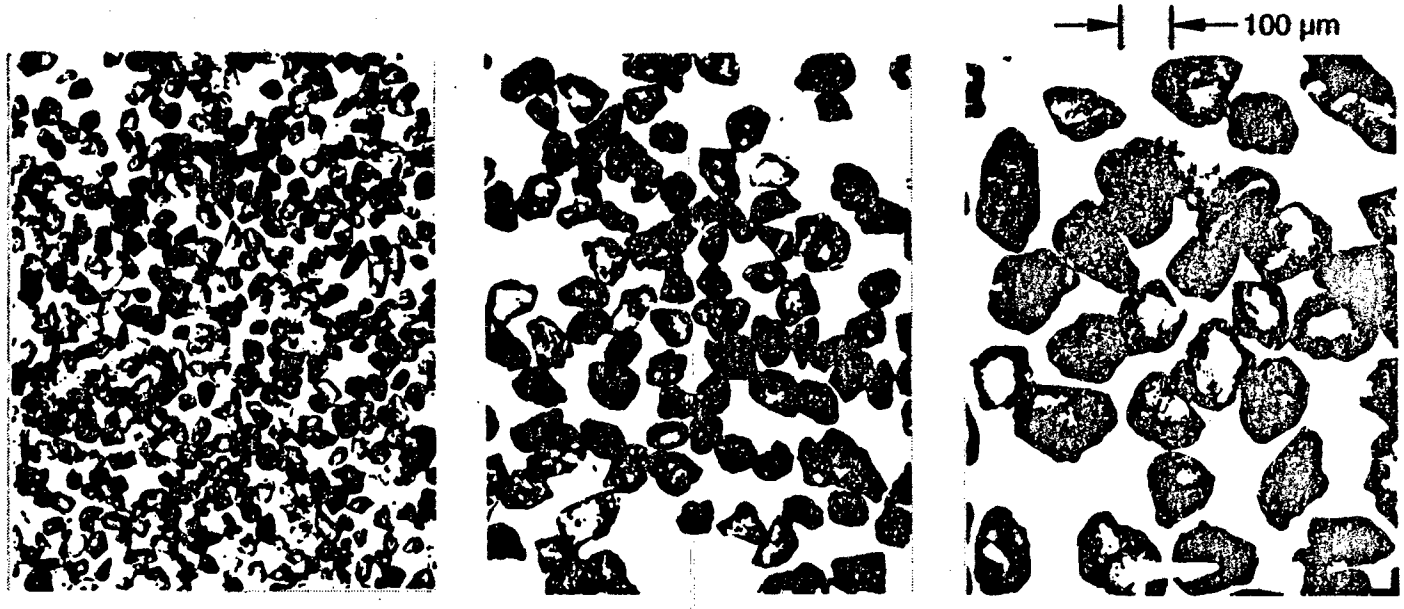


Figure 3. Photomicrographs of Crushed Silica Particles

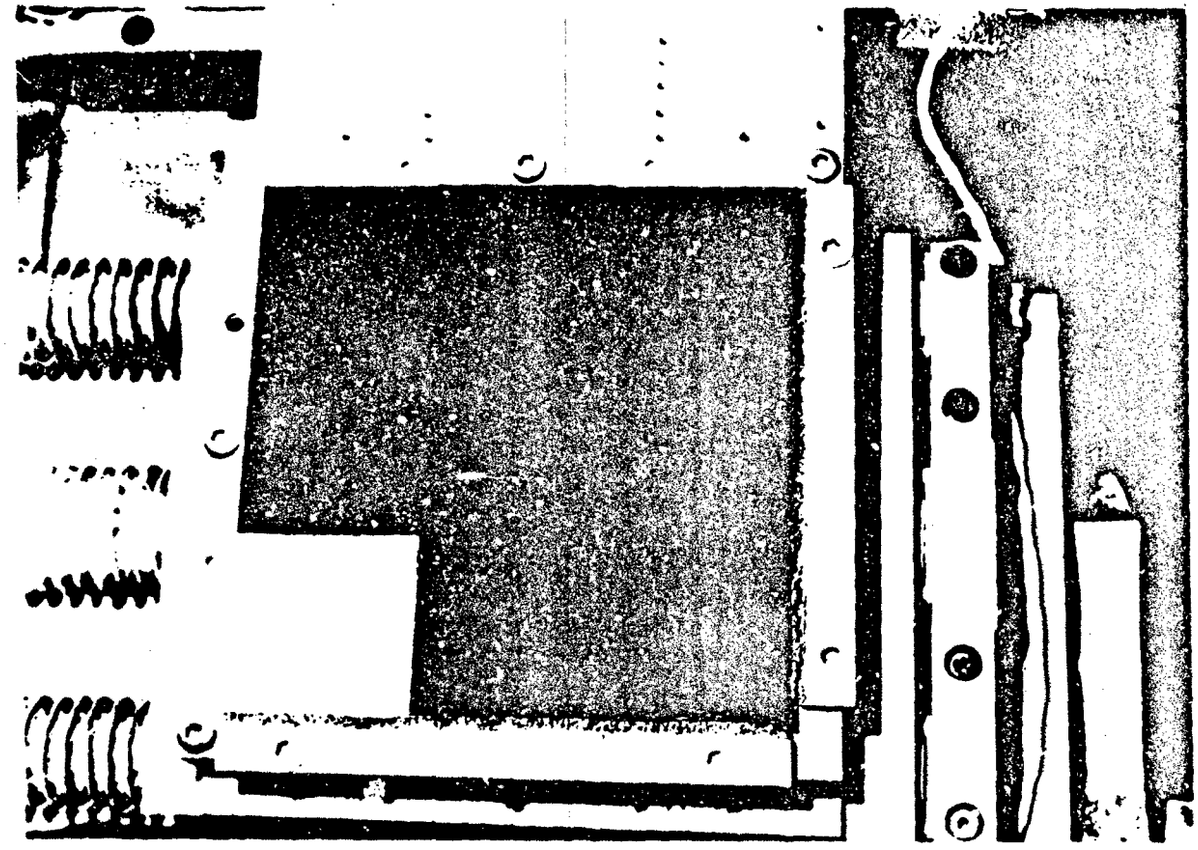


Figure 4. Test Fixture for Multi-Sample Arrays

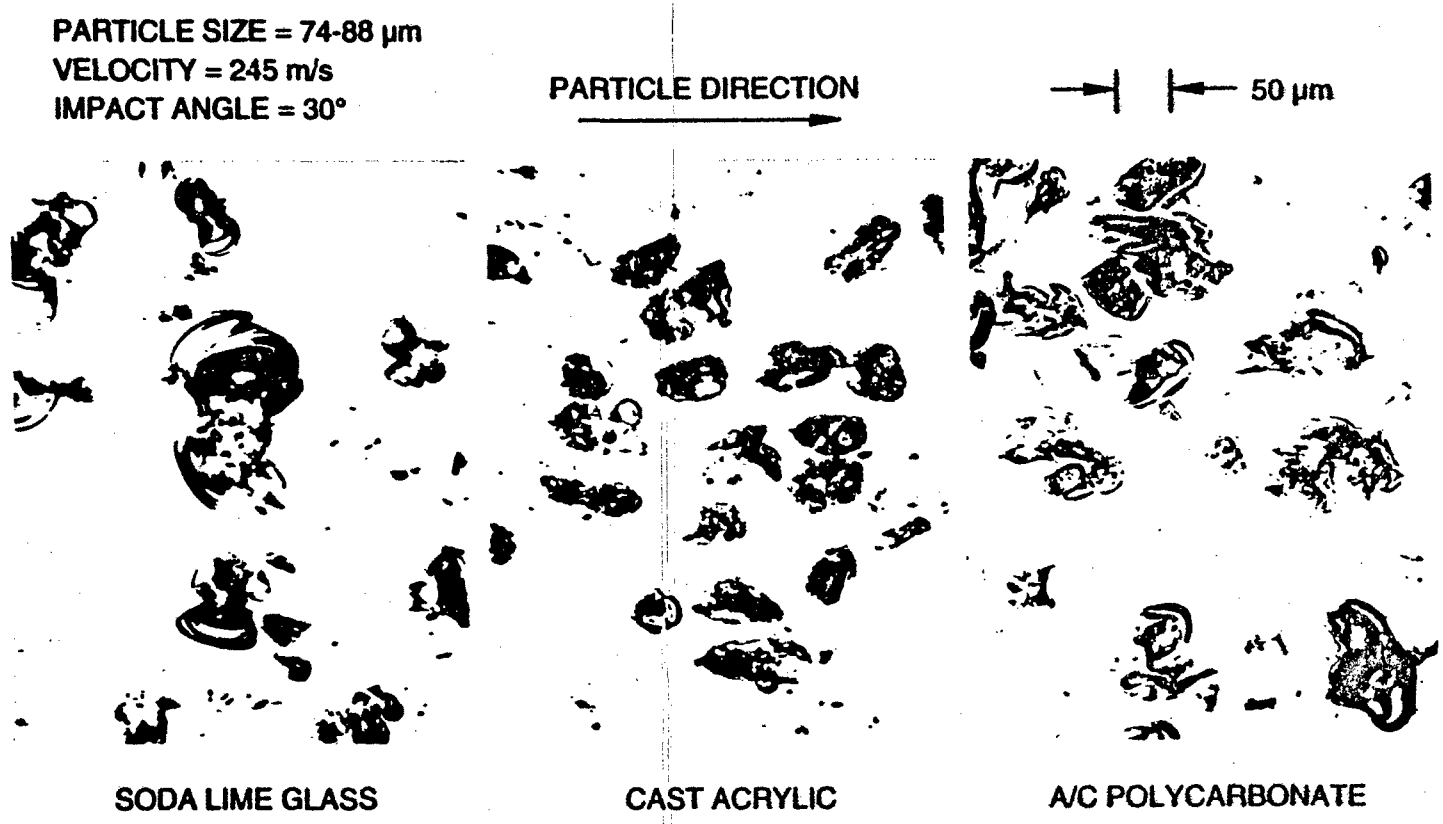


Figure 5. Photomicrographs of Particle Impact Damage

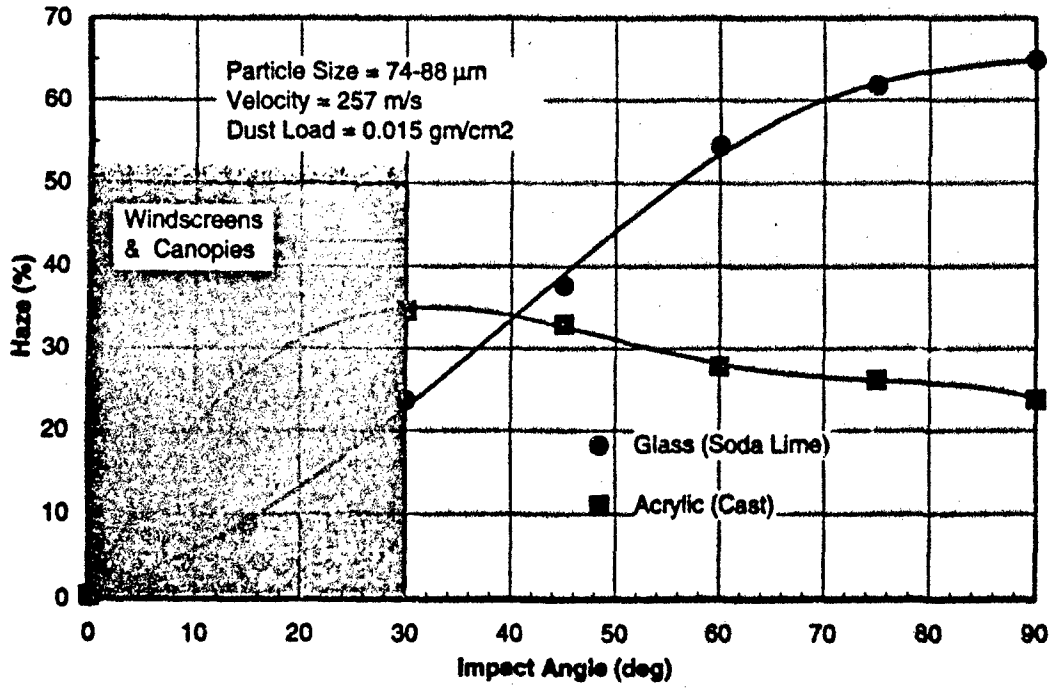


Figure 6. Effect of Impact Angle on Transparency Haze Level

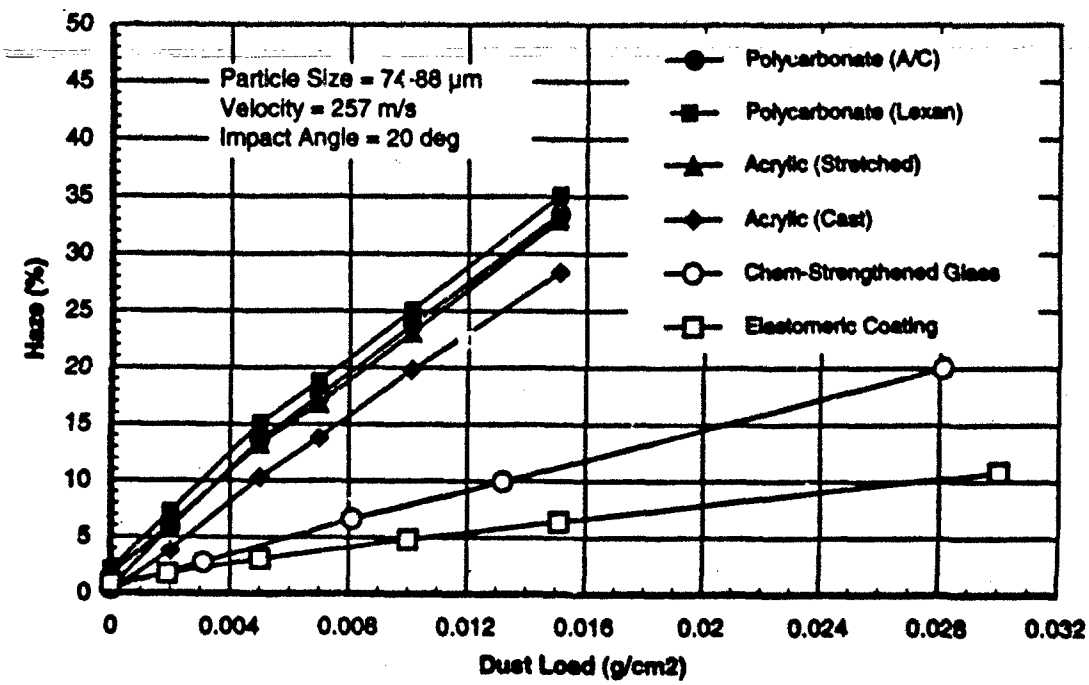


Figure 7. Dust Erosion Effects on Transparency Materials at Shallow Impact

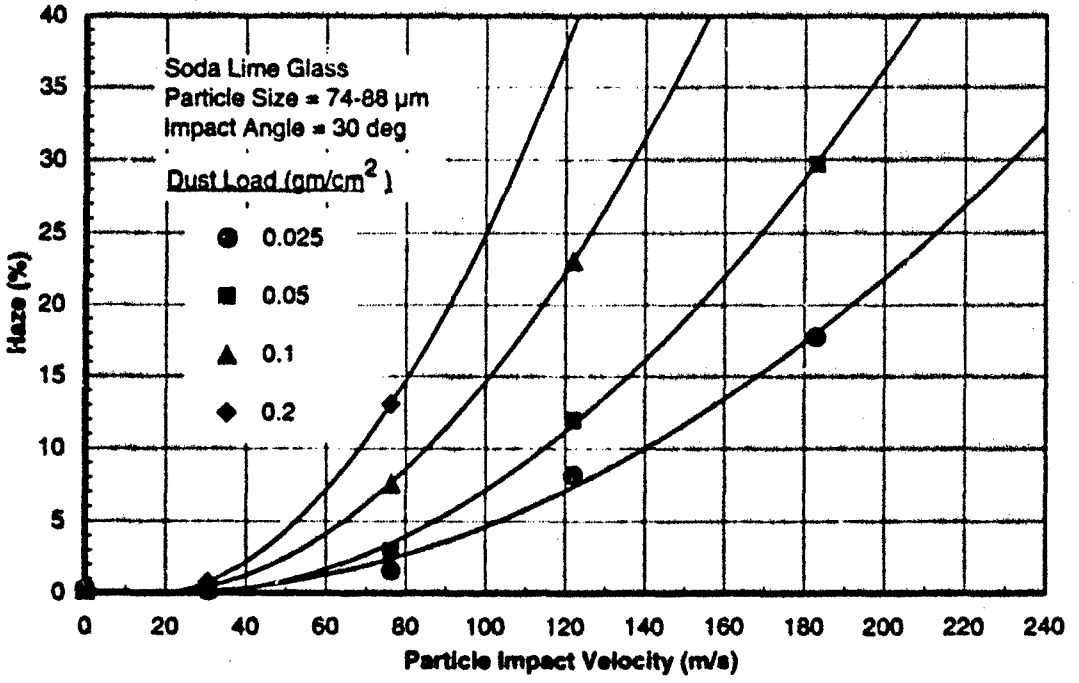


Figure 8. Haze vs Particle Velocity for Specific Dust Loads

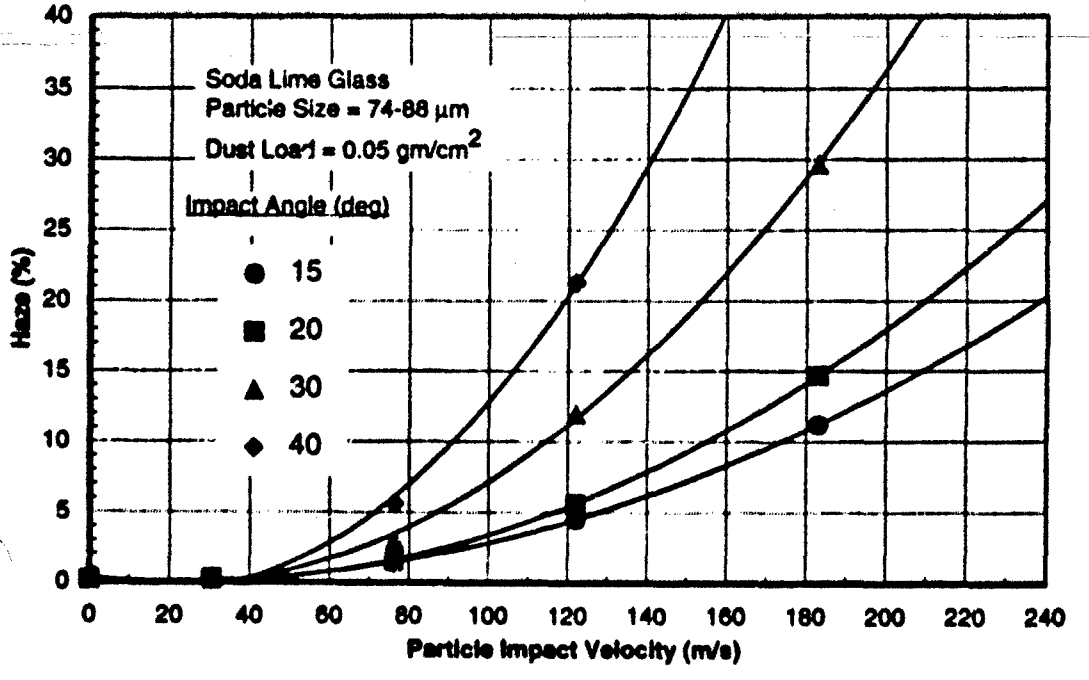


Figure 9. Haze vs Particle Velocity for Specific Impact Angles

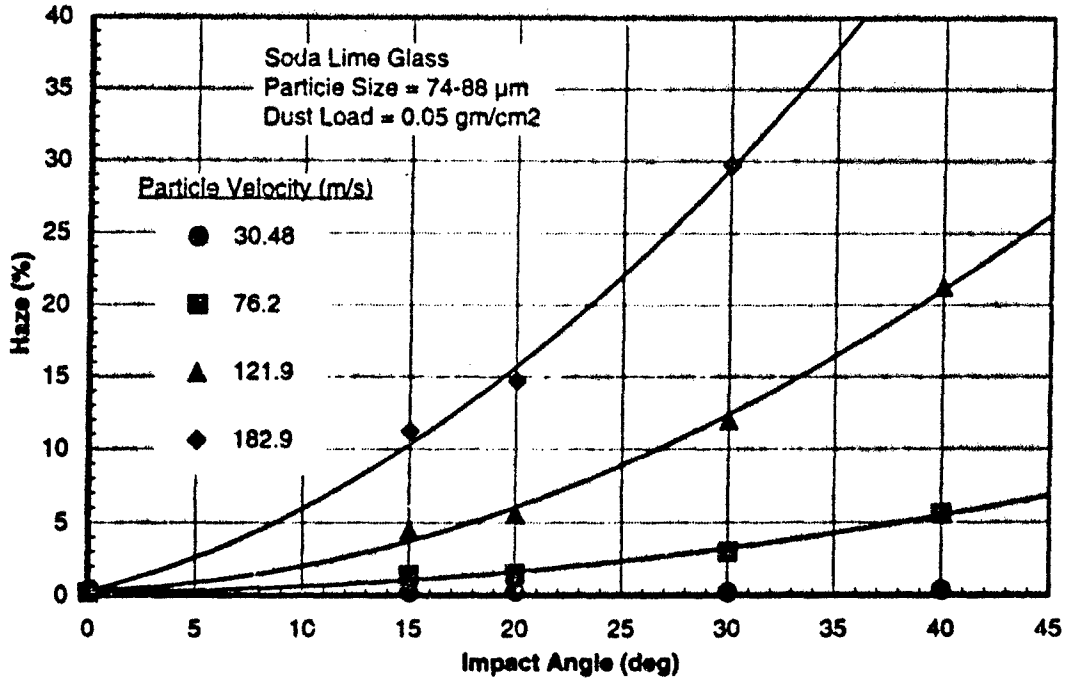


Figure 10. Measured Haze as a Function of Impact Angle for Specific Velocities

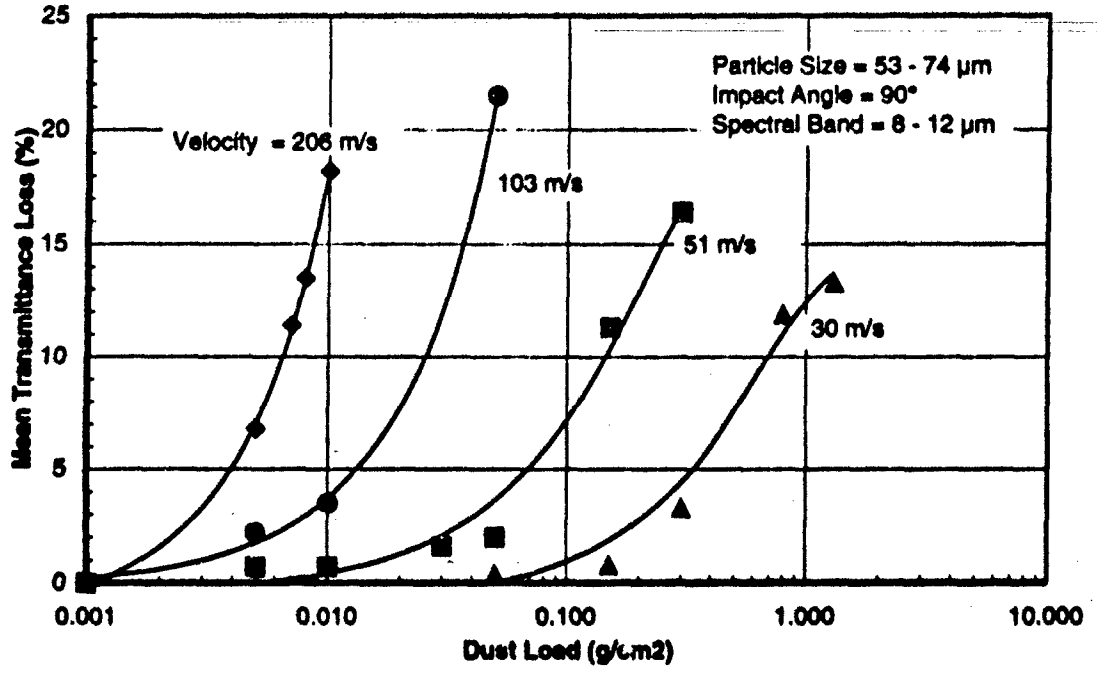


Figure 11. Particle Impact Effects on Typical IR Window Material

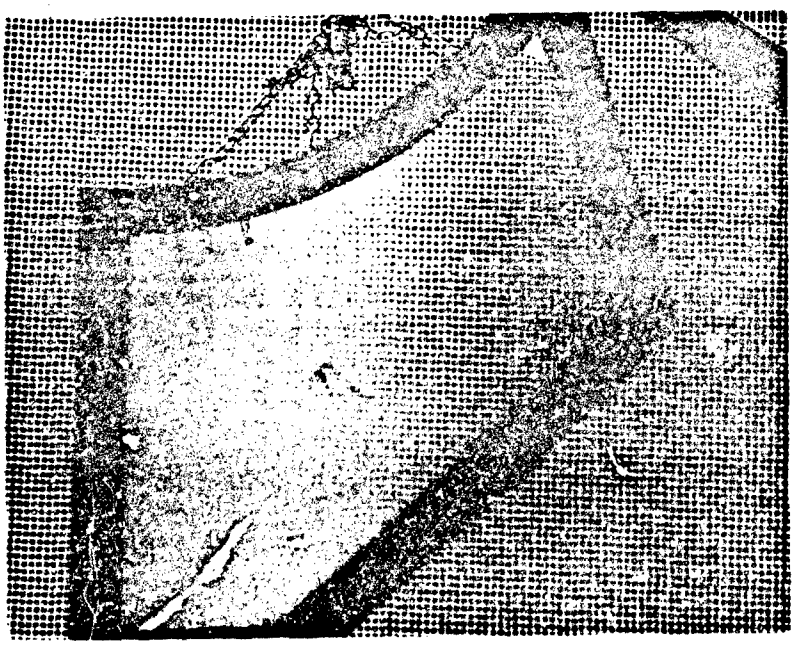


Figure 12. Grid Board Photograph of 747 Forward Windscreen

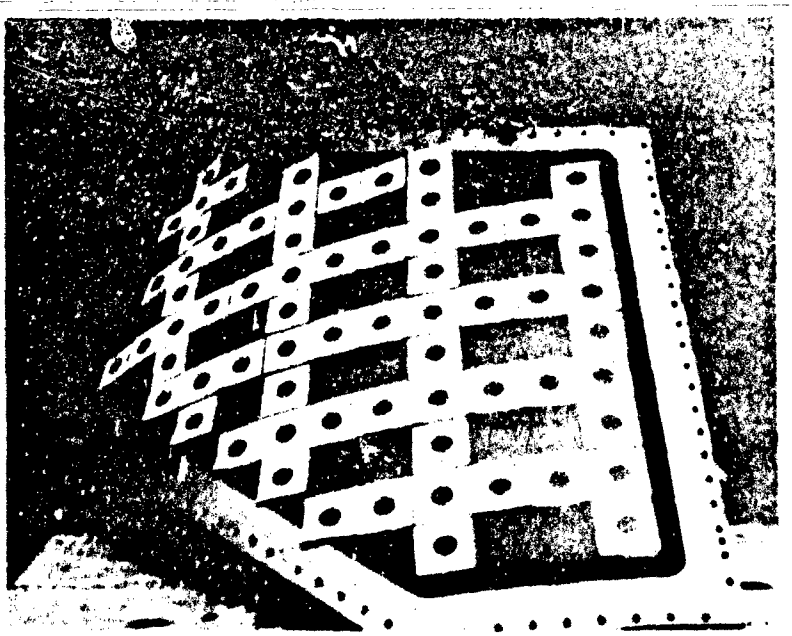


Figure 13. 747 Windscreen with Measurement Grid

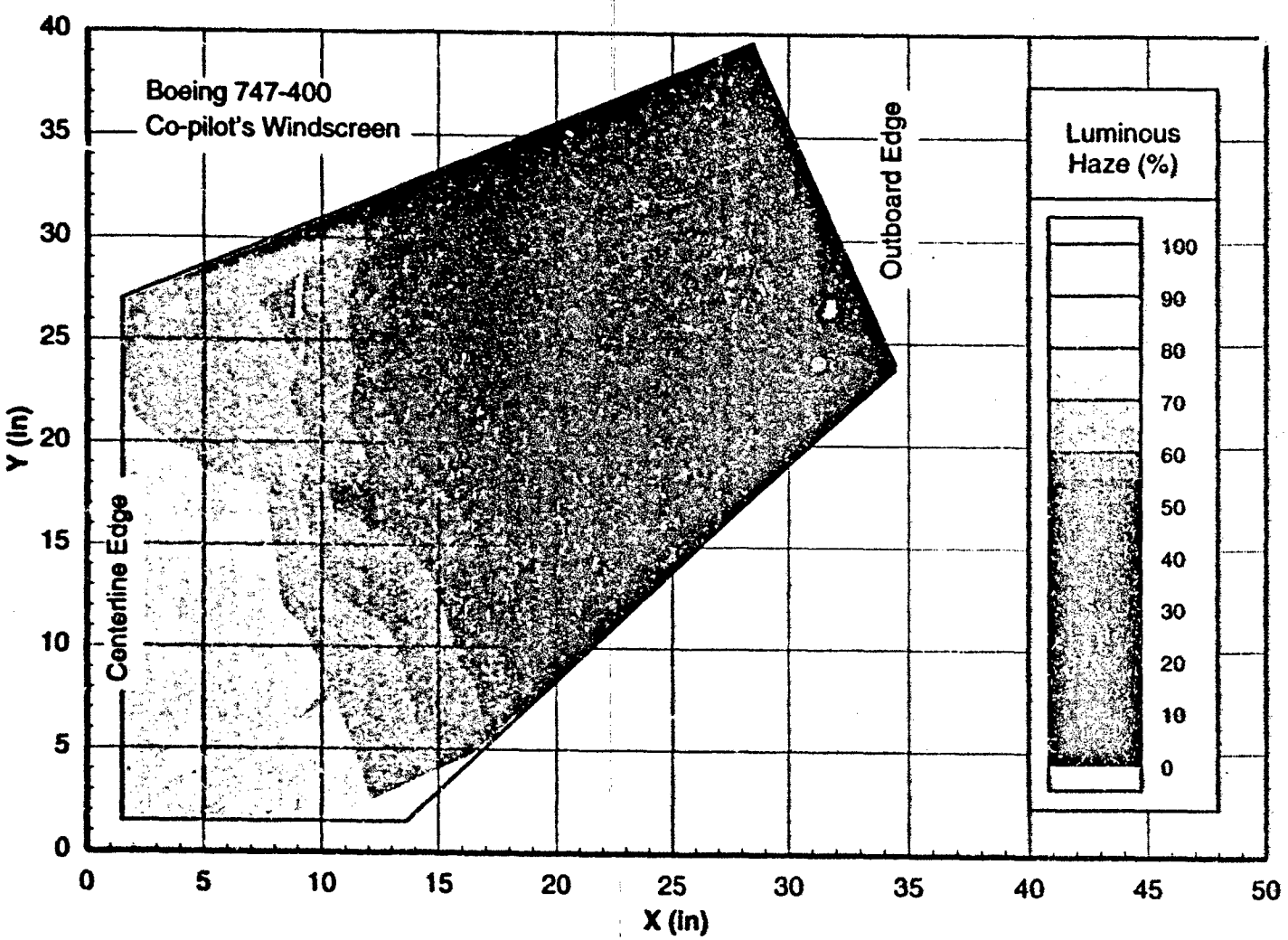
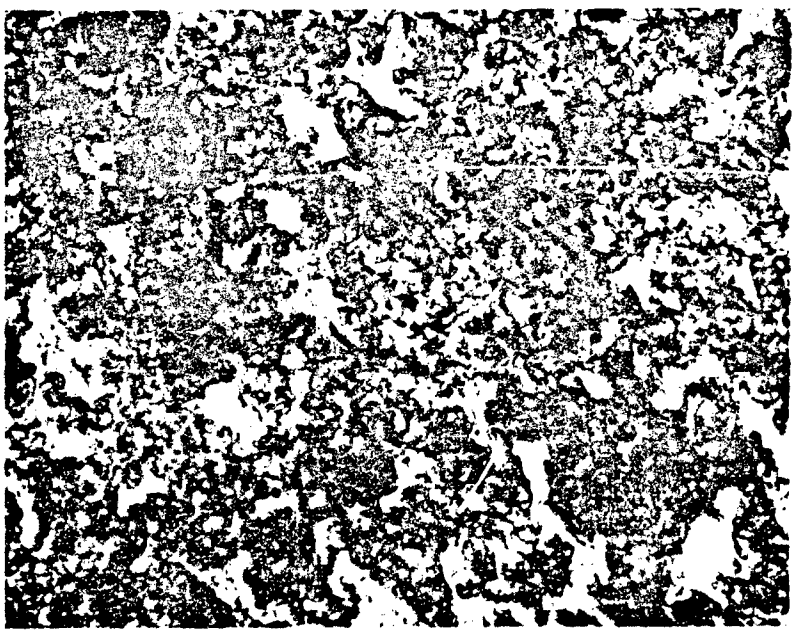
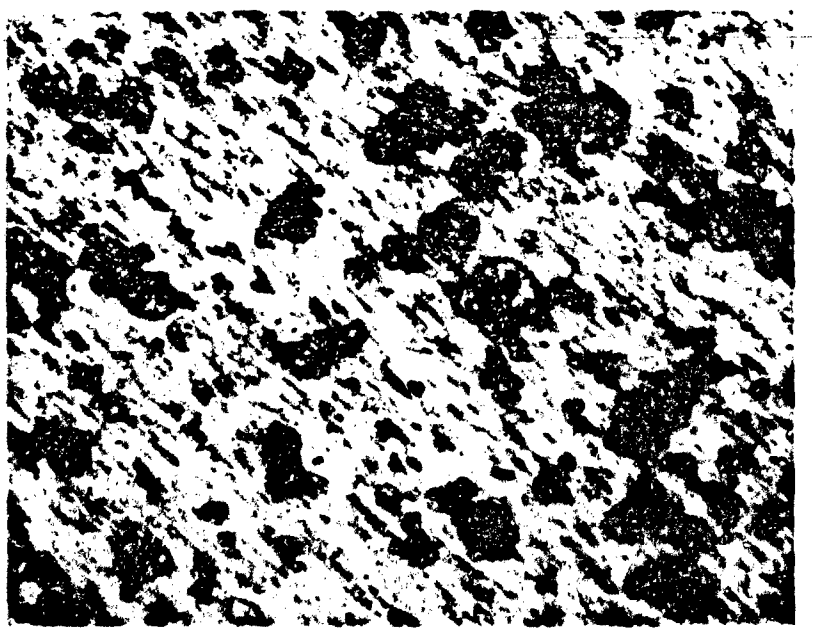


Figure 14. 747-400 Windscreen Haze Distribution



(a) Centerline Location





(b) Outboard Location

Figure 15. Photomicrographs of 747-400 Windscreen Surface

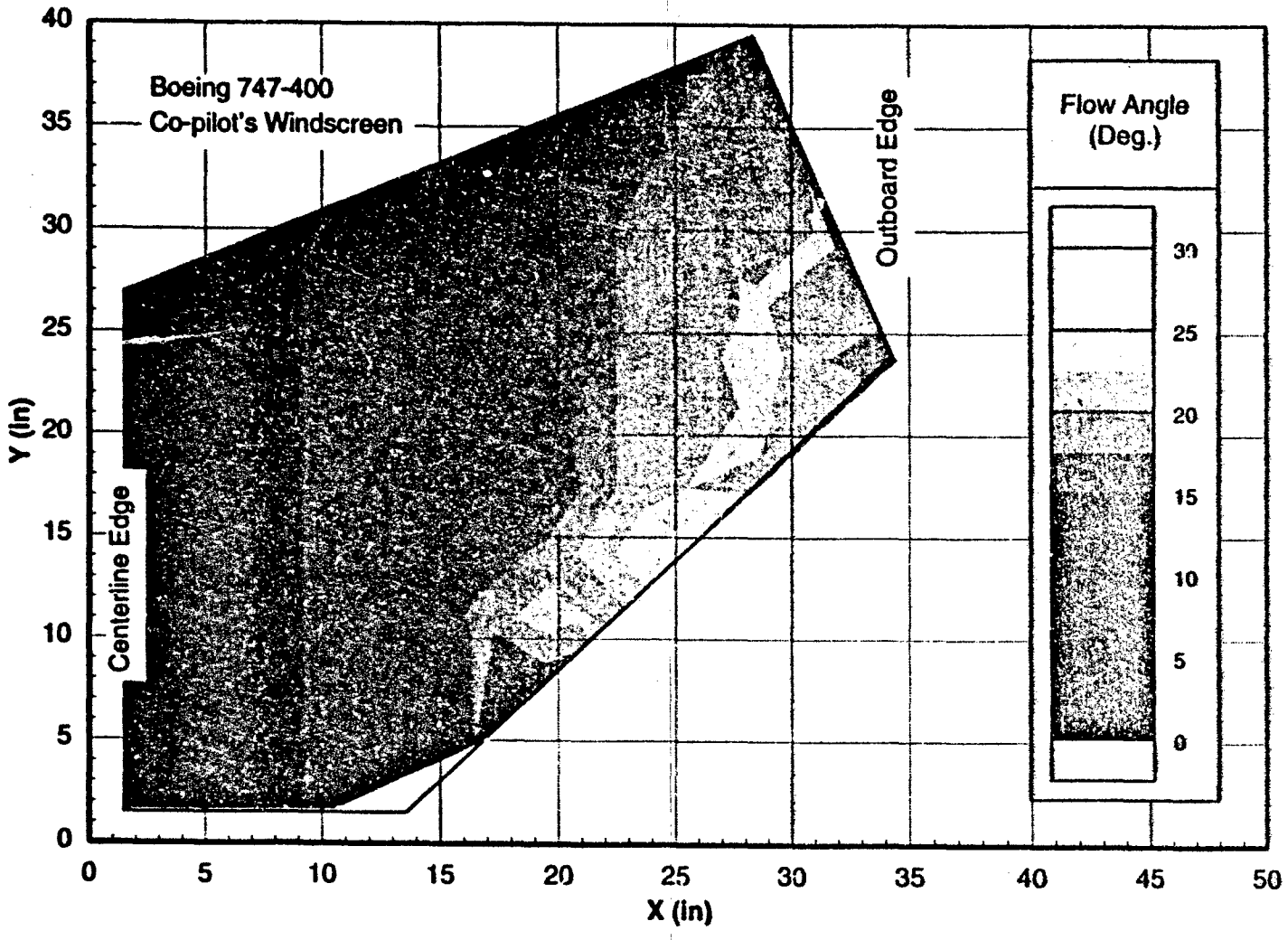


Figure 16. 747-400 Windscreen Flow Angle Distribution

HYPERVELOCITY IMPACT DAMAGE TOLERANCE OF FUSED SILICA GLASS

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Hypervelocity Impact Damage Tolerance* of Fused Silica Glass

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Abstract

A test program was conducted at the NASA/Johnson Space Center (JSC) concerning hypervelocity impact damage in fused silica glass. The objectives of this test program were: to expand the penetration equation data base in the velocity range between 2 and 8 km/s; to determine how much strength remains in a glass pane that has sustained known impact damage; and to develop a relationship between crater measurements and residual strength predictions that can be utilized in the Space Shuttle and Space Station programs. The results and conclusions of the residual strength testing are discussed below. Detailed discussion of the penetration equation studies will follow in future presentations.

Nomenclature

P_c	Crater depth, penetration depth
d_p	Projectile diameter
V_p	Projectile Velocity
ρ_p	Projectile Density
D_1	Front spall diameter
K_{IC}	Critical Stress Intensity
K_t	Stress concentration factor
a	Flaw size (depth)
a*	Flaw size computed from model
S	Residual strength
S*	Residual strength predicted by model
HVI	Hypervelocity impact

Background

In the decade since the Shuttle's first flight, four vehicles have together flown more than forty times. The Shuttle vehicle successfully performs its various missions because of its versatile design. The Shuttle windows, specifically the outermost pane, must accommodate aeronautic loads (3-5 psi burst), reentry heating (1200°F), and orbital operations (-135°F < T < 350°F; debris and micrometeoroid environment). The window system

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must safely retain cabin pressure and provide operational and recreational viewing during all parts of a Shuttle mission.

There are eleven windows in the Orbiter Crew Module/Forward Fuselage area. Nine of these are sets of three glass panes, mounted in stiff frames in the vehicle structure (the other two, between the crew module and the payload bay, have only two panes of glass). In the sets containing three panes, the outer window, or Thermal Pane, is a structural element of the forward fuselage and is made of fused silica (Corning 7940). After every flight, the Thermal Panes are inspected in detail to detect any surface damage which might cause the pane's failure during the next ascent. These inspections detect one or more damage pits on almost every flight. The dimensions of the pit are measured using a mold impression technique and a profilometer; these measurements are then evaluated by stress analysts who determine if the window pane's residual strength and remaining life are adequate for continued operation. About half of the damages result in window removal.

The procedures and specifications used in inspecting the Orbiter windows are derived from the original window specification provided to Corning Glass Works. In this specification, the maximum depth of a surface flaw allowed in the fused silica (as delivered) is .0006". The maximum flaw size in each pane is verified with a flaw screening test, also called a proof test, at the vendor. This .0006" flaw size was chosen after considering the desired window strength, its life, handling damage potential, and the inspection methods practical for this vehicle.

In designing the window and determining an appropriate initial strength, certain assumptions are made regarding the flaw size and shape. As is done industry wide in glass manufacturing, a factor of 3 is used on the size of the last grind particle to determine the depth of surface flaws after the final grind.[1] The same conservatisms are used in analyzing the surface damages found during inspections. The flaw depth measured after a flight is multiplied by 3, and the life analysis is performed using this value. Other conservatisms built into this analysis are 3-sigma flaw growth properties and conservative wind gust loads in the load-time profile. The result of this flight experience and analysis method is the replacement rate shown in Figure 1.

It is always in the interest of the Shuttle program to minimize costs when changes can be made safely. The multiple conservatisms inherent in the window system maintenance compel investigation into more reasonable maintenance techniques and replacement criteria. The Space Station Freedom program, NASA's biggest development effort of the decade, can also benefit from increased understanding of fused silica's response to hypervelocity impact; fused silica is the material being used in all Freedom windows.

The test was performed to resolve the replacement criteria issues primarily, and also to provide a larger penetration equation data base. Details of the test procedures and results follow.

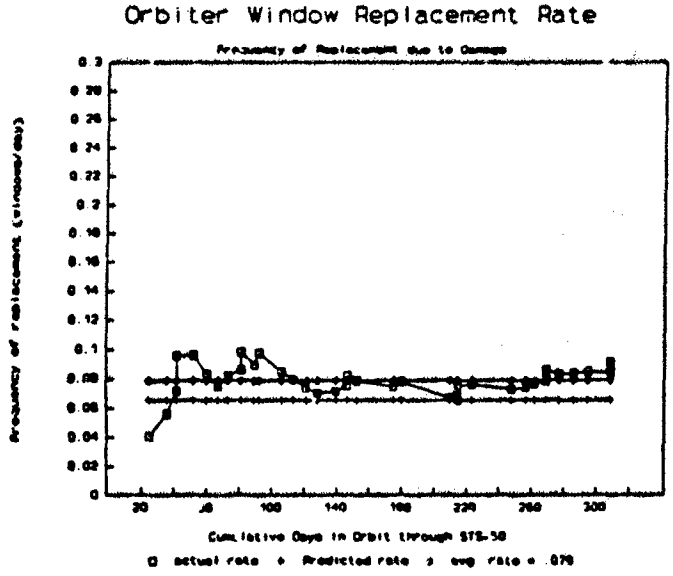


Figure 1

Test Procedures

Using six fused silica panes scrapped from the Shuttle program (damaged thermal panes), target disks were prepared at JSC. One disk from each scrapped thermal pane was cut around the pre-existing damage which had caused the pane's removal. The selection of the scrapped panes included some consideration of the damage location and relative size. In general, the panes with the largest damages were selected. One window with a pit considered a micrometeoroid damage was specifically selected to investigate the hypothesis of the source. The flight-damaged disks were delivered to a Scanning Electron Microscope (SEM) laboratory for analysis.

The remaining fused silica was cut into about twenty disks per window and, less two for strength testing control purposes, these disks were impacted at JSC's Hypervelocity Impact Research Lab (HIRL) with a variety of projectiles. The first set of targets were subjected to impacts with aluminum spheres, varying in diameter from .4 mm to 1.25 mm, and in velocity from 3 km/s to 8 km/s. Further testing included spherical garnet and glass projectiles used to vary projectile density, oblique incidence angles, and thicker target disks. Impact velocity was consistently within the range of 3 - 8 km/s; a majority of shots were done at approximately 6 km/s.

Once the disks were damaged, the craters were measured at a JSC materials lab, using the same technique that Kennedy Space Center technicians use on the Shuttle windows. This involves

taking a mold impression of the crater, and then measuring the mold dimensions under an optical microscope. The crater diameter was also measured directly on the target disk using a digital micrometer and magnification. This process had to be used because often the surface spall was not ejected, and the crater diameter was not discernable from a mold.

Following the pit measurement, the damaged disks were fractured in JSC's Structural Test Laboratory. After the first set of disks were broken, additional disk preparation procedures were added to the test program. These new procedures were devised to provide a fracture surface with a minimum of sub-critical crack growth. The process involved drying the remaining disks in a 250°F oven, then bagging them in a vacuum to prevent any moisture contact with the crack surfaces during the strength testing.

Further changes were made as the test program progressed. After an initial set of shots and disk fractures (27 total), the experiment procedures were reviewed. A new test matrix was designed using fractional factorial techniques to compare the effect of different parameters on the penetration depth, aspect ratio, and residual strength. The parameters studied were projectile velocity, diameter, and density, target thickness and obliquity of impact. After these sixteen tests were completed, eighteen more tests were performed to specifically focus on the parameters which influenced the penetration depth, the crater aspect ratio and the strength the most.

The strength data, pit dimensions and fracture analysis comprise the results of this test program. This will be described in the following sections of the article. The discussion will focus on the residual strength element of the test program, and will include penetration depth and crater aspect ratio results as they pertain to the strength issue.

Data

Penetration Depth

The depth of the damages were measured using a mold yim-pression. The molds were formed and then measured under an optical microscope. The "highest" peak of the mold was taken as the maximum depth of the crater (P_c). Figure 2 shows an example of depth vs. velocity for the Al 2017, .397 mm projectile.

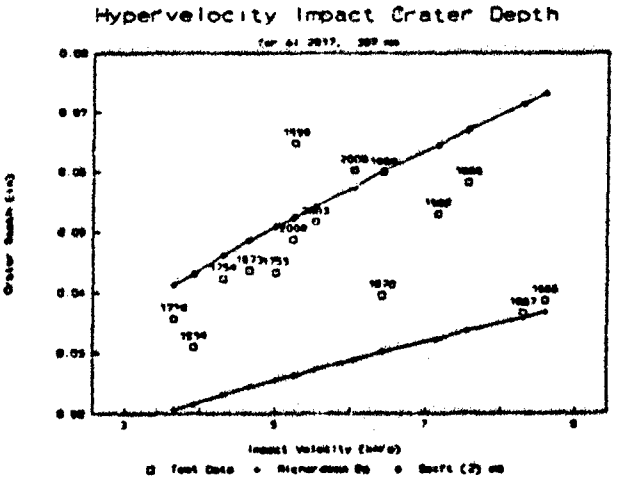


Figure 2

Crater Aspect Ratio

The front spall diameter was measured using a digital micrometer and a magnifying lens. Two measurements were taken at 90° to each other and then averaged to arrive at an approximate diameter. Figure 3 identifies the different diameters discussed in this paper.

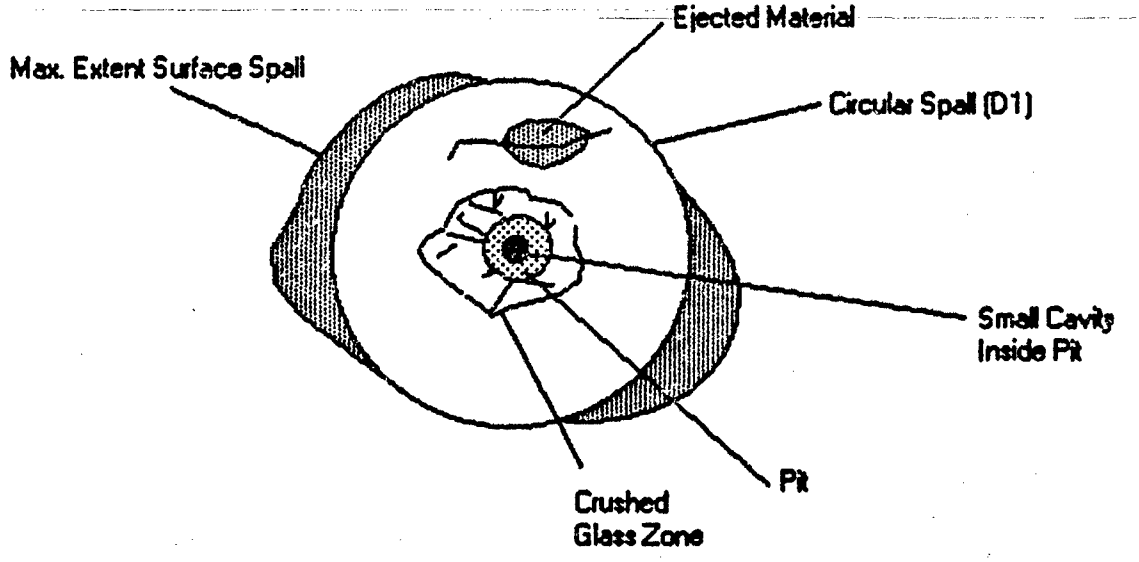


Figure 3

Several different aspect ratios were investigated in this study. One which showed a very interesting trend was the ratio

of front spall diameter (D_1) to projectile diameter (D_p). When plotted, as in Figure 4, against the energy of impact, a hyperbole can be drawn through the data.

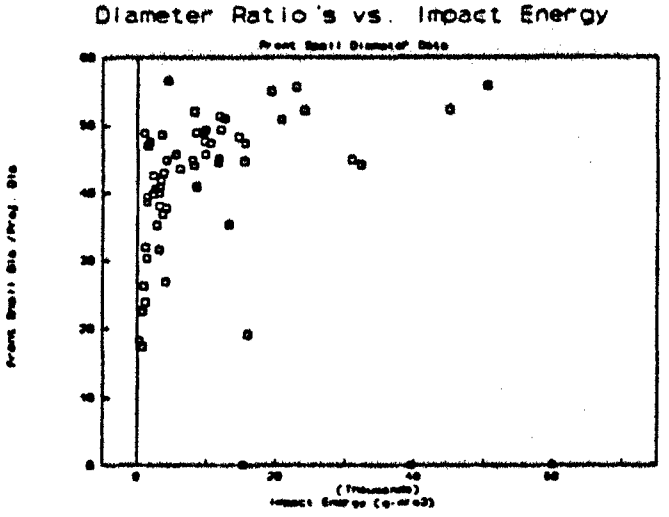


Figure 4

Another ratio that was examined was the crater diameter (D_1) to the crater depth (P_c) measured by mold impression. Much information was obtained from plotting this ratio versus different impact parameters. In one set of experiments, the projectile density and velocity were held constant while the projectile diameter was varied. Figure 5 shows the results.

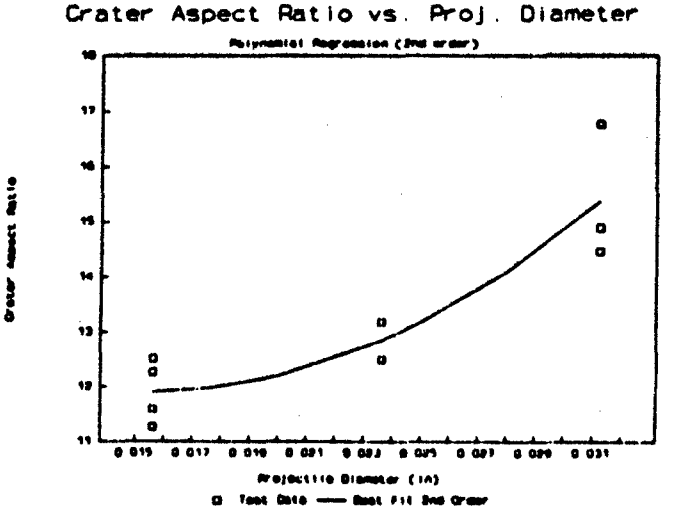


Figure 5

Residual Strength

The residual strength of the damaged disks and the control disks was measured using a concentric ring bending fixture in an Instron (see Figure 6). The concentric ring test configuration allowed any cracks beneath the damage pits to be randomly oriented, since the biaxial tension stress created by the load is a constant inside the diameter of the inner ring. Therefore, the strength-controlling flaw in each glass disk could grow to a crack in any direction.

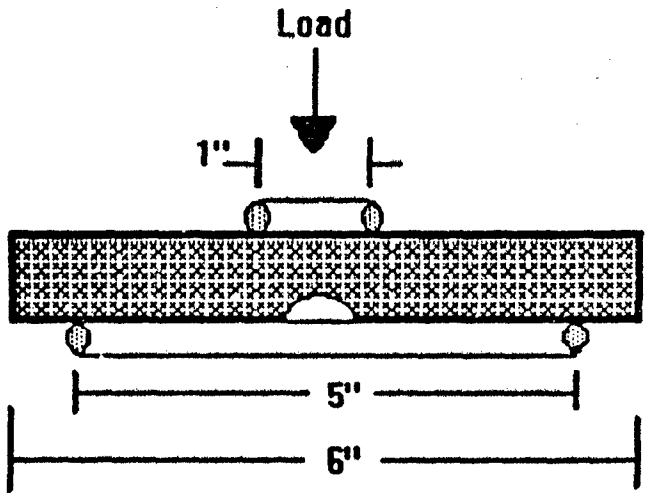


Figure 6

To verify the original condition of the window surface, the strength of undamaged disks was also tested. The nine undamaged disks tested had an average strength of 10.5 ksi, with a standard deviation of 1.5 ksi. The strength of these windows at the beginning of their life is a minimum of 8.6 ksi, as established by a flaw screening test at the vendor.

To establish a comparison between orbital experience and the test, the flight-damaged disks were fractured in this fixture. Figure 7 shows the residual strength of each flight damaged disk and the depth of the damages. (One flight damaged disk was broken accidentally).

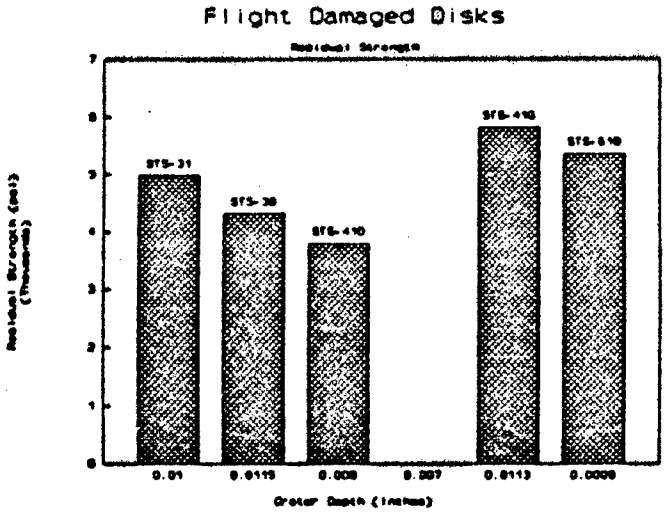


Figure 7

The complete data set from this test program can be found in Appendix A.

Analysis

Strength Prediction

The motivation for this study can be clearly illustrated by comparing the strength predicted by the Shuttle program's current techniques with the actual strength demonstrated in the lab. Early in this effort, when the first few HVI disks were fractured, the disks were 2 to 2.5 times stronger than the strength predicted by the Shuttle program technique.

For this comparison, the predicted strength was calculated using two different equations. The first and simplest, was a straightforward fracture equation,[1] with the factor of 3 on the crater depth. The other technique included subcritical crack growth in the computation of residual strength. This was done using a computer program that computes the amount of subcritical crack growth in a static fatigue condition by iterating the time under load and the crack velocity until a critical crack size is reached. The factor of 3 on the crater depth was also used in this technique to determine flaw depth. This most closely approximates the strength that would be calculated by the Shuttle's prime contractor for a flight-damaged window.

Hypervelocity Impact

The physics behind creation of the crater during hypervelocity impact are well understood for both ductile and brittle

materials. In a brittle material like glass, the dynamic load of any impact generates compressive, shear and Rayleigh surface waves. The tensile components of the surface wave extend pre-existing surface defects, initiating fracture at all of the defects above a critical size.

A hypervelocity impact, which is limited on the low end by the speed of sound in the target material, produces shock waves in both the projectile and the target. The peak stresses at the tips of the shock waves significantly exceed the material's strength, causing gross deformation, with the material behaving like a compressible fluid.[2] When the shock front meets a free surface before its energy is sufficiently depleted, surface spallation will occur. The shock waves are reflected as tensile waves, in order to meet the boundary condition of zero pressure at a free surface. Because of glass's brittle nature, these tensile stresses dominate the material strength over a relatively long period of time, resulting in more extensive damage compared to a ductile target subjected to the same impact.[3]

Crack extension in hypervelocity impact must take place. Evidence from an earlier study indicates that this crack extension might be negated by the loss of the cracked material in a spall product.[4] It can be assumed that some cracks exist in the material beyond the crater. What is not well understood is what kind of cracks these are and how they will effect the strength of the glass.

Previous studies of the residual strength of glass after impact have focused on "low" velocity damages. In one effort[5] a critical impact velocity was found above which the residual strength dropped significantly. This testing was performed at velocities between 0 and .7 km/s, so does not relate to hypervelocity impact ($V > 5$ km/s). However, another result of this effort found that at "high" ($V \approx .7$ km/s) velocity, the average fracture stress was a weak function of impact velocity. This result was corroborated in the present study of hypervelocity impact.

As described in the test procedures section, certain segments of the test program were designed to focus on specific issues: what are the strongest factors, among the impact parameters, in determining residual strength; and once those were found, what was the quantitative effect of a variation in those influential parameters.

In the first such test segment, the projectile diameter was found to have the strongest influence on residual strength. The next strongest effect was projectile mass. With this information, two more test segments were designed, where diameter and density were varied at three different levels. Regression analysis was performed on this data.

Finally, the entire database was used to define a useful relationship between residual strength and any crater parameters. Since crater depth and crater diameter are easily measured crater characteristics, while impact parameters such as projectile velocity, diameter and density must be derived from crater measurements, the relationship sought was one which would derive directly from crater measurements.

The technique used in the Shuttle program, as described in the Background section above, assumes that the crater depth itself is the "measurable surface flaw." This value is multiplied by a factor of three, as is done with glass manufacturing flaws, and the resulting "flaw depth" is used in a static fatigue analysis to predict remaining life and residual strength. The fracture relationship used in this calculation is given in equation (1):

$$\sigma = \frac{K_{IC}}{1.12\sqrt{\pi a}} \qquad (1)$$

where $a = 3P_c$.[1]

Clearly, the impact pit itself is not the strength controlling crack. A crack exists near the crater that causes the fracture of the glass at the load found in the test program. By inverting equation (1), the flaw size can be calculated using the known fracture strength and the appropriate geometry factors for the crater.

In equation (1), the flaw described is a penny-shaped crack in a semi-infinite medium. This description would certainly fit a manufacturing flaw in a glass plate, but loses its relevance when applied to the pit created by a hypervelocity impact. In research related to hypervelocity impact damage[6], a stress concentration factor was found for a hemispherical pit at a free surface in a state of plane hydrostatic tension: $K_t = 2.23$. Also related to hypervelocity impact damage, additional research[7] found that K_t is approximately 3.8 for a 45° hyperboloid cavity and 3.5 for a 60° hyperboloid cavity in a thin plate under hydrostatic tension.

To derive the size of the critical flaw causing failure in the hypervelocity impact test specimens in this test program, factors such as the above should be used, compounded with the geometry factors inherent in the fracture equation (1).

Before applying these factors to the fracture equation, however, the loading condition of the test plates was considered. These plates were fractured in bending, in a concentric ring apparatus. The two research efforts described above calculated geometry factors for plates in hydrostatic tension. Another source was found which addressed the case of bending in a plate, but examined the geometry of a deep hyperboli notch, not a central pit.[8] For the dimensions found in the test specimens in

this study, the stress concentration factor derived from this source would not exceed 3.6. For the purposes of this effort, the hydrostatic tension cases described above were used, with a conservative value of K_t=4.0 (by plotting K_t vs. hyperboloid angle given in ref. 7, it can be shown that K_t approaches 4 as the angle approaches 0°).

Additional consideration is due to the scale of the damages. For the test cases, where the hypervelocity pits are created in the lab, the smallest crater had a depth of .02 inches. Compare this to the six flight-damaged disks tested, where the largest pit had a depth of .012 inches and the smallest was .0009 inches deep. In analyzing the fracture of these disks, the thin plate assumption was used for the test disks (K_t=4), while the free surface model was used for the flight-damaged disks (K_t=2.23). The fracture equation is now:

$$\sigma_{RES} = \frac{K_{IC}}{1.12 K_t \sqrt{\pi a}} \qquad (2)$$

Now the strength controlling flaw size can be found from the failure strength.

As described earlier in this paper, most of the specimens were dried and vacuum bagged prior to the strength testing, to prevent sub-critical crack growth. In this part of the analysis, only the data from the bagged specimens was used, since some sub-critical crack growth presumably occurred in the remaining unbagged disks prior to their final failure. The strength data from the bagged disks was used with equation (2) to compute a flaw depth.

The flaw depth data were used in a linear regression analysis with the crater depth measurements and a relationship between them, in equation (3) was determined. This relationship is shown in Figure 8.

$$a(p_c) = .001608 + .01063 p_c \qquad (3)$$

Figure 9 shows several curves and data points. The ▲ indicates a tested value. The lowest curve, labeled "fatigue model," is the strength that would be predicted by the current Shuttle Program techniques. The next curve, "fracture model," is a value that would be calculated if no static fatigue was assumed. The curve labeled "S*", running through the mean of the test data, reflects the fracture strength based on a best-fit curve for flaw depth as a function of crater depth (see equation (3)).

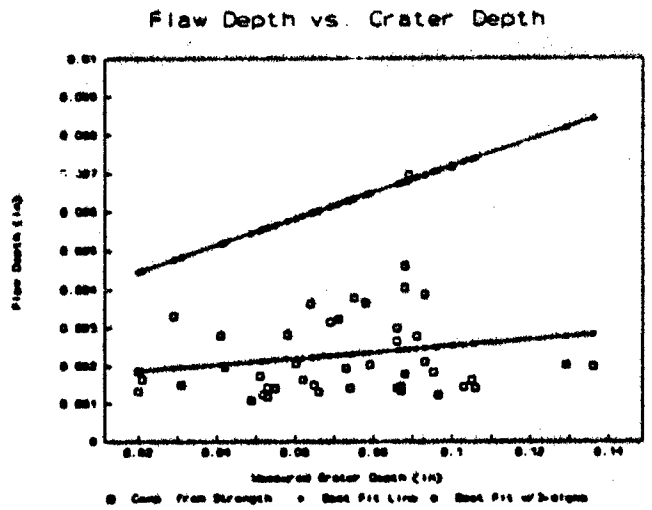


Figure 8

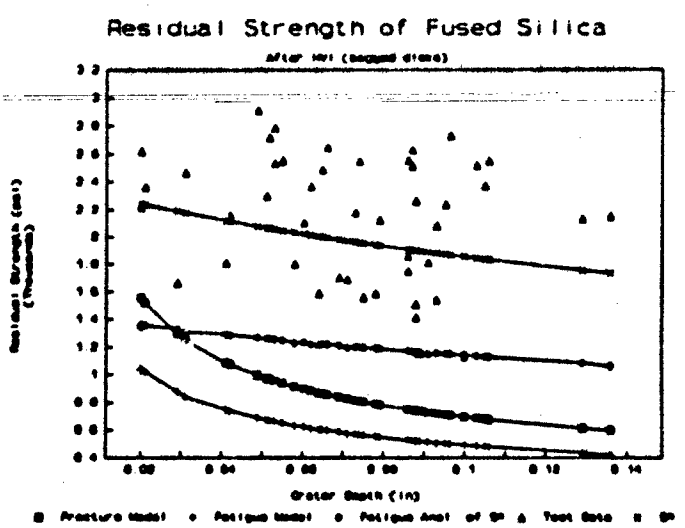


Figure 9

Equation (3) was used to calculate flaw depth a*, which was then used in a static fatigue analysis to predict residual strength, shown in Figure 9 as the curve labeled "Fatigue Anal. of S*."

Figure 9 shows that the strength of the test disks are bounded by the predicted values using a* and static fatigue analysis. Some subcritical crack growth probably occurred in this test program, even though the disks were dried and vacuum-

bagged, so the static fatigue curve for a* should be a conservative predictor for the bagged specimens' strength. This method was next tested against the unbagged specimens of the test program, using the curve of equation (3) to predict flaw depth, as shown in Figure 10.

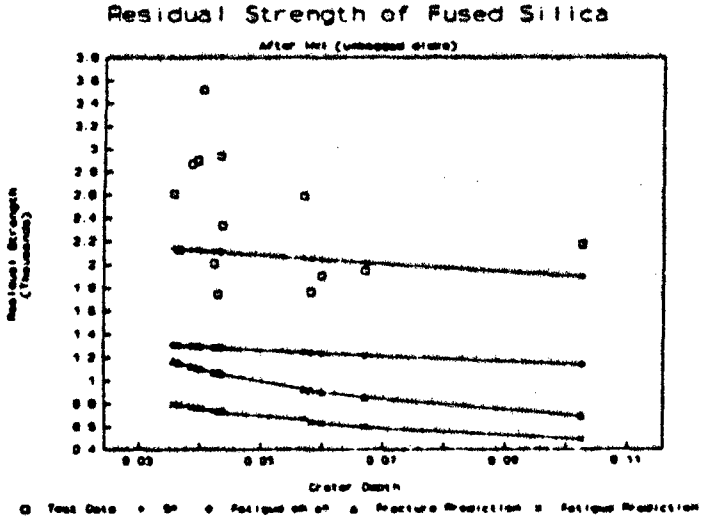


Figure 10

Finally, this technique was evaluated using the flight-damaged disks. It was immediately apparent that the difference in scale between the craters on the test disks and the flight damaged disks would make application of equation (3) impossible. The crater depth for the STS-61B damage is only .0009", while the smallest flaw depth predicted by equation (3) is .0016". The result of using this relationship would be extremely conservative predicted strengths for the very small flaws found on the Space Shuttle windows. To resolve this problem, the strength data from the vacuum-bagged flight disks were included in the derivation of the flaw depth equation.

A new equation was derived from this data:

$$a(p_c) = .000845 + .022264 p_c \quad (4)$$

This curve is compared with equation (3) in Figure 11:

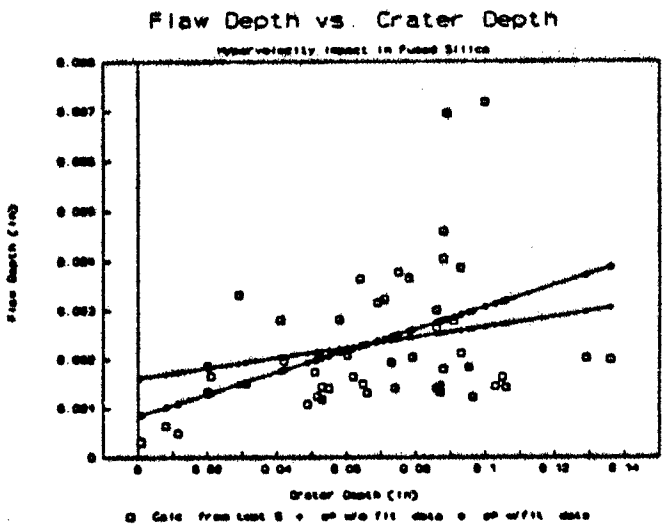


Figure 11

The strength predicted from this equation is shown in Figure 12:

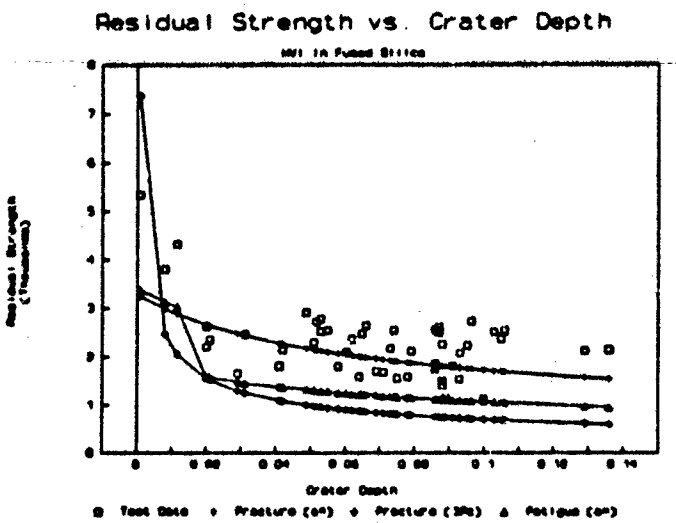


Figure 12

The curve showing estimated fatigue strength from a* has a "jog" in it for the smallest crater depths. This "jog" is attributed to the different K_t used for flight damaged disks, because the craters are so small with respect to the thickness of the plate (the free surface assumption is used here, rather than the thin plate assumption as discussed in references 6 and 7). This figure demonstrates that the test data is bounded by the fatigue curve.

This technique was evaluated by using the unbagged test specimens and flight damaged disks:

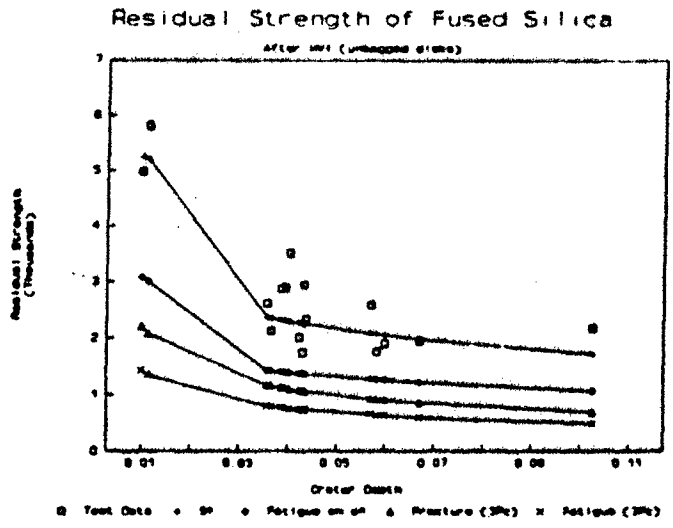


Figure 13

The critical aspect of this graph is the fact that the curve "fatigue on a*" lies below all of the test data strength values. A conservative predictor of window strength has been derived.

Conclusions

The strength of a window pane damaged during orbital operations can be conservatively predicted by following the procedures outlined below:

(1) Accurately measure the depth of the impact.
(2) Predict the size of the strength controlling flaw by using equation (4).
(3) Use a conservative fatigue analysis to calculate the residual strength and life of the Shuttle window; an additional stress concentration factor of 2.23 is used in the computation of K_{IC} for this analysis.

This analysis should result in a strength value for the window that is conservative, ie. includes a margin of safety over the actual value, but that will not be as conservative as the current techniques. This new technique will be recommended to the Shuttle Program as a change to current maintenance and failure analysis. Further testing will probably be done to establish a high degree of confidence in this method. Such testing should include more exploration into issues of scale and stress concentration factors.

This research effort has resulted in a great deal of data and analysis, a small part of which is presented here. Future papers will address the penetration equation, the possible relationships between crater aspect ratio and impact energy, and will further explore crack growth during hypervelocity impact.

Acknowledgements

The author wishes to acknowledge the employees of the NASA Hypervelocity Impact Research Laboratory, the Scanning Electron Microscopy Laboratory, the Structures Laboratory and the Materials Laboratory, all at the Johnson Space Center.

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8. Peterson, Rudolph E., Stress Concentration Factors, John Wiley & Sons, New York, 1974. Figure 33.

Appendix A

Shot #	Material	ρ_p (g/cm^3)	D_p (mm)	V_p (km/s)	Angle (deg)	rear spall	P_c (in)	D_1 (in)	σ_{res} (psi)	Load (lb)
1888	Al 2017	2.77	0.395	7.54	45	no	0.02	0.547	2217.18	968.20
1909	Al 2017	2.77	0.395	3.98	45	no	0.02	0.272	2619.74	754.10
1912	Garnet	3.8	0.395	2.47	0	no	0.021	0.285	2360.24	679.40
1805	Al 2017	2.77	0.794	3.14	0	yes	0.029	1.15	1663.35	478.8
1914	Al 2017	2.77	0.395	3.92	0	no	0.031	0.352	2464.04	1076.0
1812	Al 2017	2.77	0.6	7.3	0	yes	0.041	0.967	1808.56	520.60
1814	Al 2017	2.77	0.6	7.03	0	yes	0.042	1.06	2156.66	620.80
2008	Al 2017	2.77	0.395	5.24	0	no	0.0488	0.611	2910.17	837.70
1813	Al 2017	2.77	0.6	7.28	0	yes	0.051	1.23	2297.36	661.30
2003	Al 2017	2.77	0.395	5.53	0	no	0.0517	0.6	2716.32	781.90
1902	Al 2017	2.77	0.395	7.16	0	no	0.053	0.63	2529.77	728.20
1816	Al 2017	2.77	0.6	4.53	0	yes	0.053	0.898	2781.63	800.70
1970	Al 2017	2.77	0.794	6.25	0	yes	0.055	1.51	2553.04	734.90
1905	Al 2017	2.77	1	4.24	45	yes	0.058	1.39	1801.77	786.80
2006	Al 2017	2.77	0.395	6.04	0	no	0.0603	0.74	2103.51	605.5
1918	Garnet	3.8	0.395	3.91	45	no	0.062	0.408	2363.28	1032.0
A1481	Garnet	3.8	1	3.85	0	yes	0.064	1.76	1587.89	693.40
1998	Al 2017	2.77	0.395	5.26	0	no	0.0648	0.73	2481.13	714.20
1883	Garnet	3.8	0.395	7.14	45	no	0.066	0.756	2642.67	760.70
1803	Al 2017	2.77	0.794	4.04	0	yes	0.069	1.36	1704.34	490.60
1899	Al 2017	2.77	1	6.6	0	yes	0.071	1.77	1685.93	736.20
1820	Al 2017	2.77	0.6	4.68	0	yes	0.073	0.99	2176.46	626.30
1806	Al 2017	2.77	0.794	2.91	0	yes	0.074	0.986	2548.53	733.6
1977	Garnet	3.8	0.794	6.6	0	yes	0.075	1.74	1557.74	448.40

Shot #	Material	ρ_p (g/cm^3)	D_p (mm)	V_p (km/s)	Angle (deg)	rear spall	P_c (in)	D_1 (in)	σ_{res} (psi)	Load (lb)
1819	Al 2017	2.77	0.6	5.17	0	yes	0.078	1.06	1584.49	456.10
1973	Al 2017	2.77	0.6	7.24	0	yes	0.079	1.04	2122.27	610.90
1904	Al 2017	2.77	1	3.64	0	yes	0.086	1.8	1748.81	503.40
1795	Al 2017	2.77	0.794	5.52	0	yes	0.086	1.39	1859.98	535.40
1974	Al 2017	2.77	0.6	5.94	0	yes	0.086	1.08	2552.00	734.60
1802	Al 2017	2.77	0.794	5.03	0	yes	0.087	1.49	2507.19	721.7
1980	Garnet	3.8	0.794	6.29	0	yes	0.087	1.59	2624.95	755.60
A1478	Garnet	3.8	1	6.97	45	yes	0.088	2.2	1412.47	616.80
1933	Garnet	3.8	1	3.92	45	yes	0.088	0.758	1506.67	433.70
1967	Al 2017	2.77	0.794	6.48	0	yes	0.088	1.48	2260.88	650.80
1981	Garnet	3.8	0.794	6.68	0	yes	0.089	1.63	1148.85	330.70
1922	Garnet	3.8	0.395	8.25	0	yes	0.091	0.88	1814.60	792.40
1978	Garnet	3.8	0.794	6.06	0	yes	0.093	1.72	1540.37	443.40
2060	glass	2.3	0.756	6.2	0	yes	0.0931	1.47	2080.23	598.80
2061	glass	2.3	0.756	6.68	0	yes	0.0953	1.47	2235.87	643.60
2059	glass	2.3	0.756	6.89	0	yes	0.0965	1.52	2728.13	785.30
A1477	Garnet	3.8	1	6.58	0	yes	0.1	2.06	1130.79	325.50
1971	Al 2017	2.77	0.794	4.79	0	yes	0.103	1.53	2514.83	723.90
1903	Al 2017	2.77	1	6.67	45	yes	0.105	1.74	2369.96	682.20
1968	Al 2017	2.77	0.794	5.03	0	yes	0.106	1.53	2544.36	732.40
A1479	Garnet	3.8	1	3.84	0	yes	0.129	0	2127.83	612.50
1896	Garnet	3.8	1	6.17	0	yes	0.136	0	2147.56	937.80
1800	Al 2017	2.77	0.794	3.31	0	yes	0.048	1.18	0.00	N/A
2064	glass	2.3	0.756	6.24	0	yes	0.0874	1.53	2483.91	715.00
1758	Al 2017	2.77	0.397	3.65	0	no	0.0357	0.352	2608.97	751.00
1667	Al 2017	2.77	0.397	8.3	0	yes	0.0366	0.671	2126.78	612.2
1666	Al 2017	2.77	0.397	8.6	0	yes	0.0386	0.421	2869.18	825.9

Shot #	Mate-rial	ρ_p (g/cm^3)	D_p (mm)	V_p (km/s)	Angle (deg)	rear spall	P_c (in)	D_1 (in)	σ_{res} (psi)	Load (lb)
1670	Al 2017	2.77	0.397	6.42	0	yes	0.0395	0.622	2900.44	834.9
1764	Al 2017	2.77	0.397	4.65	60	no	0.0403	0.374	3515.69	1012.0
1754	Al 2017	2.77	0.397	4.3	0	no	0.0423	0.764	2009.71	578.50
1795	Al 2017	2.77	0.794	5.62	0	yes	0.043	1.41	1743.95	502.00
1755	Al 2017	2.77	0.397	5	0	yes	0.0433	0.474	2935.53	845.00
1673	Al 2017	2.77	0.397	4.65	0	no	0.0436	0.498	2330.71	670.9
1762	Al 2017	2.77	0.397	7.61	45	no	0.057	0.638	2584.66	744.00
1668	Al 2017	2.77	0.397	7.56	0	yes	0.0582	0.624	1764.10	507.8
1669	Al 2017	2.77	0.397	6.43	0	yes	0.0599	0.665	1901.32	547.3
1799	Al 2017	2.77	0.794	5.31	0	yes	0.067	1.48	1946.48	560.30
1767	Al 2017	2.77	1.25	6.48	0	yes	0.1025		2169.17	624.40

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8. Peterson, Rudolph E., Stress Concentration Factors, John Wiley & Sons, New York, 1974. Figure 33.

FATIGUE ANALYSIS OF POLYCARBONATE TRANSPARENCIES

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ABSTRACT

There are complaints of fatigue failures of the canopy transparencies in flights, and, therefore, there is a need to improve the canopy design and prolong its fatigue life and reliability.

The study described in the paper is devoted to this goal and includes three parts:

1. Investigation of fatigue behavior of structural polycarbonate plies on the basis of the developed long-term test procedure. Fatigue characteristics have been determined for the coupons manufactured from plies with different thickness, for the solid coupons and for the specimens with stress concentrators.
2. Accelerated investigation of fatigue parameters of the coupons from structural polycarbonate. The accelerated fatigue test procedure has been developed and permits to control a stability of the manufacturing process of the polycarbonate sheets and enables quick preliminary estimates of the design changes.
3. Investigation of fatigue failures of coupons from polycarbonate transparencies in different environmental conditions. The combination of different temperatures, weathering, and chemical actions can significantly change polycarbonate fatigue crack propagation parameters. This part of the project is in progress but some results are presented for discussion.

The paper partially combines the data published in [1,2] with the addition of temperature exposure fatigue investigations.

INTRODUCTION

There are complaints of F-16 transparency failures in flight [3,4,5]. The nature of failures is not quite clear, but some evidence implies that transparency life is limited by fatigue. At present, no experimental fatigue statistics on the composite material used for F-16 transparency were found. Therefore, today no data exists that allows life prediction for a canopy based on fatigue or crack propagation, and thus no basis for comparison of new materials and designs taking into consideration their resistance to fatigue and crack propagation.

The F-16 transparency is manufactured from a laminated composite material. Components of the composite are an acrylic face ply, a polycarbonate ply, interlayers, and coatings with some variations among the vendors. Design of the canopy allows to unload an acrylic ply and that is why the structural polycarbonate ply of the composite was the primary concern during fatigue investigation in this project.

Note that a long-term fatigue test procedure requires the breaking of 20 to 30 identically prepared specimens and 15 days to one month to complete. Thus, manufacturers do not perform a conventional fatigue test

in spite of its obvious utility. Therefore, there is a definite need for an accelerated test which can be completed in approximately one shift and the fatigue tests results gained in this study can be useful to verify the precision accelerated test procedure developed.

The project also includes some investigations of fatigue resistance of polycarbonate sheets with different thickness, and study of fatigue life under different load frequency and temperature.

MATERIAL AND SPECIMENS

The material used was supplied by Wright-Patterson Air Force Base. The testing coupons were cut from the 0.5-inch polycarbonate sheets. The sheets were extruded, pressed and polished in accordance with the military specification MIL-P-83310.

The configuration and dimensions of the test specimens are shown in Fig. 1. The specimens were cut by a fine band saw with the lowest possible speed and using a cooling liquid. The holes were also drilled very slowly with intermittent stops and using cooling liquid. The hole edge burrs were not removed to prevent invisible damage. The cut specimens were divided into two groups. In the first group the edges were left as they were after machining. All machined specimens in the second group were polished on the sides and edges by carbimet paper disks for automet attachment. The 8" x 2.5" self-adhesive back disks NO. 30-5158-120, grits 120-180, were manufactured by BUEHLER LTD. The 3/4" core-series 17-0310 scotch tape was used to protect from damage the gripping area of the specimens. The scotch tape was bonded in three layers on each end of the specimen. It took usually about 10 minutes after the beginning of the test to adjust the loading regime assigned due to formation of "bed" by hard jig rollers in the soft tape layers.

EQUIPMENT

The flexure fatigue tests were conducted on an MTS machine using four point MTS flexure system to provide pure bending. Actual flexure fixture and loading diagram are shown in Figures 2 and 3.

Note that load reading on the MTS machine controller is P, in other words the load reading on the machine device doubled comparing with that shown in Figure 3. The appropriate support and load spans were selected to provide minimum possible deflection of the specimens. The small deflections provide more stable position of the specimens between jig rollers and permit to assign greater testing frequency.

The flexure fatigue tests under different temperatures were conducted.

FAILURE CRITERIA

Different options were analyzed prior to the assign a final failure criterion.

1. Certain percentage load drop can be considered as failure during constant stroke and hence controlled specimen deflection testing. This

testing mode permitted crack growth observation and decreased creep influence. But non-uniform and inconsistent load drop registered during preliminary tests did not permit to make a clear test result interpretation. Also constant deflection conditions do not reflect the actual situation.

2. During constant load testing the complete separation or certain percentage crack propagation can be considered as failure and the preliminary testing showed that the resultant scatter was significantly less than in the constant stroke test. The constant load mode also better simulates the actual loading.

3. Crazing and therefore visibility lost definitely can be considered as failures for the canopy material.

4. A minute crack detected by eye can also be considered as a failure because there is no data on how fast an initial crack will propagate in the polycarbonate sheet tested.

Taking this into consideration the decision was made to perform the test under constant load regime and develop S-N diagrams with different failure criteria (complete separation or 80% crack propagation over the specimen width, crazing, minute crack formation).

LONG-TERM TEST PROCEDURE (ROOM TEMPERATURE)

The pulsating bending tests were conducted in the laboratory atmosphere (about 70°F and 50% relative humidity). The tests were run at least at four load levels and four specimens were tested at each load level. The regimes are given in Table 2. The ratio of minimum load over maximum load was 0.2 for all tests. The testing time per day was not more than 10 hours, hence the possible influence of stops was not considered in this project. The tests continued until specified damage were observed but not longer than 10^6 cycles. Regression analysis was used to treat the test results.

LONG-TERM TEST RESULTS AND DAMAGE DESCRIPTION

The test results of 0.5-inch coupons are plotted in Figures 4, 5, and 6. The cracks were always started at the bottom tensile zone of the specimens (Figures 7, 8). In all solid specimens with the exception of two cases the cracks propagated from the edge toward the center of the coupons tested. In all specimens with stress concentrators the cracks propagated from the hole edges toward the specimens' sides. Usually craze (minute crack) spot preceeds the crack formation and propagates ahead of the crack tip and it can be concluded that damage mechanism is very similar to that described in [6] for crack propagation in polysterene under fatigue loading. When the visible separate minute cracks were detected during the high load level testing, the massive craze zones developed after that very fast. It can be noted that the lives of specimens are significant after the massive craze spot formation until complete breakage. For low load levels no massive craze zones were observed. But again in many cases after the initiated crack was easily visible we could detect a substantial number of cycles until complete breakage of the specimen. Comparison of the S-N

diagrams in figures 4, 5, and 6 shows that the stress concentration influence on fatigue life of the tested polycarbonate specimens is significant. The specimens without stress concentration have much longer fatigue life.

Comparison of fatigue lives for the 0.5" and 0.25" thickness specimens tested under the same stress is given in Figure 9. The test results show that the life of the thinner material is less. The breakage of the 0.25-inch specimen occurred almost without any "warning." Complete separation was approached very fast after the crack initiation.

The results of fatigue tests conducted at different load frequencies are given in Table 1. The results show that the properties of the material tested are time dependent. The number of cycles until complete separation strongly depends on frequency, but specimen life in hours was almost the same for the specimens tested at the same load level and different frequencies. It can be taken into consideration to provide optimal usage and efficiency of the polycarbonate parts.

ACCELERATED FATIGUE TEST

It appears that the fatigue properties of polycarbonate sheets vary significantly from sheet to sheet. Therefore, it is important to have a mechanism which permits quality control of polycarbonate sheets, detects deviations in the manufacturing process, and enables preliminary estimates of the design changes. The accelerated procedure proposed in this project is based on Locaty's accelerated method [7] used as prototype. The major objectives of the study was development of a detailed procedure of the accelerated fatigue test for coupons cut from structural polycarbonate sheets.

The Locaty's accelerated fatigue method is based on the concept of cumulative fatigue damage [8] considering $\sum\left(\frac{n_i}{N_i}\right) = 1\text{-}5$, where n_i is the number of cycles which the specimen worked in the specified test regime, and N_i is the number of cycles which the specimen could potentially work in accordance with the fatigue curve received from the long-term fatigue tests of the same type of specimens. The loading program and the treatment of results are presented in Figures 10 and 11. Figure 10 shows three fatigue curves received from a long-term fatigue test. From Figure 10, the magnitudes of

$$\sum\left(\frac{n_i}{N_i}\right)_A \qquad \sum\left(\frac{n_i}{N_i}\right)_B \qquad \sum\left(\frac{n_i}{N_i}\right)_C$$

are determined. With these parameters and the corresponding stresses, we can find the coordinates of the points which result in the curve shown in Figure 11. Now, if according to an accepted hypothesis fatigue strength corresponds to a definite value it is possible to determine the magnitude of fatigue strength.

The long-term fatigue test regimes and results are given in Table 2, and plotted in Fig. 12. The example of program and parameters of the accelerated tests and experimental results are given in Figures 12, 13, 14 and in Tables 3 and 4. The testing lives n_i are taken from an accelerated test program which is given, for example, in Figure 15, and expected lives N_i for the curves A, B, and C (90%, 5%, and 95% probability of survival) are taken from Figure 12. The repeatability of the results is quite reasonable

and, therefore, the sum of relative lives which is determined experimentally can probably be recommended as a basic parameter to confirm whether the specimens tested during the control procedure belong to the entire population. The time of accelerated tests was never more than 6 hours. The failure damage and failure mechanism during long-term and accelerated fatigue tests were similar. The cracks always started at the bottom tensile zones of the specimens. In all specimens with stress concentrators the cracks propagated from the hole edges toward the specimens sides. Usually a minute crack spot preceded the crack formation and propagated ahead of the crack tip during the entire damage process.

TEMPERATURE EFFECTS

This part of the project is in progress and only limited results can be presented. The goal is investigate the flexure fatigue resistance of the 0.5-inch polycarbonate. Coupons exposed to temperature changes in the range from -50°C to +50°C. The test procedure, specimen design, equipment and loading diagram are the same as described for the long-term fatigue test. By this time the study under positive temperatures is completed and results are given in Figures 15, 16, 17.

The influence of increased temperature (Fig. 15, 16) is very strong, and the fatigue lives of the specimens tested are from 2 to 3 times less comparing with the test results under room temperature. It is important to note that complete fractures of the coupons tested under increased temperatures followed almost immediately after crack initiations, therefore, there is no "warning" before the break. The fracture surface topography shows (Fig. 17) that it is brittle failure and it is different from the failure under room temperature.

CONCLUSIONS

1. The 0.5-inch polycarbonate coupons have substantial fatigue life after crack initiation until complete fracture.
2. The stress concentration effect on the fatigue life of the polycarbonate coupons tested is very strong. Therefore, frameless canopies without holes for fasteners can be very promising.
3. The results of the accelerated tests have good repeatability. The procedure can be useful to control the stability of the manufacturing process and for preliminary estimate of new designs.
4. The polycarbonate sheet thickness (from 0.25" to 0.5") has a strong effect on the fatigue life of this material.
5. The fatigue life of the polycarbonate tested is time-dependent. The specimen life in hours does not depend on the testing frequency.
6. The fatigue life of the polycarbonate tested is significantly decreased as the temperature is increased (from 20°C to 50°C).
7. Programmed fatigue investigation is strongly recommended.

ACKNOWLEDGEMENTS

Sponsorship by Flight Dynamics Directorate, Wright Laboratory, Air Force Systems Command, United States Air Force, Wright-Patterson AFB, Air Force of

Scientific Research, Bolling AFB, DC and financial support from Universal Energy Systems, Inc., and Purdue Research Foundation is gratefully acknowledged.

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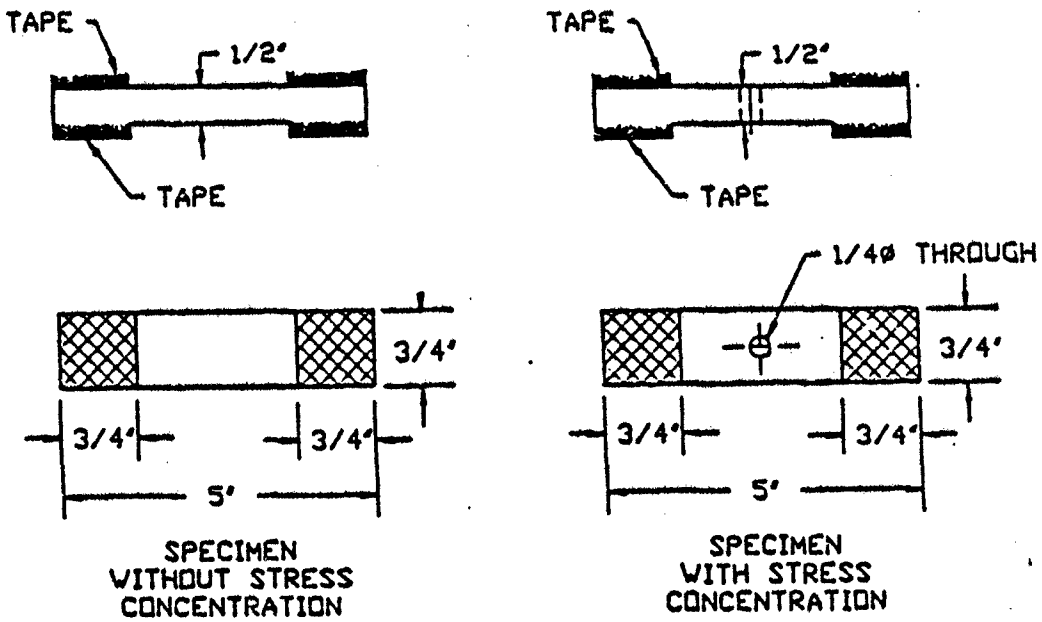


FIGURE 1. Specimens Used in Study

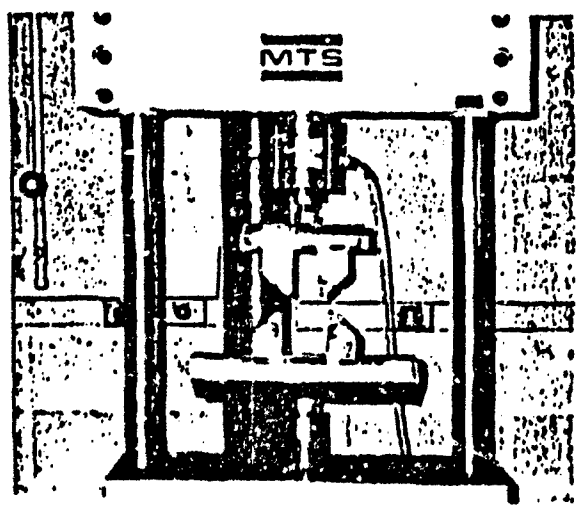
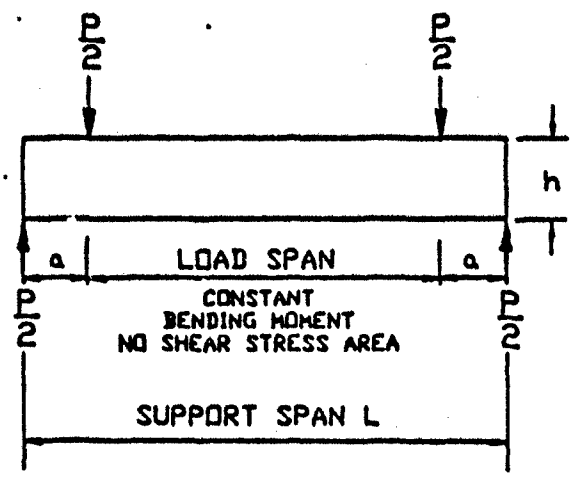


FIGURE 2. MTS Flexture Fatigue Fixture



MAX DEFLECTION:

$$Y_{MAX} = \frac{Pa}{48EI}(4a^2 - 3L^2)$$

MAX STRESS:

$$\sigma_{MAX} = \frac{3P(a)}{bh^2}$$

FIGURE 3. Loading Diagram

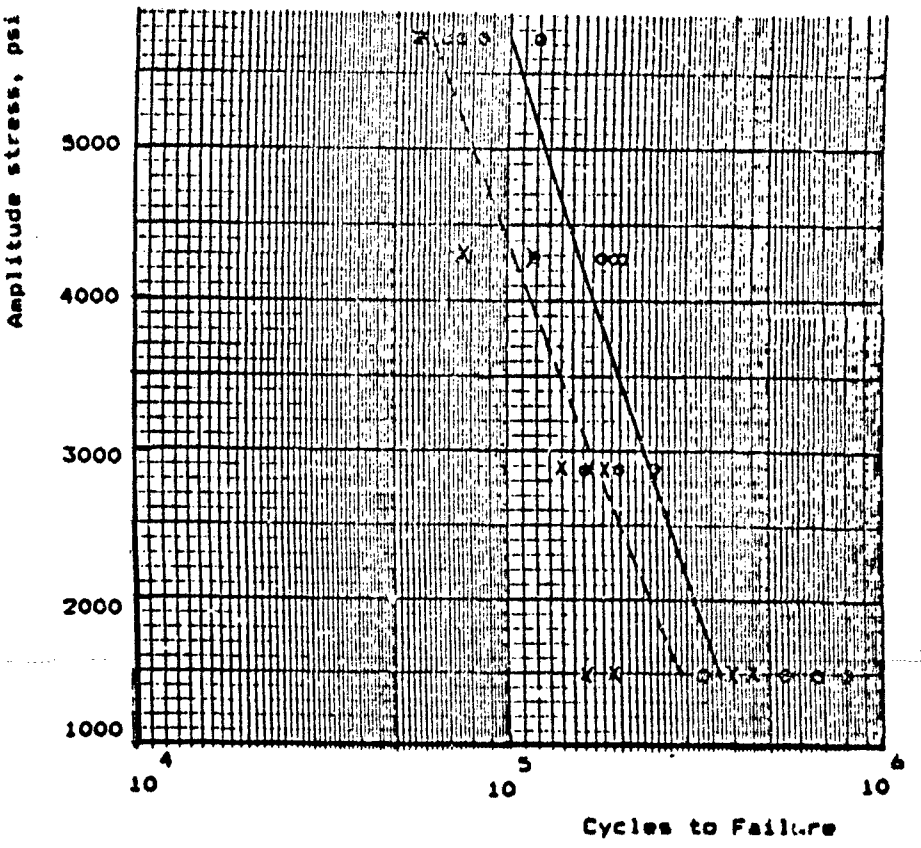


FIGURE 4 S-N Diagram plotted from the results of fatigue test for the polycarbonate specimens with stress concentrators.

O Failure
X Crack initiation

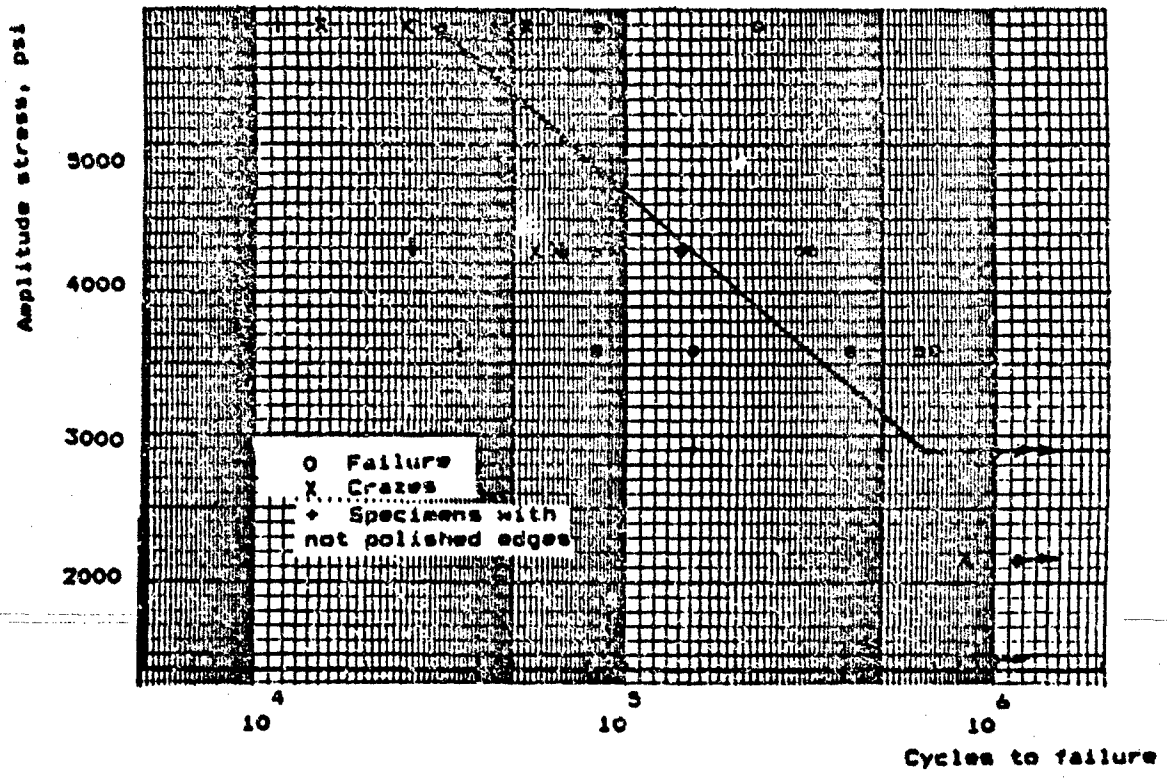


FIGURE 5 S-N Diagram plotted from the results of fatigue test for the solid polycarbonate specimens.

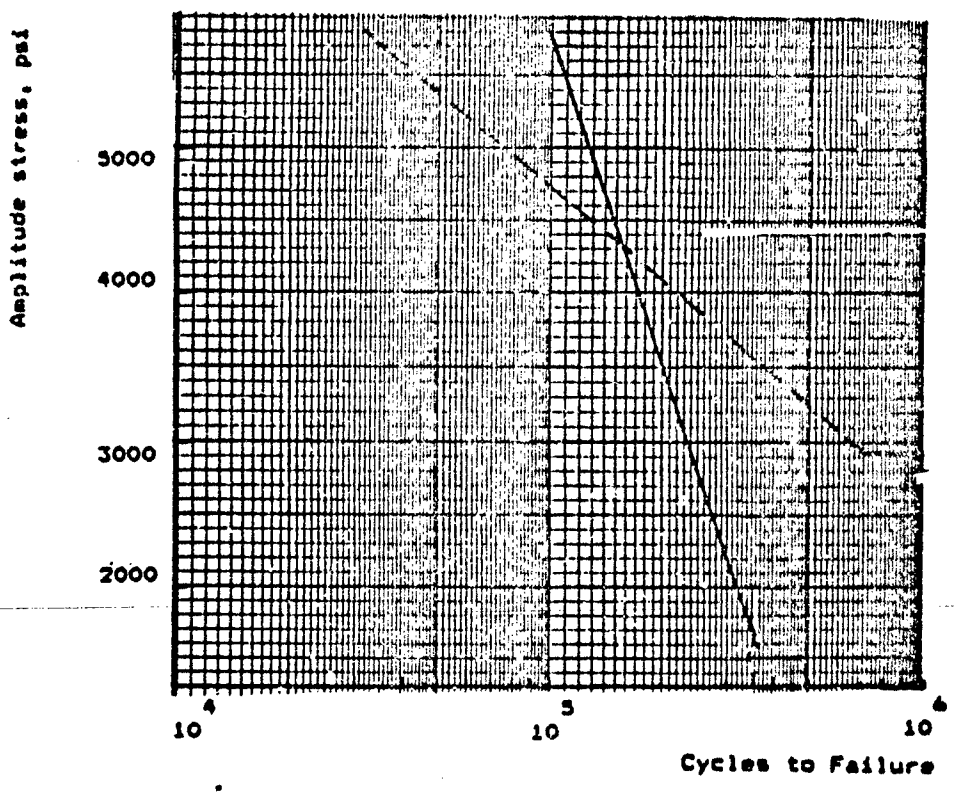


FIGURE 6 S-N Diagram plotted from the results of fatigue tests for the polycarbonate specimens with and without stress concentrators.

- - Without stress concentrator
___ With stress concentrator

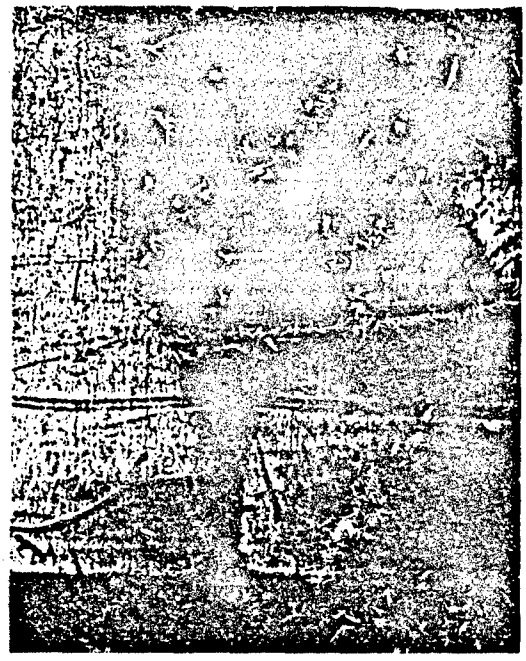


FIGURE 7. Crack at tensile zone of the solid polycarbonate specimen, 20X. Crack is initiated from the edge. Craze spot is located ahead of the crack tip.

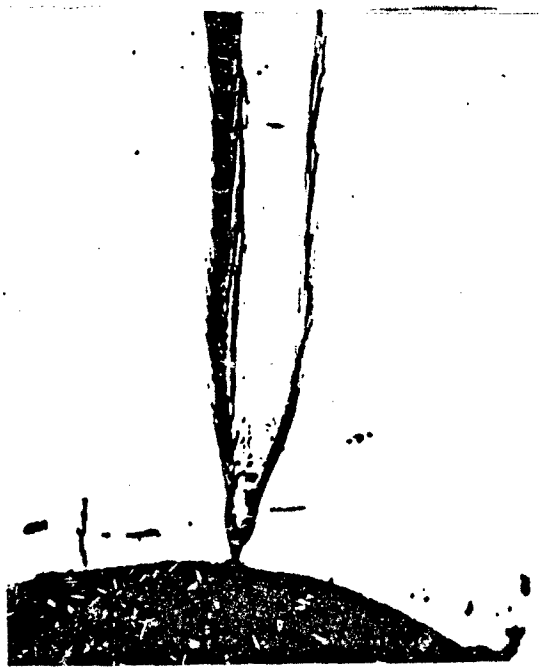


FIGURE 8. Crack at tensile zone of the specimen with 1/4" Dia hole, 20X. Crack is started from the hole edge. Craze spot is located ahead of the crack tip. 163000 cycles.

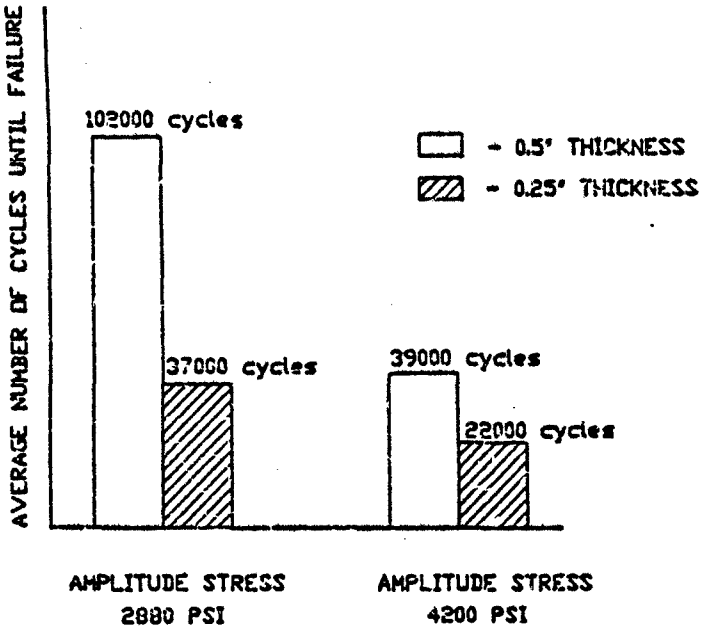


FIGURE 9 Comparison of fatigue lives of the specimens with different thickness. Variable bending. Specimens with stress concentration. Fequency 5 Hz. Five Specimens tested in each group.

TABLE 1
Fatigue lives of specimens tested at different frequencies. Specimens with stress concentration. Amplitude load is 240 Lb.

Specimen number	Frequency, Hz	Number of cycles until failure	Time until failure, h	Average failure time, h
1	8	119300	4.14	4.69
2	8	151000	5.24	
3	5	92000	5.11	5.31
4	5	99000	5.5	
5	5	115000	6.39	
6	5	100000	5.57	
7	5	72000	4.01	
8	2	40900	5.68	4.70
9	2	33700	4.68	
10	2	61300	3.74	

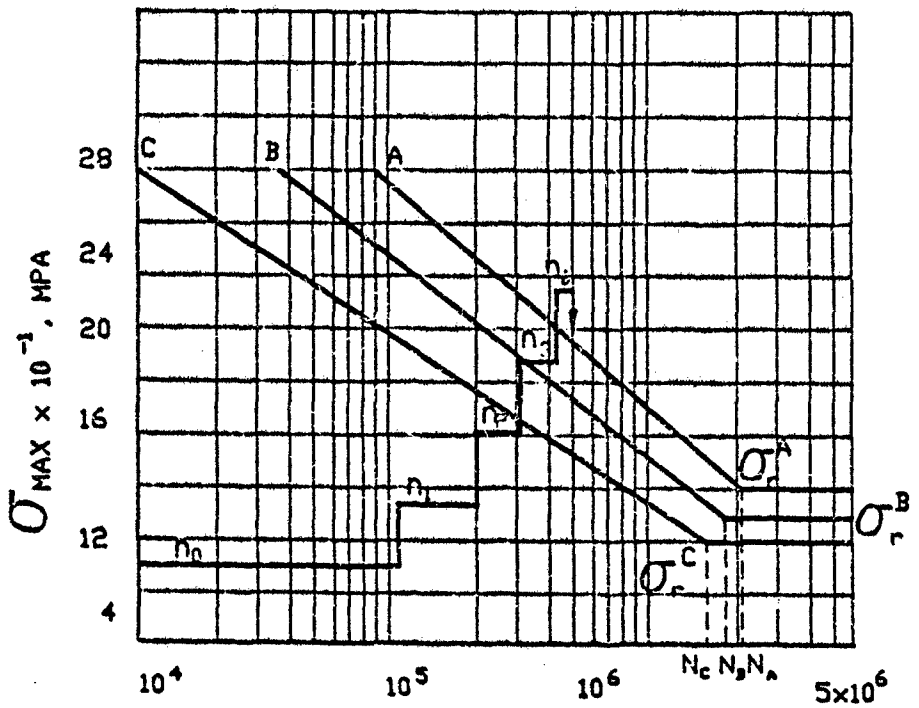


FIGURE 10. Loading program for the accelerated test. Three fatigue curves A, B, C are received from a long- term fatigue test and correspond, for example, 5%, 50%, and 95% probability of failure.

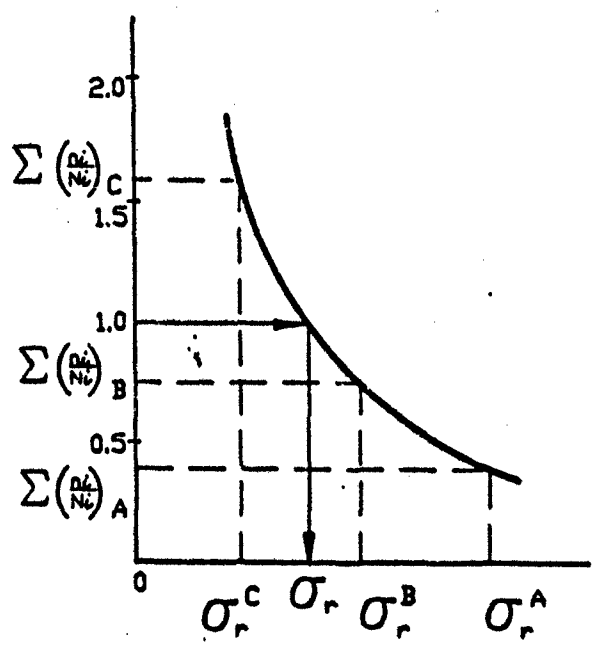


FIGURE 11. A diagram for graphical determination of fatigue strength.

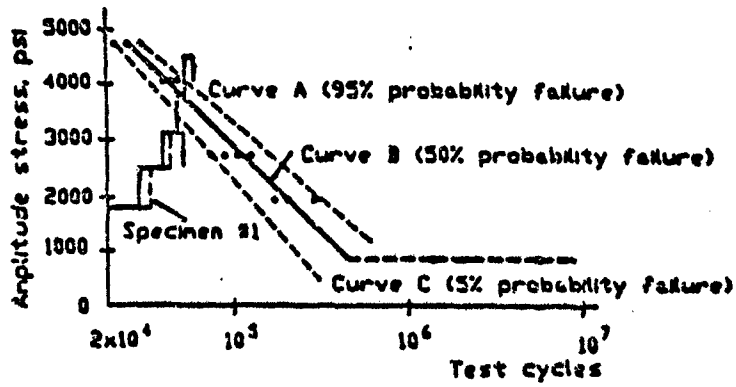


FIGURE 12. S-N diagram for polycarbonate specimens with stress concentrators. A, B, C curves plotted to calculate lives during treatment of the accelerated test results.

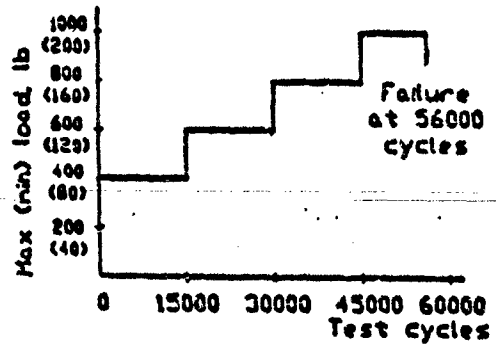


FIGURE 13. Program and results of accelerated fatigue test. Specimen #1 with stress concentration.

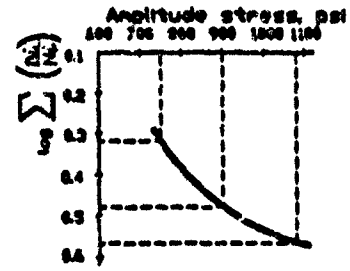


FIGURE 14. Graphical representation of the accelerated fatigue test. Specimen #1.

TABLE 2.
Testing regimes (Long-term test; variable bending).

Amplitude load, Lb		Amplitude stress, psi	Min. Stress / Max. Stress	Frequency, Hz
Plate with a hole	Solid plate			
180	240	1440	0.2	12
	360	2160	0.2	8
360	480	2880	0.2	6
	600	3600	0.2	5
540	720	4320	0.2	4
750	990	5760	0.2	3

TABLE 3. Accelerated test results.

Specimen number	Max. load increment, Lb	Number of cycles at one load level	Number of steps until failure	Number of cycles until failure
1	200	15000	4	56000
2	200	15000	5	74200
3	200	15000	4	50500
4	200	15000	3	53900

TABLE 4. Accelerated test result treatment.

Amplitude stress, psi	Curve A		Curve B		Curve C	
	Ni, cycles	n_i/Ni	Ni, cycles	n_i/Ni	Ni, cycles	n_i/Ni
1280	450000	0.030	350000	0.043	230000	0.065
1920	300000	0.050	260000	0.058	150000	0.100
2560	170000	0.088	140000	0.107	100000	0.150
3200	105000	0.105	85000	0.129	68000	0.162
		0.273		0.337		0.477

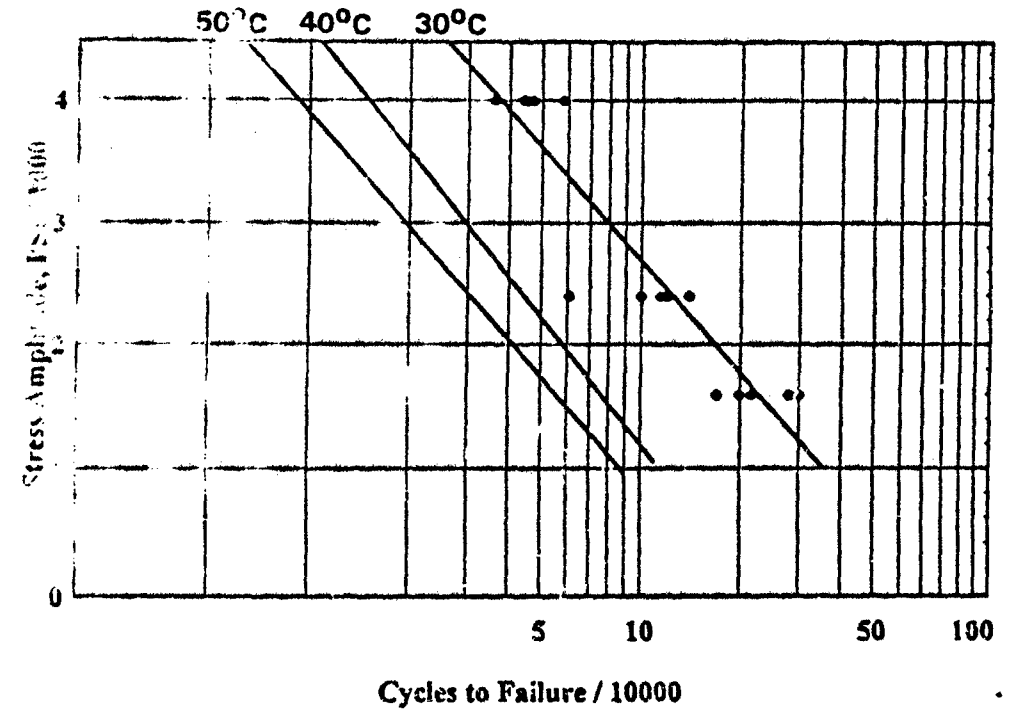


FIGURE 15 Temperature effect on fatigue life of the 0.5-inch polycarbonate specimens.

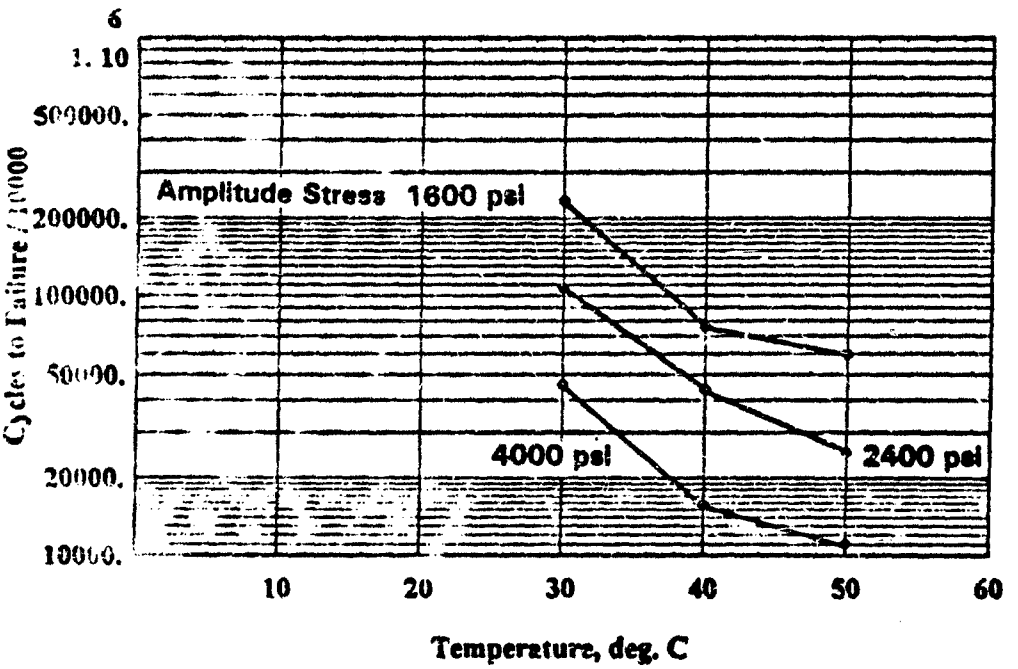


FIGURE 16 Temperature effect on fatigue life of the 0.5-inch polycarbonate specimens.

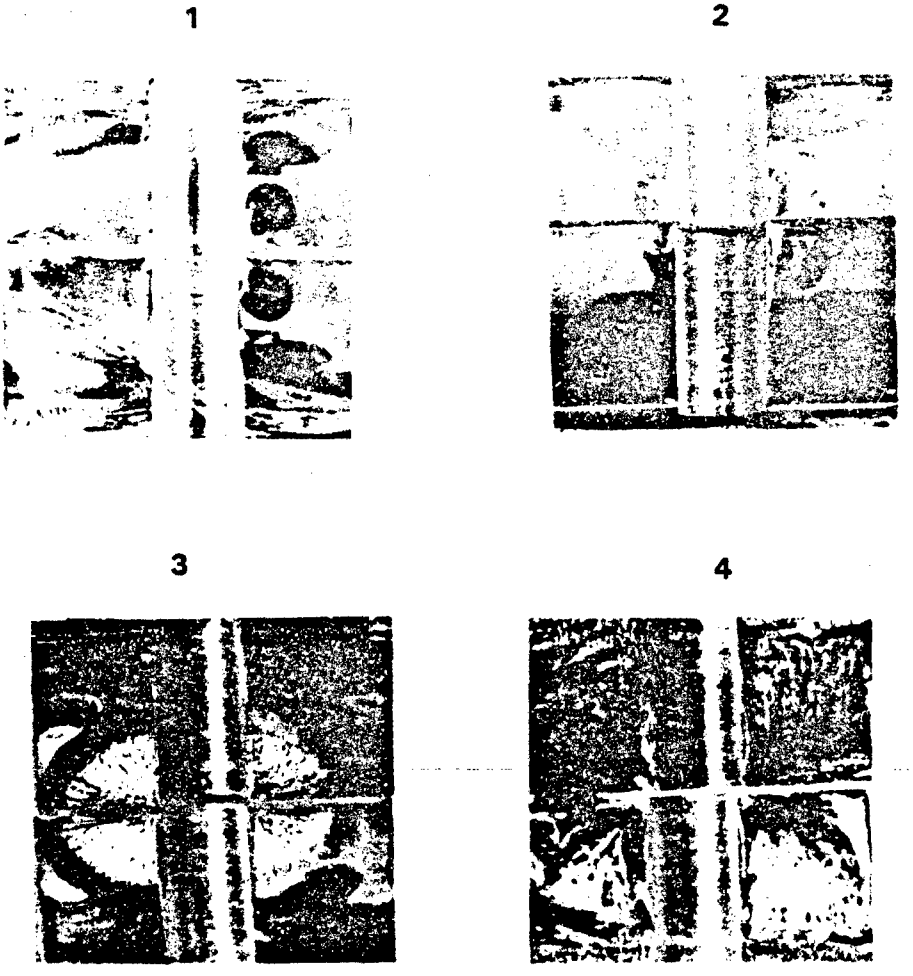


FIGURE 17 Failure surface topography for the specimens tested under different load and temperature.

1 Amplitude Load = 400 lb
Temperature = 40°C

2 Amplitude Load = 240 lb
Temperature = 40°C

3 Amplitude Load = 160 lb
Temperature = 40°C

4 Amplitude Load - 400 lb
Temperature = 30°C

PHYSICAL AGING OF POLYCARBONATE BY FREE VOLUME CONSIDERATIONS

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by

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BATTELLE

ABSTRACT

A major problem associated with amorphous polymers is that they are not in thermodynamic equilibrium after forming, and it may take a long time for equilibrium to be reached. As a result, even when the other destabilizing factors are eliminated, amorphous polymers physically age with time. This phenomena is characterized by molecular rearrangement into a thermodynamic state of lower free energy. The rate of molecular rearrangement increases with increasing temperature, and has been found to be nonlinear with time. This means that the thermal history of the material has a strong influence on the state of aging the material has experienced. The decrease in free volume of the polymer slows with time, as has been shown in long-term creep tests, and by viscoelastic measurements that show an increase in relaxation time. Laboratory experiments, in which a polymer is subjected to well-defined and known thermal histories, have shown that aging results in a number of changes in the polymer that eventually lead to changes in the macroscopic property performance of the polymer. In particular, it has been shown that aged amorphous polymers become brittle. An appropriate method for analyzing polycarbonate fracture is based on the model of a critical free volume leading to a ductile-brittle transition. It is hypothesized that a master curve of free volume versus aging time, normalized for stress level and temperature, can be developed to provide a predictive tool for determining how close a transparency is to the critical transition point. Since it has been shown that polycarbonate can be rejuvenated by reheating above its glass transition temperature and requenching to regenerate the necessary excess free volume, brittle failures can be averted in transparencies that are close to the ductile-brittle transition. Positron annihilation spectroscopy can be used to measure the free volume of as-molded and laboratory-aged polycarbonate samples. The free volume can then be correlated with the mode of failure of the polymer, and a model developed that will relate the free volume of a transparency to the condition of the aged polymer at any time in its post-molded life.

INTRODUCTION

A technical approach addressing the development of a useful testing protocol and analysis methodology for aging of polycarbonate is proposed based on the concept that the key mechanical properties of an amorphous polymer are directly related to its free volume. There are two basic classes of polymers: amorphous and semi-crystalline. Semi-crystalline polymers have a complex molecular morphology and their aging behavior is determined by changes in either, or both, phases. In contrast, amorphous polymers have no ordered morphological features. They are vitrified liquids, whose main chain molecular motion has been frozen. A number of changes can occur to amorphous polymers during aging that depend on the environment to which they are exposed.

Many of the long-term performance characteristics of polymers are not well understood. In addition, it is realized that additional stresses, which are not always well known or consistent, can be introduced by various manufacturing techniques. The variability in molded-in residual stresses makes the understanding of the material even more difficult. Yet understanding the material at the basic level is necessary for developing the capability to predict and extend service life. A number of other phenomena can also influence the long-term behavior of polycarbonate. In addition to molded-in stresses, these phenomena include residual orientation, surface defects, absorbed moisture, and UV degradation. Molded-in stresses and surface defects make the polymer susceptible to stress cracking by small concentrations of solvent. Molded-in stresses and residual orientation will slowly relax with time, but in doing so will alter the mechanical properties of the material, and can often lead to warpage of the part. Absorbed moisture and UV degradation will reduce the molecular weight of the polymer, thereby reducing its mechanical properties. The effects of all of these phenomena are to make the polymer more brittle.

Many of the above problems with molded polymers can be eliminated by quality mold design (to eliminate surface defects), close control over the molding process (to eliminate residual orientation, molded-in stresses, and absorbed moisture), and chemical modification (to provide UV protection). Even if these precautions are taken, however, the problem remains that amorphous polymers are not in thermodynamic equilibrium, and it may take a long time for equilibrium to be reached. As a result, even when the other destabilizing factors are eliminated by suitable processing and stabilization, amorphous polymers physically age with time.

Figures 1-3 show that while the tensile strength and tensile modulus of polycarbonate do not change in any significant detrimental manner during physical aging, the impact strength of the polymer exhibits a sharp reduction over a very short period of time.[1] In order to make reasonable predictions of the time at which the polycarbonate changes from ductile to brittle behavior, it is necessary to identify some property of the polymer that changes in a dependable manner and can be correlated to the onset of brittleness. Moreover, this property must be capable of being measured in a sensitive manner during short-term "aging" experiments.

THEORETICAL CONSIDERATIONS

Although the rate and extent of aging varies for each polymer, depending on the polymer type and its thermal and mechanical history, research has shown that physical aging is

thermodynamic in nature, and, therefore, the extent of aging can be described as a state function.[2] This means that the extent of physical aging of a polymer can be described simply by determining the thermodynamic state of the polymer at a given point in time. The state function that has been most closely correlated with the extent of physical aging in a polymer system is the excess free volume of the polymer system. Free volume exists in polymer systems because the randomly coiled polymer molecules cannot pack to 100 percent molecular density. The term "excess" free volume is used because there will always be unoccupied space, or free volume, available in a polymer, even at maximum molecular packing (i.e, the "equilibrium" free volume content of a totally amorphous system). The excess free volume in the polymer system allows molecular motions to occur that would be frozen out if the polymer were able to relax to its equilibrium level of free volume during cooling. It is this molecular motion that allows structural rearrangement and free volume relaxation to occur within the polymer, and leads to the process generally described as physical aging in the amorphous regions of polymer systems.

At any point in time, the rate of physical aging is typically dependent on the structure of the polymer system (i.e. how far the system is away from its equilibrium level of free volume) and the temperature of the polymer system. The farther a given system is away from its equilibrium structure, and the higher the temperature at which the process is allowed to occur, the faster the effects of physical aging will be noted. Thus, it is obvious that the thermal and mechanical history of a given polymer material, not only defines the current state of physical aging the material exhibits, but also has a strong influence on the rate at which physical aging will proceed. It had been demonstrated through numerous studies that polymers age more rapidly as the temperatures to which they are exposed approach their glass transition temperatures. This is expected since the rate of molecular rearrangement, and hence structural relaxation, increases with increasing temperature.

High excess free volume often results in more ductile behavior, as characterized by a large strain-to-break or high impact strength.[3] This type of behavior is not found in all amorphous polymers, but primarily those which exhibit low secondary (sub-Tg) transition temperatures. For example, polycarbonate, which has its lowest secondary transition temperature at -85 C, demonstrates ductile behavior (as exemplified by its high impact strength) at temperatures well below its glass transition temperature of 150 C. However, laboratory experiments, in which a polymer is subjected to well defined and known thermal histories, have demonstrated that physical aging results in a change in the free volume of the polymer that eventually leads to changes in the macroscopic property performance of the polymer. In particular, evidence exists that a shift from a ductile-to-brittle failure mode will occur when free volume decreases below some critical level of excess free volume.[4] Since the excess free volume is simply the actual free volume minus the constant equilibrium free volume at a given temperature, measurement of the free volume is all that is needed to provide an index of the degree to which the polymer is aged.[5] Furthermore, the free volume of a polymer sample of any age and history can be correlated to the strain-to-break at failure or high impact strength. Crissman and McKenna have even demonstrated that the aging of amorphous polymers under different conditions could be reduced to a single master curve of strain-to-break versus time, with appropriates shifts to account for aging temperature and loading force.[6]

The free volume of polycarbonate can be measured as a function of time at various temperatures, and those curves can be shifted to produce a single master curve of free volume versus time at the temperature of interest. That single free volume-time curve can then be used to predict the time at which the critical free volume for the onset of brittle failure will occur.

FREE VOLUME MEASUREMENT PROCEDURES

Until recently precise determination of the free volume of polymers required tedious dilatometric measurements. In the last few years an important technique has been applied to the study of physical aging that significantly reduces the time required to make this measurement and greatly enhances the precision of the data obtained. This technique is Positron Annihilation Lifetime Spectroscopy (PALS).[7,8] Figure 4 shows that this technique is quite sensitive to small changes in free volume and has been used to quantify the aging behavior of polycarbonate.[4] When positrons are injected into solid matter, some of the positively charged particles combine with electrons to form ortho-positronium, which becomes trapped in local regions of free volume. Positron annihilation spectroscopy measures the time for ortho-positronium to decay, and the intensity of the decay event. The decay time of ortho-positronium is between 1 and 5 nanoseconds. This decay time is proportional to the average size of the free volume sites in a polymer sample. The intensity of the decay is proportional to the number of free volume sites in which ortho-positronium is trapped. Therefore, the free volume is simply the product of the void size times the number of void sites. The decay events are measured from gamma rays emitted from the source (at one energy level) and the positronium (at another energy level). A typical positron lifetime curve is shown in Figure 5. This decay curve in polymers is typically a composite of four contributions, which can be expressed mathematically as:

$$Y(t) = Ae^{\alpha t} + Be^{\beta t} + Ce^{\gamma t} + D \qquad (1)$$

Y(t) is the number of positron annihilations recorded at time, t. D is the magnitude of the background decay recorded by the instrument. A, B, and C are zero time intercepts of the three curves that comprise three modes of positron decay. α, β, and γ are reciprocals of the slopes of these lines. The curve $Ae^{\alpha t}$ is related to the decay of para-positronium, a short-lived positron-electron complex and other short lifetime events. The decay time associated with this type of first component is on the order of 0.1-0.3 ns. Curve $Be^{\beta t}$ is related to the decay of free positrons with electrons. Free positrons have a lifetime between 0.3 ns and 0.8 ns. The curve $Ce^{\gamma t}$ is related to the decay of ortho-positronium, a relatively long-lived, (1-5 ns), positron-electron complex. Ortho-positronium is postulated to become trapped in free volume sites prior to decaying, thus the lifetime of ortho-positronium is related to the free volume in a sample. The size of the free volume sites is related to the reciprocal of slope of the $Ce^{\gamma t}$ curve, $\tau_3 = 1/\gamma$. The area under each of the three decay curves can be calculated to produce three quantities; I_1, I_2, and I_3. I_3 is related to the magnitude of the free volume in the sample.

ANALYSIS OF AGING BEHAVIOR

An important element in analyzing the performance of polycarbonate is the way in which it is evaluated to determine its lifetime in a given service environment. Specific evaluation tests must be conducted on samples removed from an accelerated aging environment, usually at elevated temperatures, and correlated with data taken from samples in the actual service environment. The objective of all the evaluations is to determine the projected lifetime of the polymer for use in the service environment.

Several mechanical properties are particularly sensitive to the performance of polycarbonate parts. These include elongation-to-break and impact resistance. As shown in Figures 1 and 2, the degree of aging of polycarbonate cannot always be determined by tensile strength and modulus data. The polymer may not show evidence of visual or mechanical property changes during a limited aging investigation. In addition, chemical or physical changes in the polymer may take excessively long to manifest themselves as visual or mechanical property defects, even under accelerated conditions. Thus, it would be useful to have a technique that can provide early detection of significant changes in the material.

The proposed testing procedure is designed to provide specific information about the performance of polycarbonate, and its anticipated performance in the future. The goal is to provide information that can be used to either detect when a part will fail, or project how long it can be relied upon to perform satisfactorily. The most basic testing conducted on polymers is to measure their stress-strain response under tension and flexure. A number of important pieces of information can be obtained from the stress-strain response of a material. The stress-strain curve that is generated provides a tangent modulus, yield stress, breaking strength, failure strain, and work expended to break the sample (area under the stress-strain curve). All of these factors are important in determining the performance of the polymer at any point in time after molding, but in particular, the elongation-to-break and area under the stress-strain are indicators of brittle behavior. These characteristics of the polymer will change with either chemical or physical changes that occur during aging. They are also sensitive to changes in the surface of the material, where localized changes can lead to premature initiation of failure in the bulk material. The only shortcoming of mechanical testing is that mechanical changes are not always directly proportional to the physical or chemical changes that can occur in a material. They are essential, however, since the failure of the material will be in response to a stress, whether it is a mechanical stress imposed on the material, or an internal stress developed by chemical or physical changes.

The tensile properties of a plastic provide basic information about its bulk mechanical behavior to an applied stress. Flexural tests are often utilized, in addition to the tensile tests, because they impose a more complex stress on the sample than a simple tensile load. Flexural loading imparts a tensile stress on one side of the sample and a compressive stress on the other. In some instances an aged material may be more affected by one form of these two basic stresses than the other. By measuring both tensile and flexural behavior, any distinction in response can be determined. As an example, if the surface is attacked by chemicals and produces microscopic etches on the surface of the polymer, the tensile strength may not be affected until the etching

process has proceeded quite far. Flexural failure may then occur sooner, since the etches provide notches for the initiation of failure in this stress mode.

It has been demonstrated that the primary failure mechanism expected of aged polycarbonate is brittle failure. While brittle behavior is exhibited as a reduction in the elongation-to-break in a tensile or flexural stress-strain test, brittle failure is most dramatically evident in impact tests. The important difference between tensile and flexural stress-strain tests and impact tests is that the rate of application of the stress is much higher in an impact test than in a stress-strain evaluation. The impact behavior of plastic changes very sharply as the material under goes a ductile-to-brittle transition during the aging process. This is shown in Figure 6, which mimics the anticipated elongation-to-break response of a ductile amorphous polymer, such as polycarbonate. The time required to reach the ductile-brittle transition is too long for reasonable experimentation. Fortunately, this ductile-to-brittle response can be produced by changes in the basic short term environment of the plastic; e.g. lower temperatures. This is shown in Figure 7, where the hypothetical elongation-to-break is plotted against free volume, achieved by measuring the strain-at-break at progressively lower temperatures. The ductile-brittle transition occurs at a critical temperature as polycarbonate is cooled. This graph identifies the critical free volume associated with the ductile-brittle transition. Aging experiments, in which the free volume is monitored as a function of time, at various temperatures between room temperature and just below the glass transition temperature, can now be generated to produce the time-temperature shifted master curve of free volume versus time at room temperature. This shifted master curve enables confident extrapolation to be made to the time at which the free volume reaches the critical value for the onset of brittle behavior. Figure 8 is an example of such a hypothetical master curve. The molecular conditions responsible for both occurrences of the brittle-ductile transition are the same, namely changes in free volume.

SUMMARY

Recent studies of aging in amorphous polymers have shown that knowledge of the free volume in a polymer at any time-after-manufacture can provide the necessary information for quantifying the "age" of the polymer, thus providing a methodology for predicting the lifetime of polycarbonate, an amorphous polymer. Data can be generated to measure both the free volume and mechanical properties of test coupons subjected to an accelerated aging procedure. That data can then be correlated to short-term laboratory experiments in which the critical free volume for the onset of brittle behavior is identified. As discussed previously, there will be minimal change in the tensile modulus and strength properties of the candidate plastic materials during any reasonable aging period, so that impact strength and elongation-to-break are the key mechanical properties of interest.

The strain-at-break of even brittle polymers can be changed without aging through various thermal conditioning, e.g. making the samples colder. The conditions that make a plastic brittle under these circumstances are the same conditions that make a plastic brittle after aging. The reason for this is that the response of the plastic is determined by its thermodynamic status, which is independent of its actual age. Therefore, it is possible to study the conditions in the laboratory that will make polycarbonate brittle, and measure its free volume at that condition, to identify values of these parameters that can be considered critical.

In the aging study a free volume versus time master curve can be generated by aging the polymer at temperatures up to the glass transition temperature. That curve can then be used to extrapolate to the time at which the critical free volume is reached for the change from ductile-to-brittle behavior, thereby providing an estimate of the expected life of the plastic. This premise is based on the evidence that the physical "age" of polycarbonate is not a chronological age, but rather a thermodynamic age.

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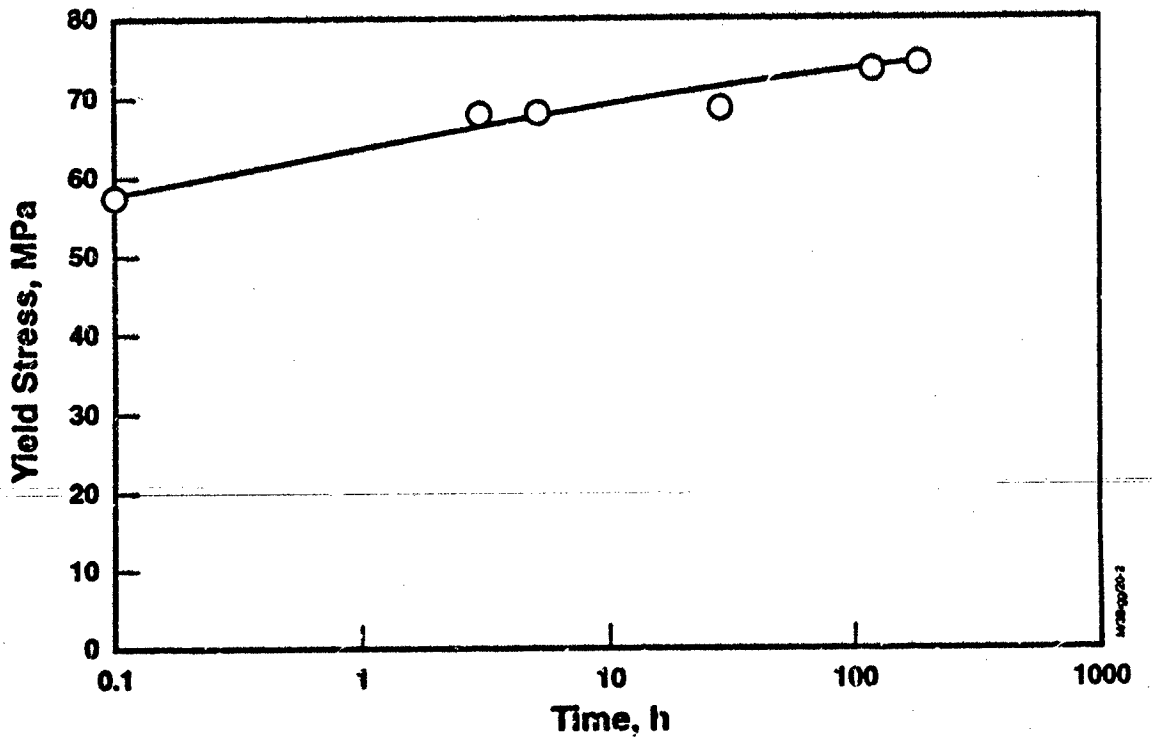


Figure 1. Tensile yield stress as a function of aging time at 135 C for polycarbonate. Graph taken from Reference 1 by permission of John Wiley & Sons, Inc.

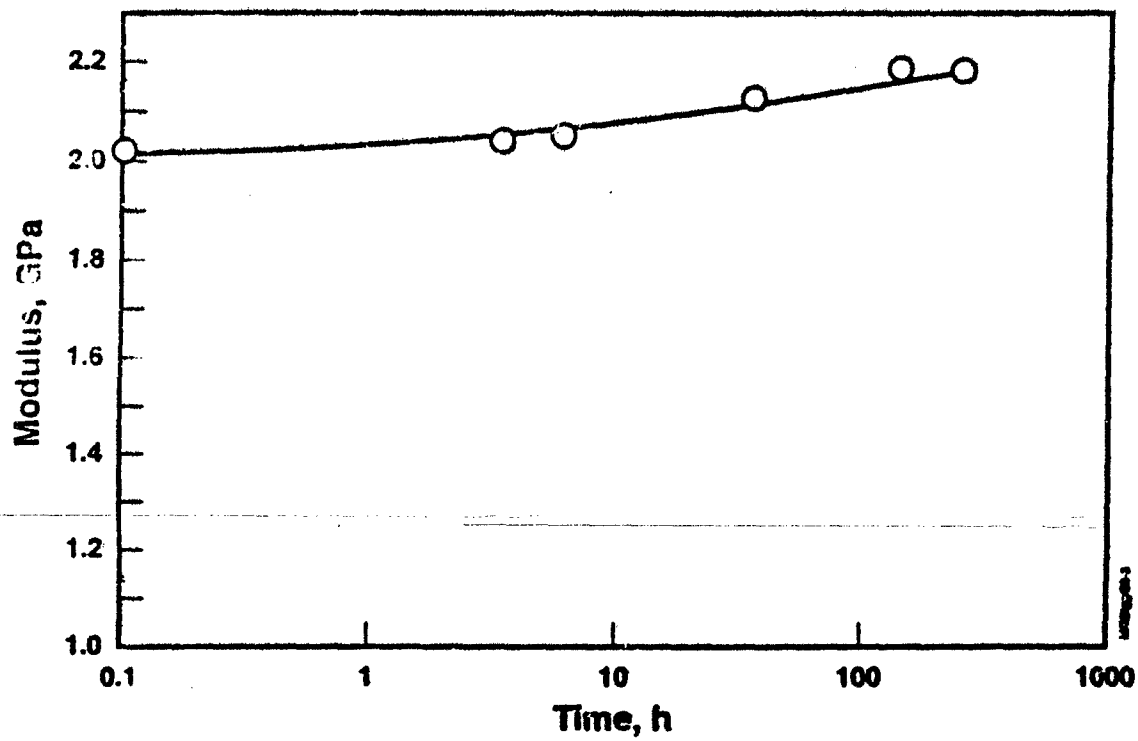


Figure 2. Tensile modulus as a function of aging time at 135 C for polycarbonate. Graph taken from Reference 1 by permission of John Wiley & Sons, Inc.

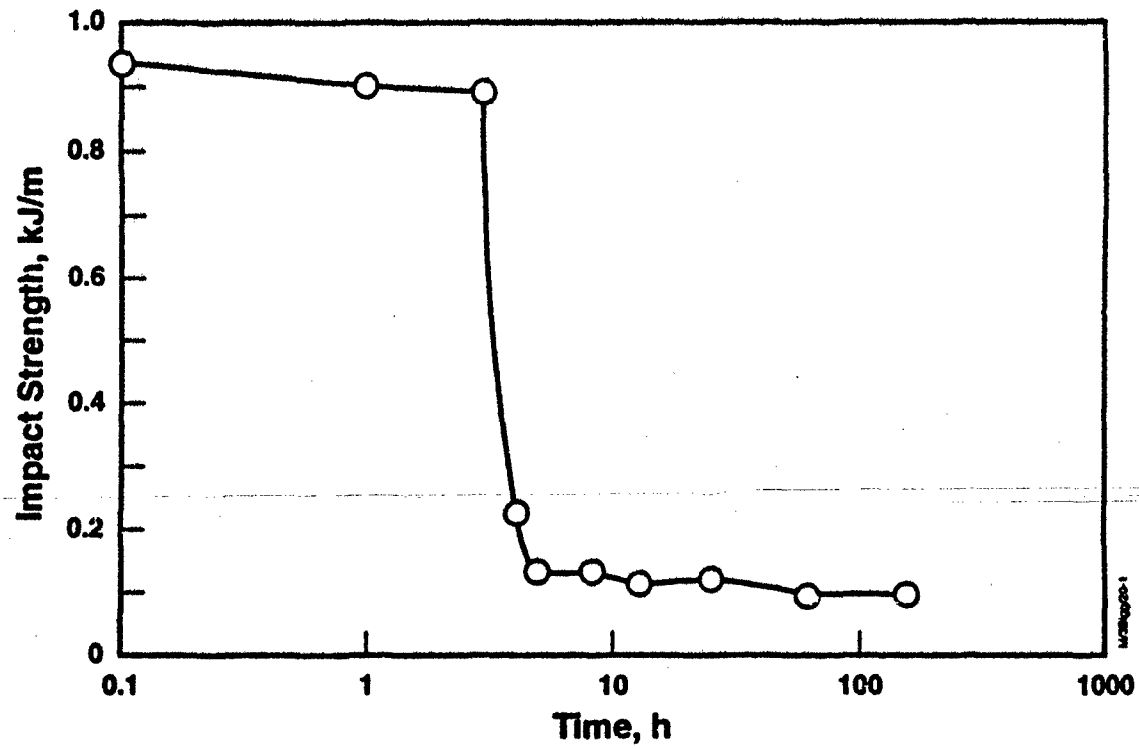


Figure 3. Impact strength as a function of aging time at 135 C for polycarbonate. Graph taken from Reference 1 by permission of John Wiley & Sons, Inc.

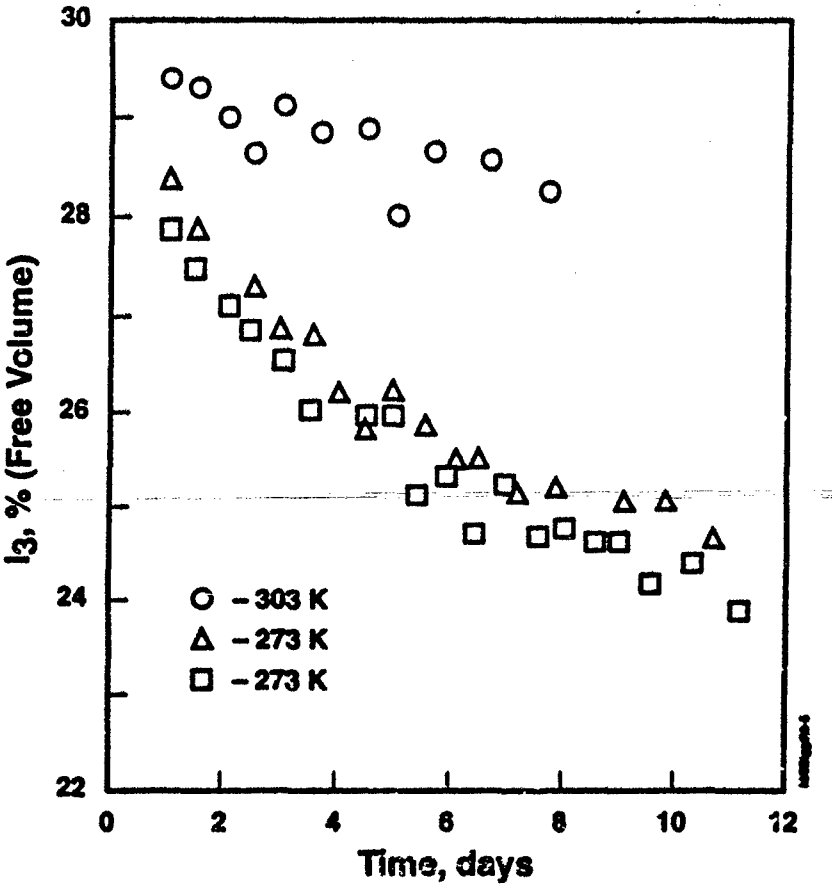


Figure 4. Free volume concentration (I_3) as a function of aging time as measured by PALS for polycarbonate. Graph taken from Reference 4 by permission of John Wiley & Sons, Inc.

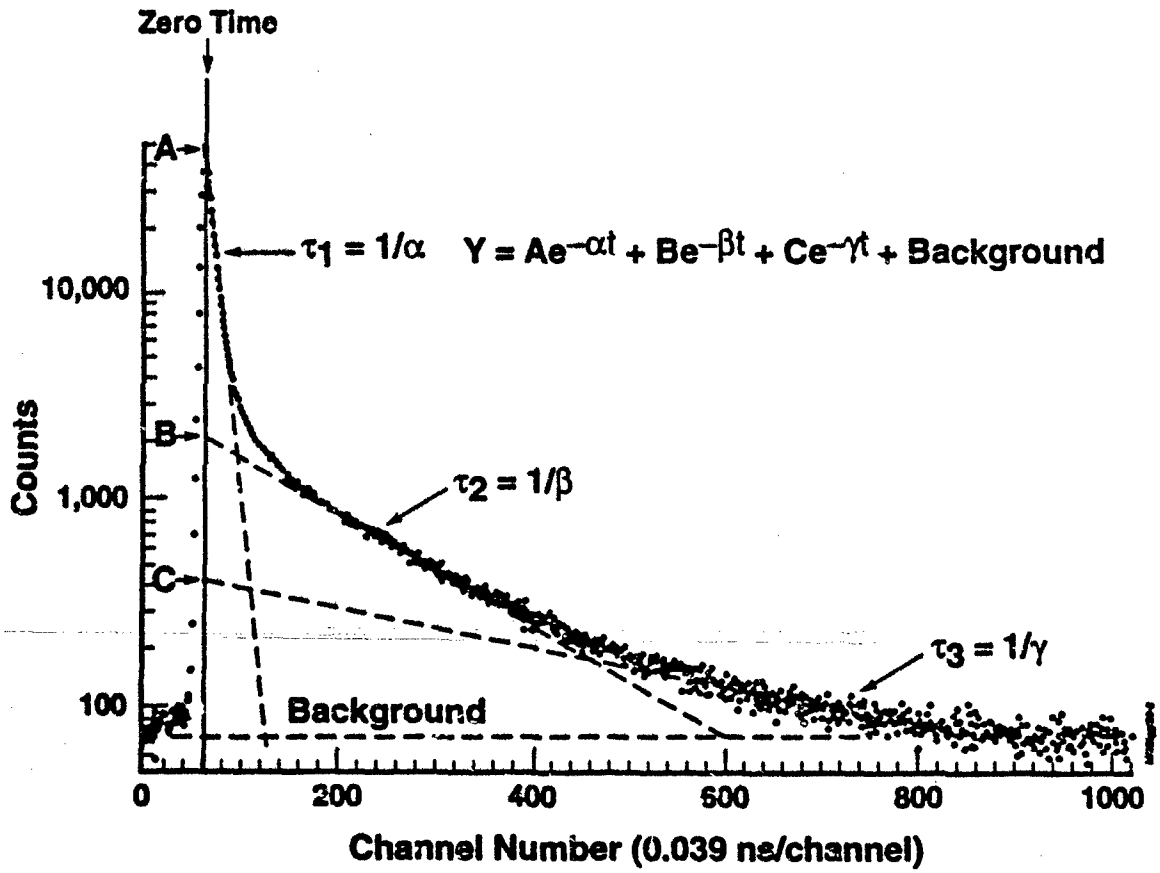


Figure 5. Typical PALS decay curve showing relevant parameters.

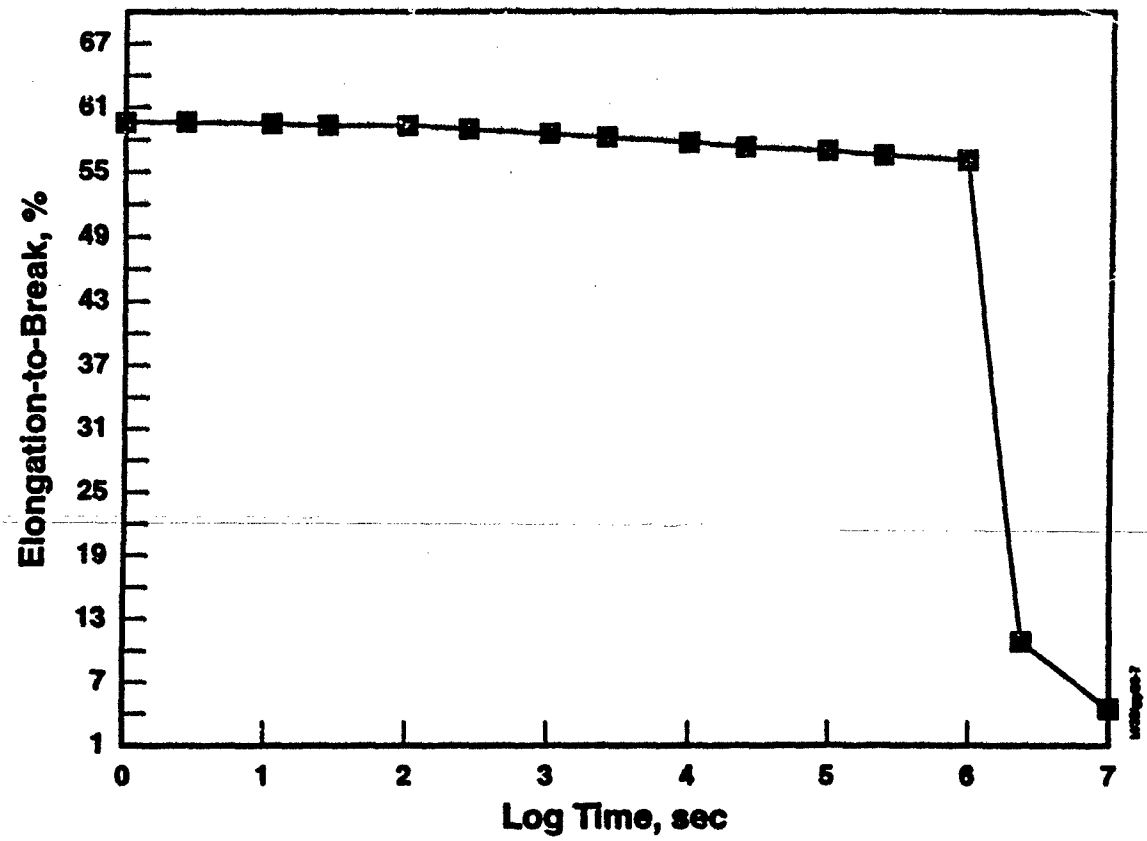


Figure 6. Hypothetical elongation-to-break versus aging time curve for polycarbonate.

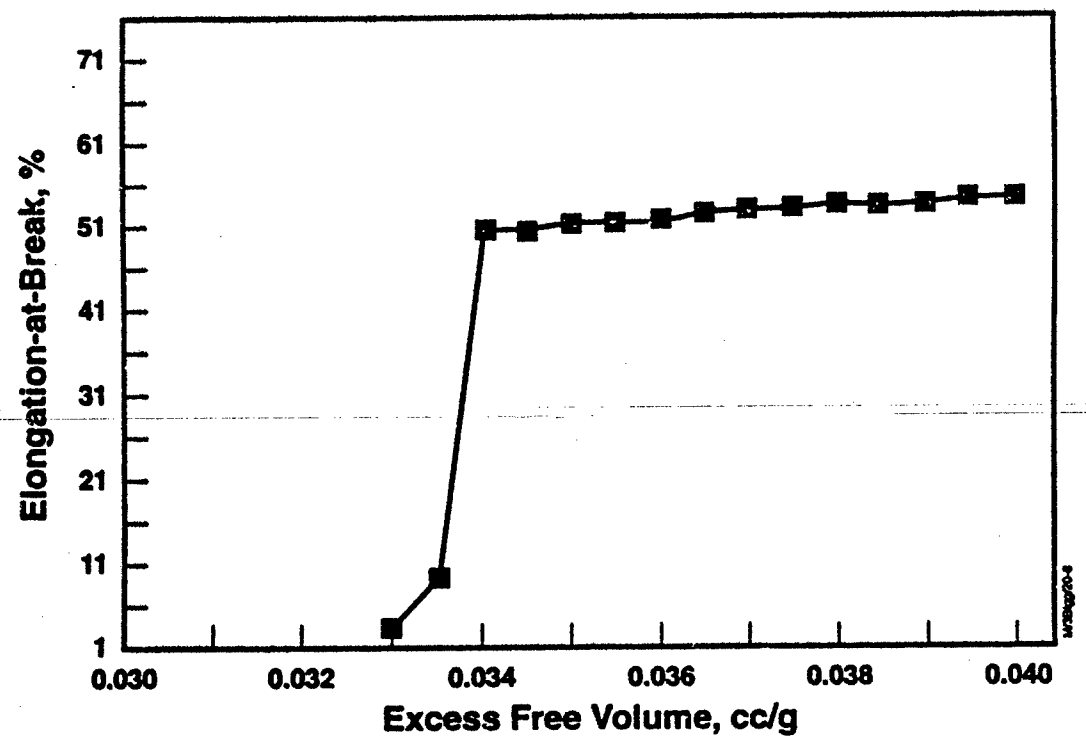


Figure 7. Hypothetical elongation-to-break versus free volume curve for polycarbonate.

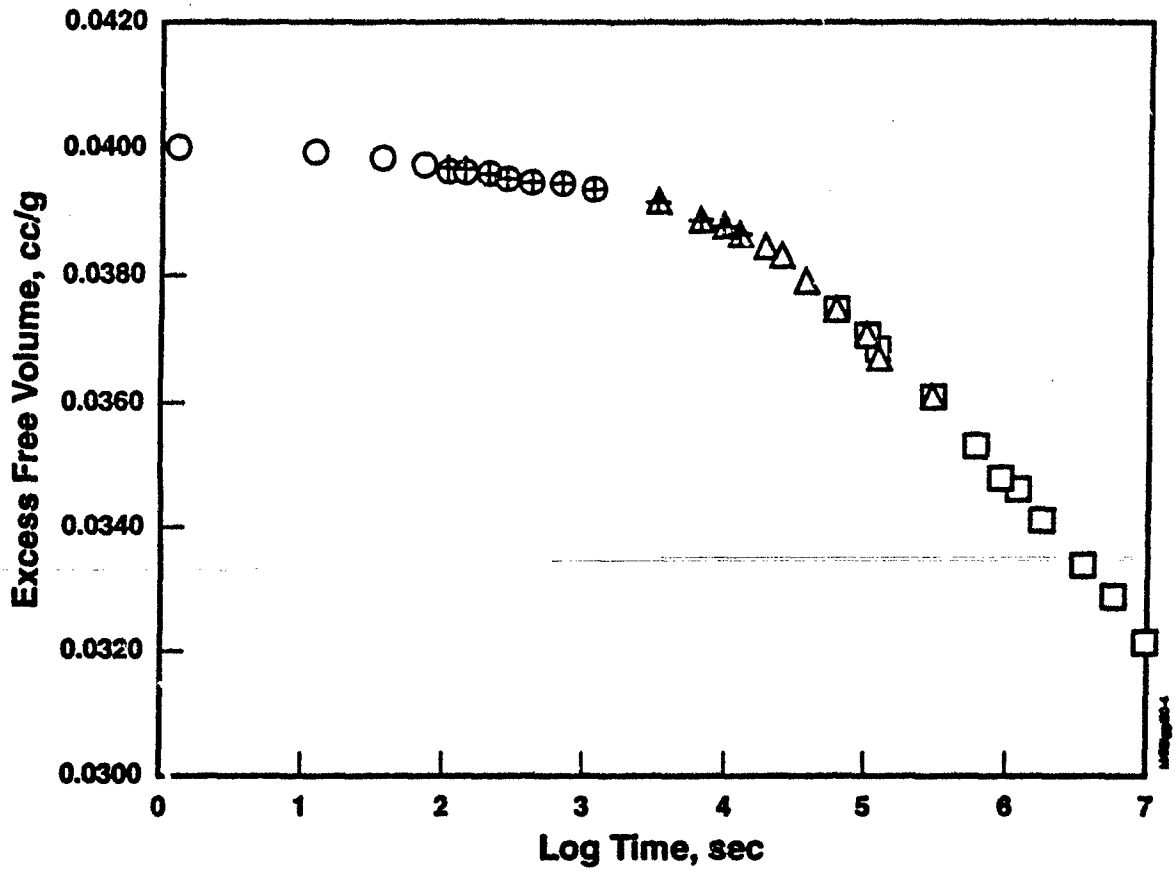


Figure 8. Hypothetical master curve of free volume versus aging time for polycarbonate generated by superposition of data taken at various temperatures indicated by different plotting symbols.

RELIABILITY ANALYSIS AND CRACK INITIATION STUDY IN CANOPY GRADE POLYCARBONATE

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Reliability Analysis and Crack Initiation Study in Canopy Grade Polycarbonate

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ABSTRACT

The methodology for canopy reliability analysis is outlined in this paper. It incorporates three elements. The first is the observation and characterization of an existing population of defects recorded during the inspection. The second is the analysis of the stress, strain, temperature fields the canopy is exposed to in service. And the third is the formulation of the fracture criterion.

An inspection tool and inspection manual have been developed to monitor the existing crack population and its evolution. The stress and temperature fields in a canopy under the service condition are obtained either experimentally or by numerical simulation.

The search for an adequate fracture toughness parameter leads to recognition of various failure scenarios depending on the stress level, thickness, temperature, rate of loading, environmental conditions, etc. We have identified five basic micromechanisms of fracture initiation in polycarbonate. These mechanisms were summarized in the form of fatigue crack initiation map for various

stress levels and thicknesses of the polycarbonate plate. The appearance of each mechanism is illustrated on the map by micrographs. The map can be used as a guideline for optimal design. An implementation of the reliability analysis for the canopies presently in service and the crack initiation map for the advanced design of the next generation canopies will be discussed.

I. INTRODUCTION

Canopies are designed to sustain high speed impact, chemical aggression, UV radiation, temperature and humidity cycles, low and high frequency mechanical stress, etc. An unfavorable combination of the above factor during the service time lead to material degradation and to a significant danger of sudden catastrophic failure. At present time the design philosophy does not include the lifetime and reliability factor. In this paper we outline a rational for a new design philosophy as well as a simple remedy for the canopies already in service.

There are three progressive stage of polycarbonate failure: 1) damage nucleation and growth leading to crack initiation, 2) stable (quasi-static) crack growth and 3) transition to unstable (dynamic) crack propagation. The third stage take just a fraction of a second. Thus the lifetime of a component depends on the duration of the first two stages. Much effort has been place in the studies of crack propagation. The summary of the conventional approach can be found in [1,2]. A relatively new crack layer analysis of PC fracture is reported in [3-5]. Fatigue crack growth in

transparency grade PC is reported in [6,7]. Comparing with the crack propagation, crack initiation has not been paid enough attention. However, the prediction of service lifetime and the reliability assessment require the formulation of an adequate criterion for crack initiation, the kinetic equation for slow crack growth and the crack stability criteria.

The three progressive stage of polycarbonate failure are presently being conducted at the Fracture Research Laboratory, UIC. Meanwhile, the practical recipes are developed for the reliability assessment of the canopies already placed in service. As result of the visual inspection, the presence of the cracks at the canopy's perimeter has been reported. It means that some region of the canopy have already passed the first stage of failure process (crack initiation). Therefore the remain canopy's lifetime prior to a catastrophic failure is the major concern. The criterion for the transition from slow to dynamic crack growth is employed as the rejection criterion for the canopies in question.

Below, we report 1) the crack initiation scenarios which summaries the canopy failure analysis, 2) a new inspection tool and inspection manual for the canopy field inspection together with the rejection criterion, and 3) a failure crack initiation map for polycarbonate. The latter may serve as a basis for the advance design philosophy for next generation of canopies.

II. FRACTURE SCENARIOS

The present study is based on the examination of a canopy removed from the plane and place in the Fracture Research Laboratory at UIC. The sketch of canopy is shown in Figure 1. All the crack recorded are associate with the bolt holes along the canopy's perimeter. There are very small crack (less then 5.0 mm length) emitting from the bolt hole with on preferable orientation. The N-W crack shown in Fig. 2 exemplifies this type of cracks. The fate of this crack is not clear at the early stage. The local stress field and material degradation determine the trajectory of fracture development of the crack. Figure 3 displace the two more mature cracks. The south bound crack runs to the edge of the canopy and does not constitute any dangerous, since the metal frame support. The second N-W crack aims to center of the canopy. The detail stress analysis is needed to estimate the risk level associated with this crack. A higher risk of failure is associated with the crack connected two or more bolt holes, since they act as one long crack. Such scenarios is exemplified in Fig.4. The sketches shown in Fig. 2, 3 and 4 result from an inspection tool developed in FRL (see Fig.5). The geometry of the inspection tool is design to allow one to observe and measure the length and orientation of the cracks along the canopy perimeter covered by the metal frame.

During the visual inspection an inspector fills up a 'yes-no' table. There is a computer program which converts this table data into the particular configuration of crack distribution in the canopy in question. Once the number of cracks, their size and distribution are known. The values of fracture mechanics parameters, such as Stress Intensity Factor K,

Energy Release Rate G and Crack Opening Displacement δ, can result from stress analysis. Stability conditions for brittle crack lead to the critical value of ERR or SIF. This results in the rejection acceptance of the canopy with the existing pattern of cracks for the next mission. The developments described above aimed at inspection and maintenance of the canopies in service. In the next section, we discussed the mechanisms of crack initiation in PC to assist in the design of the new generation of canopies.

III. THE FATIGUE CRACK INITIATION MAP

Experimental procedure

Polycarbonate sample with molecular weight Mw=29,000 g/mole was kindly provided by the Dow Chemical Company. After drying in a vacuum oven at 130°C for 24 hours, the plaques were compressed to smaller thickness under the following conditions: preheat 240°C under no load for 10 minutes, under 30 tons ram pressure for 20 minutes then cooled to 200°C by air followed by water to 30°C while still maintaining pressure. Rectangular specimens 80 mm x 20 mm were cut from the sheets and a 60° V-notch milled into the center of one long edge with notch length 1 mm and notch radius 0.01 mm. For tensile testing, dumbbell shaped specimens of gage length 50 mm x 10 mm were machined from the 3.1 mm plaques and pulled at an initial strain rate of 0.02% S^{-1}. The tensile yield stress, σ_y, was determined as 68 MPa.

The tension-tension fatigue experiments were conducted on a servohydraulic Instron Testing System at room temperature. Sinusoidal waveform loading with frequency 1.0 Hz was used for fatigue testing. The range of stress levels was (σ_{max}/σ_y) 0.35 - 0.75 and minimum to maximum stress ratio, R ($\sigma_{min}/\sigma_{max}$) = 0.4.

For evaluation of the number of cycles to crack initiation, N_i, under the various conditions, the crack is considered to have initiated when it reached 0.2 mm from the notch on the specimen surface. This convention was adapted for ease of detection of the crack and hence consistency of measurement with duplicate specimens.

Results and discussion

Figure 6 shows the typical crack surrounded by a process zone. Such a system is refer to as Crack Layer (CL) [3-5]. When the extensive process zone exists in the fracture, it is called cooperative fracture. In polycarbonate, the process zone may consist of shear bands (ductile micro-mechanism), craze or micro-cracks (brittle micro-mechanism). From these appearance of process zone, we can identify three basic crack initiation mechanisms and two mixed ones.

A typical example of *cooperative ductile* is seen in Figure 6. The cooperative ductile mechanism is so named because the process zone associated with crack initiation consists of yielded material as reflected by a large thinning ratio and hence representative of

significant cooperative ductile processes. This mechanism appears analogous to necking behavior in polycarbonate [8,9].

Figure 7 shows a typical fracture surface in the vicinity of the notch. The process zone apparently consists of a large number of micro-cracks or crazes and so we call the mechanism of initiation *cooperative brittle*. At this scale of magnification we were unable to determine whether these micro-features were actually crazes as suggested by [8] or cracks, as described by [10]. In our studies the main crack did not initiated in the mid-plane of the process zone, but followed a tortuous path through the process zone. The rough appearance of the fracture surface seen in Figure 6 clearly indicates the main crack initiated through a cloud of micro-cracks or crazes.

Figure 8 illustrates an example of a mechanism we have named *solo-crack brittle* . Solo-crack brittle is a well-known failure mechanism of cross-linked polymers such as epoxies [11,12] with very little thinning. The process zone is extremely small and observable only at the corners of the notch tip. As the name suggests, only solo crack (no process zone) is initiated and propagates immediately within one cycle to ultimate failure. Consequently, the surface is seen to be "mirror-like".

Under certain intermediate conditions, some of the mechanisms described above are in competition with each other. For example, in Figure 9, the outer fracture surfaces of the specimen process the characteristics of the cooperative ductile fracture whereas in the core

of the specimen a flat planar surface, reflects the solo-crack brittle fracture.

The above observations of the various crack initiation scenarios are summarized in the form of a map shown in Figure 10. On the map, five different crack initiation mechanisms are classified by various applied stress and specimen thickness. The orders of values of N_i are also shown in the plot.

For PC lifetime analysis, it is useful to see the relationships between stress-level and the thickness of the sheet which determines the failure mechanism and the time to initiation. In Figure 11 is plotted the normalized applied stress versus $\log(N_i)$ for the various thicknesses denoted, where Ni is the number of fatigue cycles to initiate the 0.2 mm surface crack. The solid lines represent the connection of experimental data with the same failure initiation mechanism. The dashed lines represent the extension of a single mechanism and the dotted lines a transition in mechanism. From a general perspective, the values of N_i decrease with increase in thickness and increasing stress levels. However, with thickness greater than 1.2 mm, the number of cycles to initiate a crack will increase with decreasing applied stress. For certain stress level, the transition in initiation mechanisms from cooperative ductile to solo-crack brittle is very sudden with increasing thickness whereas transition from cooperative ductile to cooperative brittle is less well

defined. This latter phenomenon is similar to that observed in fracture toughness evaluation of polycarbonate [13,14].

IV. CONCLUSIONS

1. The fracture scenarios of canopies are characterized and result a new developed inspection tool for field inspection. Those works lead to the rejection criterion of the canopy with the existing pattern of crack for next mission.

2. Novel fatigue crack initiation mechanisms maps are under development for polycarbonate. An illustrative example with polycarbonate is given where the effects of thickness (0.5 - 4 mm), and applied tensile loading at frequency 1 Hz on single-edge notched specimens are examined.

3. The number of fatigue cycles to initiate a crack were found strongly dependent on the micro-mechanism of deformation which occurs at the notch tip prior to crack formation and hence not to be a simple function of the applied stress.

Acknowledgement

The authors wish to thank Dr. Amkee Kim, University of Illinois at Chicago, and Dr. Kalyan Sehanobish and Dr. Che-I Kao, The Dow Chemical Company, for their helpful discussions. Thanks also to the

Wright-Patterson AFB and the Dow Chemical Company for the financial support of this work.

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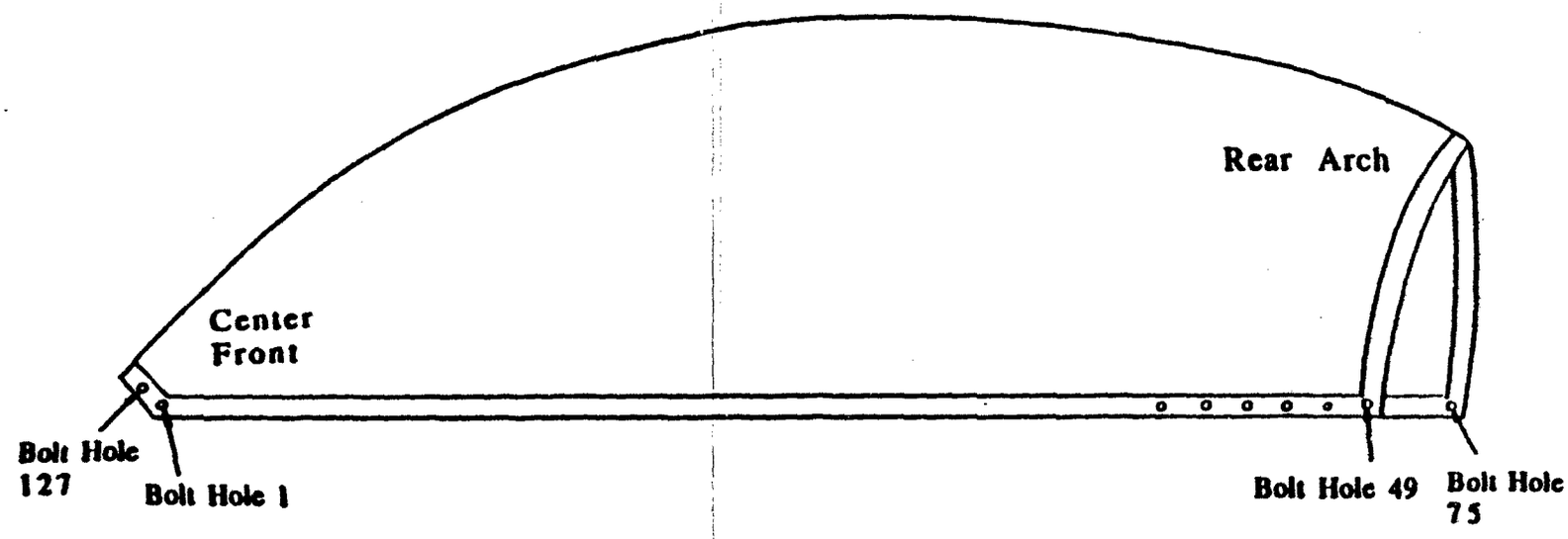


Figure 1 Sketch of the canopy without installed metal frame.

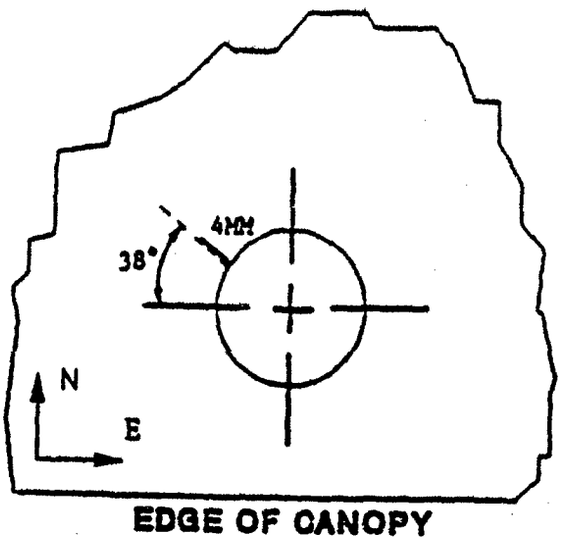


Figure 2 Scenario 1.

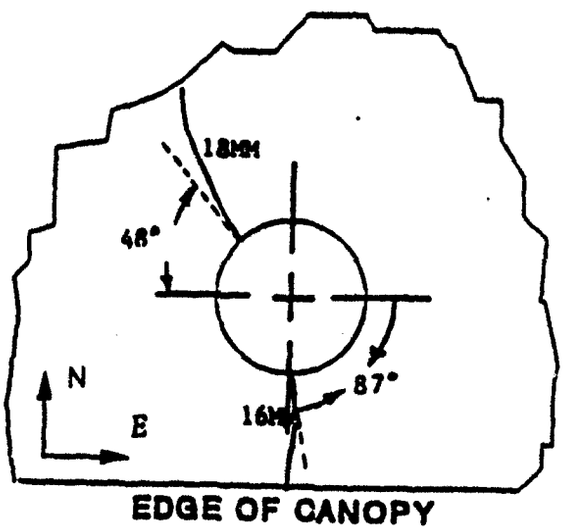


Figure 3 Scenario 2.

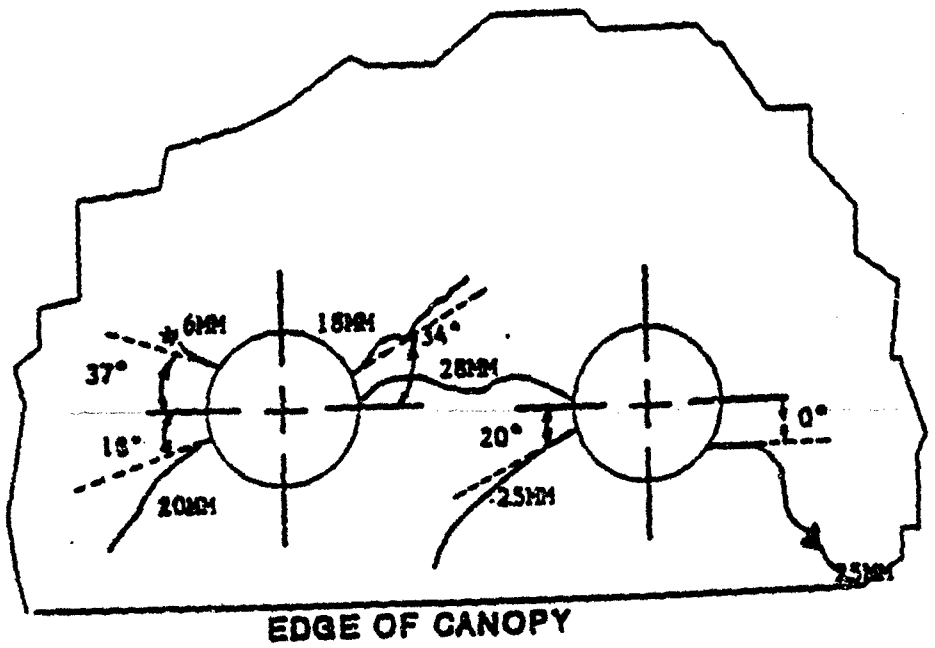


Figure 4 Scenario 3.

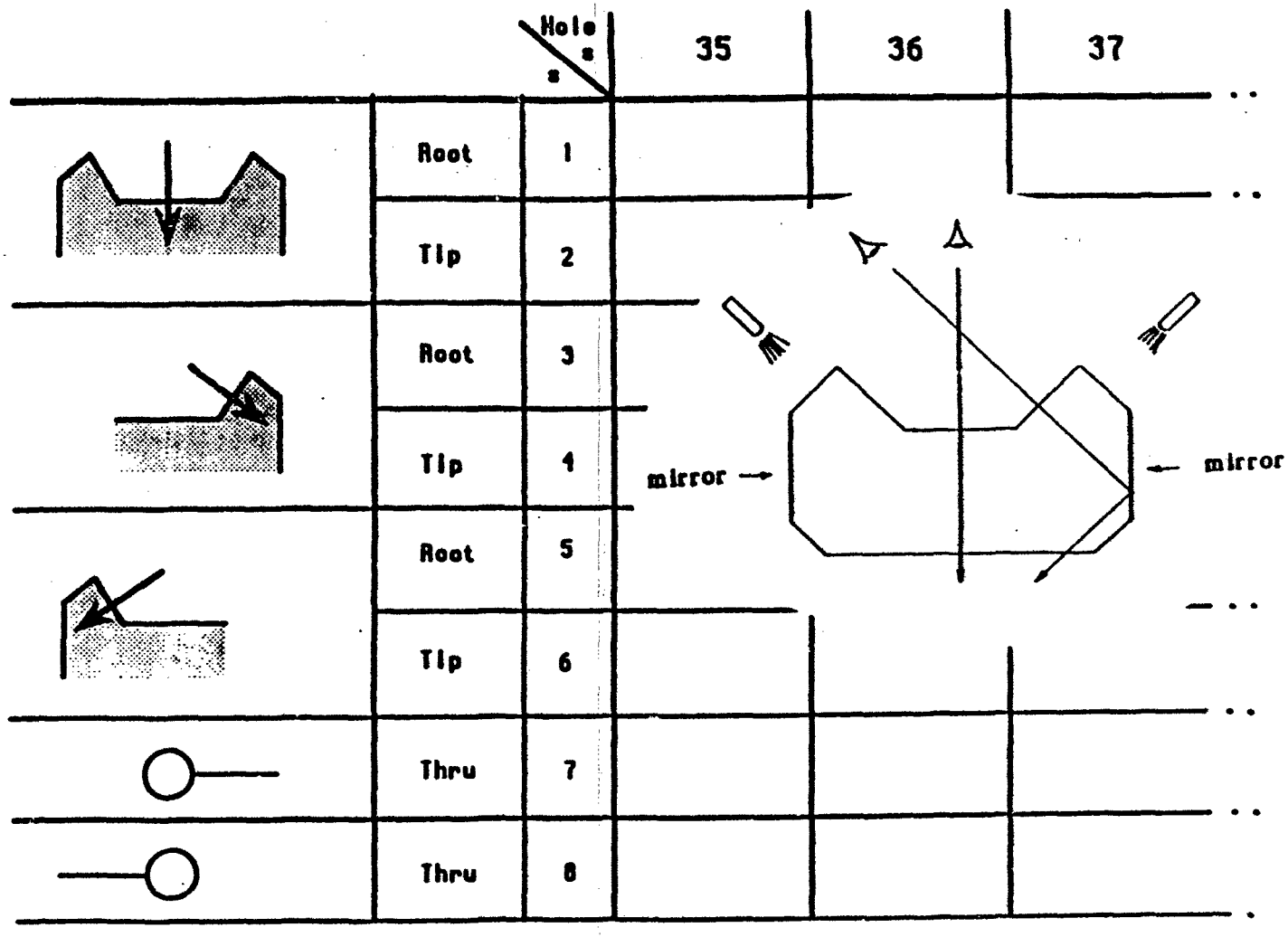


Figure 5 The inspection tool for the canopy with installed frame.

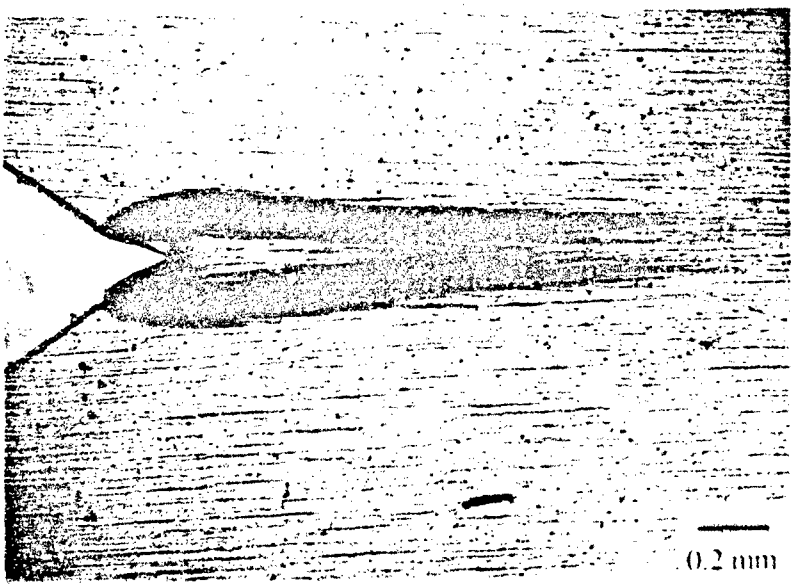


Figure 6 Cooperative ductile fatigue failure initiation in PC. σ_{max}/σ_y = 0.75, thickness 0.5 mm.

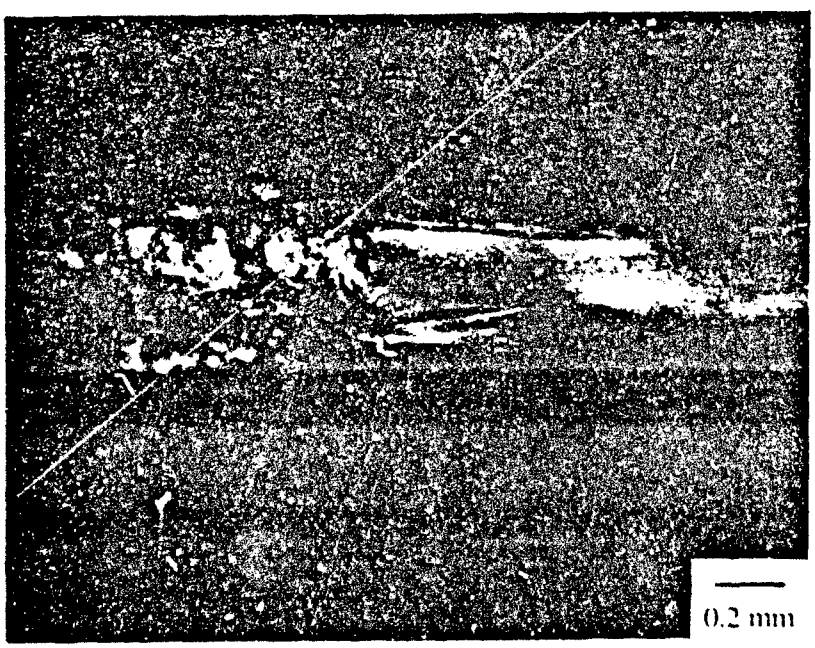


Figure 7 Cooperative brittle fatigue failure initiation in PC. σ_{max}/σ_y = 0.45, thickness 0.5 mm.

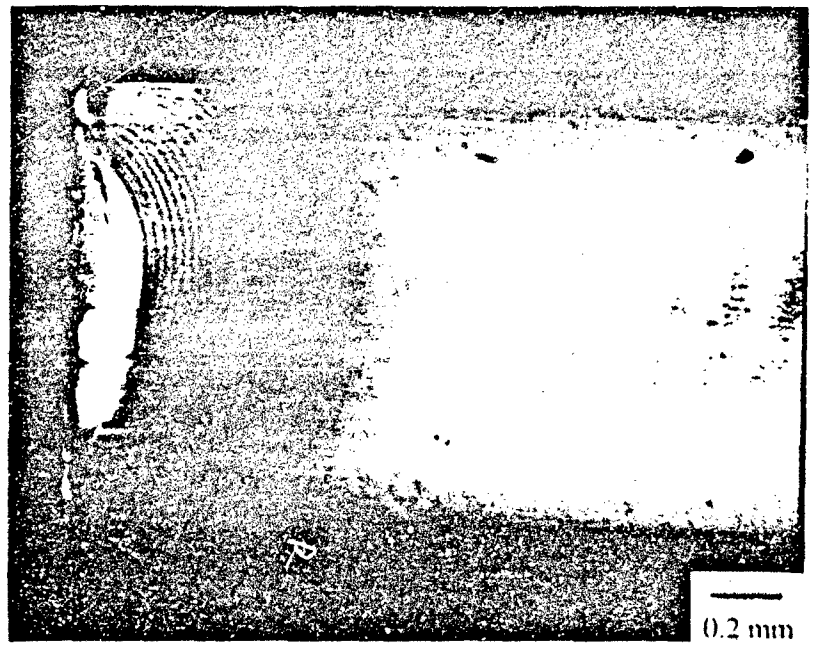


Figure 8 Solo-crack brittle fatigue failure initiation in PC, σ_{max}/σ_y = 0.45, thickness 1.2 mm.

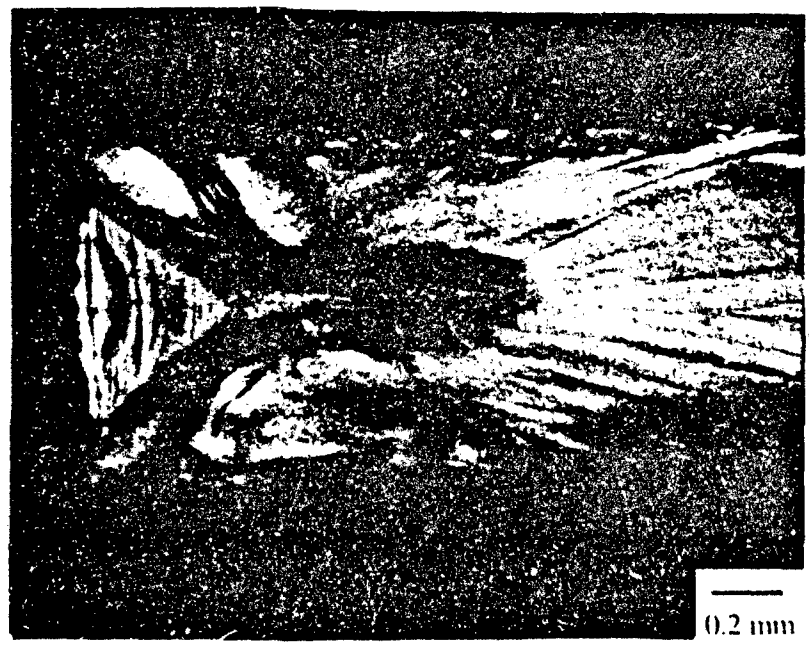


Figure 9 Mixed cooperative brittle and solo crack fatigue failure initiation in PC, σ_{max}/σ_y = 0.45, thickness 1.4 mm.

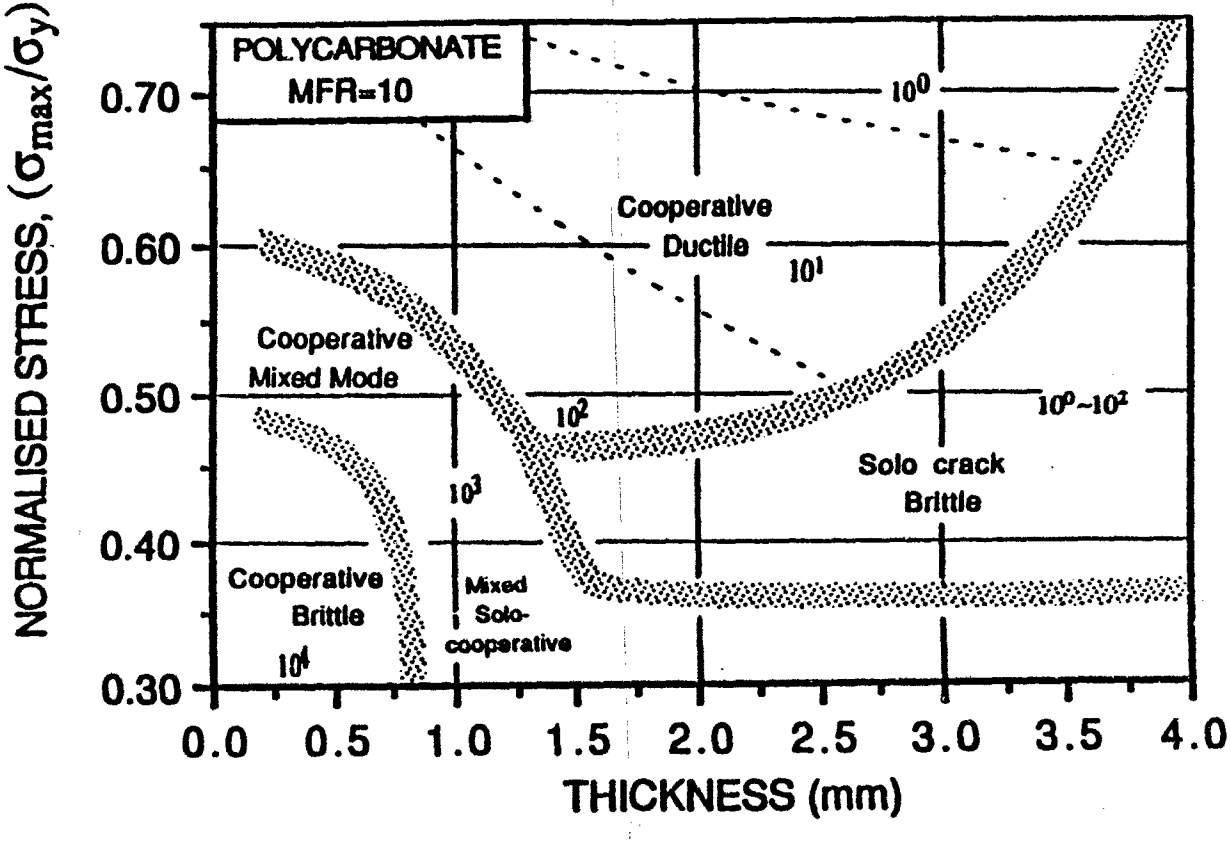


Figure 10 A fatigue crack initiation mechanism map for PC, Mw 29,000 g/mole. The orders of values of N_i are shown in the plot.

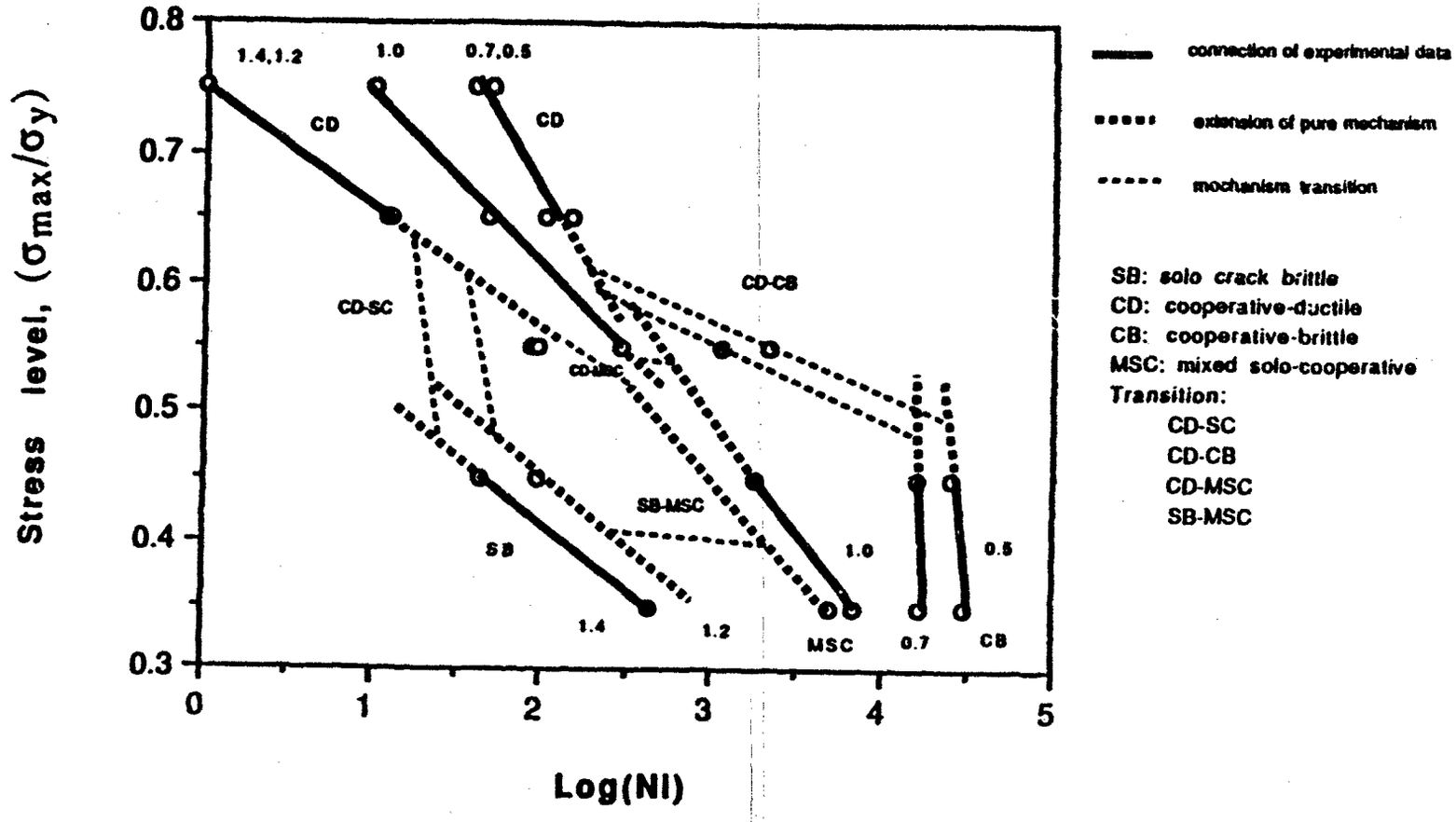


Figure 11 σ_{max}/σ_y versus $\log(N_i)$ for the various thicknesses.

EVALUATION OF CRAZE INITIATION CRITERIA FOR CAST ACRYLIC

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EVALUATION OF CRAZE INITIATION CRITERIA FOR CAST ACRYLIC*

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ABSTRACT

Chemical crazing is directly responsible for many aircraft transparency removals. Laboratory chemical stress craze testing can be used to evaluate the effects of different chemicals on aircraft transparencies. Most craze testing to date has been uniaxial, while the stress state in an installed aircraft transparency is biaxial. The uniaxial craze test is easier to conduct and requires less and more simple fixturing than the biaxial craze test. It is desirable to be able to use uniaxial data to predict the effects of a biaxial stress field on crazing. The focus of this effort was to develop an empirical time-to-craze criteria for cast acrylic applicable for both uniaxial and biaxial states of stress. Uniaxial tests were conducted using tensile specimens with isopropyl alcohol to determine the effect of creep and stress relaxation on time-to-craze. Results indicated no time-to-craze dependence on creep, but increasing time-to-craze as stress relaxed. Uniaxial cantilevered beam tests and biaxial craze tests (with a pressurized acrylic disk) were then conducted to determine time-to-craze as a function of stress. Results plotted in stress space (tension-tension quadrant only, due to the stress distribution in a pressurized disk) show that critical stress, for a given time-to-craze, can be represented by an inclined ellipse. The axes of the ellipse are smaller for longer craze time, larger for shorter craze times. The elliptical criteria do not agree with other criteria suggested in the literature.

*Performed under Contract F33615-84-C-3404 for the Flight Dynamics Directorate, Wright Laboratory, Wright-Patterson Air Force Base, Ohio.

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ABSTRACT: Chemical crazing is directly responsible for many aircraft transparency removals. Laboratory chemical stress craze testing can be used to evaluate the effects of different chemicals on aircraft transparencies. Most craze testing to date has been uniaxial, while the stress state in an installed aircraft transparency is biaxial. The uniaxial craze test is easier to conduct and requires less and more simple fixturing than the biaxial craze test. It is desirable to be able to use uniaxial data to predict the effects of a biaxial stress field on crazing. The focus of this effort was to develop an empirical time-to-craze criteria for cast acrylic applicable for both uniaxial and biaxial states of stress. Uniaxial tests were conducted using tensile specimens with isopropyl alcohol to determine the effect of creep and stress relaxation on time-to-craze. Results indicated no time-to-craze dependence on creep, but increasing time-to-craze as stress relaxed. Uniaxial cantilevered beam tests and biaxial craze tests (with a pressurized acrylic disk) were then conducted to determine time-to-craze as a function of stress. Results plotted in stress space (tension-tension quadrant only, due to the stress distribution in a pressurized disk) show that critical stress, for a given time-to-craze, can be represented by an inclined ellipse. The axes of the ellipse are smaller for longer craze time, larger for shorter craze times. The elliptical criteria do not agree with other criteria suggested in the literature.

INTRODUCTION

Background

The US Air Force has been and continues to be concerned with aircraft transparency life-cycle costs and overall durability. As part of this concern, the Air Force has funded programs to study transparency materials, evaluate transparency durability, and develop durability test methods. Acrylic plastics are frequently used for aircraft transparencies. Acrylic is subject to a phenomenon known as crazing. Crazes appear to be small cracks in the surface of the material, although they are not. Crazing is a form of yielding in polymers characterized by a spongy void filled fibrillar structure. The density of the material in the craze changes, causing a change in the index of refraction, which causes light to be reflected off of the crazes. Crazing occurs when tensile stresses are present, and is accelerated under the presence of certain chemicals and when temperature is increased. Crazing generally occurs perpendicular to the direction of the largest principle tensile stress [1]. The significance of crazing of acrylic is that it degrades transparency optics and often is the cause for transparency removal and replacement.

The current method of evaluating transparency chemical craze resistance, is the uniaxial cantilever beam craze test (reference ASTM F 484). This test method has been used almost exclusively in the transparency industry. The advantages of the cantilever beam craze test are that it is simple, it requires minimal equipment, and it is relatively inexpensive. The disadvantage of the cantilever beam craze test is that it does not simulate real world stress conditions. Aircraft

*Performed under Contract F33615-84-C-3404 for the Flight Dynamics Directorate, Wright Laboratory, Wright-Patterson Air Force Base, Ohio.

transparencies are typically under a biaxial state of stress.

This paper is divided into two parts. The first Part deals with a set of uniaxial craze tests, conducted to deterine whether craze is a stress or strain driven phenomenon. Part II details a set of biaxial and uniaxial craze tests run to evaluate chemical craze initiation criteria, and to evaluate the use of uniaxial craze tests to assess resistance of transparencies to craze.

Objective

The objective of Part I was to establish a basis for a craze initiation criteria in acrylic by determining whether craze is a stress or a strain dependent phenomenon. The objective of Part II was to investigate the relationship between uniaxial and biaxial chemical stress crazing of cast acrylic, and to develop a better understanding of the crazing phenomenon. The development of a relationship between uniaxial and biaxial crazing would validate the use of the inexpensive uniaxial chemical craze testing to evaluate the effects of various chemicals on aircraft transparencies.

Part I: Effect of Creep and Stress Relaxation on Crazing

Developing craze criteria in order to enhance aircraft transparency durability requires a fundamental understanding of the craze phenomena. The abundant literature which describes crazing agree that specific factors, such as temperature and solubility of the craze agent, influence crazing. Virtually all previous work also recognizes the effect of internal and external forces on craze [1]. However, past research on this effect is not consistent: some refer to the stress in the material, while others refer to the strain [2,3]. For a purely elastic material, the terminology is interchangeable, since the two quantities are related by a constant. For a material such as aircraft transparencies at service temperatures, the terms "stress" and "strain" cannot be interchanged because of the material's time dependent properties. This inconsistency represents a void in the basic understanding of craze. It inhibits development of a craze initiation criteria, since the basis for the criteria must first be established.

Part I Testing

To evaluate craze dependence on stress and strain as independent quantities, one quantity was held constant, while the other changed with time. A material held under constant stress will experience an increase in strain (creep), while a material held under a constant displacement will experience a decrease in stress (stress relaxation). Decreases in time-to-craze for material under constant stress would indicate a strain dependence of craze, since the strain increases with time. Increases in time-to-craze for material held to a constant displacement would indicate a stress dependence, since the stress decreases over time.

For the purposes of this discussion, the term "creep" specifically refers to the increase in strain experienced by a viscoelastic material under a constant stress. Creep is not the same as "stress relaxation," which refers specifically to a decrease in stress experienced by a viscoelastic material under constant strain.

Part I Test Specimens

Figure 1. shows dimensions for the dogbone specimens used in this study. Samples were machined from 3/16" Polycast acrylic "Poly 84." This material conforms to MIL-P-8184 and possesses low moisture absorbing properties. Butyl rubber applied to the test surface prevented the crazing solution from penetrating the edge of the sample and producing invalid edge initiated crazes. It also defined the test area for each craze test. Sanding of all specimen edges prior to testing also helped to prevent edge initiated crazing. Strain gages installed at the underside center of each specimen provided strain data during the test.

Part I Test Plan

Twenty tests were run under constant load, while five tests were run under constant strain. Table 1 gives the test matrix for the constant load tests. Table 2 shows the constant displacement test matrix. Creep data obtained by Bouchard [4] for laminated transparency material with acrylic surface plies formed the basis for stress level selection. These levels were also chosen to provide enough variation to make the creep effect on craze evident.

The length of each sample permitted 4 craze tests to be conducted on each sample. Each craze test was conducted after the sample had been allowed to creep for a specified length of time, as shown in Tables 1 and 2. Based on test results as the program progressed, some samples were allowed to creep for longer periods of time.

Based on information available in the literature [5], conversations with other workers in the field [6], and past craze testing at UDRI, the results were expected to show a craze dependence on *strain*. The high number of constant load tests reflects this expectation. The anticipated result of the constant displacement tests was simply verification of the strain dependence through no change in time-to-craze. Consequently, fewer constant displacement tests were conducted.

Part I Test Fixture and Hardware

Figure 2a displays the creep fixture employed to conduct the constant load tests. A lever arm, wire rope, and dead weight assembly applied the load to the specimens, which were rigidly fixed at the opposite end. The lever arm provided an 8x increase in the load applied to the specimen, thereby reducing the amount of dead weight required to run the test. Nylon blocks inserted between specimens prevented the samples from rotating as the load was applied.

Figure 2b shows the fixture used for the constant displacement tests. Samples were loaded between rigid aluminum plates by loading bolts. A full-bridge strain gage array installed on each load bolt provided load data during these tests. The gages were calibrated by the UDRI Structural Test Lab with standards traceable to NITS. The gages installed on the specimens monitored strain to ensure that load reduction during the test was not due to slip in the fixture.

A Measurements Group, Inc., P3500 Strain Indicator and SB10 Switch and Balance Unit provided measurement capabilities for both the specimen gages and the load bolt gages. Strain

gaged Polycast acrylic plates served as temperature compensation in the bridge circuitry for the specimens.

Part I Test Procedure

All tests were conducted at 70° F and 50% RH. Constant load and displacement tests were monitored for creep and stress relaxation, respectively, for ten minutes after the load or displacement was first applied. Strain (constant load test) and stress (constant displacement) were then monitored every 48-72 hours thereafter until the final craze test was conducted.

Craze tests were conducted roughly at the intervals shown in Tables 1 and 2. Isopropyl alcohol (99% by volume) was applied to the test area and reapplied when necessary to prevent evaporation from drying out the surface. Time-to-craze was defined as the time when crazing was first apparent by visual inspection. A Precision Instruments Co. timer, accurate to 0.1 seconds, provided accurate indication of the time for the craze to appear.

Each craze test was terminated when one of the following conditions occurred: (a) the first sign of crazing appeared in the test area; (b) edge craze (due to seeping under the butyl rubber) threatened to cause specimen failure; (c) a significant amount of time (generally 10,000 seconds) elapsed with no crazing apparent. The alcohol and butyl rubber were removed from the specimen and the time-to-craze (if any) noted. The strain (or load) of the specimen being craze tested was also recorded immediately after each craze test.

Part I Results and Discussion

Figures 3 shows creep data plotted for one sample from each group of stress levels tested in the constant load tests. Figure 4 shows stress relaxation for sample 1, 2, and 3 of the constant displacement tests. In general, Figure 3 shows creep ranging from 70% to 90% over the duration of the test. Figure 4 shows stress relaxation of 45% to 48%. The definition of percent creep is given by

$$\%\text{creep} = 100\left(\frac{\varepsilon_f - \varepsilon_i}{\varepsilon_i}\right) \tag{1}$$

in which ε_f is strain at any time and ε_i is the initial strain. Percent relaxation is defined in a similar manner:

$$\%\ \text{relaxation} = 100\left(\frac{\sigma_i - \sigma_f}{\sigma_i}\right) \tag{2}$$

The creep and relaxation appear to be quite considerable and represent significant changes in the strain or stress in the material over time.

Figures 5 shows time-to-craze data for the constant load tests. The graph categorizes the data parametrically into groups with the same nominal stress. Variations of the actual stress from

the nominal averaged 95 psi for all specimens. The scatter in Figure 5 is typical of crazing data and is a function of several factors, including variations in surface roughness and flaw size across the surface of the sample, and variations in test area. The data suggests that creep does not influence crazing to a significant degree, as the time-to-craze is relatively constant over time, while significant creep has occurred. The slight decrease in time-to-craze for some groups may suggest a weak dependence on stress, but the small number of tests and wide scatter prevent more detailed analysis of this trend.

Figure 6 displays time-to-craze data for the constant displacement tests. Results from all five samples are plotted independently. The initial strain in these samples represents a stress of approximately 2000 psi. The data show an increase in time-to-craze as the time under a constant displacement increases. Although the low number of samples may suggest this trend was simply due to the natural dispersion seen in craze data, the increase is much larger than the widest scatter band in the constant load data. The increase in time-to-craze was attributed to the relaxation of stress.

In both Figures 5 and 6, several of the data points indicate that tests were stopped before crazing initiated. These points were plotted to indicate that the lack of crazing in these tests supports the observed trends. In particular, several of the later tests in the 1500 psi stress group did not craze even after 14,000 seconds.

In comparing the creep curves and time-to-craze curves (Figures 3 and 5) , it appears that the greatest percentage of creep occurred within the first 200 hours under load. While a substantial number of data points exists prior to 200 hours, the majority represent tests conducted after 200 hours of creep. Future programs should consider conducting more tests sooner, or allowing the specimens to creep for a substantially longer periods of time. Stress relaxation data (Figures 6) past 800 hours were not available due to instrumentation problems, so a determination of percent relaxation past 800 hours is not possible. However, it does appear that craze times continue to rise as time under displacement increases, despite apparent reduction in the rate of stress relaxation. The small number of data points prevents the formulation of a more specific relationship between time-to-craze and stress.

Part I Conclusions

The craze tests conducted during this program suggest that criteria for craze initiation in acrylic should be based on the stress in the material. However, a statistically small number of tests were conducted, preventing more detailed evaluation of the relationships between time-to-craze and stress.

The general conclusion has several implications for loading mechanisms in aircraft transparencies that are considered craze inducing. Constant displacement states, such as installing a canopy or windshield into an aircraft, may not be primarily responsible for craze since the stress associated with installation presumably relaxes over time. Residual stresses due to manufacturing may also relax over time and therefore become less significant to crazing. Crew module pressurization may increase craze susceptibility in a transparency if the induced stress is high

enough, but the resulting creep will have no effect.

The result also has several implication for the criteria itself. A stress formulation implies that craze is a function of the strain energy in the material. As the stress relaxes, the strain energy is dissipated by viscous flow at the molecular level. Extending an energy based criteria to biaxial stress implies a tension/compression field is more susceptible to craze than a tension/tension field. It also implies that maximum principal stress may not be a good criterion for predicting craze initiation, since the presence of a second stress field alters strain energy in the material. A tension/compression stress state produces more strain energy than a uniaxial tension state, and therefore may increase craze susceptibility.

A chemical craze test has been developed to evaluate the effect of biaxial stresses on crazing, using a circular plate with clamped edges and a uniform pressure load. While this biaxial craze specimen is more simple to fabricate, test, and analyze than those used by other researchers to study biaxial crazing, the test is more complicated and more time consuming than the uniaxial craze test and requires special fixturing.

PART II: BIAXIAL TESTING AND CRAZE INITIATION CRITERIA

This Part of the program consisted of craze initiation theory development and craze testing. A series of uniaxial and biaxial craze tests was conducted at various stress levels in conjunction with isopropyl alcohol. Isopropyl alcohol was the chosen chemical craze agent because it is a representative chemical which is often used for cleaning of aircraft transparencies. The results of this testing were analyzed to develop craze initiation criteria which apply to uniaxial and biaxial crazing. This effort has also been published as Reference 7.

Theoretical Development of Craze Initiation

Craze initiation criterion are analogous to stress yielding criterion. Stress yielding criterion describe the necessary conditions (state of stress/strain) for yielding to occur. Stress yielding criterion which may apply to chemical stress crazing include: maximum principal stress, maximum principal strain, maximum shear stress (Tresca), distortional energy (von Mises), strain energy, and combinations of these, deviatoric stresses, and/or flow stresses. These yielding criterion were considered as a starting point for the development of chemical stress crazing criterion.

While there is extensive information in the literature concerning stress yielding criterion (although most of it has not been applied specifically to polymers), there is limited information available in the literature concerning chemical stress crazing of polymers. The majority of the research which has been conducted has been concerned only with stress crazing, not chemical stress crazing. Two basic craze initiation criteria have been proposed. Sternstein and Ongchin [8] proposed a critical stress bias criterion for surface stress crazing of polymethylmethacrylate (PMMA, acrylic) as follows:

$$\sigma_1 - \sigma_2 \geq \frac{A}{(\sigma_1 + \sigma_2)} + B \tag{3}$$

where σ_1 and σ_2 are the principle biaxial stresses, and A and B are functions of time and temperature. The difference between σ_1 and σ_2 represents a stress bias or flow stress (this is equal to twice the maximum shear stress), and the quantity of $\sigma_1 + \sigma_2$ represents twice the first stress invariant or the mean stress. This criterion, along with the von Mises criterion for yielding (which has been shown by the same authors to be fairly representative of yielding behavior for acrylic) is plotted in biaxial stress space in Figure 7. Sternstein and Ongchin based their conclusions on cylindrical specimens under tension with internal pressure, and on combined tension/torsion tests, all at elevated temperatures (50°, 60°, and 70°C).

A second similar criterion, based on critical strain, has been developed by Oxborough and Bowden [9] for polystyrene, as follows:

$$\sigma_1 - \mu\sigma_2 = \frac{A}{(\sigma_1 + \sigma_2)} + B \tag{4}$$

The only difference between this and the previous criterion is that the left side of the equation represents the maximum strain in this case, where μ is Poisson's ratio. Oxborough and Bowden based their conclusions on combined tensile and compressive tests, at room temperature, conducted on rectangular annealed polystyrene specimens with a hole in the center. This criterion plotted in stress space is similar to Figure 7.

Uniaxial Chemical Craze Testing

Uniaxial chemical stress craze testing was conducted using ASTM F484-83 as a guideline. The craze beam specimens were 1 inch x 7 inch x 1/8 inch thickness. Polycast Mil-P-8184 Type II (low moisture uptake) cast acrylic from the same lot was used for all testing. The craze tests were conducted at 75 ± 10°F. The cantilever craze beams were loaded to produce a maximum stress at the fulcrum of 2000, 3000 and 4000 psi. The underside of the beams were marked at 0.25 inch intervals. After the load was applied, the beams were allowed to stabilize for ten minutes before the test chemical was applied to the beam surface. The edges of the beams were protected with a butyl rubber sealant to prevent the chemical from coming in contact with the machined or cut edges and causing premature crazing. Isopropyl alcohol (99% pure) was applied to the top surface of the beams as required to maintain a wetted condition. Time to craze initiation and location (corresponding to a discrete stress level) were recorded during the tests. The uniaxial chemical craze test setup is shown in Figure 8.

Notice that using craze beams in uniaxial testing is fundamentally different than using tensile bars, as in Part I. The craze beams have a stress gradient across the surface, while the tensile bars have a constant stress. The area available to craze at any given stress level is infinitesimal for the craze beams, while the area is finite for tensile bars. One would expect different craze times for these types of specimens due to the area effect, even if material and craze

chemical are identical.

The results of the uniaxial craze tests are summarized in Figure 9. The uniaxial craze results plotted in Figure 9 indicate that there is a linear relationship between the log of time to initiation of craze and applied stress.

Biaxial Chemical Craze Testing

The biaxial craze tests were conducted using the general guidelines of ASTM F1164-88. The biaxial craze specimens were 8.5-inch diameter, 3/16-inch thick plate specimens. Polycast Mil-P-8184 Type II (low moisture uptake) cast acrylic from the same lot was again used for all testing. The test fixturing included a pressure cell, a precision pressure regulator, and a pressure test gauge with accuracy of 0.075 psi. The test setup is shown in Figure 10. The pressure in the test cell was used to induce equal principal biaxial stresses of 2000, 3000, and 4000 psi at the center of the plates. Concentric rings were drawn on the underside of the plate to facilitate location of the crazes. The components of the principal stresses (the radial and tangential stresses) were determined from:

$$\sigma_r = \frac{3pR^2}{8t^2}\left[-(1+\mu)+(3+\mu)\frac{r^2}{R^2}\right] \tag{5}$$

$$\sigma_t = \frac{3pR^2}{8t^2}\left[-(1+\mu)+(1+3\mu)\frac{r^2}{R^2}\right] \tag{6}$$

where:

σ_r = radial stress (psi)

σ_t = tangential stress (psi)

R = plate radius (inches)

t = plate thickness (inches)

μ = Poisson's ratio

p = pressure (psi)

r = radial dimension from center to point of interest (inches)

Figure 11 is a plot of the radial and tangential components of the stress in the biaxial plate specimen. After the pressure load was applied to the plate, the plates were allowed to stabilize for ten minutes before the test chemical was applied. Isopropyl alcohol (99% pure) was applied to the top surface of the plates as required to maintain a wetted condition. Time to craze initiation and location (corresponding to a discrete stress condition) were recorded during the tests.

The biaxial and uniaxial test data is presented in Figure 12. A typical tested biaxial

specimen is shown in Figure 13. It is believed that the spread in the data is due, in part, to the fact that each plot does not represent a discreet instant in time, but represents a time interval.

All of the types of yield initiation criterion listed above evaluated. None of these criterion fit the data in the forms that they have been used to describe yielding. The elliptical shape of the von Mises and strain energy criterion showed promise, but did not fit the uniaxial and biaxial data generated by test.

Equations 3 and 4 which are semi-empirical, were also evaluated. Because of the limitations of biaxial stress combinations which can be obtained from the biaxial plate specimens (the biaxial plate is only effective for measuring tensile-tensile stress loads of limited combinations; see Figure 11) it is difficult to determine by visual inspection if the shape of the craze initiation surface in stress space is cusp shaped as shown in Figure 7, or if it is some other shape.

The parameters A and B from equations 3 and 4 were determined as follows:

For the uniaxial stress state, Equation 3 (stress bias criterion) reduces to

$$\sigma = \frac{A}{\sigma} + B \tag{7}$$

A least square fit of the data in Figure 9 provides a relationship between time to craze and uniaxial stress

$$\log t = 3.5057 - 7.711\times10^{-4}\sigma \tag{8}$$

or, rearranging to solve for stress in terms of time,

$$\sigma = \frac{(3.5057 - \log t)}{7.7113\times10^{-4}} \tag{9}$$

Substituting Equation 9 into Equation 7, and solving for B,

$$B = \frac{(3.5057 - \log t)}{7.7113\times10^{-4}} - \frac{A}{(3.5057 - \log t)\left(7.7113\times10^{-4}\right)} \tag{10}$$

Equation 10 is then substituted into Equation 3, leaving A, σ_1, and σ_2 as the only unknowns.

$$\sigma_1 - \sigma_2 \geq \frac{A}{(\sigma_1 + \sigma_2)} + \frac{3.5057 - \log t}{7.7113\times10^{-4}} - \frac{A}{(3.5057 - \log t)\left(7.7113\times10^{-4}\right)} \tag{11}$$

Equation 11 is rearranged to solve for *A*, and the biaxial test data is then input into the equation to determine *A* for each test data set σ_1, σ_2, and time *t*. The corresponding value for *B*

is determined from Equation 10. The values of A and B are then plotted versus time, see Figures 14 and 15, and a least square fit provides a relationship between the value A and time, and the value B and time. Note that the coefficients of determination, R, for A and B are shown on Figures 14 and 15. The coefficient of determination, R, is a measure of the standard error associated with the least square fit to the data. Possible values range from 0 to 1. The closer the R value is to 1, the smaller the standard error is for the straight line fit to the data. Equation 3 (stress bias criterion) plotted in the first quadrant of stress space (tension-tension) with the functions for A and B shown in Figures 14 and 15, is shown in Figure 16.

The parameters A and B for Equation 4 (maximum strain criterion) are solved for in a similar manner and, along with corresponding R values, are shown in Figures 17 and 18. Equation 4 (maximum strain criterion), plotted in the first quadrant of stress space with the functions for A and B shown in Figures 17 and 18, is shown in Figure 19.

Most accepted yield criterion are elliptical in shape (e.g., von Mises and strain energy). In fact, the plots of biaxial and uniaxial results for the later time periods (after 15 minutes) appear to be elliptical shaped. The general formula for an ellipse oriented at 45° to the x and y axis is

$$\frac{\sigma_1^2 + 2\sigma_1\sigma_2 + \sigma_2^2}{A^2} + \frac{\sigma_1^2 - 2\sigma_1\sigma_2 + \sigma_2^2}{B^2} = 2 \qquad (12)$$

where the parameters A and B are functions of time. A and B are solved for in a manner similar to that shown above. The parameters A and B are plotted versus time in Figures 20 and 21. A family of empirical elliptical shaped craze initiation criteria curves, plotted using Equation 16 and the equations for A and B shown in Figures 20 and 21, are shown in Figure 22. A plot of this craze initiation criteria in biaxial stress and time space is shown in Figure 23. This surface represents the threshold between uncrazed and crazed material. Inside the surface there is not sufficient energy to cause crazing. The craze surface (and condition) can be reached by increasing the available energy; the available energy is increased by moving up the time scale, increasing the stresses, and/or increasing the temperature.

Table 3 presents the equations for each of the three proposed criterion, the values of the parameters for each equation, and the corresponding coefficient of determination, R, for each parameter. The elliptical stress craze initiation criterion provides the best fit to the data obtained, with R values for the two parameters of 0.8 and 0.9.

Part II Discussion, Conclusions, and Recommendations

The results of this program indicate that there is a definite relationship between uniaxial and biaxial chemical stress crazing with isopropyl alcohol. Three possible chemical stress crazing criterion have been presented. Two represent adaptations of criterion which have been developed for pure stress crazing (where the craze agent is air), and the third criterion represents an empirical elliptical criterion. The elliptical craze initiation criterion provided the best fit to the data obtained.

The choice of a circular plate specimen prevented studying craze in all regions of the biaxial stress state. Even though the biaxial craze specimen design used in this effort is more simple to fabricate, test, and analyze than those used by other researchers to study biaxial crazing, it is not possible to study all of the combinations of principle biaxial stresses of interest with this specimen. Therefore, a different type of specimen is required for future analysis of biaxial craze. To better define a multiaxial chemical stress crazing criterion, other tests should be conducted, with different combinations of principle tensile stresses, and with combinations of tensile and compressive stresses.

It is recommended that future work also include analysis of the effects of other chemicals (in addition to isopropyl alcohol) on crazing. In addition to conducting more tests with different combinations of biaxial stresses and with different chemicals, it is recommended that future work also take into account area effects. The testing on this program was conducted with time to initiation as the measured parameter. If future testing were to be conducted with the measured parameter being time to a specified craze density (i.e. number of crazes per surface area) instead of time to initiation of first craze, it would allow a better comparison of different types of tests. Time to initiation of first craze is a function of the surface area at a given stress level. Crazing occurs sooner on larger areas than smaller ones. The cantilever beam has a given surface area of material at each stress level, while the area at each stress level for the biaxial plate specimen is a function of the radial location in the plate and is not equal to the area for the cantilever beam specimen. In general, area effects have been ignored by researchers.

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Table 1. Constant Load Test Matrix

Number of Samples	Nominal Stress (psi)	Nominal Time Under Load (hrs)			
		Test 1	Test 2	Test 3	Test 4
5	1500	24	168	672	1344
5	2000	24	168	672	1344
5	2500	24	168	672	1344
5	3000	24	168	672	1344

Table 2. Constant Displacement Test Matrix

Number of Samples	Nominal Strain	Nominal Time Under Strain (hrs)			
		Test 1	Test 2	Test 3	Test 4
5	1500	24	168	672	1344

Table 3. Summary of Proposed Craze Initiation Criterion

Craze Initiation Criterion	Equation	Parameter A	R*	Parameter B	R*
Stress Bias Criterion	$\sigma_1-\sigma_2 \geq \frac{A}{(\sigma_1+\sigma_2)}+B$	$A=5.115\times10^7-1.881\times10^7 \log t$	0.730	$B=-6.8519\times10^3+1.8488\times10^2 \log t$	0.033
Max Strain Criterion	$\sigma_1-\mu\sigma_2=\frac{A}{(\sigma_1+\sigma_2)}+B$	$A=2.1768\times10^7-6.3262\times10^6 \log t$	0.446	$B=-1.7425\times10^{-2}-1.3776\times10^3 \log t$	0.326
Elliptical Criterion	$\frac{\sigma_1^2+2\sigma_1\sigma_2+\sigma_2^2}{A^2}+\frac{\sigma_1^2-2\sigma_1\sigma_2+\sigma_2^2}{B^2}=2$	$A=5.357\times10^3-2.028\times10^3 \log t$	0.910	$B=4.095\times10^3-8.95\times10^2 \log t$	0.798

*Coefficient of Determination

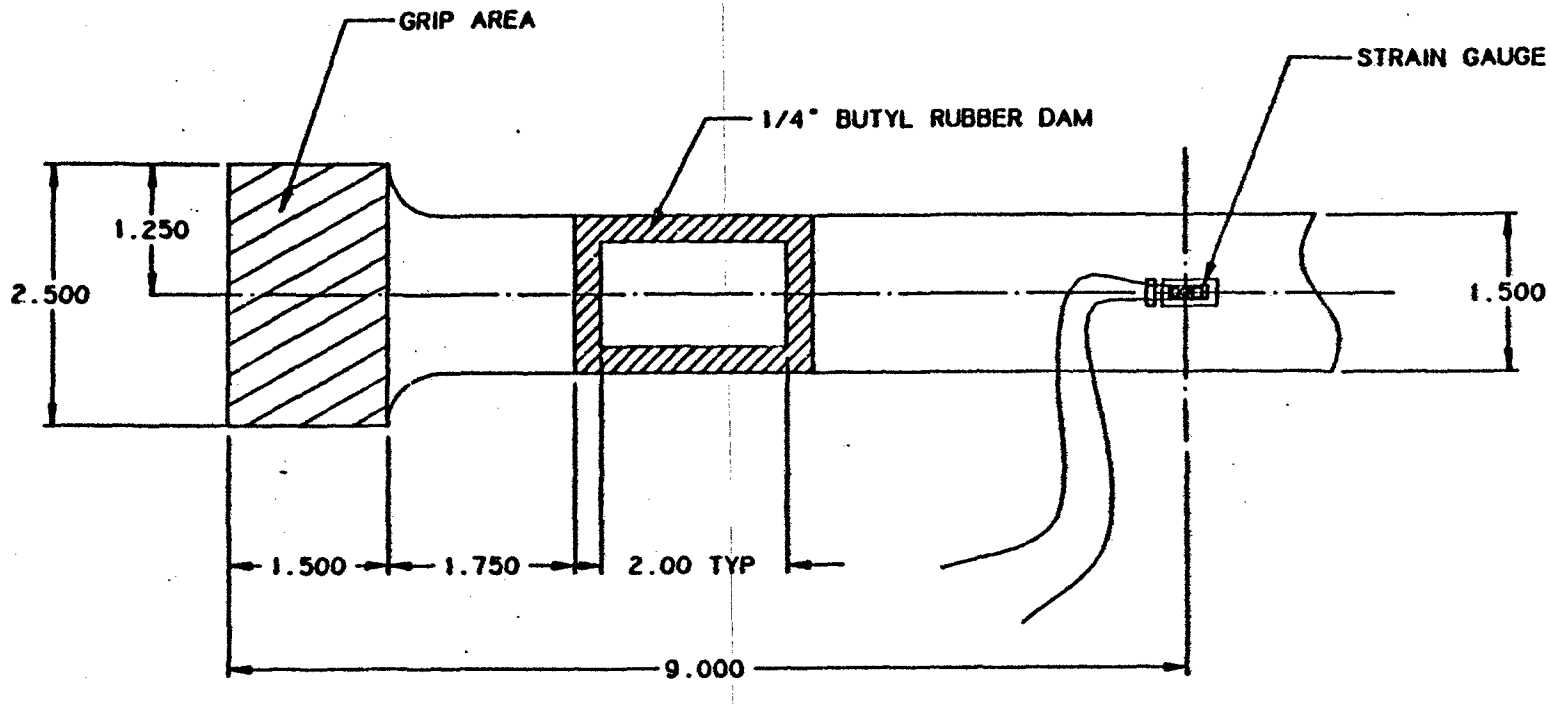


Figure 1. Schematic of Craze Test Sample

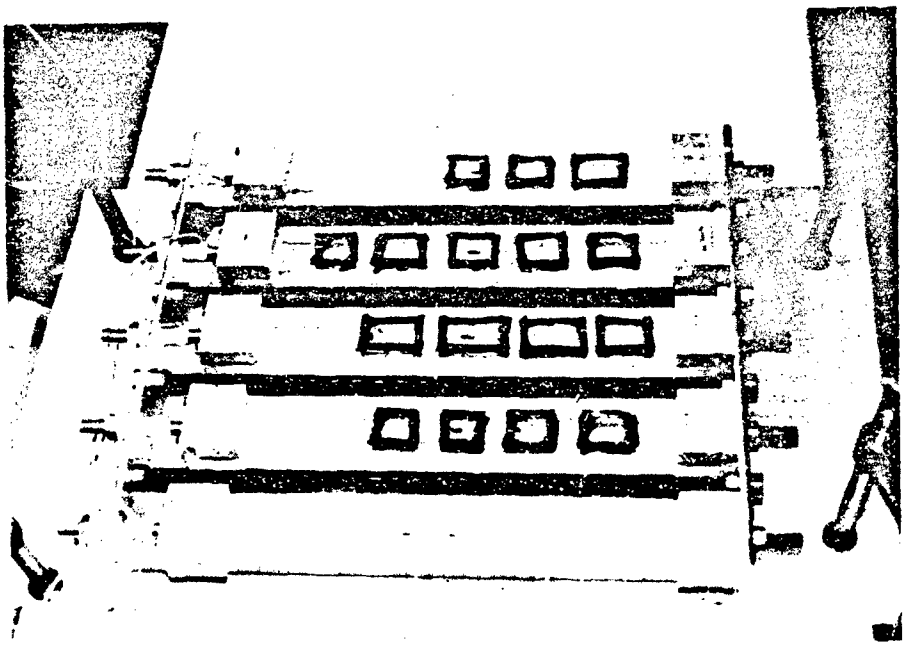


Figure 2a. Creep Fixture

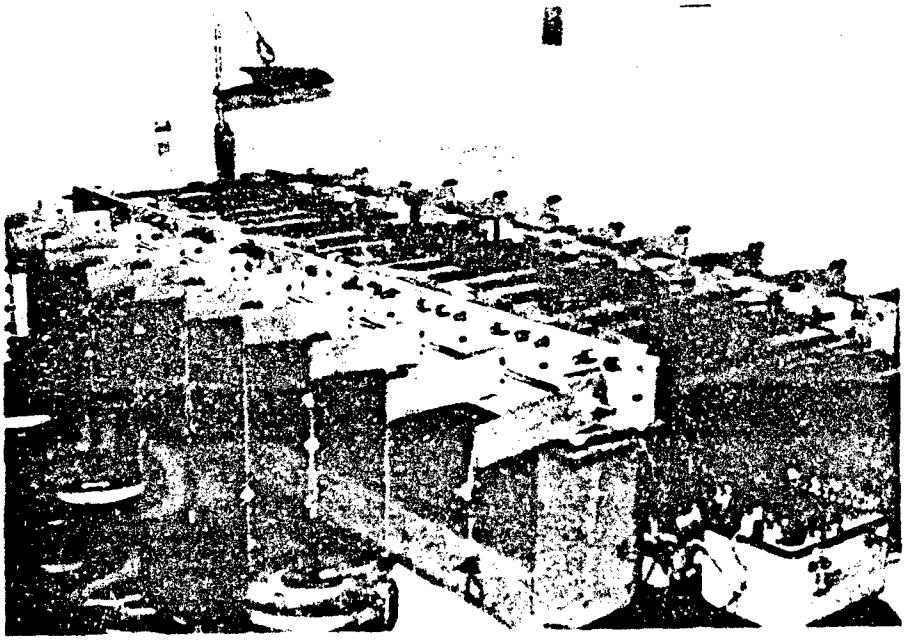


Figure 2b. Stress Relaxation Fixture

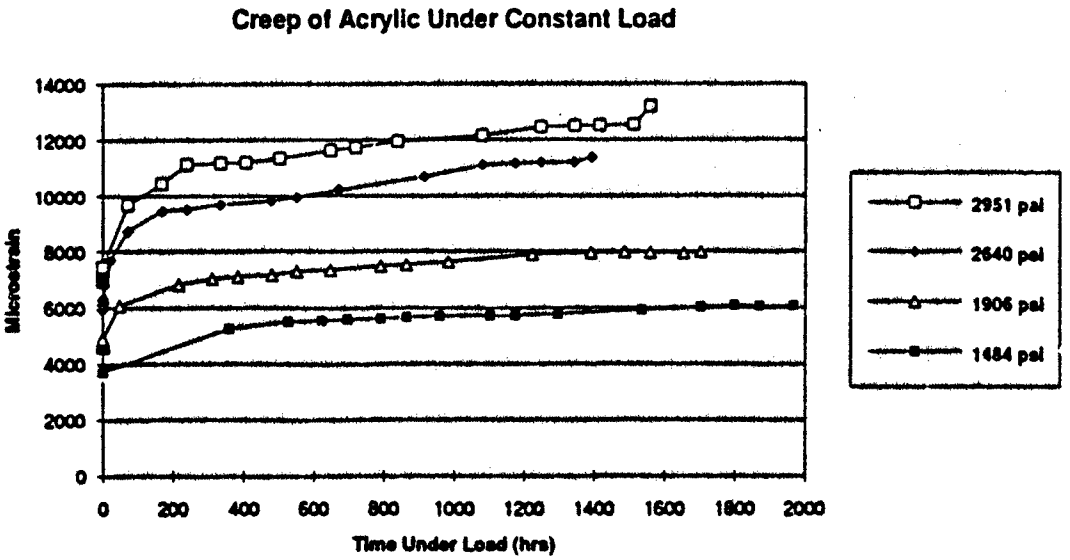


Figure 3. Creep Results from Constant Load Tests.

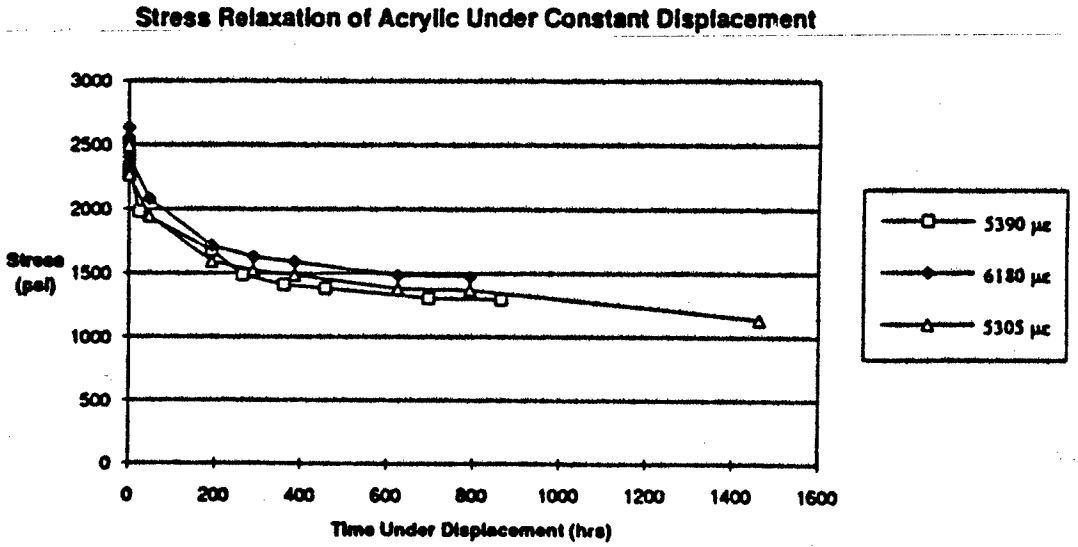


Figure 4. Stress Relaxation Results for Constant Displacement Tests.

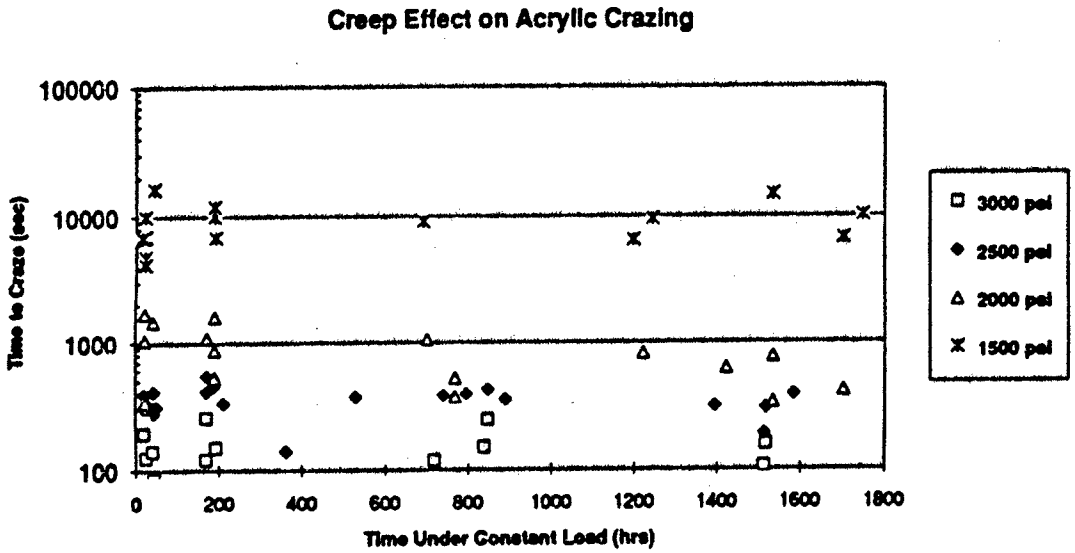


Figure 5. Craze Results for Constant Load Tests.

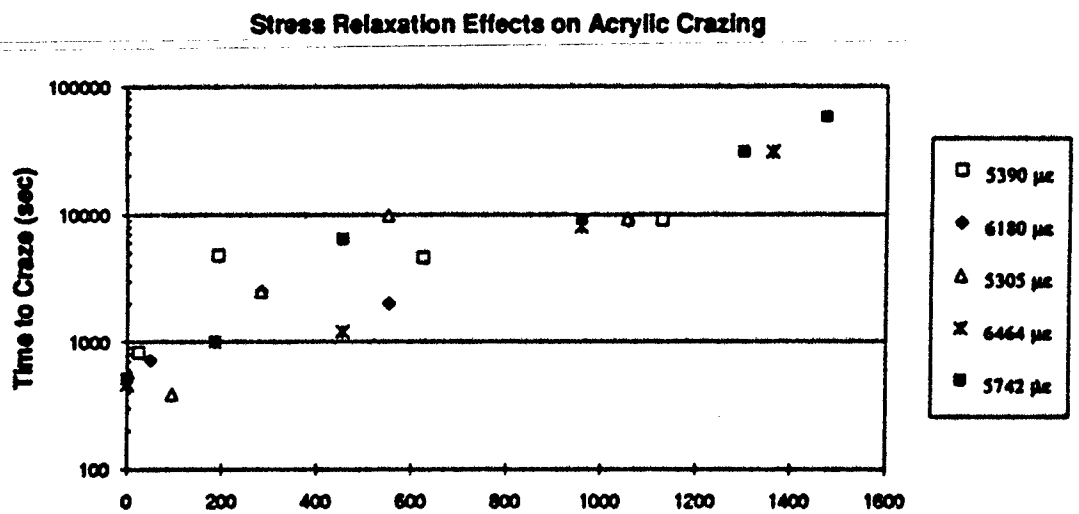


Figure 6. Craze Results for Constant Displacement Tests.

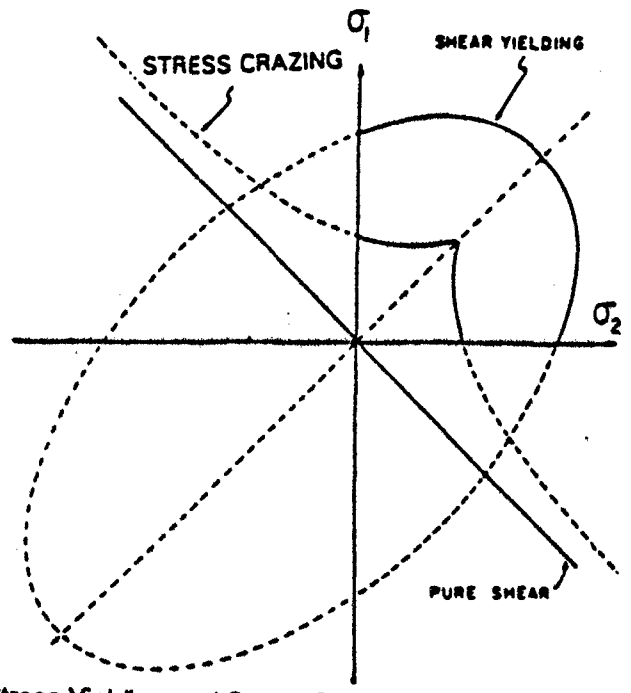


Figure 7. Biaxial Stress Yielding and Stress Crazing Curves for PMMA (from Ref. 8).

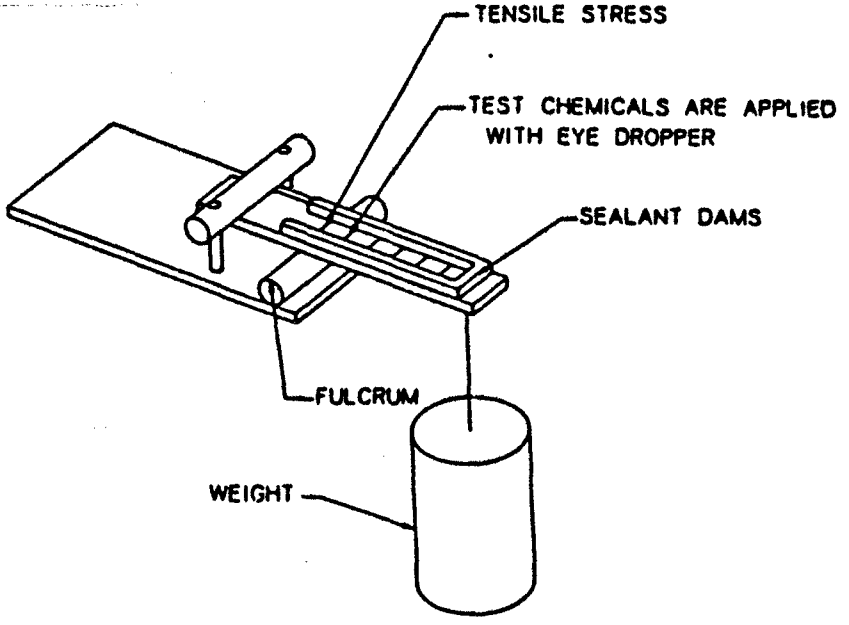


Figure 8. Cantilever Beam Test Setup.

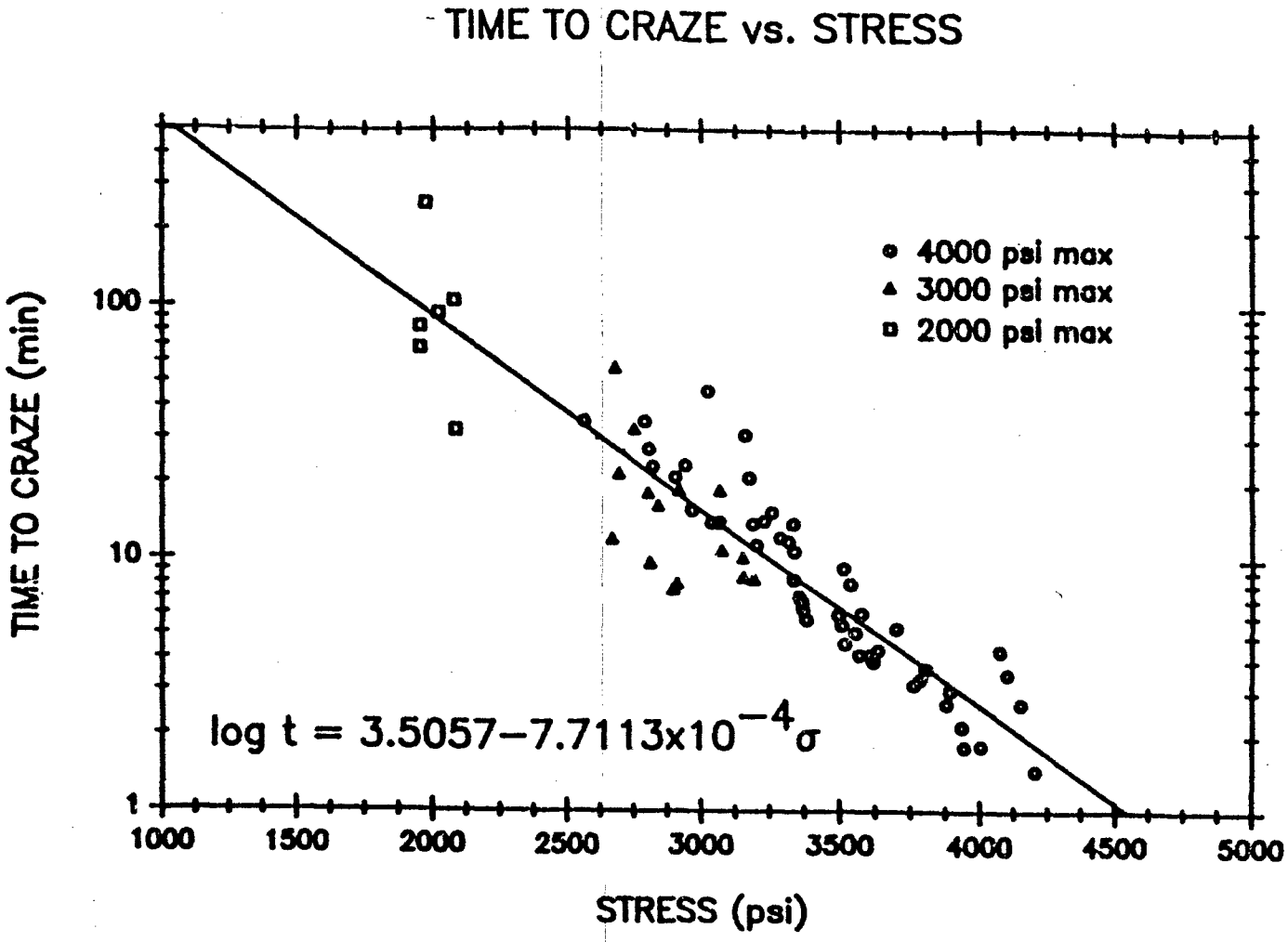


Figure 9. Uniaxial Craze Test Results.

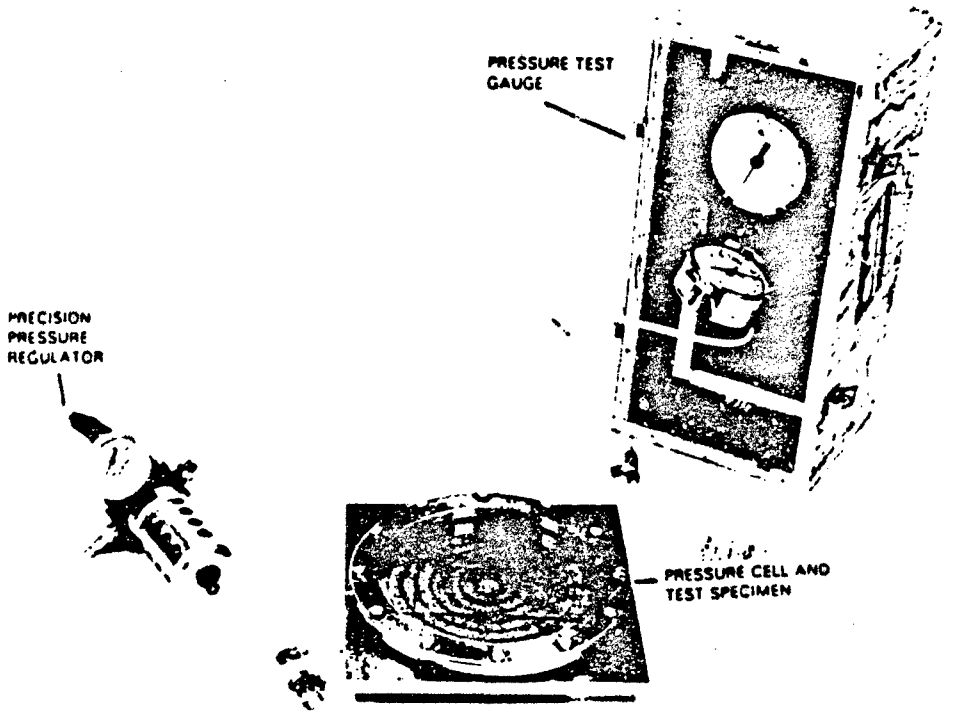


Figure 10. Biaxial Chemical Stress Craze Test Fixture.

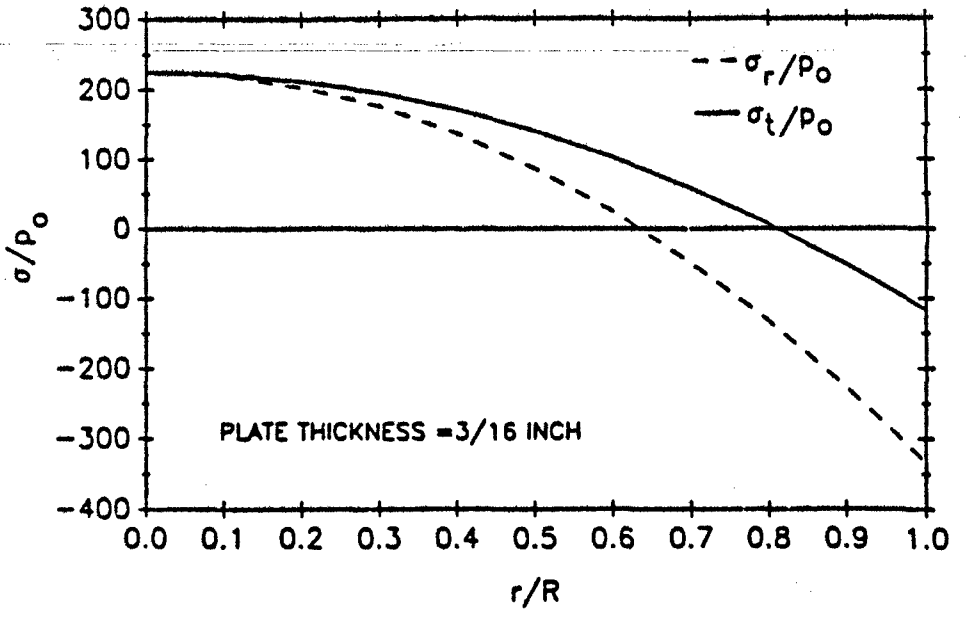


Figure 11. Radial and Tangential Components of the Stress in the Biaxial Plate Specimen.

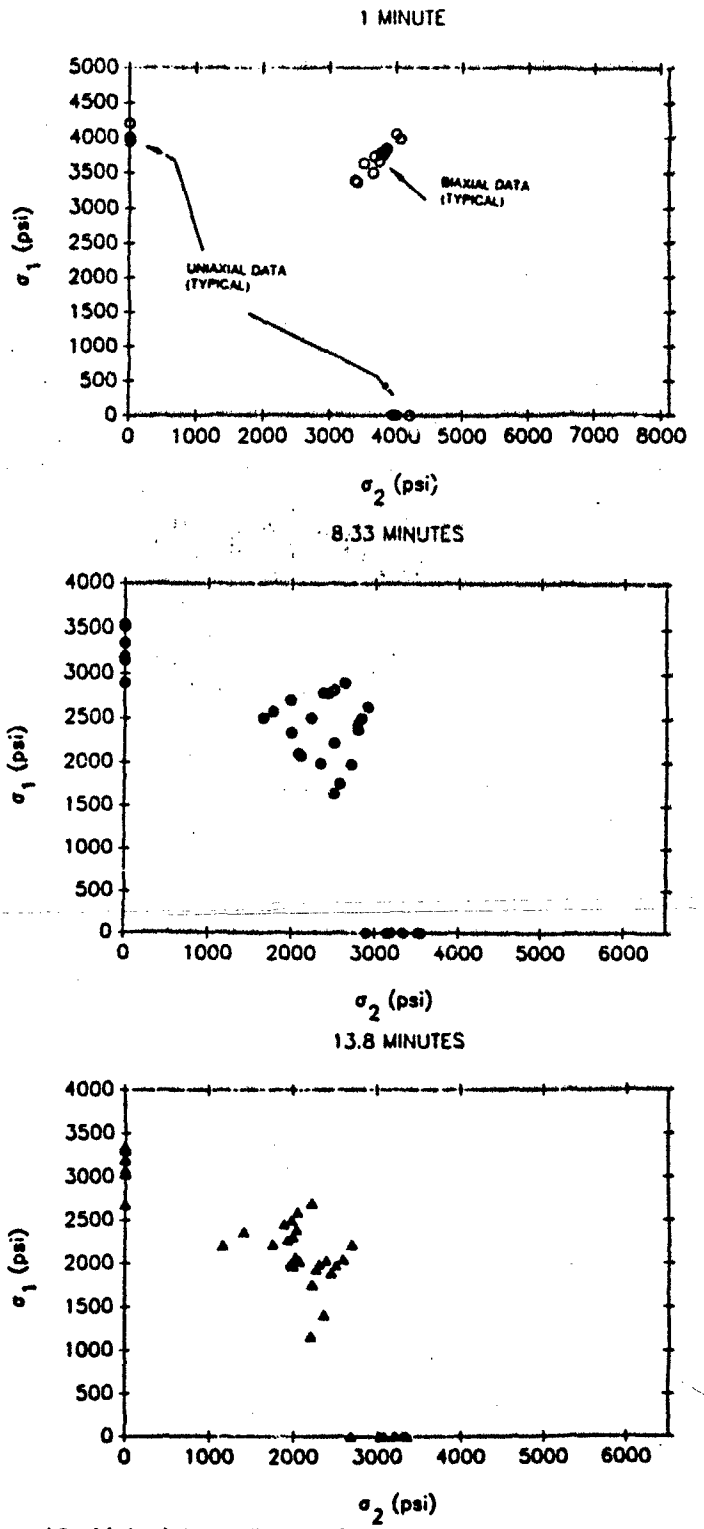


Figure 12. Uniaxial and Biaxial Stress Craze Data at Various Times.

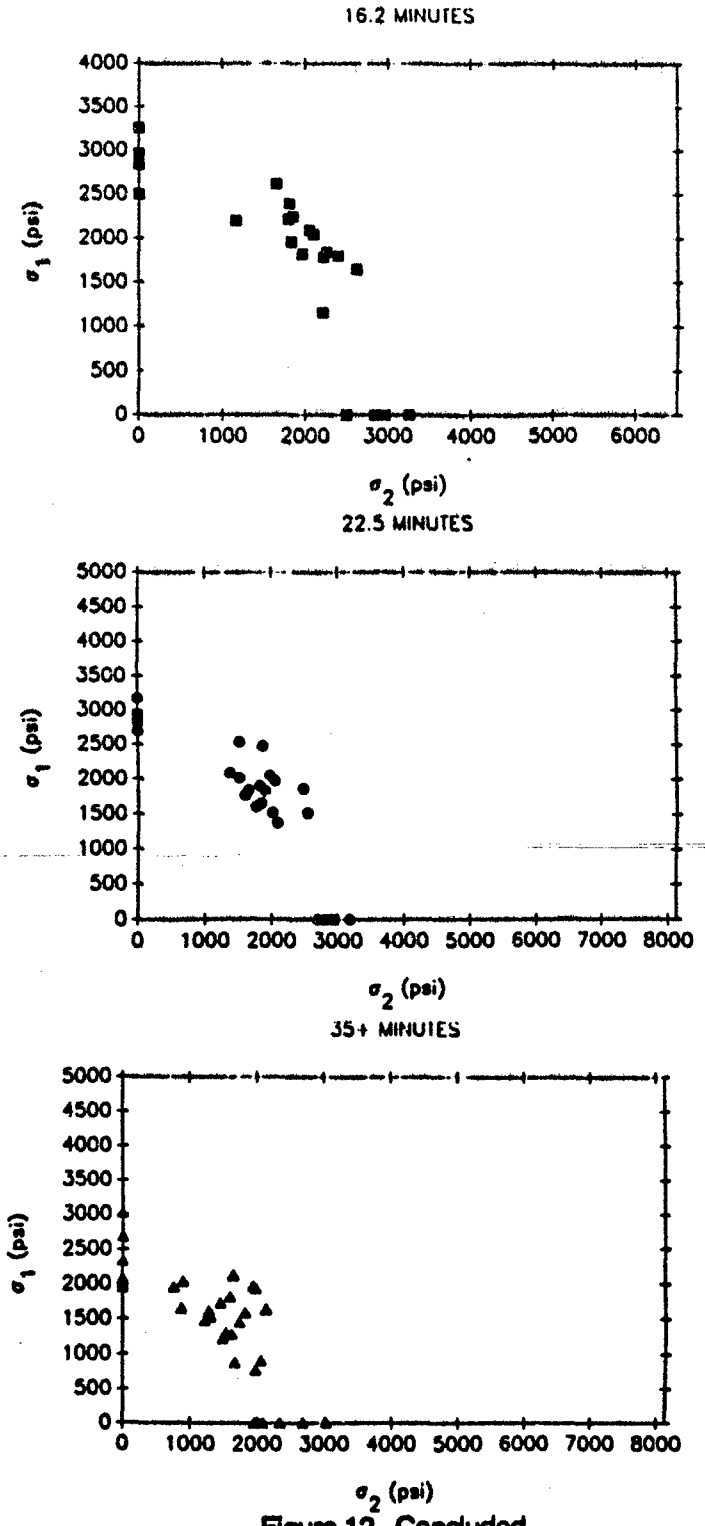


Figure 12. Concluded.

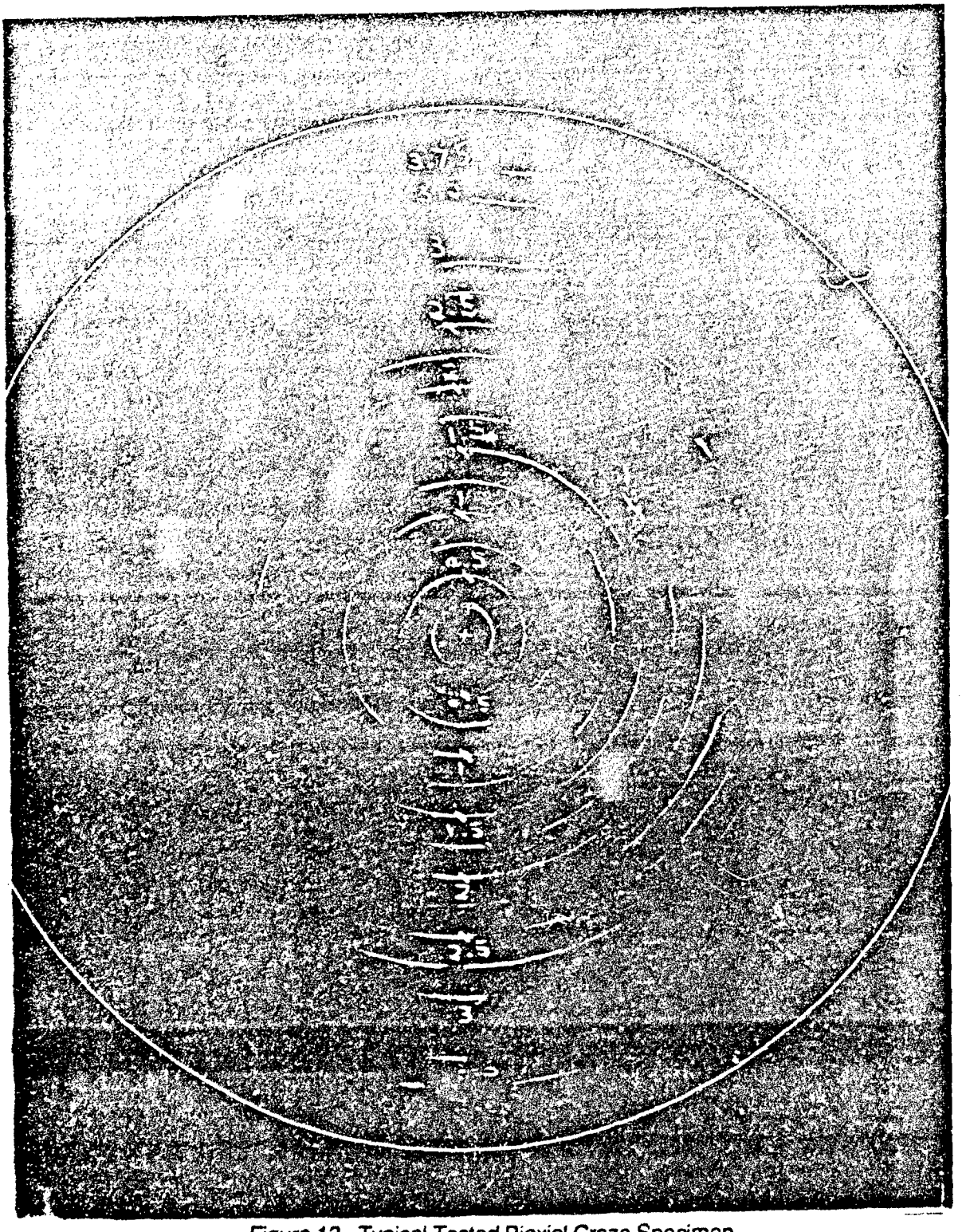


Figure 13. Typical Tested Biaxial Craze Specimen.

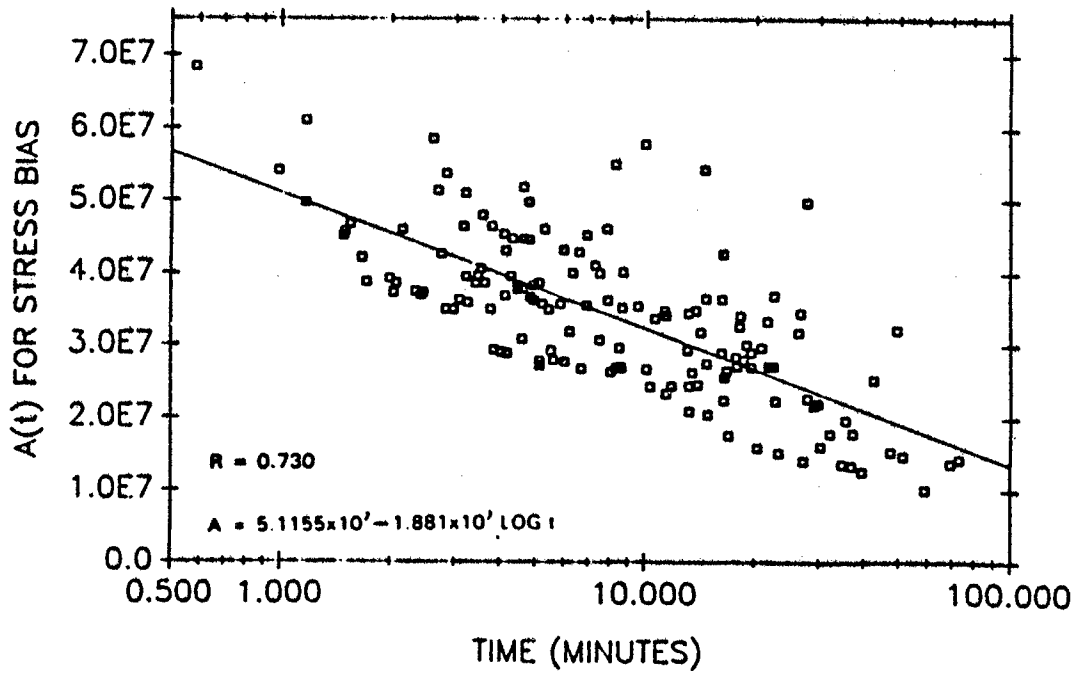


Figure 14. Parameter A as a Function of Time for Stress Bias Criteria.

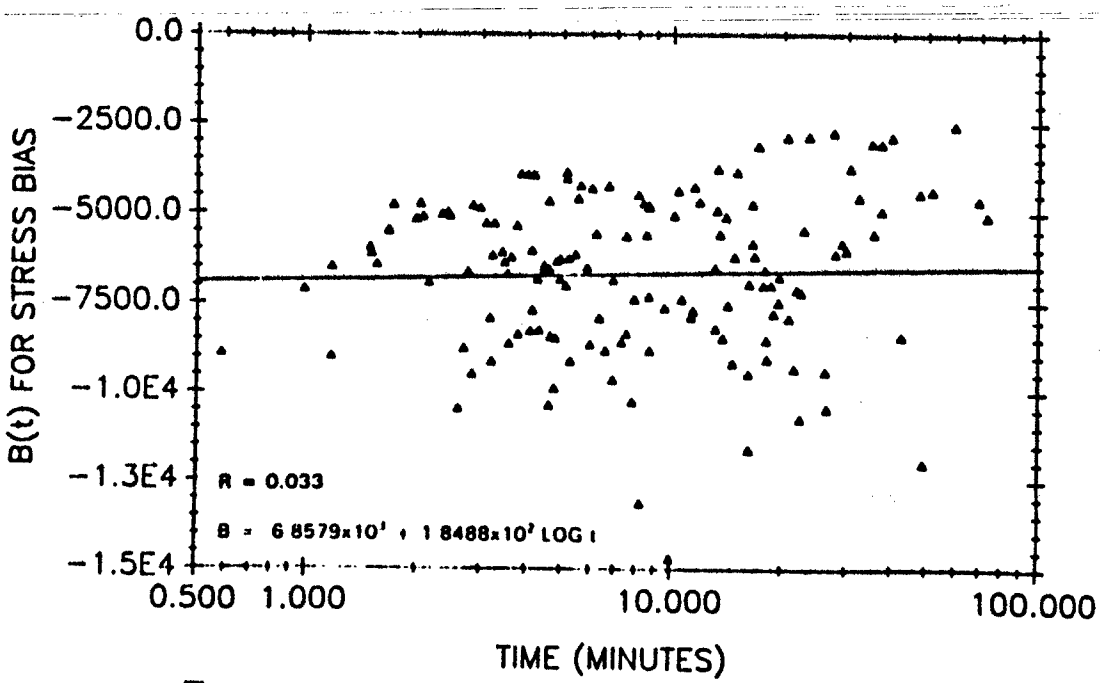


Figure 15. Parameter B as a Function of Time for Stress Bias Criteria.

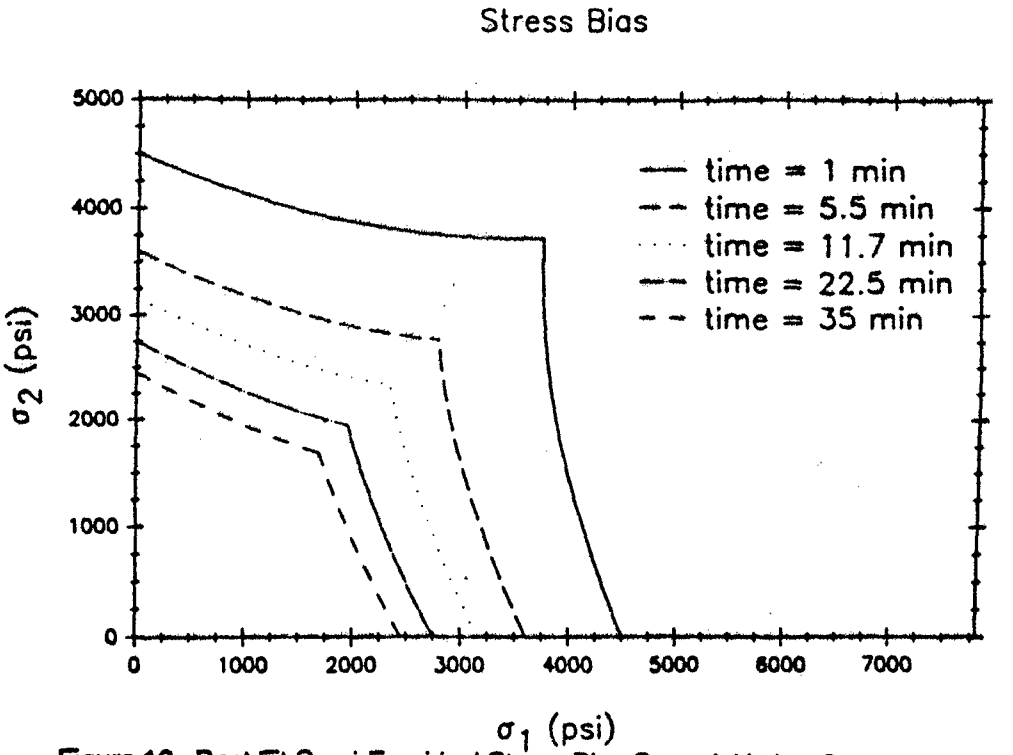


Figure 16. Best Fit Semi-Empirical Stress Bias Craze Initiation Criteria for Uniaxial and Biaxial Chemical Craze Data.

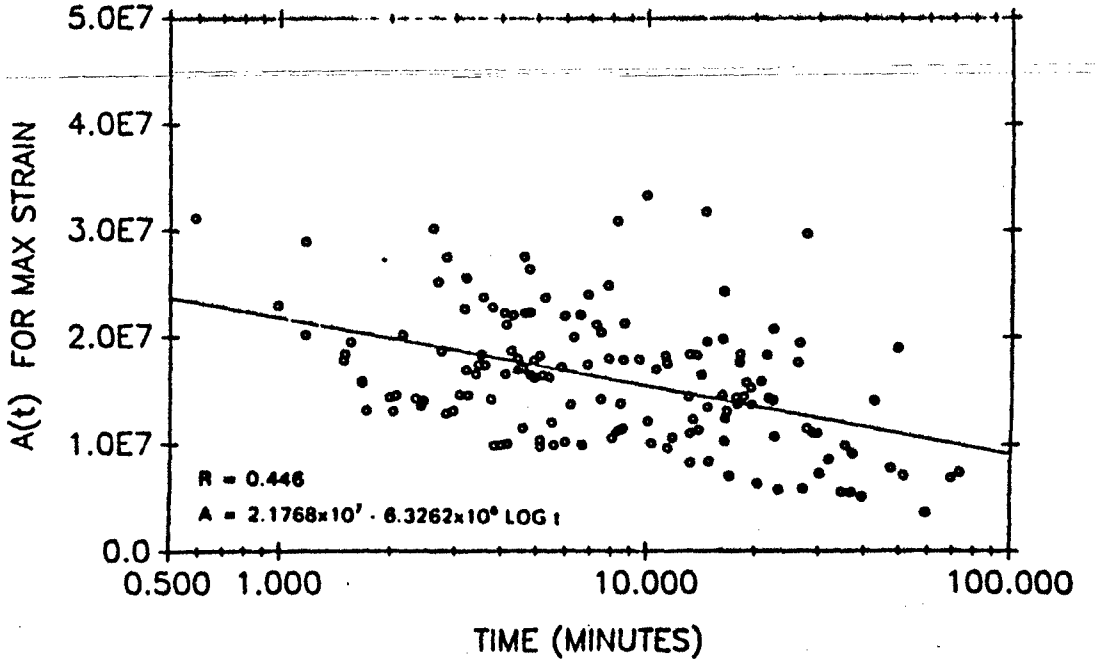


Figure 17. Parameter A as a Function of Time for Maximum Strain Criteria.

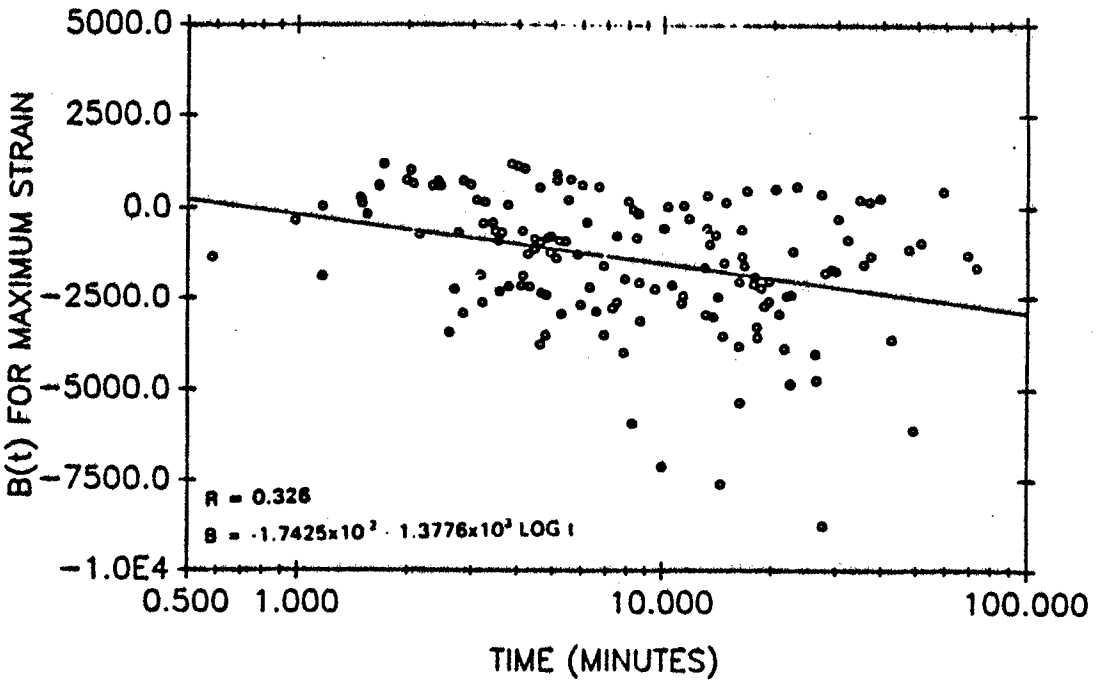


Figure 18. Parameter B as a Function of Time for Maximum Strain Criteria.

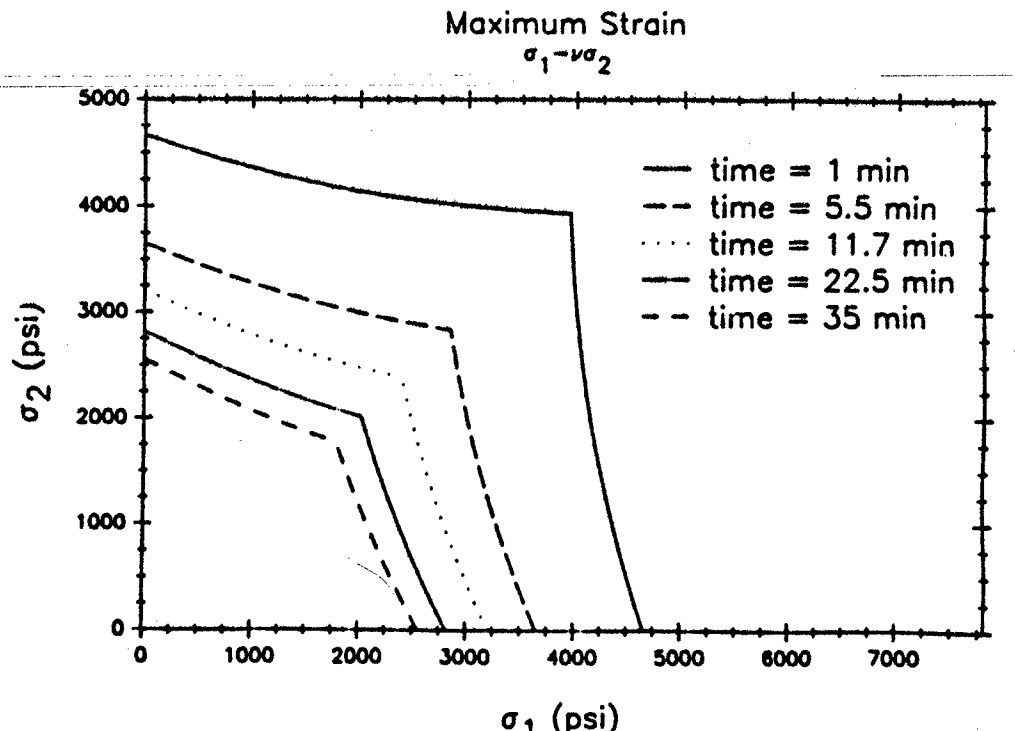


Figure 19. Best Fit Semi-Empirical Maximum Strain Craze Initiation Criteria for Uniaxial and Biaxial Chemical Craze Data.

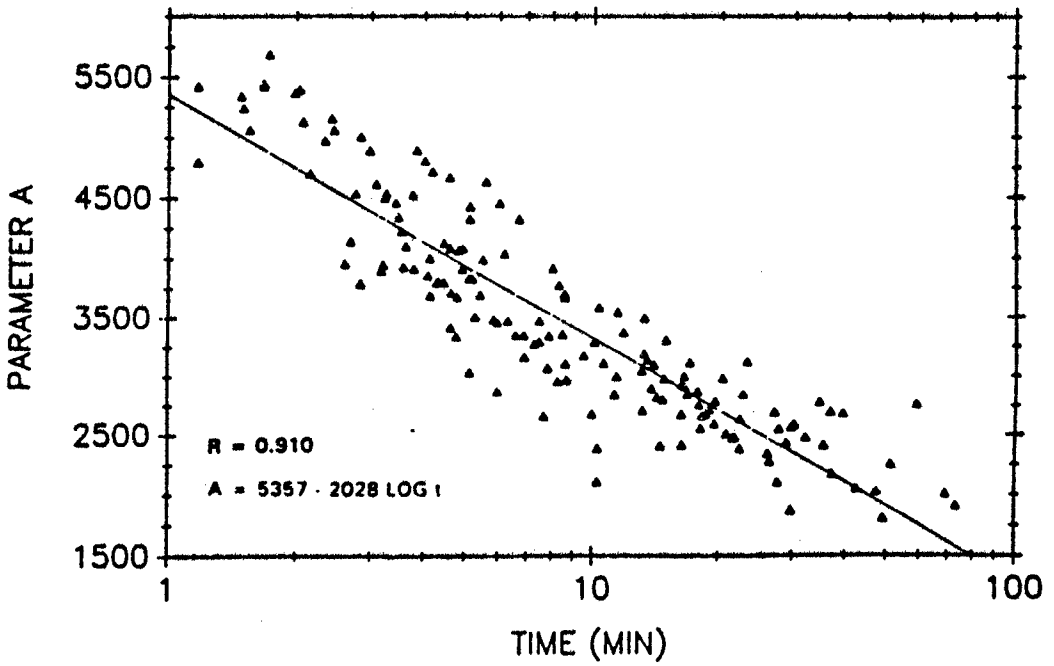


Figure 20. Parameter A as a Function of Time for Elliptical Shaped Craze Initiation Criteria.

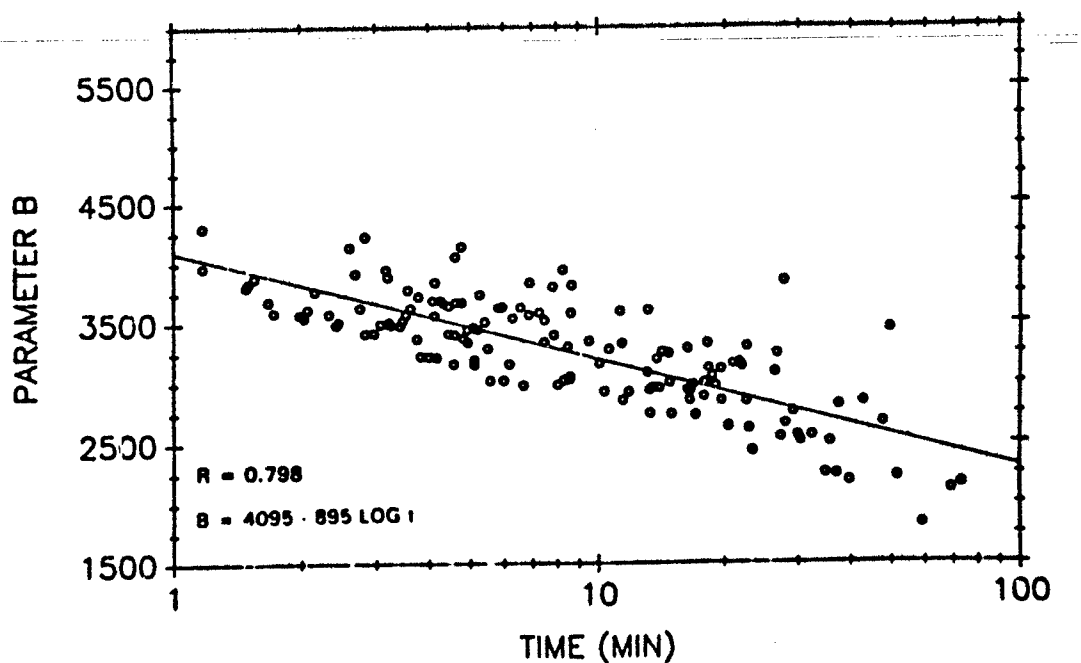


Figure 21. Parameter B as a Function of Time for Elliptical Shaped Craze Initiation Criteria.

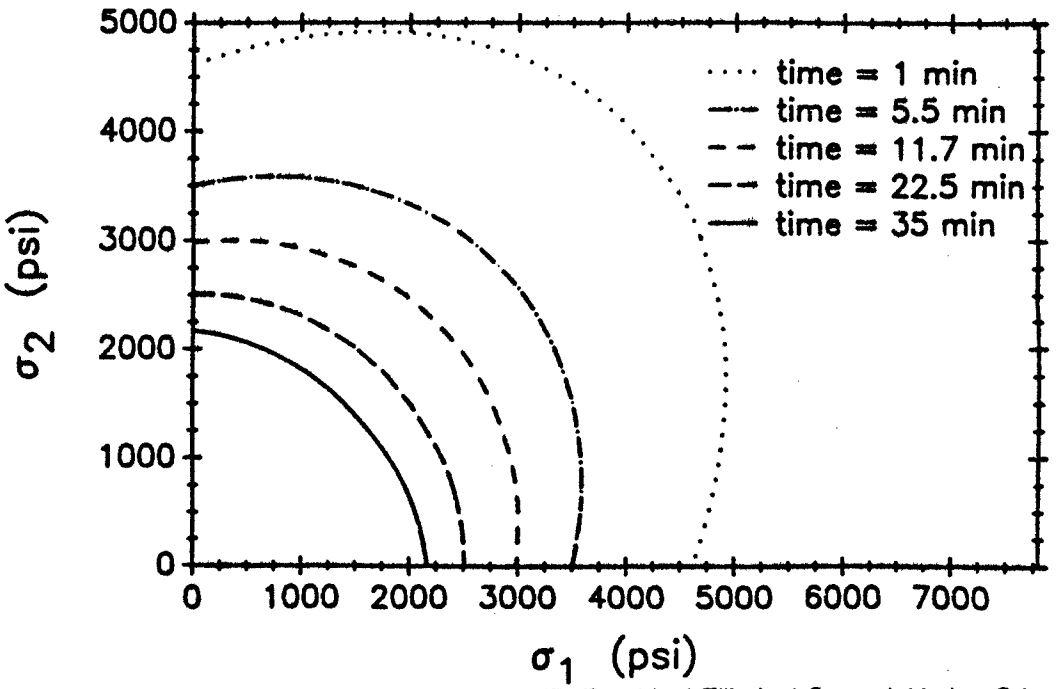


Figure 22. Best Fit Empirical Elliptical Craze Initiation Criteria for Uniaxial and Biaxial Chemical Craze Data.

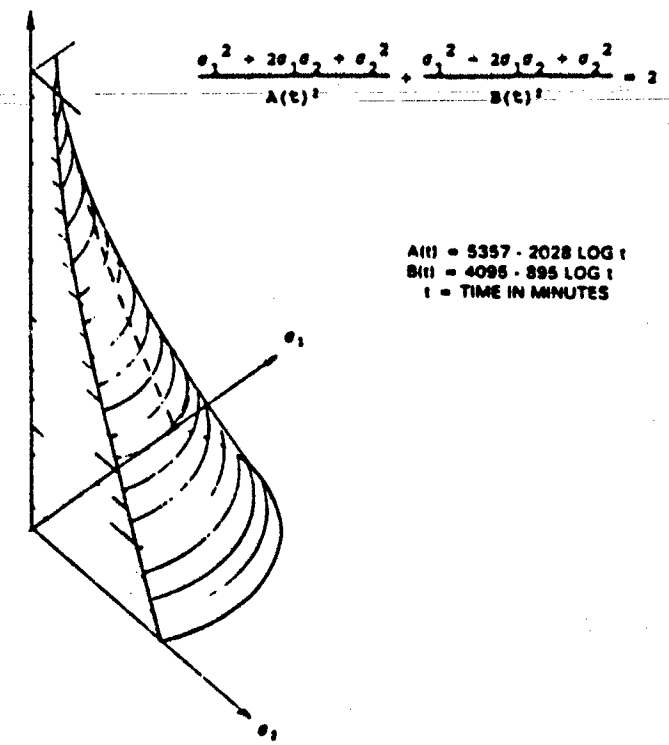


Figure 23. Elliptical Craze Initiation Criteria in Biaxial Stress and Time Space.

STRESS MEASUREMENT IN STRUCTURAL PLASTICS BY L-cr WAVES

Nisar Shaikh
Analytic Engineering Company

STRESS MEASUREMENT IN STRUCTURAL PLASTICS BY L-cr WAVES

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ABSTRACT

Longitudinal waves at the critical incidence (L-cr) angle have been utilized to measure residual stress in glass and plastics used for high performance aircraft. The acoustoelastic effect is most profound in this mode. The novel feature of this research is the development of transducers that are patterned after acoustic microscopy. The reflected specular and leaky wave signals are received by separate ceramics. The phase comparison with specular reflection allows better measurement of the acoustoelastic (velocity changes) effect, as compared to the practice of using the input signals. Rayleigh waves can also be used in the same fashion. Use of water as acoustic couplant is not suitable for plastics due to the low phase velocities. Instead, a silicone rubber medium is used in a manner similar to wedge transducers. This arrangement has an additional benefit of dry contact scanning, an attractive feature for field measurements. The data are acquired using a PC and they are cross-correlated with the reference signal to determine precise transit time. Two additional measurement systems are developed using an electronic counter and spectral technique. The stress measurements are made on aircraft grade acrylic specimens.

INTRODUCTION

The use of acrylics and polycarbonates as structural members is particularly desirable as transparent enclosures for high performance aircraft. The measurement of the state of stress during fabrication and operation in these members is critical for assessing their durability and reliability.

Among existing stress measurement techniques, X-ray diffraction is applicable only to metals and is generally limited to smaller samples for laboratory use. While the hole-drilling technique is the most common method used for field or in-situ measurement, there are two major problems with using this procedure on plastics. The local heat produced while drilling the hole causes material changes that distort the existing stress

field. Also, due to large creep in plastics, there is no unique stress and strain relationship. Thus, the measure of strain does not unequivocally relate to the state of stress.

A viable nondestructive evaluation (NDE) technique is needed to measure both the residual stress induced during manufacturing and applied stress incurred during operation. Ultrasonic stress measurement, often referred to as acoustoelasticity, has shown great promise as an NDE technique [1-3]. Earlier efforts in stress determination in polymers were marred, due to the lack of a transducer capable of launching surface waves in plastics [4,5]. Recently, Obata *et al* [5] have measured stress successfully in acrylics by using an acoustic microscope. Their method has some limitations: it requires the sample to be immersed in a fluid, and the cost of acoustic microscope used is fairly high. Shaikh [6] has developed surface wave transducers and techniques for the detection of flaws in structural polymers. These transducers and technique are applied to nondestructive measurement of stress, as well as creep and stress relaxation in plastics.

In the past, Surface stress measurements exclusively used Rayleigh waves [3]. Recently, the surface skimming compressional wave, better known as the critically refracted longitudinal wave (L-cr), has gained wider acceptance [2]. The L-cr wave is most suitable for stress measurement, since the stress induced effects are the largest in this mode. This mode also has the highest propagation velocity, approximately twice that of the Rayleigh wave. On account of its speed, the L-cr signal is the first to arrive at the receiving transducer, and is not cluttered by various reflections and extraneous modes.

ACOUSTOELASTICITY

The acoustic stress measurement technique is termed "acoustoelasticity," in analogy with "photoelasticity." The acoustoelastic effect is characterized by the fact that the propagation velocity of an acoustic wave in a solid changes in proportion to the change in stress in the material. The equation below illustrates the dependence of stress on the velocity of a longitudinal wave propagating along a uniaxial stress direction.

$$\frac{\Delta V}{V} = L\sigma \qquad (1)$$

Where V is the phase velocity in the unstressed material, and σ is the uniaxial principal stress along the direction the wave travels. For most materials, the above equation is linear and thus the slope of the line provides the value of L. The value of L is an acoustoelastic property of a material, similar to the familiar "elastic modules." The value of L should be known beforehand or must be calibrated from tests for each batch of the material in which the stress is to be measured. For the purpose of calibration, uniaxial tension test is conducted on a material to determine its value of L. A typical value of L for acrylics is 5×10^{-6} per psi.

STRESS MEASUREMENT SYSTEMS

The basic measurement technique involves precise determination of the time of wave transmission that is dependent upon the stress. The stress acoustic effect is comparatively small; under 3% for the yield stress of acrylic. Typical measurements require the accuracy of one part per thousand, or typically within 20 nano seconds. Three different measurement systems are being developed to provide a range of instrumentations of varying levels of sophistication and price range.

1. Digital Ultrasonic Instruments

Cross Correlation of Tone bursts: One of the measurement techniques that we have developed to a great extent is digital data acquisition by a PC. It measures the delay time by computing the cross-correlation function. The software is written in a Windows environment with visual C++ and the algorithm is based on Fast Fourier Transform (FFT) [7].

The correlation function is the integration of the product of two waveforms with the argument of one displaced by the next value of τ. This function takes an optimum value when the value of τ equals the delay between the signals. A simple way of calculating involves taking two waveform arrays of size n and summing up the product point by point for each value of τ. This computation requires n^2 multiplications where n represents the number of digitized points of the signals. Therefore the computation complexity would be $O(n^3)$. Presently, even with fast computer processing time, this is not desirable for large waveforms (1024 or more points) operating in a real-time environment.

An alternative to this is to use the FFT. It has been shown that by transforming the two waveforms by an FFT, then multiplying one of these by the complex conjugate of the other, and finally using the inverse FFT on this product, the correlation function could be acquired. Although this procedure sounds a bit involved, the resulting computation is much more efficient than the direct multiplication approach. Therefore, this has been chosen as the basis for our correlation function [7]. Figure 1a shows the two waveforms, reference and the transmitted signal, acquired from a digital oscilloscope; one is delayed from the other by the amount of time the wave travelled through the test specimen. The correlation function is shown in Figure 1b. The correlation result was compared to the time delay read manually on the oscilloscope. The calculated value was exactly the same for the waveforms. When the waveforms are not similar, but have some distortion, the manual technique becomes ambiguous, thus requiring a technique such as cross-correlation.

2. Time Interval Measurement Technique

Determining the precise time of travel of ultrasonic waves through the specimen is the single most important factor in stress measurements. In this approach, a counter, pre-amplifier and a gate circuit are needed. A Universal Counter, which measures the time interval between two waveforms with an accuracy of 2 nano seconds is used. A

pre-amplifier is necessary since the transmitted signal may not always be of the minimum amplitude required by the counter. The need for the gate circuit is a result of signal ambiguities. One typical problem in measuring the time delay is the distinction between the noise and the beginning of the signals. This ambiguity is removed by providing a gate that can be positioned at the appropriate position and instructing the counter to start measurements from the first peak from that time onward. Figure 2a shows the gate on the reference signal. A similar gate is also placed on the transmitted signal (not shown). Figure 2b shows the schematic for our measurement scheme. A tone burst is sent to the transmitting transducer and two signals are received, one by the reference and one by the receiver (through the sample). Each of the signal is pre-amplified as necessary and sent to the Universal Counter. The gate sends an arming signal to the counter and its position is set with the help of an Oscilloscope. Studies are continued to establish the efficacy of the method, including accuracy and limitations. The technique will be compared with the digital correlation technique.

3. Sweep Frequency Technique

During the initial development stage, special wedge transducers were designed [8] to launch and receive L-cr and Rayleigh waves in plastics such as acrylics and polycarbonates. The transducers were made of wedges of high grade silicone rubber whose wave velocities is smaller than those of acrylics and polycarbonates.

Success with the wedge transducers motivated further extension of the technique of acoustic microscopy to stress measurements. A line focus transducer was designed for acoustoelastic measurements with a center frequency of 5 Mhz. Transducer frequency variation can easily be controlled and measured to the accuracy of the signal generator. For our signal generator, the accuracy is in increments of 100 Hz. Thus the measurement of the phase and the time delay by this technique can offer an accuracy of one part in 10,000. Figure 3a and 3b shows the classic V(z) curves of acoustic microscopy on acrylics and aluminum respectively. In acrylic, the leaky wave is longitudinal, with a phase velocity of 2.71 km/sec, while in aluminum the leaky wave is Rayleigh wave with a phase velocity of 2.97 km/sec. Typical of this research is the measurement of V(z) curves at a constant value of the de-focus distance z. Figure 4a shows the plot at the focal point (z=0) which essentially shows the frequency response of the transducer. When the transducer is moved closer to the focus point by a distance z the leaky waves are generated along with the specular reflection. Figure 4b shows the signal variation versus frequency; a sharp null is seen at 4.997 Mhz. This null frequency is unique to the phase velocity at that point and thus corresponds to the state of the stress. The transducers and technique are being developed to measure the null frequency in scanning mode to measure the stress field.

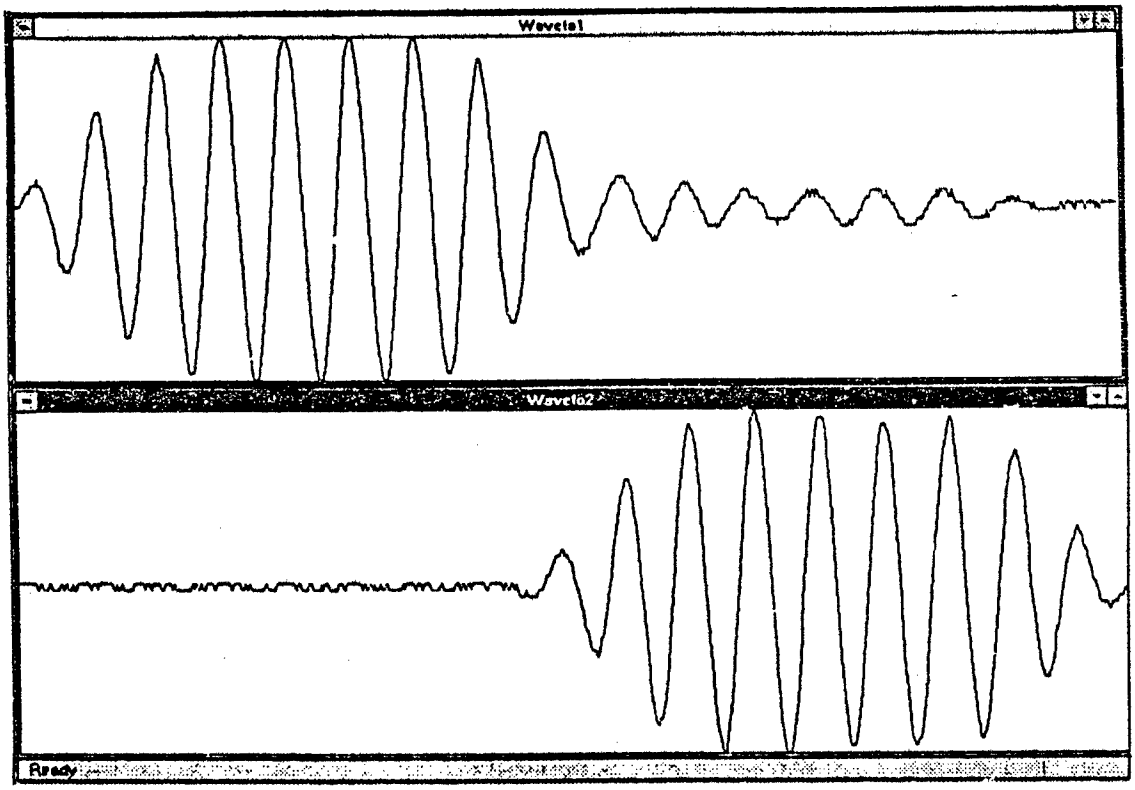


Figure 1a

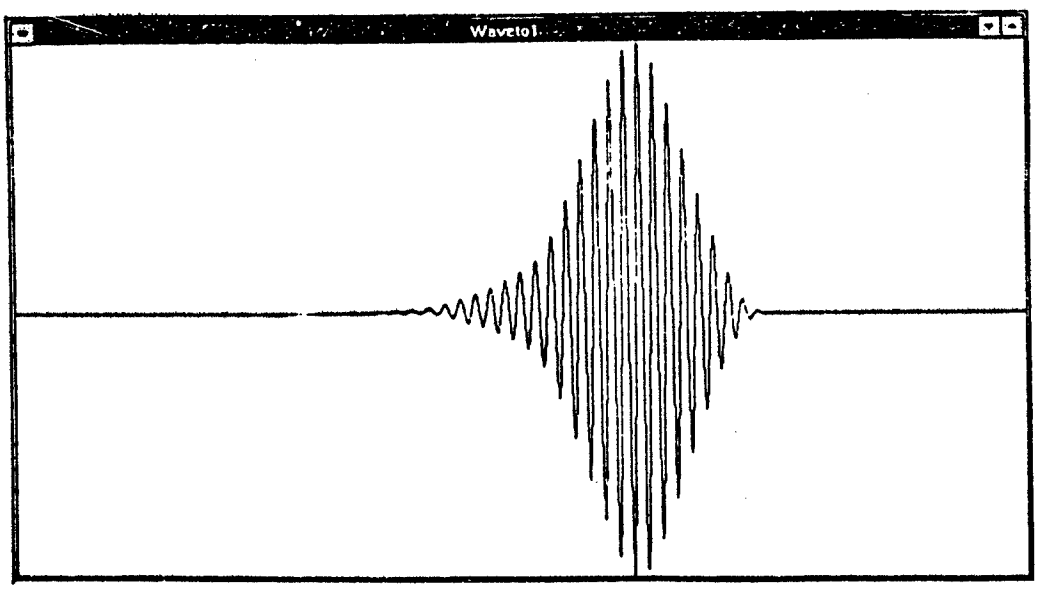
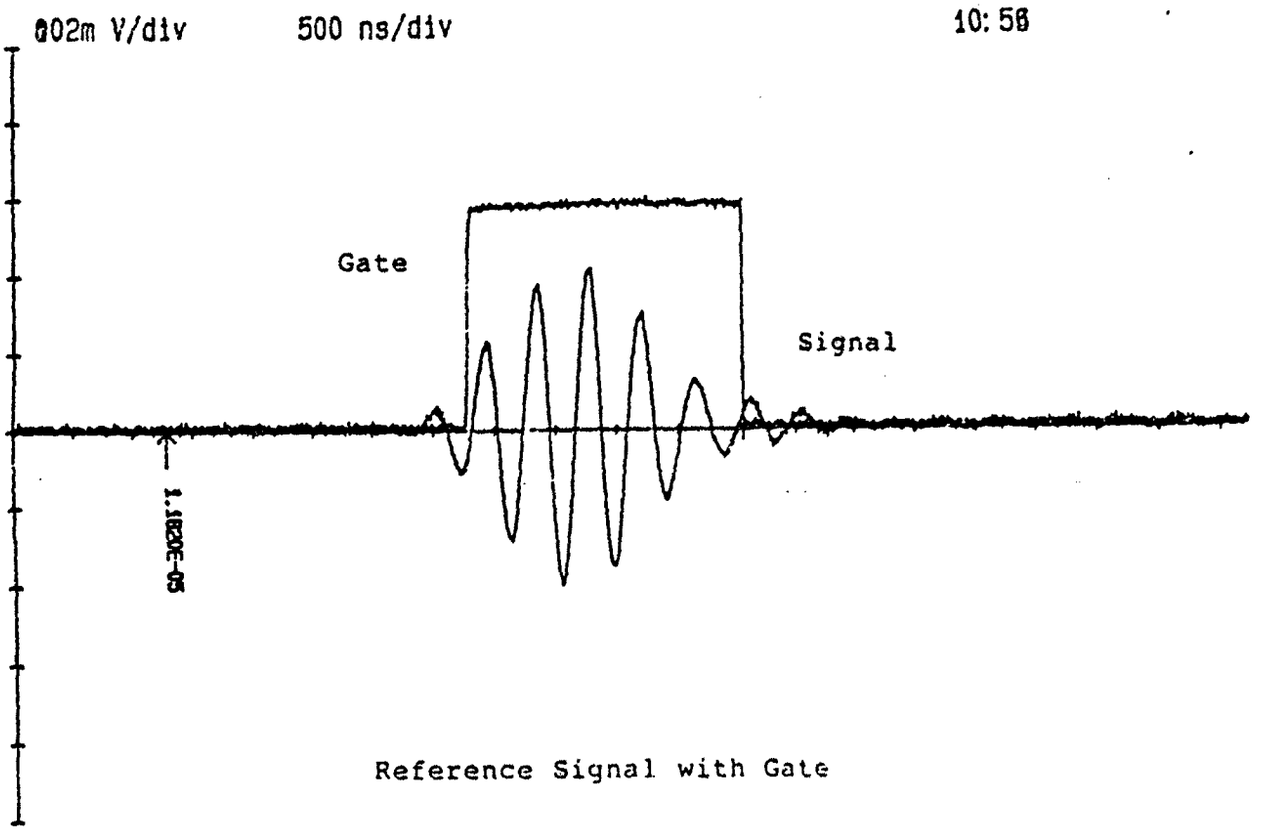
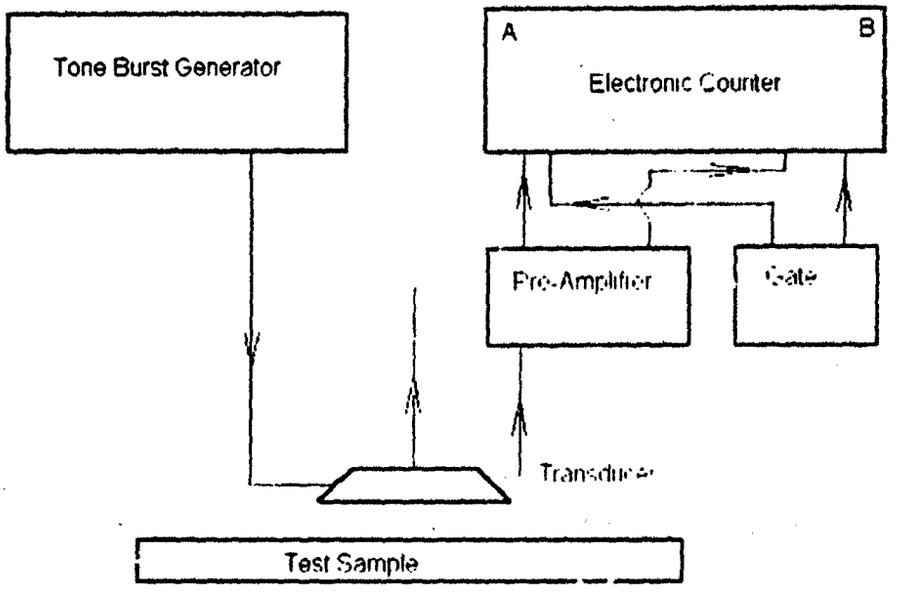


Figure 1b



Reference Signal with Gate

FIGURE 2a



Time Delay Measurement System

FIGURE 2b

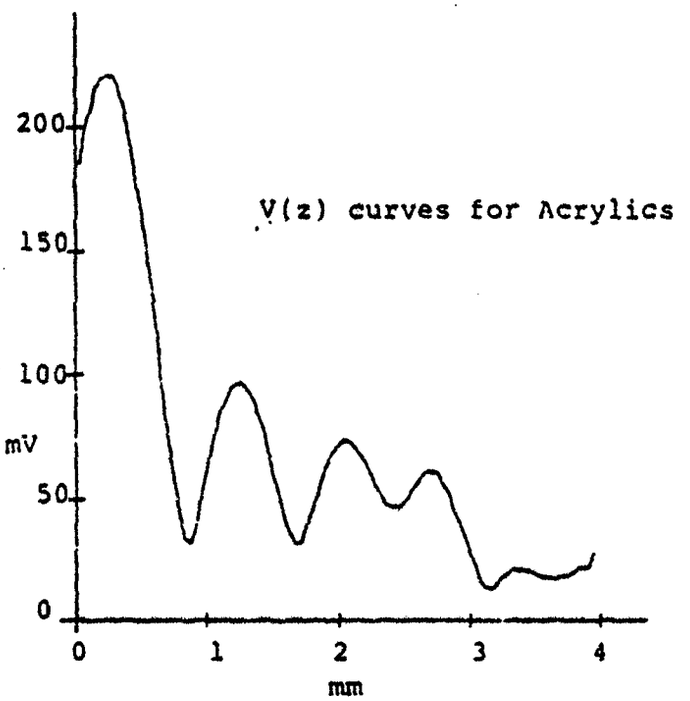


FIGURE 3a

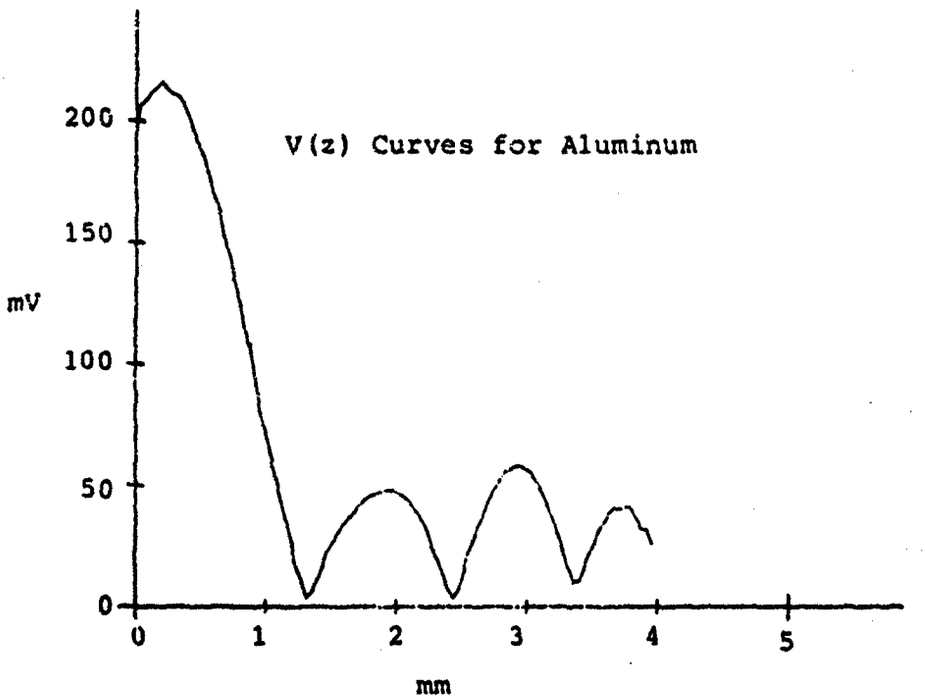


FIGURE 3b

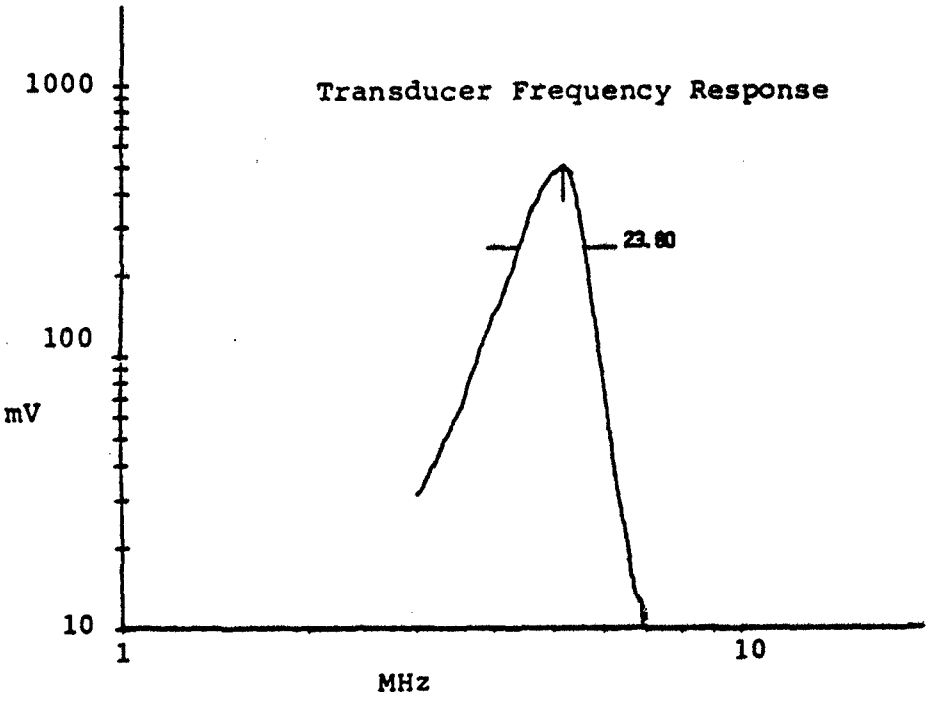


FIGURE 4a

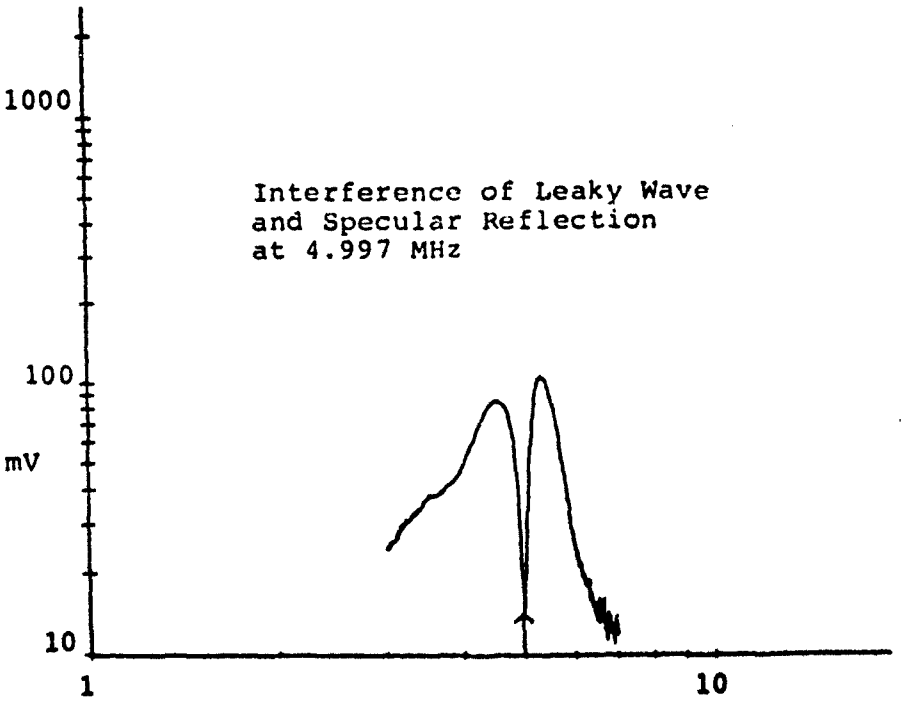


FIGURE 4b

CONCLUSIONS

The feasibility of the transducers and techniques along with the necessary instruments is shown for measurement of stress in plastics and glass used for transparent enclosure of high performance aircraft. The technique and hardware are being developed for the field measurement for quality control at manufacturing stage and durability at operation.

ACKNOWLEDGMENTS

This work was sponsored by the Vehicle Subsystems group of the USAF, WRDC/FIVR under SBIR contract F33615-91-C-3408

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AIRCRAFT WINDOW PROBLEMS
AIRCRAFT MANUFACTURER'S ASSISTANT - HELP OR OBSTRUCTION?

Klaus W. Edwald
Lufthansa German Airlines

One element of aircraft structure deviates quite a bit from the extensively used more or less durable and strong metal construction. This element is made from a very sensitive and transparent material, which is normally handled with some respect and even more care, because everybody has already realized its greatest disadvantage - it breaks quite easily. Just think about your experience - a drinking glass falling down to the floor or just falling over, or remember the story of the football and your neighbour's window pane.

Aircraft visual transparencies, particularly the glazing material, are subjected to a wide variety of much higher loads. That nobody, under everyday conditions, would normally ever take into considerations.

These loads are: internal pressure, aerodynamic pressure, bird impact (your football shot is just a gentle stroke compared to the impact of a four-pound bird at aircraft's design cruise velocity up to an 8000 foot altitude) foreign object damage, thermal expansion and contraction, transverse thermal gradients, airframe torsion and bending, fatique, etc. - not one load at a time - no, all at the same time.

With that knowledge, and the realization of a high cost burden, people should expect that this structural element is getting the required attention within aircraft operations.

Just the opposite seems to be the fact, mainly due to the following:

1) Lack of interest and knowledge by the airlines.
2) Defectiveness in realizing problems and coordinating them between airlines and window manufacturers by the aircraft manufacturers.
3) Lack of sufficient in service experience reports due to 1) and 2) by the window manufacturers.

Traditionally all aircraft operation problems are reported by operators to, and discussed with, the aircraft manufacturers. Direct contacts between part manufacturers and airlines, at least in the past, were quite rare and seemed to be actively discouraged.

The considerable amount of worldwide information generated and submitted to the aircraft manufacturers - even not always of best scientific and technical content - in connexion with the awareness of a critical, safety related airframe component, should raise the expectation of early decisions, recommendations and modifications, as required.

Even when close examination of aircraft window failure modes and/or damage disclosed an inadequacy in transparency design, resulting in unacceptable transparencies - urgently needing redesign - soon after their introduction into service, nothing happened - sometimes for years, even when flight safety was directly affected.

Windows were replaced under warranty, at no charge by the aircraft manufacturers. Their responsible engineers very often were not involved and had no idea of the large number of returned windows and, quite often, had not the slightest idea of the different failure modes. The window manufacturers got absolutely no information what happened in service.

Were they not interested? Or did some strange contract between aircraft and window manufacturers hinder airline and service experience being coordinated ?

On June 21, 1982 the area sales manager of one of the leading window manufacturers, during a meeting with an aircraft manufacturer, made the following statement:

"When you visit our facility next Friday, be aware that you will get a shop tour, only. Nobody will discuss any technical item except somebody from the aircraft manufacturer will accompany you."

Since that historic day, improvements are clearly noticeable and there was a certain, very successful period of about 6 years when some airlines and some window manufacturers solved problems without the more or less negative influence of the aircraft manufacturers.

In the meantime more and more airlines and window manufacturers have realized that close cooperation and experience exchange is essential.

But still, even more modification/improvements or just recommendations were not introduced because of a certain influence exerted by aircraft manufacturers - even though a window is not considered to be an aircraft manufacturer's proprietary item.

However, some mystery must be involved when windows, demonstrating a certain failure mode, are returned to the aircraft manufacturer at their request and, on return, the following statement is received:

"The window we had planned to examine was inadvertently scrapped. We regret the loss of this window as we had hoped to learn more regarding the failure condition."

This was not too long ago. This statement was received June 22, 1990.

To answer the question: "Aircraft manufacturer's assistance - help or obstruction?" four typical failure modes are presented and discussed in detail.

The first defect/failure mode in question occured due to an aircraft manufacturer's enforced modification to the transparency, which was later explained to be the most successful change to reduce the probability of another defect.

This "other" defect was pre-dominant on four of the ten window positions.

The modification actually created a more severe, safety related defect on all windows.
The windows in question are flat and utilize chill-tempered soda lime glass for the primary structural ply, with a substantial amount of polyvinyl butyral (PVB) bonded to the glass. The PVB serves the purpose of a fail-safe diaphragm, as well as being the ultimate bird-stopping agent. This windshield design relies on the bird bagging concept.
For this method the glass pane fractures on impact and allows the PVB interlayer to deform and absorb energy.

With the realization that aircraft and birds will always share the same sky, and collisions between birds and aircraft are a major flight safety problem, the allowable limit for the new type of defect, which involves the PVB, was of course "zero". We are talking about PVB cracking - a defect, present on every window of a certain type, after some time in service, for more than 20 years, but in many or most cases never detected by most of the operators.





Between 1970, when the first cracks were detected, and 1990 when the first real fix was introduced to avoid PVB cracking, an incredible, never ending story of non-observance, inactivity, mismanagement, and ignorance was experienced.

The new failure mode was acknowledged by the aircraft manufacturer, all windows returned under warranty were replaced at no charge, no information to other operators, or changes to the windows were made.

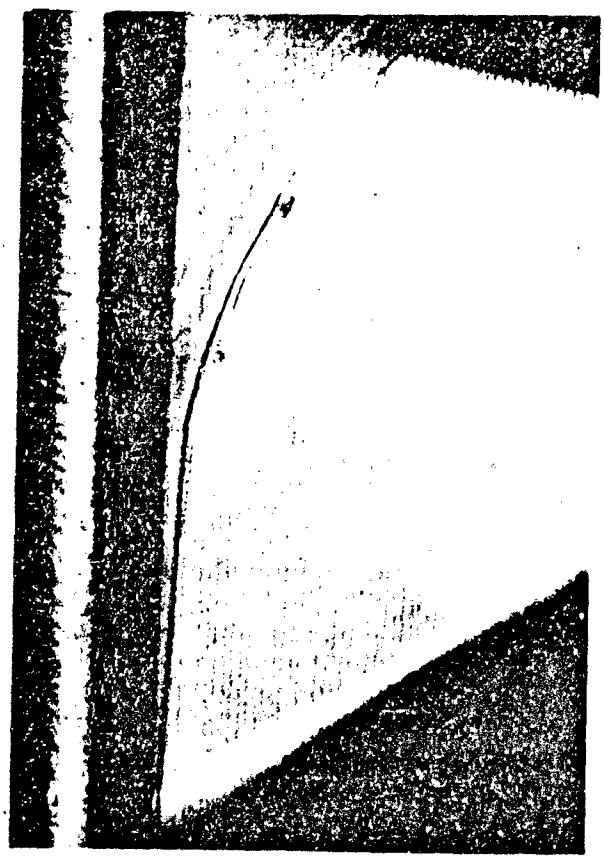
About two years later, due to the increasing number of windows removed due to PVB cracking, a repair in the form of recycling of used windows in an autoclave process was explored, in close cooperation between aircraft manufacturer, window manufacturer (one of two) and one airline. Initially, the results appeared to be promising.
However, a closer review revealed other problems and imperfections.
Because recycling did not appear to be cost effective for the results gained, this process was abandoned as a means of improving window durability.

The first, at least partly effective modification to reduce PVB cracking was developed in close cooperation with airlines and a small specialized manufacturer - by the addition of a heater bonded to the inside of the window.

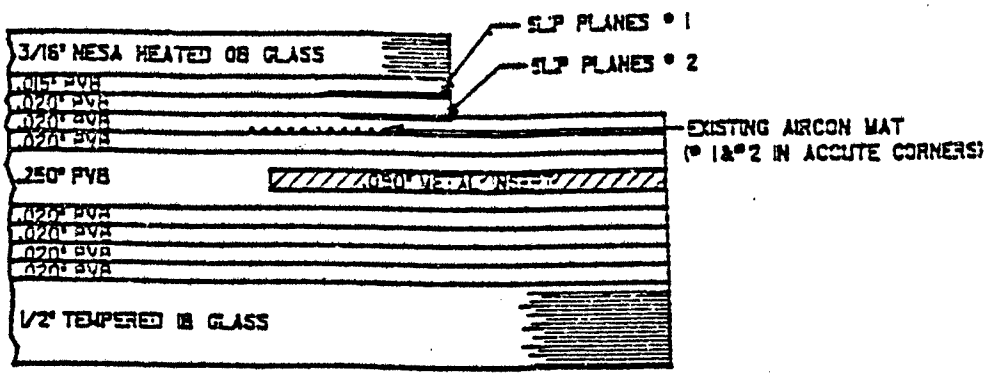
It took the responsible part of that business two years to introduce something similar or even slightly better - , but the problem never disappeared.

In 1981 the following was published by the aircraft manufacturer:

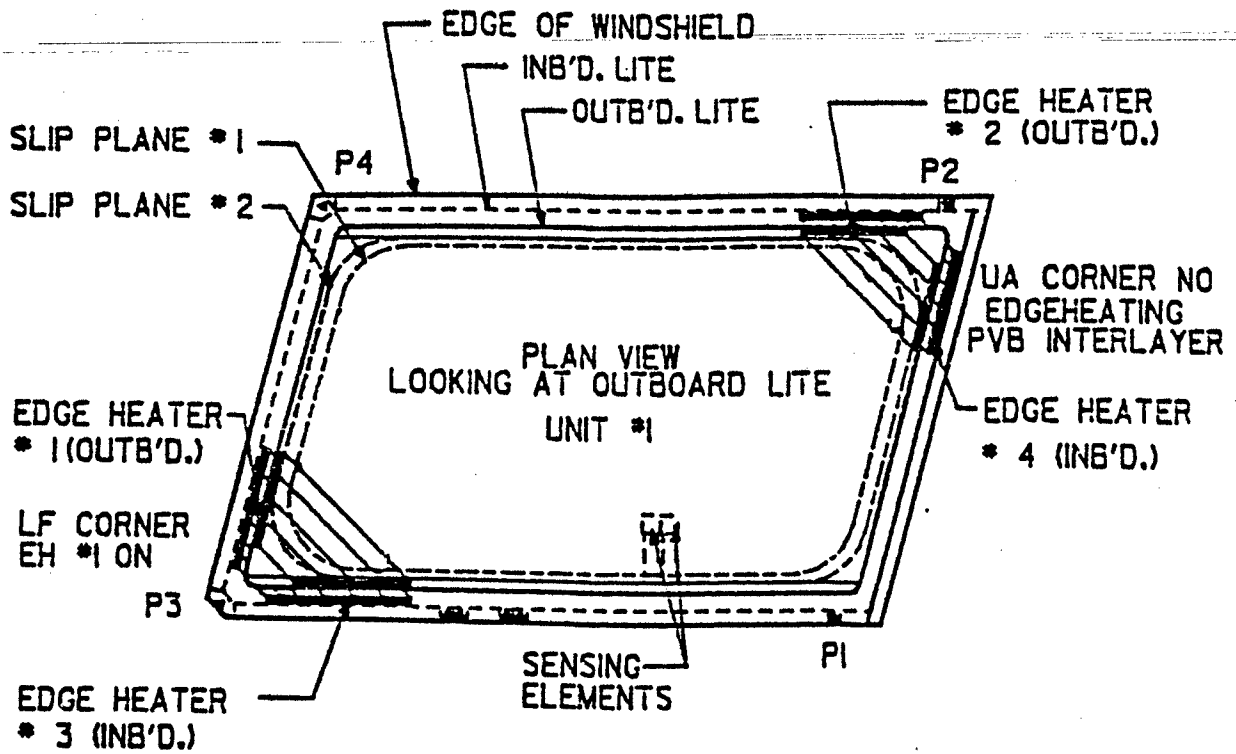
"The cracking of the vinyl interlayer in the number 1 window (it seems that they still had not realized that cracking occured on all window positions no. 1 to 5. left hand and right hand) was attributed to cold areas in the window. Production improvements achieved more uniform heating by relocating the heater wires and making them from a heavier gage.
The heater wires were moved from the forward (inboard) and aft (outboard) edges to the forward lower and upper aft corners of the window.



The wires in the vinyl interlayer were also relocated so that they would be nearer to the outer glass ply rather than the inner glass ply. Each new heater is divided into two parallel elements.



Should one element burn out, the other will still heat at 50 percent of the applied power."

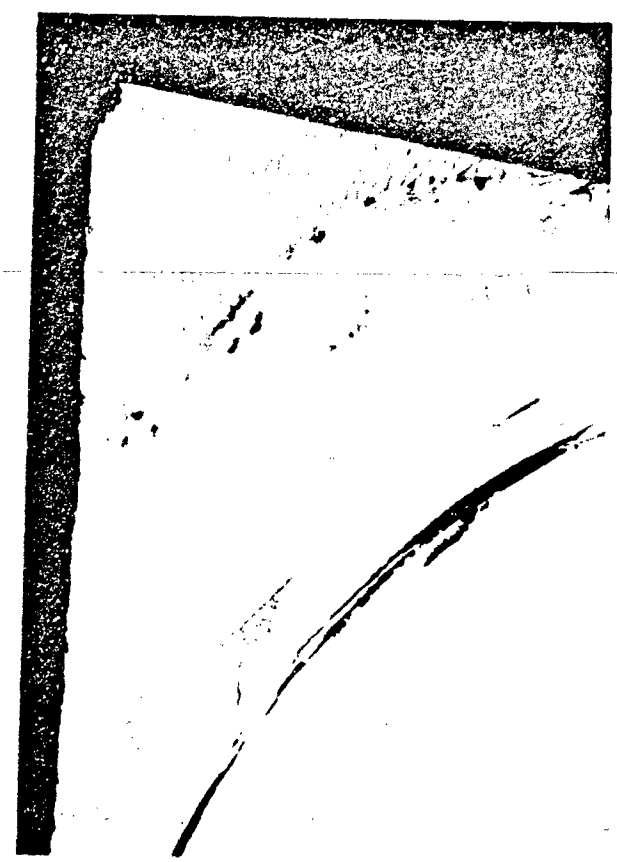


Sound's good! ?
I still have some doubt whether this change was introduced as an improvement or due to production cost reduction

This change became effective July 1979 at only one window manufacturer.

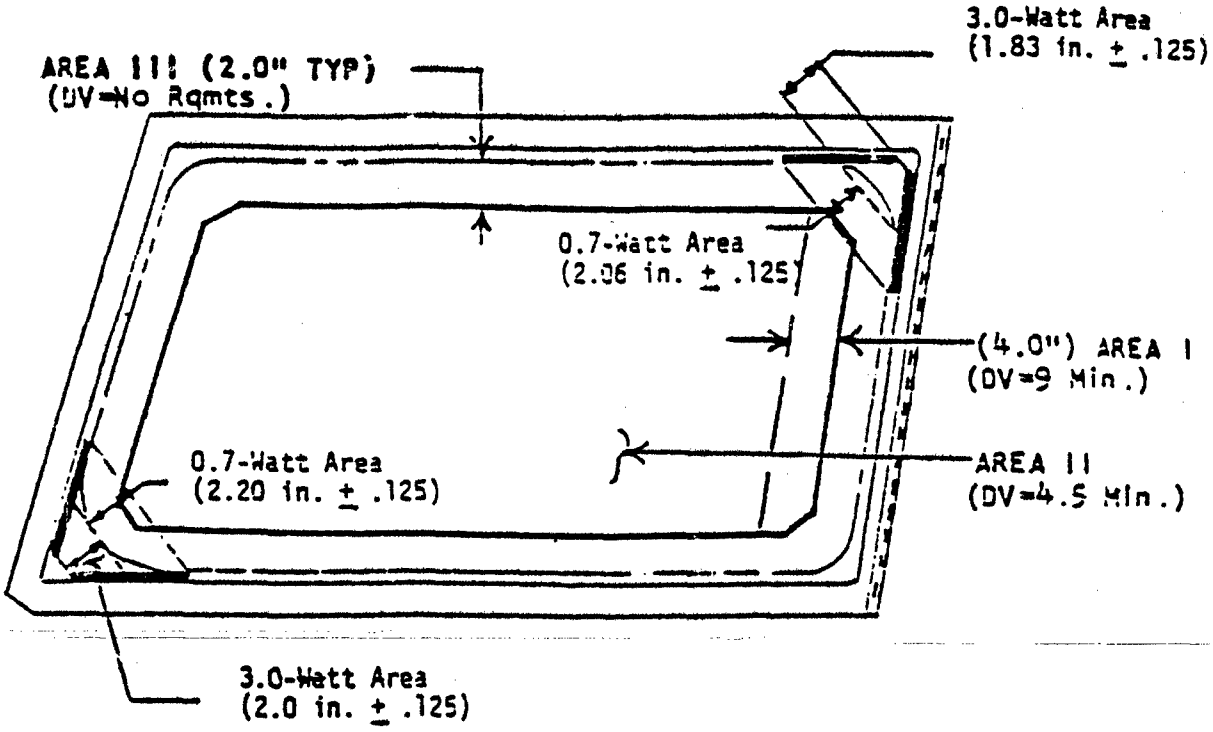
It was absolutely no surprise, that shortly after introduction of this improvement a new type of defect was to be found in the area of the new heating element, and - PVB cracking was found in an increasing number after a very short time in service, again

Due to severe, PVB shrinkage and warping, just in the area of the heating element, wires had to fail, causing arcing and local overheat - the new defect, but by then the heating element was completely inoperative and we were back to square one - a completely cold corner, with high stresses and consequent PVB cracking.



But why was such severe shrinkage and warping occuring? The new heating element should have been capable of supplying sufficient heat.

Sufficient heat? That's the question!



There was a certain aircraft manufacturer's edge heater check procedure published for the window manufacturer:

"While unit is powered, check against gridboard to ensure the edge heaters are working. Broken wire in the 0.7-watt area is acceptable. Broken wire in the 3.0-watt area is to be rejected."

Defective windows leaving the factory with aircraft manufacturer's approval !!

For the first time in history, 1987 - 17 years after the first vinyl crack was found - the aircraft manufacturer published details on vinyl cracking and the area of occurance!

Nearly everything in that publication was wrong or at least misleading !!

Due to legal problems with the two established window manufacturers - they were working too closely together with the airlines - the aircraft manufacturer in 1987 began working with a third manufacturer (known for excellent performance with other types of windows) to develop improved windows.

The first windows successfully completed full certification testing and were FAA certified in May 1988.

Shortly after in-service evaluation installation of some of these windows at one airline we received the first reports on PVB cracking !!

Nevertheless on May 18, 1989 the aircraft manufacturer proudly announced:

"We are pleased to announce the incorporation of improved windshield assemblies. Development testing and in-service evaluations have been successfully completed and the new windshield assemblies have been committed to production.
The new windshield configuration applied to heated window positions numbers 1, 2, 4 and 5 left and right side. Improvements included revised window edge details, and a new type conductive coating that is currently being used on other model aircraft.
These improvements will minimize glass edge chipping, delamination, and vinyl interlayer cracking."

One window position is not mentioned: no. 3. This is normally an unheated window. Together with a few other operators we are using heated windows at that position. Since the decision was made to use the heated window in 1981, this window, orginally installed on a long range aircraft, was a real disaster on a short range hopper. Nearly no window ever reached a thousand flight hours.

In January 1985 an airframe manufacturer influenced new design window was installed on new delivered aircraft. After less than one year none of the delivered 12 windows was still in service.

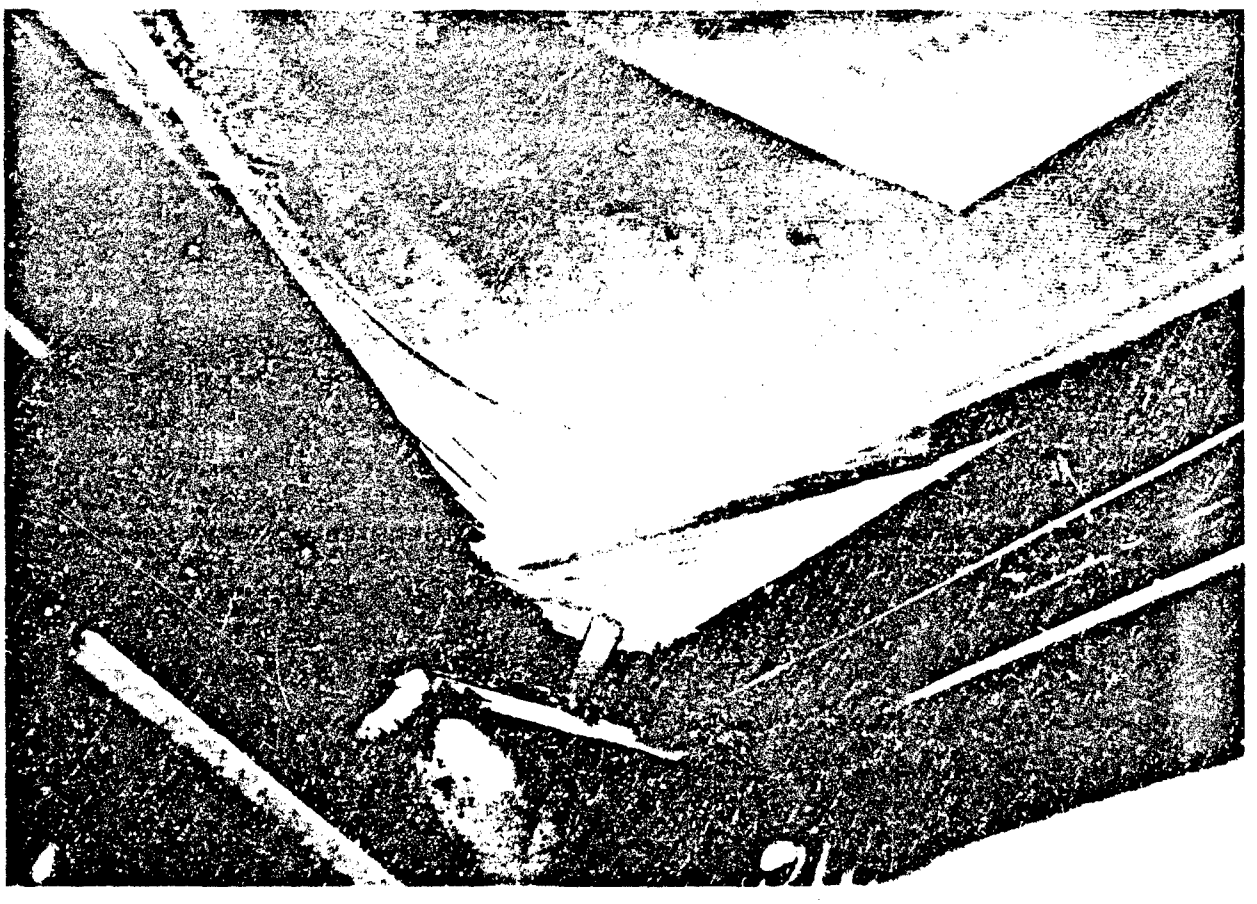
In close cooperation with the window manufacturer a new design entered service in August 1986. More than 200 no. 3 windows are now in service at Lufthansa, with just a very few justified removals in the meantime.

Coming back to the aircraft manufacturer's announcement on May 18, 1989:

Just 18 days later on June 7, 1989 a new message was received:

"The first telex informed you of our intent to incorporate improved windshield assemblies on your new airplanes.
Recent data from our in-service evaluation has caused us to reconsider the commitment of these windshield assemblies to production.
We are currently evaluating the situation and will advise you further by the end of June 1989."

Nothing was received by end of June 1989, but new aircraft were delivered in December 1989. What a surprise (but as expected) we found the windows in question installed! Following the delivery flight, after the aircraft had flown in total 7 hours 18 minutes (8 cycles) in flight test, and 10 hours 42 minutes (2 cycles) on the delivery flight, vinyl cracks were already apparent !!



As warranty replacements we received so called new design from another vendor

When they were returned to the aircraft manufacturer after 1420 flight hours, the following comment was received:

"We concur that the upper aft corner contains a vinyl crack. The discrepancy in the lower forward is probably a vinyl crack but is not as well defined.
We plan to return the window to the supplier after a warranty disposition has been made.
We appreciate your bringing this condition to our attention since this is the first report on vinyl cracks in a new configuration window."

This was the last and final comment on such a window.

The window manufacturer by the way never received the window.

Nevertheless, vinyl cracking is no longer a problem.

Most of the design featurers involved in the earlier mentioned number 3 windows, are already in use in all other positions. These new generation windows were made available directly to the airlines from 1989 and have been in use at the aircraft manufacturer for quite a while.

Aircraft manufacturer assistance - help or obstruction?

This seems to be a clear case of obstruction.

The second case in question involves a modern design window which relies on bird bouncing - on rigid material to repel a bird.

Two primary, high strength, light weight, chemically strengthened glass plies provide fail safe and bird repel capability.

Already the first issue of the window manufacturer's component maintenance sheet under "Limits of defects permissible in use" mentions an important fact, deviating quite a bit from other windows:

"Scratches: Structural component:

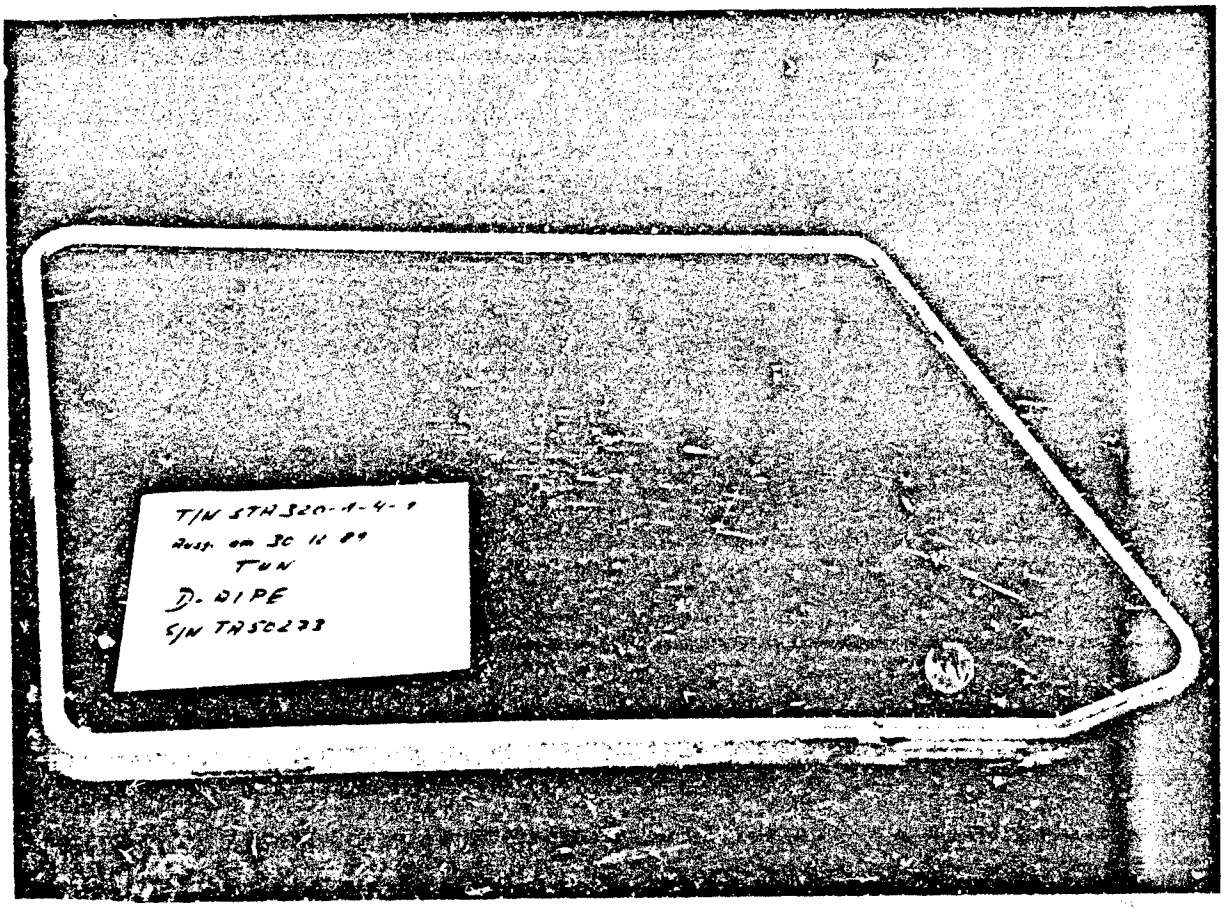
-	No scratches greater than	0.2 mm	(..0079	in)	in width
	or	0.02 mm	(..00079	in)	in depth

shall be allowed."

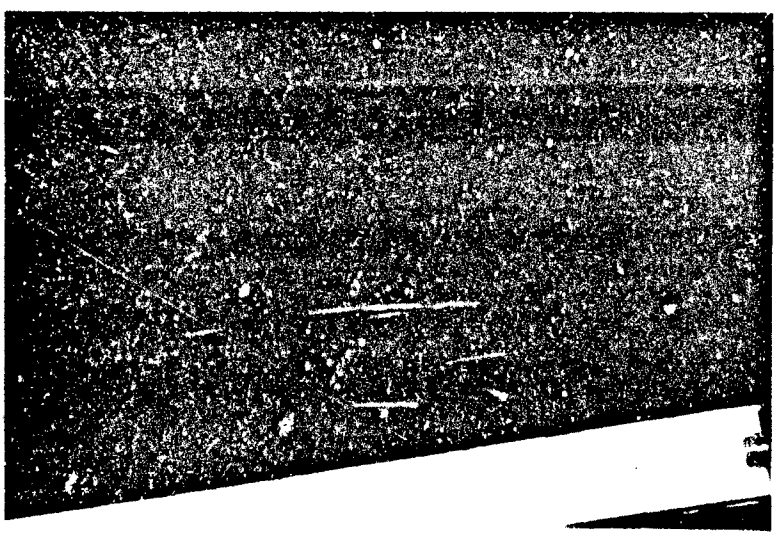
An indication that something must be wrong when such incredible tight limits are made mandatory ?

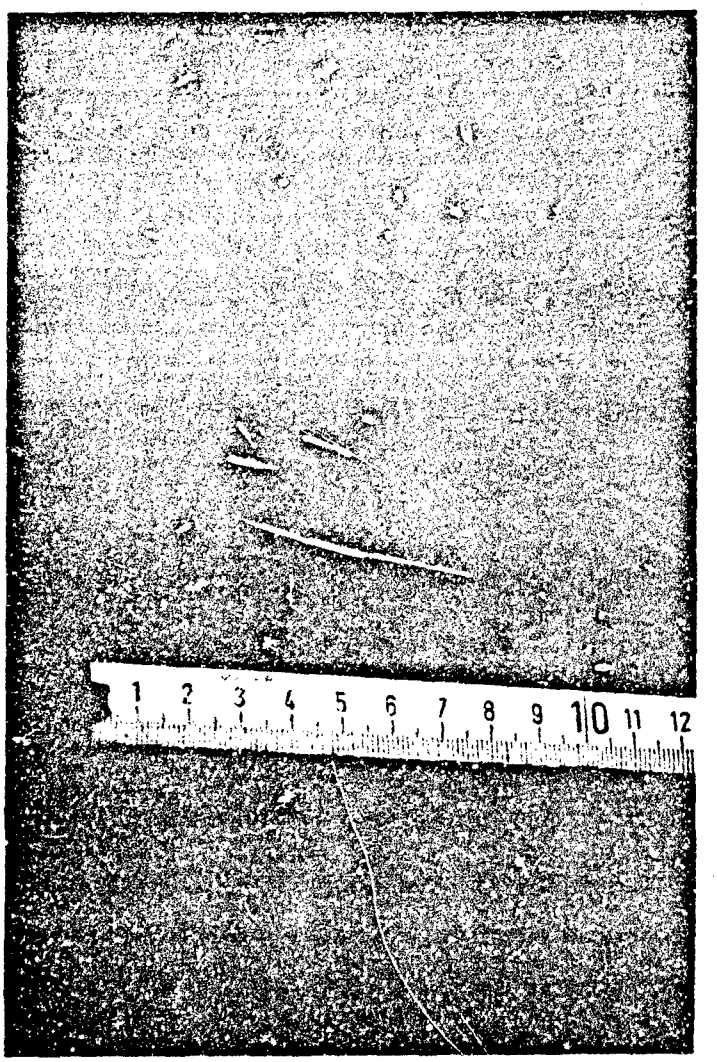
75 days after delivery of the first of such windows on a new aircraft type on January 30, 1989 further suspicion was raised that something must be wrong with the new window, with the new type of glass.

37 flight hours (48 cycles) after delivery of the aircraft, during cruise flight, when flight level 370 was, for a short term increased to 390, and 2 minutes after coming back to flight level 370, the inner main load carrying structural glass ply of the left hand windshield shattered, without any premature indication/warning.



Aircraft descended to flight level 250 with a delta p of 5 psi. During further delta p reduction for landing, the inner pane partly disintegrated with tiny glass splinters detaching from the inner glass surface into the cockpit. The pilot reported that he was lucky to be wearing sunglasses.





On request of the aircraft manufacturer the windshield was returned to the window manufacturer for investigation

The results were submitted March 1, 1990

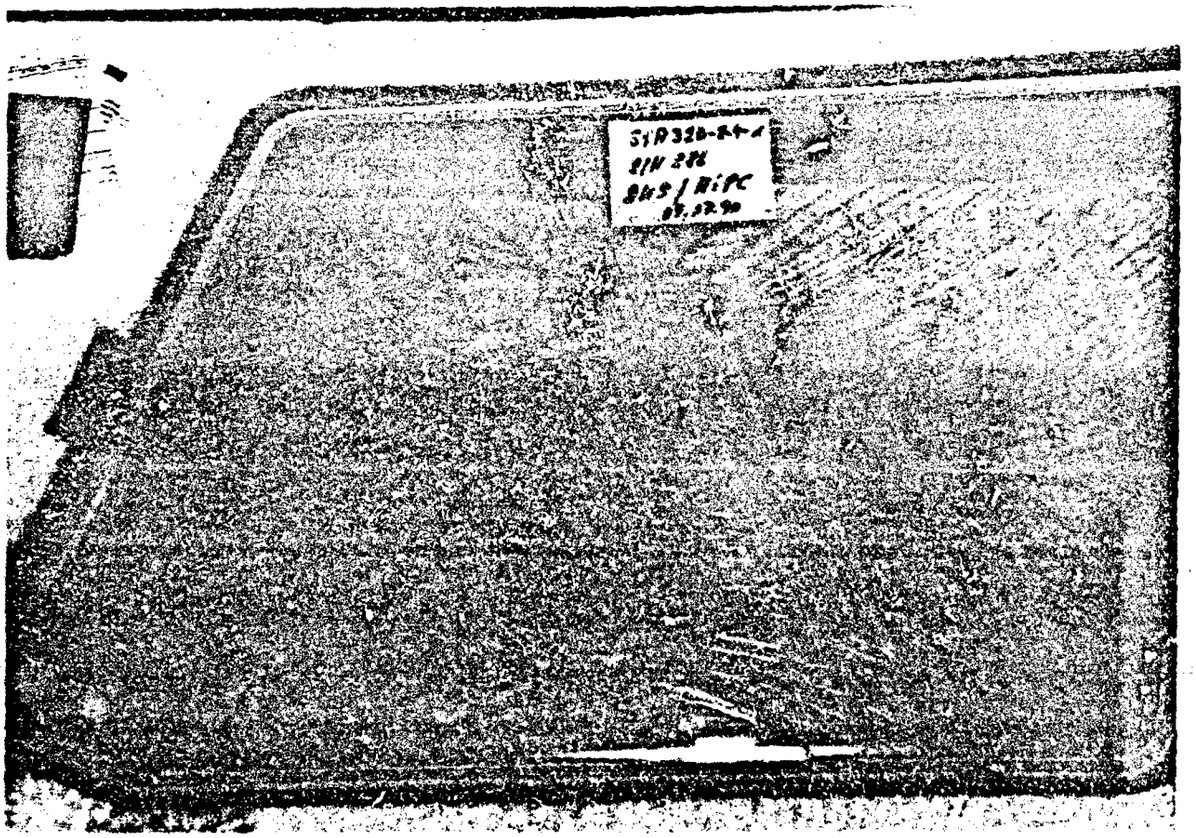
inner ply cracked (under pressurization) starting at one edge

fracture analysis shows a small chip; it might have been done at one edge

replacement under warranty

This replacement window will show up again later in our story

On July 9, 1991 during the first flight after installation of a right hand windshield the middle main load carrying structural glass ply shattered.

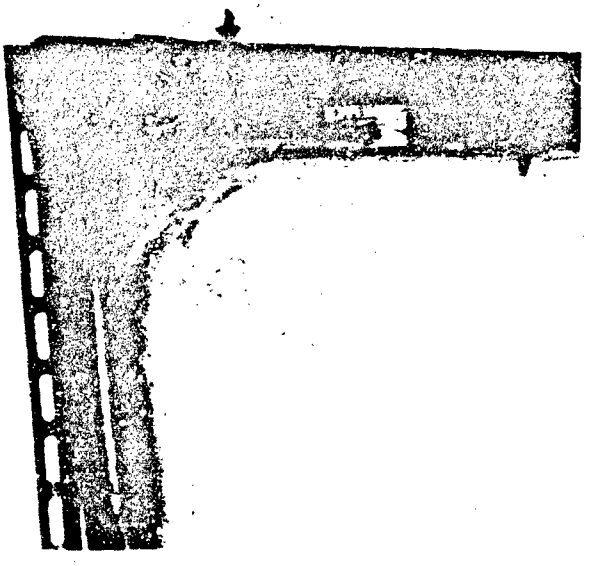


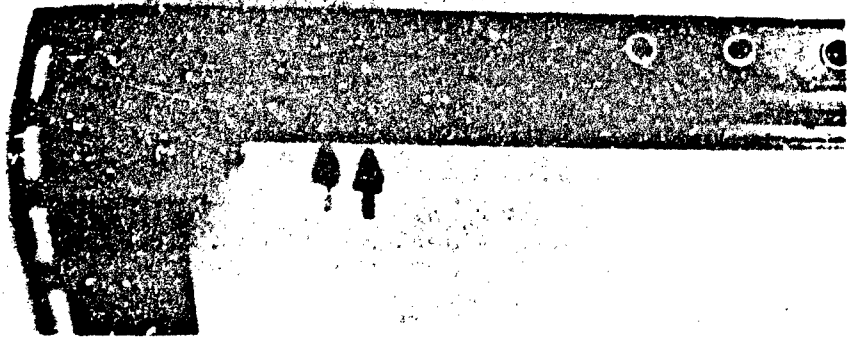
Aircraft manufacturers comment:

"We are very concerned by the window failures reported "

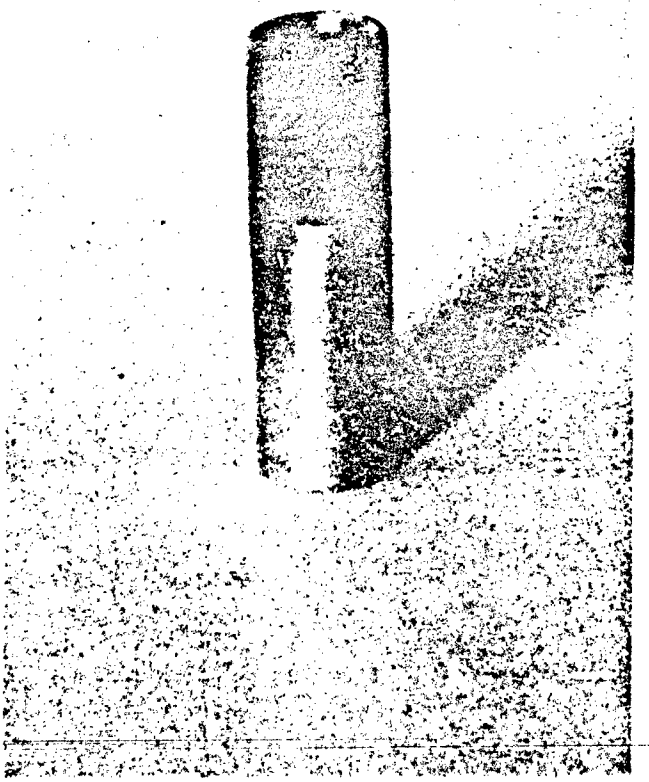
Window manufacturers conclusion:
"Crack initiated from an impact on one edge of the ply
The findings clearly show that the origin of the cracking is a shock on the glass ply which could have happened during the windshield assembly or before."

It was later found to be melted/fused. The window, which had been installed in production prior to aircraft delivery on February 25, 1991, had been removed and will be sent back on warranty.
On removing the window, burn evidence was noted on the first mounting bushing inboard of the upper power attachment point. The mating structure was also burned and required blend out.





Two previous cases of this condition were experienced with much less dramatic accompaniment:
On January 9, 1992, the right hand number one window was removed for a window heat failure. The window had been installed on June 18, 1990, and had accumulated 4051 flight hours and 2849 landings. The same bushing was found burned.



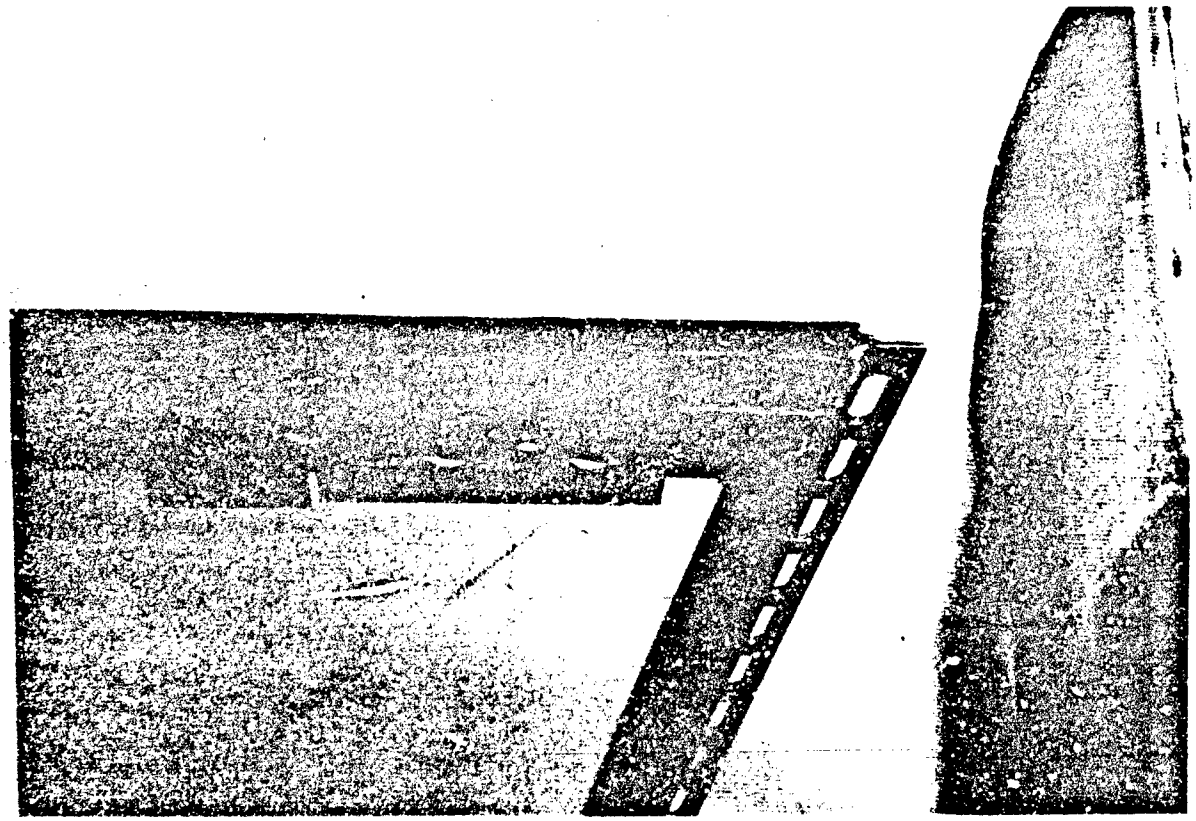
The bushing removed from this window was mailed for examination.
The first case of this condition had occured some 18 months previously.

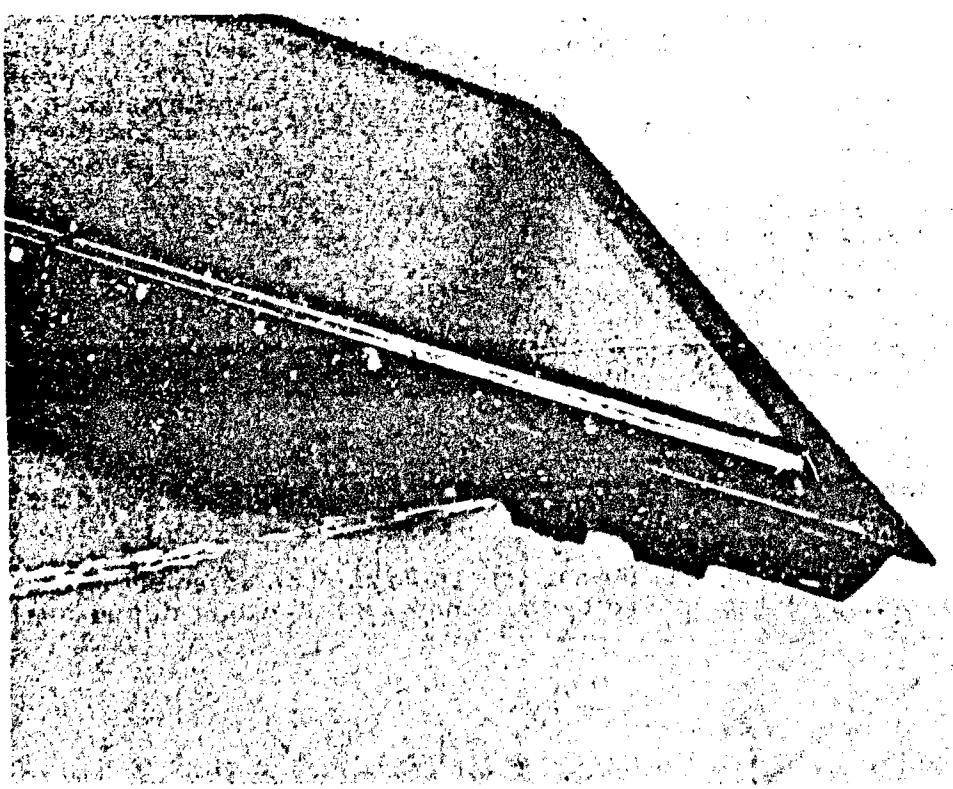
This message was sent primarily for information. However, any information on other cases of the condition and any corrective action planned would have been appreciated.

As a response, the subject circuit breaker was requested 20 days later.
5 months later on August 13, 1992, the following comments were received:
"Our investigation of the subject window is continuing. We currently plan to examine another window in our laboratory to obtain additional information. We will provide results or status of this investigation on or before November 4, 1992." In the meantime the window manufacturer, who was informed the same day, in March, had already verbally submitted the explanation for the failure mode.
On November 5, 1992, we had another occurence where a shower of sparks originated from the upper left corner of the captain's window number one, when placing window heat on during aircraft preparation for the next flight. Again the mounting bushing was found burnt.

Corrective action was promised by the aircraft manufacturer on November 17, 1992, which would be incorporated by the window manufacturer during first quarter of 1993.

On January 9, 1993, there was another occurence where an arcing/flame plus a bang originated from number 1 right hand window, two minutes after heat "ON". The circuit breaker had popped and the mounting bushing inboard of the power supply terminal was severely burned. The window seal was black and showed slight damage





Subsequent to this last report, two additional cases of burned window mounting bushings were experienced:

1) On Ferbuary 23, 1993, during climb out from Munich, the crew reported a short circuit with flames and smoke from the right hand number 1 window with window heat "ON". The crew elected to make an air turnback and returned to Munich.

2) On February 24, 1993, while on ground in Munich, there was arcing, flames and a bang originating from number 1 left hand window, shortly after window heat "ON".

Due to concerns about these latest occurences including the air-turnback, and the lack of information available on the reasons for this failure mode, plus the absence of advice on operator action necessary to prevent reoccurence and the lack of any other action taken, at the time, a fix for the problem was requested urgently from the aircraft manufacturer.
The answer was a real surprise - although typical, for that particular aircraft manufacturer. Their response was as follows:

Q 1) What is the reason for the failure mode ?
A 1) In response, the windshield arcing was most likely due to improper installation or the transparency shows evidence of moisture ingress, variation in the terminal block location or voids in the terminal block location.

Q 2) What action can the operator take to prevent reoccurence ?
A 2) In response, we suggest verifying proper windshield installation as given in the maintenance manual.

Q 3) What action has the aircraft manufacturer taken to date and what fix is being made available for the airlines ?
A 3) In response, please refer to our earlier message which defines improvements to the transparency. We anticipate that improved windshields will be available by the 4th quarter of this year".

The window manufacturer's answer differs quite a bit from the aircraft manufacturer's comments and their earlier description of corrective measures:
"Analysis of units returned by you which demonstrated arcing and melting of the bushing in the terminal block area has resulted in a solution to the problem. In the old design, the bushing extended through the terminal block to the inside surface of the retainer and was insulated with epoxy. Cracking or breakdown of the epoxy permitted arcing to the bushing.
The problem has been corrected by using a shorter bushing, and insulating both sides of the retainer with phenolic".

This answer was received 6 days before we received the aircraft manufacturer's answer.

So, to ask our question again - "aircraft manufacturer's assistance - help or obstruction ?"

Another clear case of obstruction appears to be the only answer !

The fourth and last case deals with major subject, affecting all airlines - more related to passenger comfort and economical aspects - but it also has a severe safety aspect, as clearly stated in an NTSB report, dated March 8, 1993:

"On July 30, 1992, an aircraft was destroyed by fire after the crew executed a takeoff followed by an immediate emergency landing.
The Safety Board's continuing investigation of the accident developed evidence that corrective action was necessary to ensure that flight attendants have a clear view through windows that are installed in the exit doors.

The flight attendant who was responsible for the L-2 emergency exit was unable to assess conditions outside the airplane, using the exit's prismatic window, because the window's outside pane was either sratched or crazed.
The flight attendant had to leave her exit and move to a passenger window in order to see the conditions outside the exit. After assessing the condition through the passenger window, she found it impossible to return to her exit because passengers blocked the aisle leading to that exit. Fortunately, another flight attendant assumed her position at the exit and, when told by the L-2 flight attendant that it was clear outside, opened the door and the passengers successfully escaped from the burning airplane.
The Safety Board believes that door windows must be properly maintained in order to provide flight attendants the best possible view of the exterior of the airplane through the door. However, as found during this investigation, if the window panes are scratched or crazed, flight attendants may not be able to accurately assess the condition outside a door.

The Safety Board believes that because this condition may exist on other airplanes that have prismatic windows in exit doors, the FAA should ensure that these windows are airworthy and otherwise adequate for their intended purpose."

These unsatisfactory conditions exists on nearly every aircraft, not only on prismatic windows, but on all types of windows made from acrylic.

More recently, from around about April/May 1992 all airlines, worldwide, regardless of aircraft type and also regardless of the routes flown, are being faced with another crazing epidemic.
Within a couple of weeks from installation of new windows no external vision is possible.
This epidemic differs quite a bit from anything seen before within the last 25 years.

One aircraft manufacturer, disregarding most of the actual findings, on August 17, 1992, published the following misleading and partly incorrect information:
"This Service Letter informs operators of recent reports on premature crazing of passenger window outer panes and provides interchangeability information to assist in replacement of existing "standard moisture craze resistant" passenger window outer panes with "improved moisture craze resistance" outer panes.
Major volcanic eruptions inject both chemical and particulate pollutants into the upper atmosphere. Contact with these pollutants can cause premature crazing of flight deck and passenger acrylic windows. The chemicals, particularly sulfates, attack the acrylic window material. The particulates abrade the window pane surface, primarily in the forward fuselage, allowing chemicals to penetrate, causing more rapid crazing.
This can affect all airplanes, but is most common on those that fly long duration flights at high northern latitudes."

The aircraft manufacturer recommends the use of "improved moisture craze resistance "acrylic outer panes as replacements. "Improved moisture craze resistance" panes will minimize the affects of chemicals as a cause of crazing.
When you think about the resources available to a large aircraft manufacturer, I have to conclude that they failed completely to investigate the problem properly and to inform their customers correctly.
Almost nothing in that Service Letter is a true statement of the facts. In the time this statement was made, it was already obvious that:

1) the problem was occuring worldwide-primilary on long range aircraft initially, but with strong indications that short range aircraft were being affected, also,

2) chemical and particulate pollutants ejected from volcanoes, except perhaps the particulate pollutant component, which may cause abrasion to cockpit windows when flying through a cloud of them, have never been responsible for any crazing problems.
Chemicals which later form in the atmosphere or stratosphere may be, at least, partly responsible for some types of crazing.

3) Sulfates do not attack acrylic, but some of their components may cause crazing before the sulfates are formed.

4) The connection to only long range flights in northern latitudes could not be confirmed because all aircraft flying all routes worldwide are affected.

5) The recommendation to use "improved moisture craze resistance" acrylics to solve the problem was totally misleading, because some of these acrylics were even more affected than "standard moisture craze resistant" materials.

One window manufacturer was even better at providing misleading information. The following advertisement could be found in most aerospace magazines just recently:
"Fact: 550 active volcanoes - many blasting sulfuric acid aerosols into the stratosphere are adversely affecting flying conditions for aircraft. It is a fact that early, severe window crazing is largely attributed to the destructive force of sulfuric acid."

Of course they had a fix for that problem available.
But by looking more closely and seriously at the problem, namely the windows, the type of crazing and the possible factors causing the problem, it was already obvious that something else must be responsible.
Materials which are relatively resistant to sulfuric acid attack were being severely affected, while materials normally destroyed by sulfuric acid were only slightly attacked.
Incidentally, the volcanoes never blasted sulfuric acid aerosols into the stratosphere. The blast contains various sulfuric compounds, which are released into the lower atmosphere, later progressing upwards into the stratosphere, where sulfuric acid forms.

This example is a clear demonstration that close cooperation between all involved parties-airlines, window manufacturers, material manufacturers and aircraft manufacturers is essential.

But there is real doubt whether everybody is willing to be involved in close cooperation.

While nothing is known about the real factors or chemical elements causing these problems, one major aircraft manufacturer has already offered three options for future window designs.
At the same time another aircraft manufacturer has stated that, up to now, they have had no information that a problem even exists.
But it would be totally unfair to blame the aircraft and window manufacturers alone.
What kind of help and assistance can you expect from them, when airlines do not properly report their facts and findings ?
What can you deduce when they get reports from an airline, that wonders about, and report crazing solely on only one side of an aircraft parked at an airport with one side lit by the sun and the other side in the shade ?

From my experience, it would be better if airlines discussed their problems directly with the window manufacturers (and material manufacturers, if desired) and also kept the aircraft manufacturers informed.
It is essential that the airlines handle window problems more seriously and not treat them as just another maintenance nuisance.

And finally, coming back to my original question:

"Aircraft window problems - aircraft manufacturer's assistance - help or obstruction ?"
I will conclude by saying that in most cases, problems could have been resolved more easily and faster, if the aircraft manufacturers had acted more responsible within their area of expertise.
Instead of trying to solve the problems alone, the real experts, the window manufacturers should have been directly involved. At the outset this direct contact between airlines and window manufacturers is essential, if problems are to be quickly and efficiently resolved.

In conclusion my answer to the question, unfortunately, has to be "Obstruction".

SESSION IV

EMERGING CAPABILITIES - PART A

Chairman: R. E. Colclough
Flight Dynamics Directorate
Wright Laboratory

Co-Chairman: G. J. Stenger
University of Dayton

Coordinator: K. Alexander
Flight Dynamics Directorate
Wright Laboratory

SURFACE SEAL™ COATED GLASS

G. B. Goodwin
William E. Heidish
PPG Industries, Inc.

Surface Seal™ Coated Glass

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Abstract

A new coating system, developed by PPG for glass aircraft windshield facing, repels water and dramatically improves visual acuity during flight and taxiing operations. Incorporation into the design of a new aircraft may allow elimination of other types of rain removal systems and offers a significant weight savings (the coating will weigh a few milligrams). The Surface Seal™ Coating System applies a thin coating which changes only the wettability and lubricity of the surface and not the optics, hardness, chemical resistance, strength, or other property of the glass. The coating exhibits an extremely low surface free energy which reduces interactions between water and the surface to the point where the water beads up and rolls off due to its cohesive nature. This coating has a very high degree of durability, and is expected to provide enhanced vision for thousands of flight hours.

Introduction

For many years, transparencies have been modified to provide enhanced vision under rainy conditions. Some modifications are based upon silicone polymers which are applied to the outboard surface like a wax and generally have durability for a few days. Another system, the RainBoe® rain repellent system, sprays a thin coating on the outside surface during flight which can be reapplied upon demand (generally within minutes). The RainBoe system uses Freon 113® chlorofluorocarbon as a solvent/carrier which poses environmental hazards. Freon 113 exposure to pilots has also been reported to cause physical impairment of the pilots and has been cause for litigation. (Public documentation of problems has occurred in The Seattle Times, December 16, 1990, and a TV show "Now It Can Be Told" July 10, 1992.) Reported here is a new system for imparting long term water shedding properties to the surface of the glass and allows for enhanced visibility in heavy rains. The treatment system chemically attaches a fluorinated polymer to the exterior glass surface which renders the surface water-repellent or hydrophobic.

The coating imparts high water repellency to clear rain quickly from the surface. Poor vision in rain is a result of a film of water with varying thickness that distorts vision through the transparency due to the formation of varying lenses. If the water cohesively retracts from an extremely repellent surface, the majority of the surface is completely dry with normal vision through those areas. This surface is more repellent to common contaminants and is therefore an easier to clean surface.

The coating is extremely thin (< 200 Å) which means the film cannot be visibly scratched. The film has never been observed to crack, peel, chip, craze, or haze. The thinness is also responsible for the lack of change in optical properties such as index of refraction, transmission, reflection, or color. Finally, the strength of the glass is unchanged. The only properties that change are the wettability (surface energy) and the coefficient of friction (the COF is reduced).

The Surface Seal™ Coating System is based on two materials, the Surface Prep and Coating Solution mixtures. When applied to the glass properly, the surface becomes repellent to water and other substances. Other materials which appear to wet the surface, such as insect residues, are much easier to remove from the surface.

Water repellency is based upon the cohesive forces within a water drop which are in competition with adhesive forces to any surface. A normal glass surface exhibits a high affinity for water and the drop spreads. A plastic surface has fewer sites in the surface which attract water and more sites that have no attraction for water, therefore, the drop does not spread as much on plastic compared to normal glass. Surface Seal Coated Glass has essentially no water-attractive sites and therefore, the cohesive forces within water cause the drop to self-attract, bead up, and roll off. The relative tendency to spread or bead is thus dependent on the surface in question. This tendency is best evaluated by measurement of the contact angle of the vector tangent to the profile of the water drop originating at the surface. Higher contact angles relate to a more water repellent surface. Thus, Surface Seal Coated Glass exhibits a contact angle of ~115°, temporary water repellent treatments ~95°, plastic surfaces 50-75° (dependent on type of plastic and aging of the surface), and clean glass 0-30° (depending on the concentration of surface contaminants), Figure 1.

Durability

The usefulness of repellent coatings is very much dependent on the durability of the coating. Coatings which must be refurbished often add cost and maintenance problems. A coating which lasts the life of the part is the ultimate goal but no such coating exists yet. The Surface Seal Coating System is the only known system which is expected to be efficient for an extended period of time to allow maintenance and refurbishment at the regular maintenance intervals of the aircraft. Durability is evaluated via several accelerated weathering tests and flight test programs.

Wiper Abrasion Resistance. Testing was performed by Grimes Aerospace Co. (Urbana, OH) on a Boeing 767 right hand No. 1 windshield at 43° from vertical. Wiper arm pressure was monitored and adjusted to 16 pounds pressure. Water flow rate was adjusted to 8 gallons per hour per square foot of glass area. Contact angles were measured periodically. At the conclusion of the test (1,544,530 strokes), contact angle averages of the four measured areas decreased only to 86° and the wiper blade was in good shape. That value is above the value range of 50 to 60° that is considered to exhibit acceptable vision, Figure 2.

Rain Erosion. A treated coupon was mounted on a propeller blade and spun at high speed with a water spray equivalent to ~4 inches per hour. This test is extremely severe and has been noted to erode glass and aluminum. The contact angle remained at ~90° throughout the test which has been correlated to be roughly equivalent to several years of actual exposure, Figure 3.

UV/Humidity Resistance. A QUV® tester (The Q-Panel Co., Cleveland, OH) fitted with UVB-313 lamps was used in this test. The test conditions were cycles of 8 hours of relatively dry UV at 65 °C followed by 4 hours of dark, high humidity (near 100%) at 50 °C. The glass was Herculite® II glass (PPG), tin surface. Ten other materials which are temporary hydrophobic treatments (or claim to be water repellent) were also evaluated in side-by-side fashion. Of these treatments, seven are believed to be silicones, one is believed to be a hydrocarbon wax, most do not specify composition. Periodic contact angle measurements were taken. Surface Seal Coated Glass is shown to exhibit relatively high durability, Figure 4.

Humidity Resistance. A Q-C-T® Condensation Tester (The Q-Panel Co., Cleveland, OH) was operated at 60 °C vapor temperature near 100% relative humidity. The glass was Herculite® II glass (PPG), tin surface. The backsides of the coupons were exposed to the normal indoor environment such that moisture inside the chamber continuously condensed and ran down the treated side of the coupon. The same ten other materials described in UV/Humidity Resistance, above, were also tested. Periodic contact angle measurements were taken. Surface Seal Coated Glass is shown to exhibit relatively high durability, Figure 5.

Humidity Resistance (MIL-STD-810E). Humidity testing was done according to MIL-STD-810E, method 507.3, Procedure III, aggravated. Test conditions ranged from 60 °C to 30 °C at 95±5% RH (85% RH minimum during ramps) for a total period of at least 240 hours. Before and after contact angles were within measurement accuracy which shows essentially no degradation, Figure 6.

Modulus of Rupture. Coupons were carefully selected and randomized from a single piece of glass of each type and tempered. The thermally semi-tempered glass was Herculite® I glass and the chemically tempered was Herculite® II glass (both are products of PPG). The tin surfaces of each were tested. The OEM and refurbishment processes were compared to uncoated controls. Statistically, the differences observed were not significant. Thus, the coating had no effect on the basic strength of the glass, Figure 7.

Chemical Resistance. Coupons were exposed to several materials common to the aircraft industry. These included 1% nonabrasive soap, methyl ethyl ketone, Jet-A jet fuel, Skydrol® 500 B-4 hydraulic fluid, Windex® glass cleaner, Coca-Cola® soda, 75% sulfuric acid/water, Type I deicing fluid, and Type II deicing fluid. Coupons of the latter two fluid tests were partially immersed and partially exposed to the vapor above the fluid (no difference was found in the vapor-exposed portion). The other

samples were exposed according to a variation of the test method FTMS 406, method 6053 (30 minutes exposure via saturated cotton pad, unstressed since low stress levels are not significant for glass substrates). The differences are all considered minor and not significant, Figure 8.

Flight Experience. Several flight programs have been started. Details follow:

1. Cessna Citation VII
   - W/S Installed 1/91
   - Production W/S since 8/91
   - FAA Certification 1/92
   - First Citation VII delivery 4/92
2. BAe 146 (Air Nova)
   - LH, RH #1, 2 W/S installed 4/91
   - Inspected: 7/91 at 903 hours
     - 11/91 at 1410 hours
     - 7/92 at 3035 hours
     - All windows remain effective
3. CANADAIR RJ Test Aircraft
   - LH #1,2 installed 5/91, refurbished 7/93
4. deHAVILLAND DASH 8
   - LH #1 W/S installed 7/91 (Henson)
   - Inspected: 7/92 at 2718 hours
     - W/S remains effective
   - LH #1 W/S installed 9/91 (Tyrolean)
     - Favorable pilot's comments 11/91
     - Favorable pilot's comments 7/92
     - Favorable pilot's comments 4/93
5. PIPER CHEYENNE (Corporate Aircraft)
   - LH #1 W/S installed 9/91
     - Favorable pilot's comments 4/92
6. AIRBUS A320 (Northwest Airlines)
   - LH #1 W/S installed 3/92
   - Inspected: 2/93 at 3106 hours
     - Coating is providing enhanced vision
7. BOEING 777 test aircraft (Ansett 757)
   - RH #1 W/S coated on aircraft 5/92
     - Favorable pilot's comments
8. FAIRCHILD METRO (Conquest Airlines)
   - LH #1 W/S coated on aircraft 6/92
     - Favorable comments 2/93
9. AIRBUS A340 Test Aircraft
   - RH #1 W/S installed 9/92
     - Favorable pilot's comments

10. GULFSTREAM II and IV SP (Gulfstream)
    LH, RH #1 W/S coated on aircraft 10/92
    Favorable pilot's comments 3/93
11. BOEING 737 (Linjeflyg/SAS)
    Four RH W/S coated on aircraft 1/93
    Favorable comments 4/93
12. BOEING 737 (Lufthansa)
    RH W/S coated on aircraft 1/93
13. SIKORSKY S-76 Helicopter (Chevron)
    W/S installed 3/93
    Favorable pilot's comments 7/93
14. BEECHJET 400A (Fish Eng. and Constr.)
    LH #1 W/S coated on aircraft 3/93
    Favorable pilot's comments

Status

The Surface Seal Coating System has been commercialized on Cessna Citation VII for some time. With the Boeing Company, the coating system has been selected as baseline for the 777 aircraft to replace the RainBoe® rain repellent system. A qualification program has been completed, FAA certification has been granted for the 737/757/767 aircraft, and an in-service flight evaluation is scheduled on several different models to commence in the third quarter of '93. Upon approval, all current Boeing models are to be certified with Surface Seal™ Coated Glass. OEM's and operators are being introduced to the product on a timely basis.

Summary

An enhanced vision system for aircraft transparencies is now available that will serve as a rain removal system. This new system does not require the installation of feed lines, valves, or canisters. The weight of the system once applied to the glass is on the order of a few milligrams which is less than one-millionth the weight of the RainBoe system (generally 15-20 pounds). The coating eliminates the hazards associated with the use of Freon 113 in the RainBoe system. The coating can be supplied on an OEM part or applied to installed aircraft transparencies and has been shown to be durable to weathering and chemicals for an extended period of time. The only glass properties which change are the wettability and the coefficient of friction (no changes to strength or optics). The system is currently available through PPG's Aircraft Products business unit.

RainBoe® Rain Repellent System is a trademark of the Boeing Company, Freon 113® chlorofluorocarbon is a trademark of DuPont, QUV® tester and Q-C-T® Condensation Tester are trademarks of The Q-Panel Company, Herculite® I and Herculite®II glasses are trademarks of PPG Industries, Inc., Skydrol® 500 B-4 hydraulic fluid is a trademark of the Monsanto Company, Windex® glass cleaner is a trademark of the Drackett Co., Coca-Cola® soda is a trademark of the Coca-Cola Company.

Contact Angles

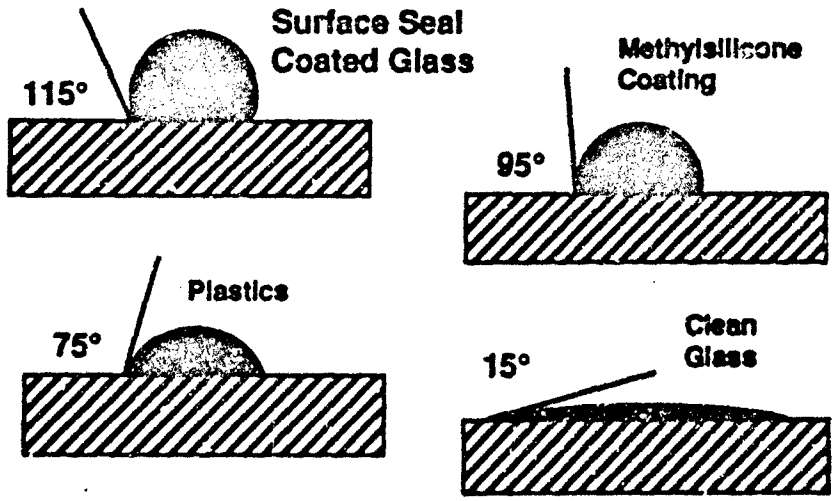


Figure 1. Contact angles of water on glass and plastics. Each material has unique surface properties that cause different wettabilities.

Wiper Abrasion Resistance

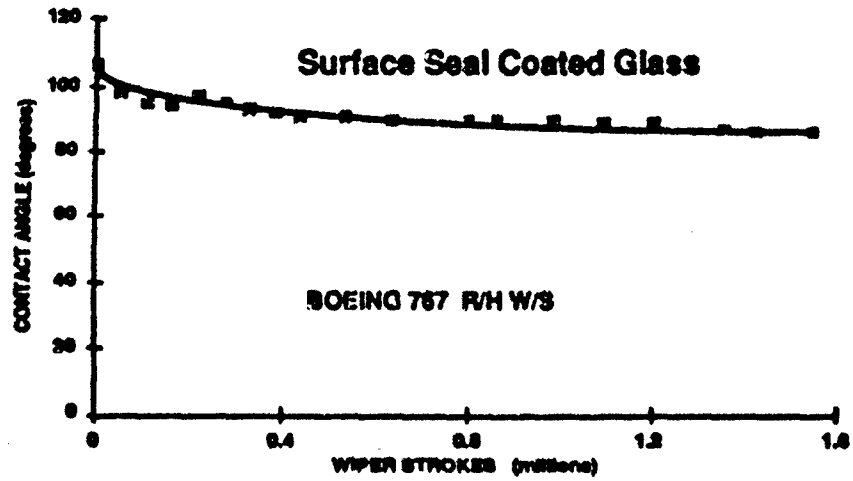


Figure 2. Wiper Abrasion Resistance is illustrated by contact angle stability up to 1.5 million wiper strokes.

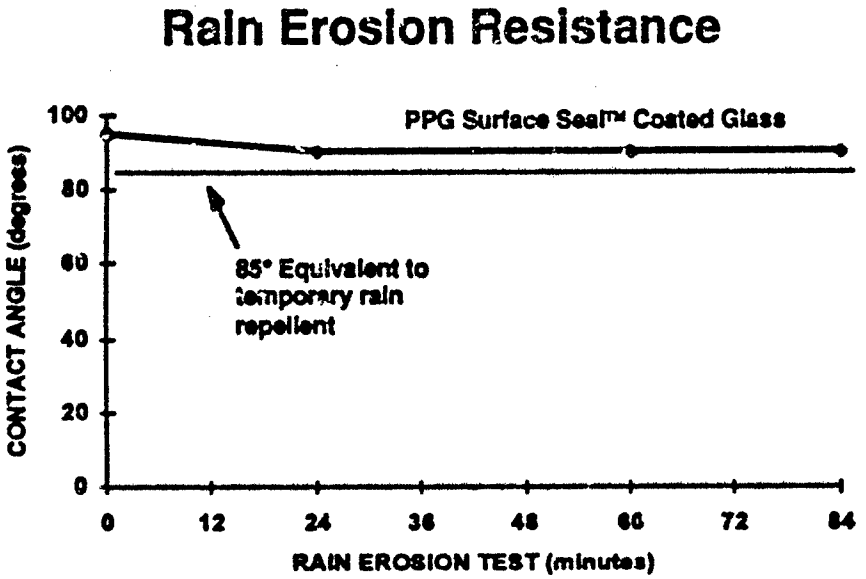


Figure 3. Resistance to rain erosion simulation. Also shown is a straight line which is representative of the wettability of a typical rain repellent immediately after application without weathering.

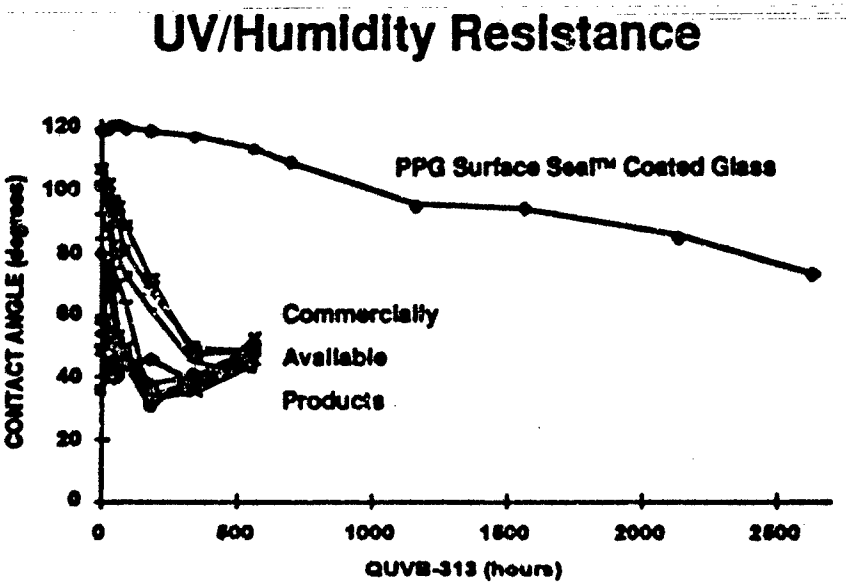


Figure 4. Comparisons of durability of Surface Seal™ Coated Glass and various common hydrophobic coatings under the action of combined UV light and humidity. Testing on Herculite II glass (tin side).

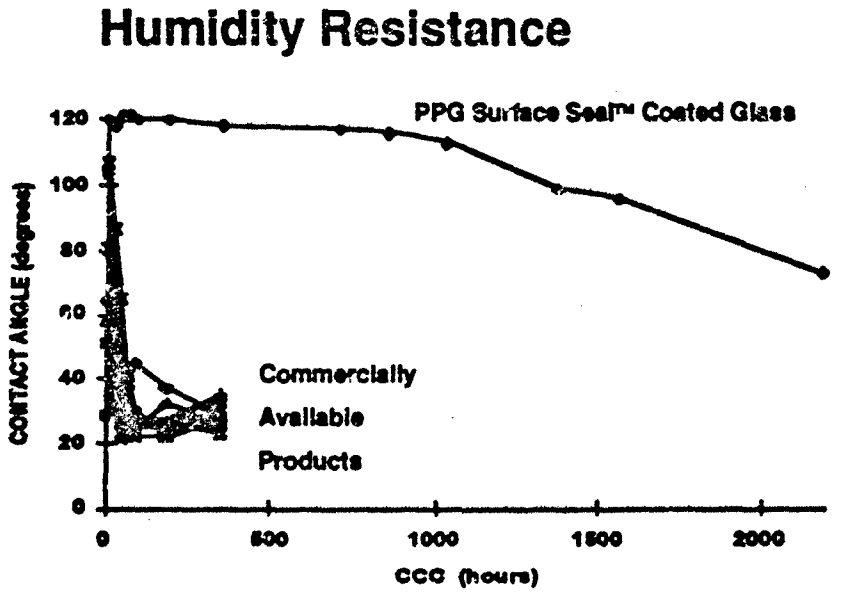


Figure 5. Comparisons of durability of Surface Seal™ Coated Glass and various common hydrophobic coatings under humidity only accelerated testing. This is a severe test for many coatings. Testing on Herculite II glass (tin side).

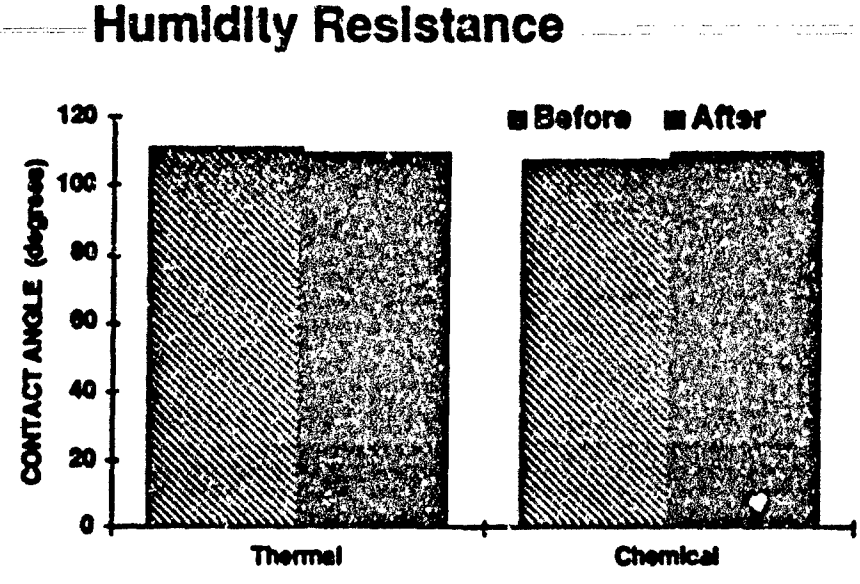


Figure 6. Cyclic humidity testing of Surface Seal Coated Glass on Herculite I (thermal semi-temper) and Herculite II (chemically tempered) glasses (both tin and atmosphere sides, both OEM and refurbishment processes used). Test according to MIL-STD-810E, Method 507.3, Procedure III, aggravated.

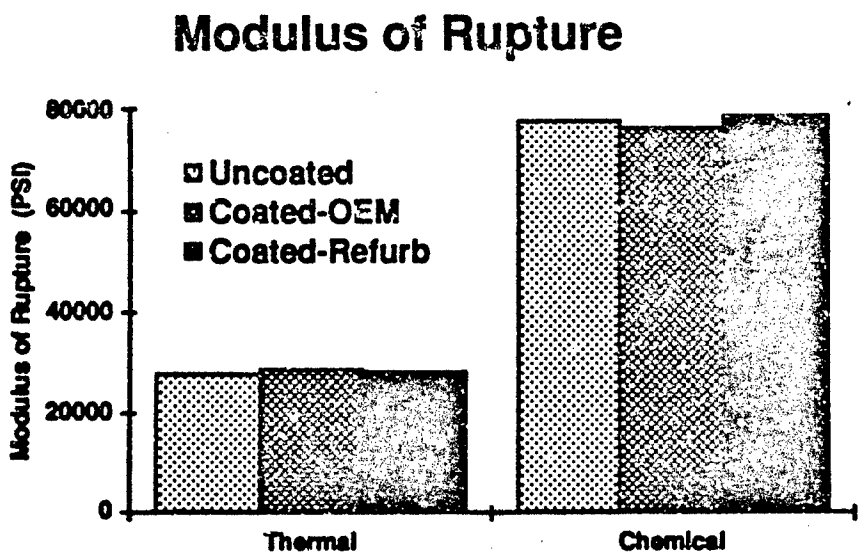


Figure 7. Modulus of rupture data (units of PSI) of uncoated, OEM-coated, and refurbishment-coated Surface Seal Coated Glass on Herculite I (thermally semi-tempered) and Herculite II (chemically tempered) glasses (tin side only since this side is generally the weaker side).

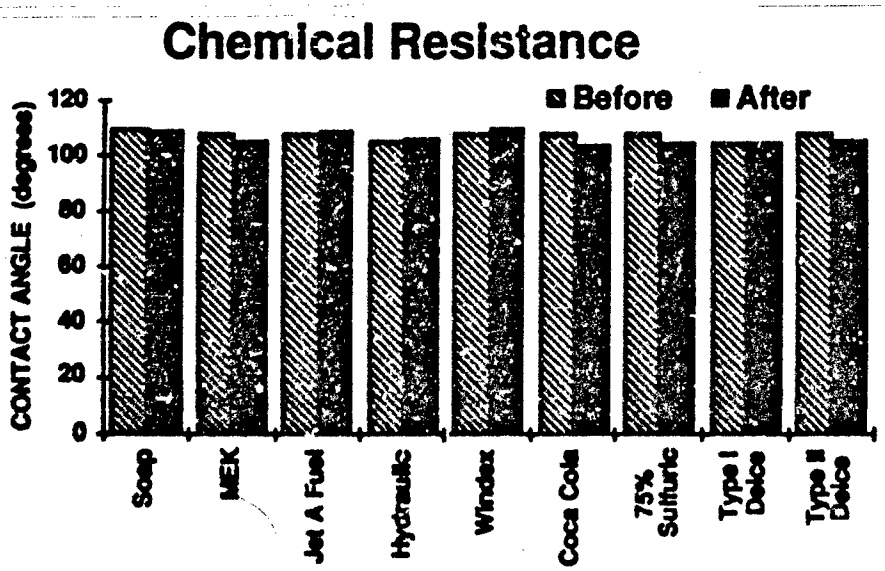


Figure 8. Exposure of Surface Seal Coated Glass for 30 minutes to various chemicals via a cotton pad. For simplicity, the data represents the average of Herculite I and Herculite II glasses (both tin and atmosphere sides, both OEM and refurbishment processes used).

SCRATCH RESISTANT COATINGS FOR AIRCRAFT TRANSPARENCIES PREPARED BY PLASMA POLYMERIZATION

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Scratch Resistant Coatings For Aircraft Transparencies Prepared By Plasma Polymerization

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Abstract

Good adherent, transparent coatings were deposited in a plasma polymerization process on PMMA and PC substrates. Therefore a reactor with 1 m^3 volume was used.

HMDSO was provided as the monomer gas and by the admixture of O_2 the composition of the coating was varied and a gradient structure obtained. This was investigated by IR - spectroscopy and XPS measurements.

Abrasion resistance was tested with the taber abraser method. The failure mode was recorded by SEM- micrographs.

A superior performance of coatings with gradient structure (600 cycles: Haze = 6 % for PMMA, 5 % for PC) compared to homogeneous films is obtained.

Introduction

The technology of low temperature plasmas offers a large area of applications for polymers. One field of application are scratch resistant hard coatings on polymer substrates. For an improved scratch resistance it is possible to generate coatings with gradient structure in a plasma polymerisation process. Furtheron the direct combination of plasma activation and cleaning processes with film deposition in the same reactor seems to be advantageous.

In the past, few attempts have been made to use plasma polymerized films for abrasion resistant coating of polymers. But only homogeneous films were tested. Because such a coating should have high hardness at the outer surface and should be ductile and good adherent to the substrate in the interface region, a film with gradient structure promises improved abrasion resistance over homogeneous films.

Plasma polymerized coatings

In Fig. 1 the schematically set up of the plasma reactor used is shown. In this reactor the plasma is generated by an RF-discharge between two parallel electrodes. With the aid of a directly coupled mass spectrometer and with optical emission spectroscopy (OES) it is possible to have a look at the plasma polymerization reactions when the plasma is active. The monomer/gas inlet is operated by gas flow controllers that are computer controlled.

Plasma polymerized films were deposited on stretched acrylic (PMMA 249, Röhm) and polycarbonate (PC Makrolon, Bayer) substrates. For both types of substrates the following process steps were performed:

- o cleaning and rinsing in water based solutions
- o drying (in vacuum)
- o plasma surface activation in an O2/Ar - atmosphere
- o deposition of plasma polymerized film, HMDSO-monomer + O_2

Two types of plasma polymers were generated: a homogeneous film and a film with a gradient structure. For the gradient structure the gas composition as well as the plasma parameters were changed in several steps. The first layer of the film has to fit best the mechanical properties of the substate with best adherence, whereas the deposited top layer should be as hard as possible. Since such a structure is better adjusted to the needs of scratch resistance an improved behaviour is expected compared to any type of homogeneous film.

Infrared spectra

In Fig. 2 the IR-spectra of a plasma polymerized film from HMDSO-monomer without oxygen admixture and a film from HMDSO-monomer with oxygen admixture are shown. Strong differences between these spectra are recognized. Without oxygen admixture a lot of characteristic features of the starting monomer HMDSO are present.

These IR-bands indicate, that the degree of fragmentation of the monomer is low. The plasma-polymer consists basically of a low fragmented HMDSO-structure that is highly cross linked.

In the IR-spectrum of the film with oxygen admixture several of the above mentioned IR-bands are missing. Especially the bands of the methyl group and its fragment CH_2 are completely absent. The remaining structures may be attributed to a strong Si-O band and a silicon-carbon structure that may contain some remaining hydrogen.

Elemental composition

The elemental composition for the coatings was determined by XPS. Fig.3 gives the composition of the same type of coating with oxygen admixture that corresponds to the IR-spectrum in Fig.2 (bottom). Hydrogen is excluded in the calculation because it is not detected by XPS.

The oxygen content of this plasma polymer is much higher than that of the starting HMDSO-monomer and it is similar to quartz glass, indicating the strong inorganic character of this coating type.

At the same time the carbon content of 67 at % in the monomer is decreased to < 10 at % in the coating. From XPS detail scans it can be concluded that the remaining carbon in the coating belongs to Si-C like chemical bonds.

These results correlate very well with the IR-measurements and in summary it was found that the HMDSO-plasma polymer with oxygen admixture has a highly cross linked structure, mainly with $Si-O_2$ like bonds and a small

content of Si-C like bonds. Both types of bonds are interconnected and form an amorphous structure that contains some (li`tle) remaining hydrogen from the starting monomer.

Abrasion resistance

The abrasion resistance of coated PMMA- and PC-samples was determined by the Taber abraser method. This test was done in accordance to DIN 52 347E, using CS-10F wheels and a load of 2.7 N.

In this test the sample surface is exposed to a complex combination of shear and tensile stresses. For coatings with a thickness of a few µm the stress on the surface of the coating is transferred to the boundary between the original substrate and the deposited thin film. This boundary layer is heavily loaded and a scratch resistant thin coating requires good adhesion and must fit to the mechanical properties of the substrate.

In order to record the type of failure that occurs in the Abraser test, SEM-micrographs were taken from the surface of the samples. Figs.5 and 6 provide typical micro graphs of coated PMMA an PC samples. For both types of samples a strong influence of the substrate's properties on the failure mode must be recognized. Obviously the scratches were propagated from the surface to the substrate material. Scratches that are only present in the coating were not observed.

The failure mode of the coating on PMMA is loss of adhesion. Nevertheless the scratch resistance is good (see below) and the coatings have passed the tape adhesion test by 100 %. If the samples were loaded with more cycles

% after 600 cycles, whereas uncoated PMMA shows already 16 % Δ Haze after 50 cycles in this test. The improvement in scratch resistance obtained by the application of plasma polymerized, graded films on PMMA is quite high and our preliminary results are encouraging. With respect to the low film thickness of only 1,5 µm a further improvement in scratch resistance seems to be possible.

Almost the same improvement in scratch resistance is observed for coated PC with gradient structure of the coating system. In Fig. 10 the results of Δ Haze measurements for such a system are given. After 600 cycles in the abrasion test, Δ Haze amounts to 5%, which is a good scratch resistance for PC .

Conclusions

- ◊ A plasma polymerized coating generated from HMDSO monomer + O_2 was developed with a gradient film structure
- ◊ Coating thickness ranges below 2 µm
- ◊ Typical deposition rate is about 10 nm/min
- ◊ Stretched acrylic and PC substrates were plasma activated and immediately coated with plasma polymer
- ◊ Between the RF- electrodes a good homogeneity in thickness is obtained in a large fraction of the inter electrode distance (10- 90 %)
- ◊ Taber abrasion resistance of PMMA and PC is strongly enhanced by the coating

- ◊ Plasma polymers with gradient structure are superior to homogeneous plasma polymerized films
- ◊ Further work of development is necessary to take advantage of the technical potential of plasma polymers for transparencies

Fig. 1

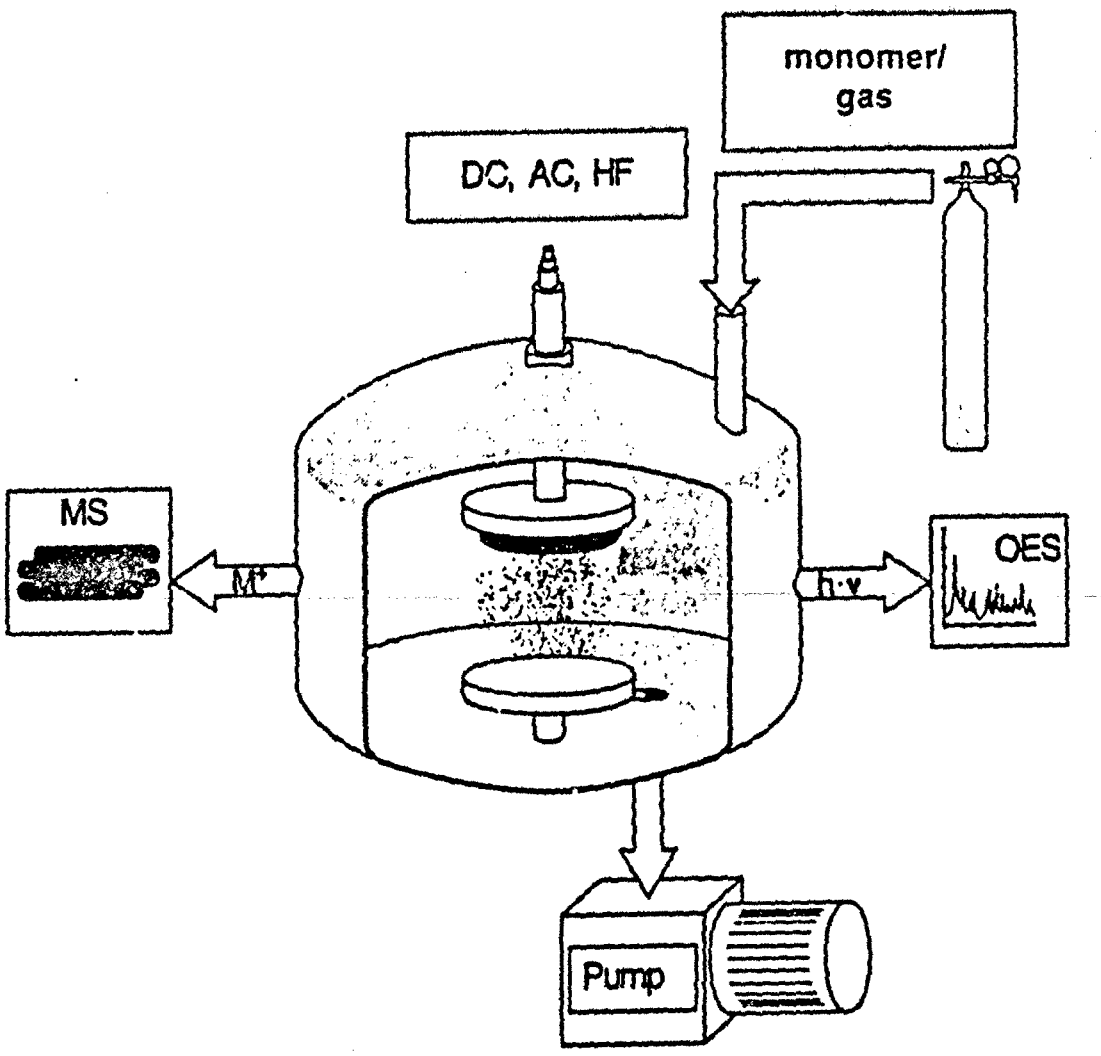


Fig. 3

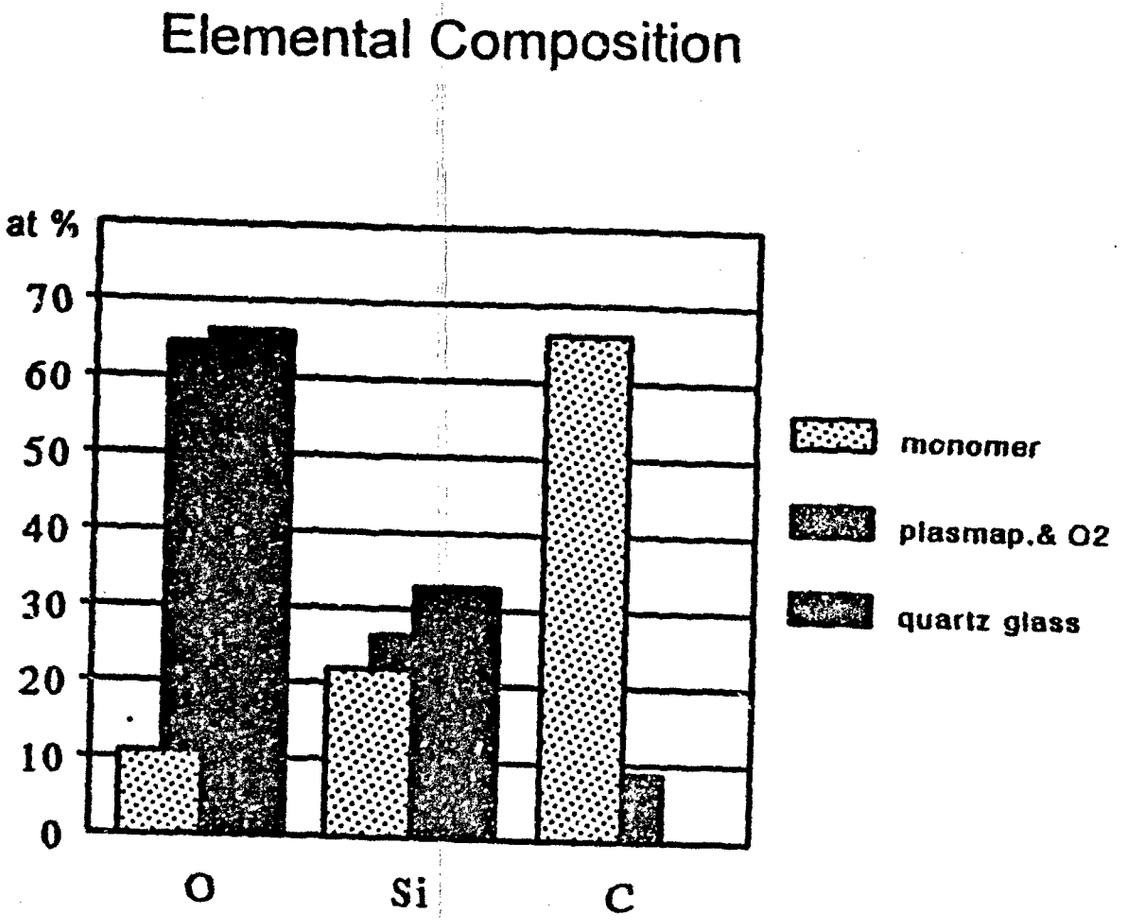

Elemental Composition
at %
70
60
50
40
30
20
10
0
O
Si
C
monomer
plasmap.& O2
quartz glass

Fig. 4

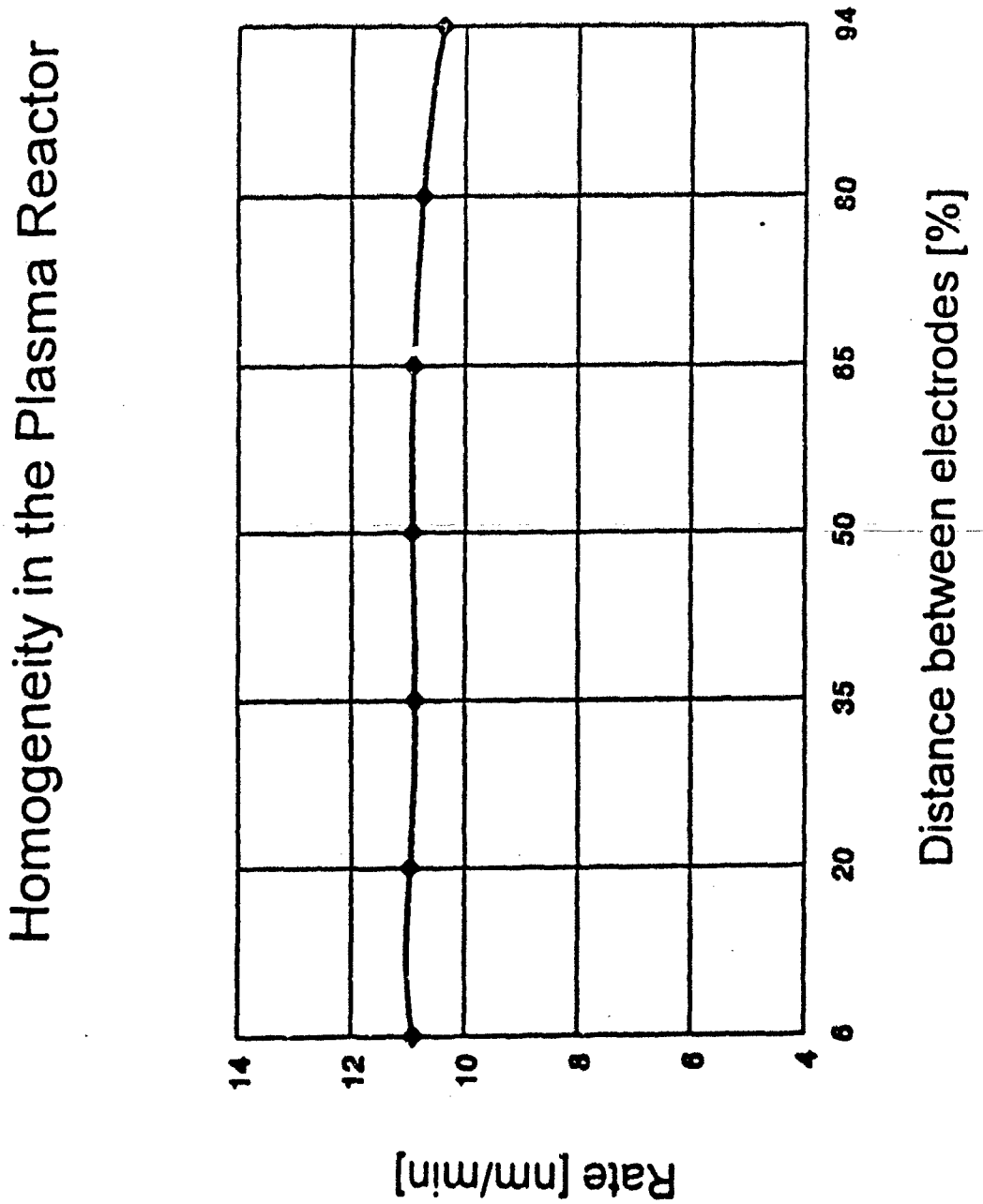
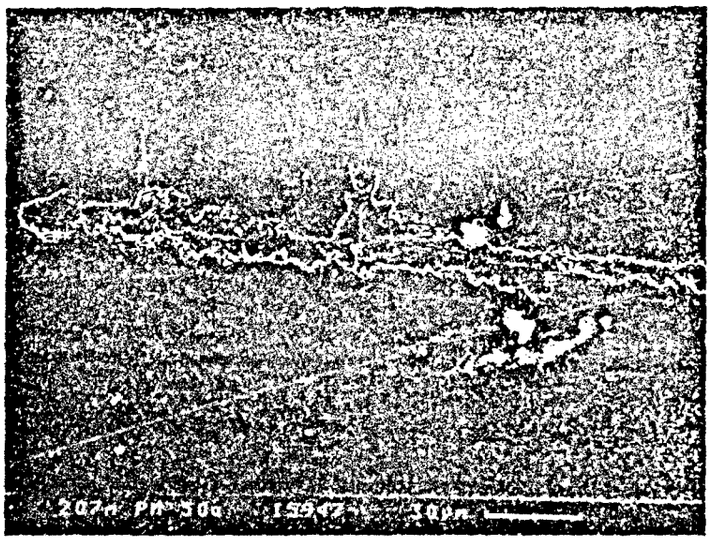
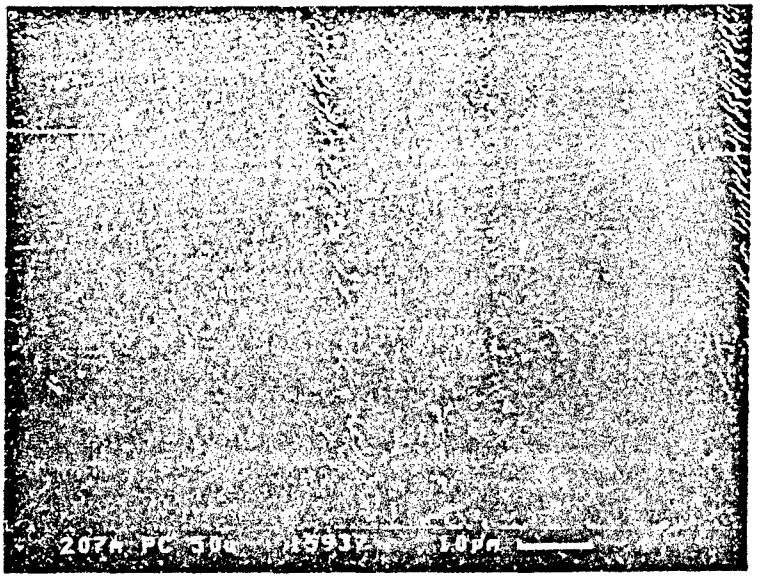

Homogeneity in the Plasma Reactor
Rate [nm/min]
14
12
10
8
6
4
Distance between electrodes [%]
6
20
35
50
65
80
94

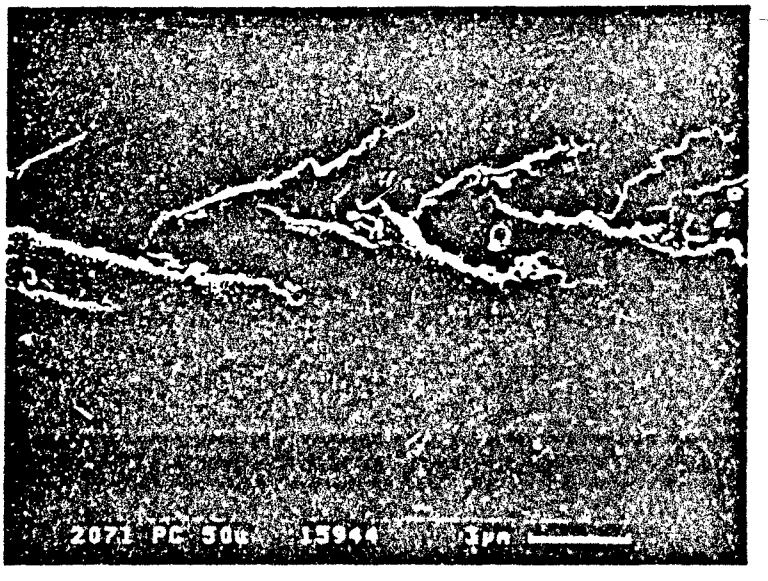
Fig. 5



PMMA, coated HMDSO + O_2
Taber Abraser, 2.7 N load, 50 cycles

Fig. 6





PC, coated HMDSO + O_2
Taber Abraser, 2.7 N load, 50 cycles

Taber Abrasion Resistance
PMMA with plasma polymerized (graded)
HMDSO + O_2

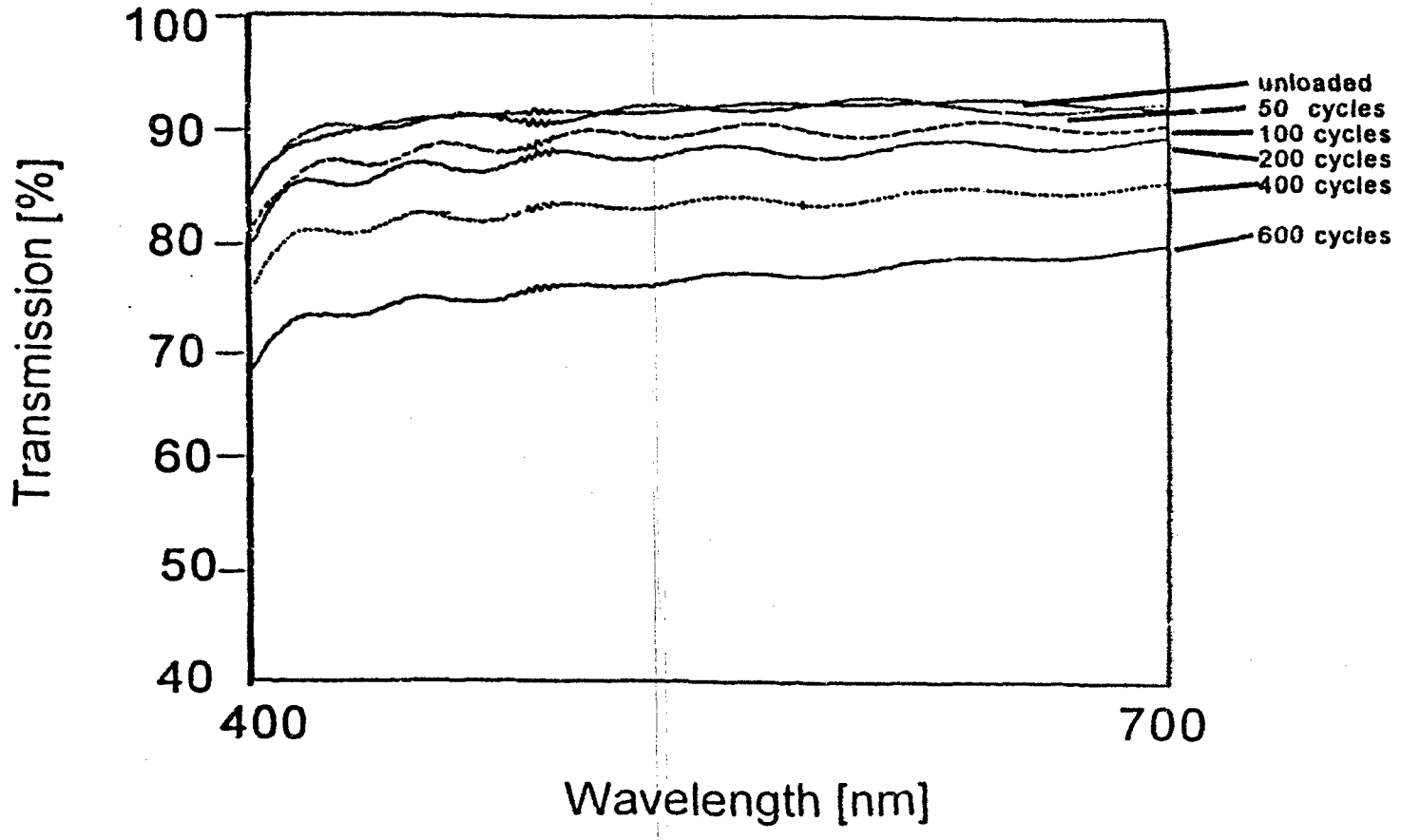


Fig. 7

505

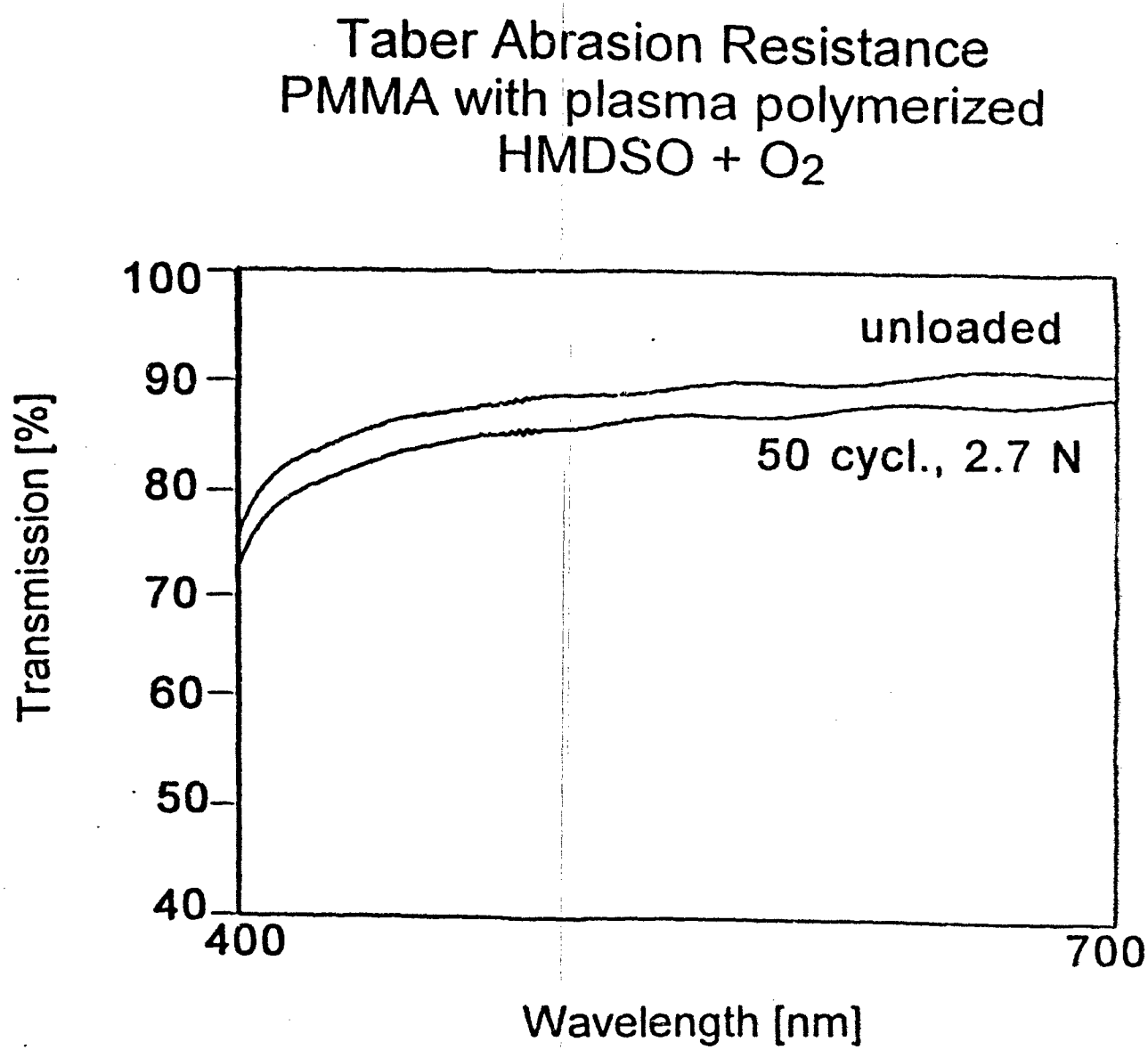
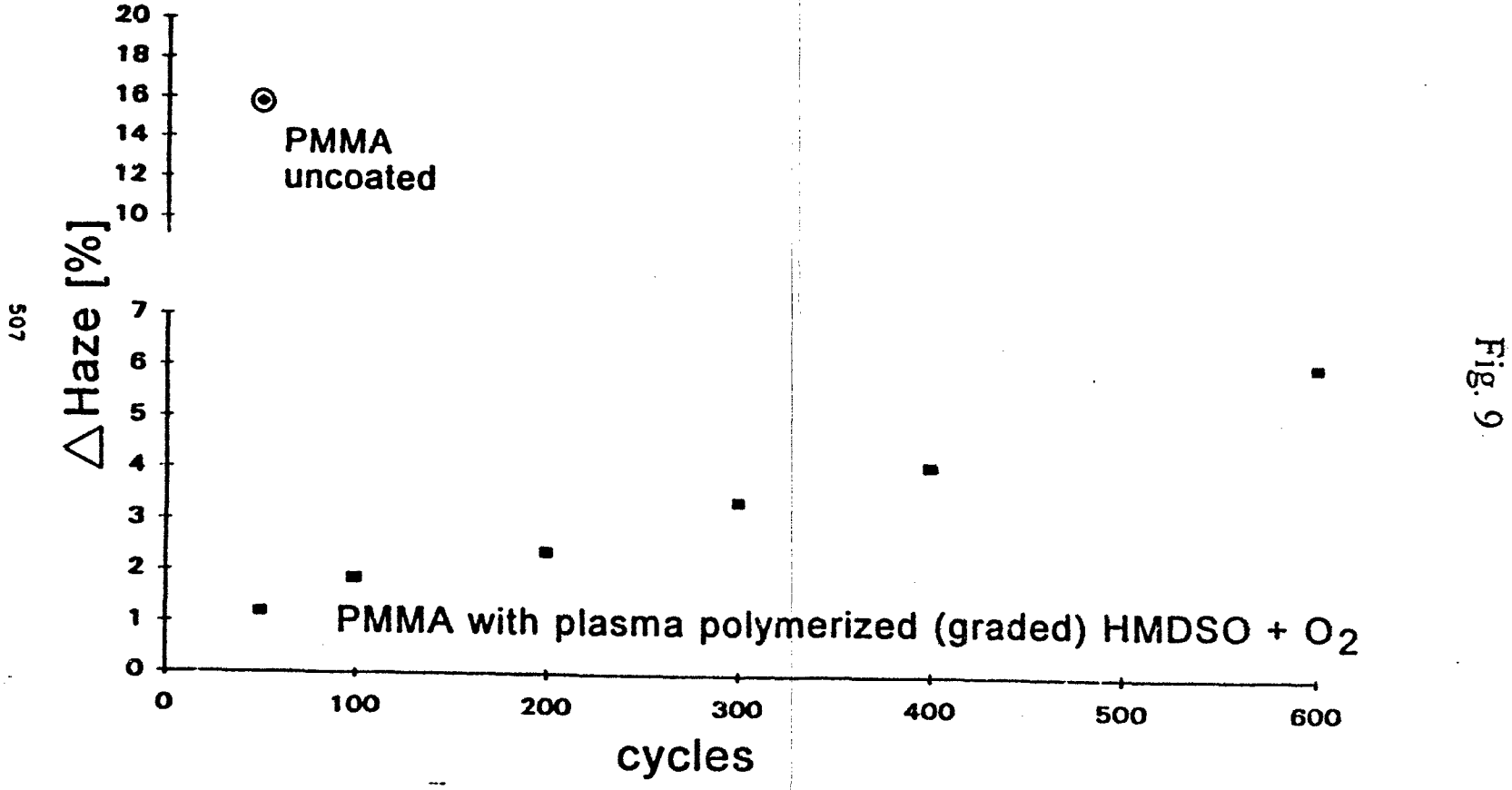

Taber Abrasion Resistance
PMMA with plasma polymerized
HMDSO + O_2
100
90
80
70
60
50
40
Transmission [%]
unloaded
50 cycl., 2.7 N
400
700
Wavelength [nm]

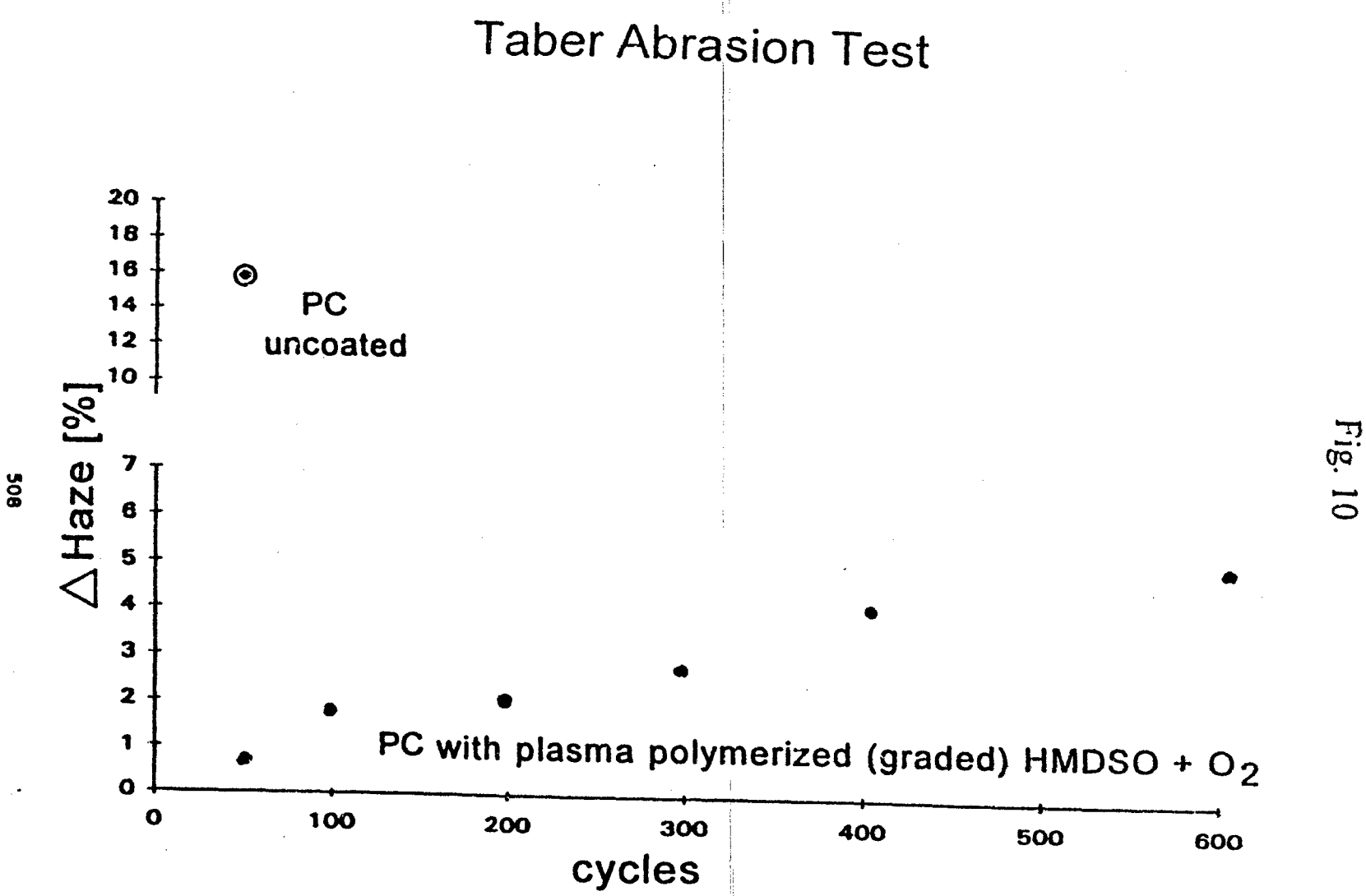
Fig. 8

906


Taber Abrasion Test
20
18
16
14
12
10
PMMA
uncoated
7
6
5
4
3
2
1
0
ΔHaze [%]
PMMA with plasma polymerized (graded) HMDSO + O2
0
100
200
300
400
500
600
cycles

507

Fig. 9


Taber Abrasion Test
20
18
16
14
12
10
PC
uncoated
7
6
5
4
3
2
1
0
Δ Haze [%]
PC with plasma polymerized (graded) HMDSO + O2
0
100
200
300
400
500
600
cycles

508

Fig. 10

VERSATILE APPLICATIONS OF S-243 PROTECTIVE COATING SYSTEM ON AIRCRAFT TRANSPARENCIES

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J. A. Ruffo
Sierracin/Sylmar Corp.

VERSATILE APPLICATIONS OF S-243 PROTECTIVE COATING SYSTEM ON AIRCRAFT TRANSPARENCIES

by

Alexander Z. Bimanand
John A. Raffo

Sierracin/Sylmar Corporation

ABSTRACT

Sierracin/Sylmar's S-243 is a weatherable, ductile polyurethane-based protective coating system. The system provides in-service protection for the electrically conductive coatings such as gold and indium tin oxide, as well as the outer acrylic and polycarbonate surfaces of advanced aircraft transparencies with excellent maintainability and optics. S-243 offers significant advantages over the acrylic and polycarbonate substrates. The advantages include superior weatherability and excellent resistance to abrasion, solvent/ chemicals and crazing. The test results, properties and performance of S-243 with respect to its superior adhesion, craze resistance and environmental protection of aircraft transparencies are discussed.

INTRODUCTION

Coatings have been used for surface modification of metallic materials with great success in aerospace and other high technology applications for many years. Metals have been treated with various coatings to modify the original surface to improve hardness, wear resistance, corrosion resistance and to provide resistance to chemical attack. In these applications the surface coating has enhanced the properties of the underlying material by addition of a physically similar but chemically superior material to the surface or by chemically changing the initial metal surface through chemical reaction.

In the case of coatings for plastic materials, the use of a physically or chemically similar material to protect the substrate to be modified has proven unacceptable since it is these very properties which are undesirable in the basic substrate. One approach has been to provide plastics with a 'glass-like' hard coating in hopes of imparting the desirable properties of glass to plastics. Unfortunately, the very nature of the coating concept, i.e. a thin (less than one mil) surface covering with substantially different properties to the substrate, has produced less than the desired results. The hard coatings do substantially modify the surface but are too thin to protect the underlying softer plastics from damage from puncture, cuts and physical impact. The coatings are too thin to offer real mechanical protection and are subject to fracture due to differences in thermal expansion between coating and substrate. These materials are, however, able to improve weatherability, abrasion, and solvent resistance due to their glass-like chemistry. Softer coatings, such as those based on acrylate, polyester and polyurethane chemistry, applied by thin surface applications are unable to provide long lasting protection from all of the environmental hazards endemic to modern, high performance aircraft transparencies. These softer coatings are more extensible and abrasion/impact resistant than their hard counterparts; but, they suffer from mediocre solvent and rubbing abrasion resistance. In general, the softer materials have not performed as well when exposed to exterior environments, namely ultraviolet light and humidity.

These deficiencies in coating materials for optical transparencies have caused the industry to look to laminated outer plies of acrylic or glass, with some success, for protection of structural components. The mystique of the thin coating concept, with substantially simpler processing, savings in weight, and the potential for improved optics, has survived in hopes of technological improvements which would make the concept viable. Recent advances appear to have answered this need.

Sierracin/Sylmar Corporation's S-243 coating system has been developed to combine the beneficial effects of the prior coatings while eliminating the technical drawbacks exposed in field service of those same materials. S-243 was developed to be versatile, i.e.

it can be applied to a variety of substrates, acrylic, polycarbonate, and metals, and to perform optically at increased thickness compared to prior coatings. Since no one material can be expected to fulfil the requirements of adhesion to a variety of substrates, the coating system was engineered to combine elastomeric, extensible, chemical and abrasion resistant topcoat with various undercoats, primers or tiecoats, to achieve the versatility goal. The data presented in this study validates the technical approach taken to provide the advanced transparency industry with a superior and cost/weight effective product.

S-243 PROTECTIVE COATING SYSTEM

The S-243 protective coating system is comprised of a proprietary universal primer, FX-217, and an aliphatic, highly crosslinked urethane topcoat, FX-216, which is covalently bonded to the FX-217 primer. The benefits of the primer in the S-243 system are:

(a) To promote the adhesion of FX-216 urethane coating to plastic and/or metallized plastic surfaces and to retain the adhesion after environmental exposures.

(b) To enhance weatherability of the underlying layers and the substrate.

(c) To improve the solvent resistance of FX-216. This effect is more pronounced when the thickness of FX-216 is in the range of 0.5 to 3 mils.

(d) To protect the substrate surface from the aggressive materials and/or solvents of FX-216 during the casting and/or flow coating application.

(e) To increase the surface energy of the substrate in order to enhance the surface flow and wetting properties of FX-216. This increase results in superior leveling and cosmetics of FX-216.

The primer is applied onto the substrate by flow coating technique and cured before the application of FX-216.

The FX-216 topcoat is responsible for most of the protective properties of the S-243 system. The benefits of the topcoat are:

(a) Excellent abrasion resistance

(b) Good dust erosion resistance

(c) Excellent hydrolytic stability

(d) Excellent ultraviolet light (UV) stability

(e) Excellent chemical/solvent resistance

(f) Excellent chemical stress craze resistance

(g) Good elongation and ductility

(h) Good rain erosion resistance

After a short review of polyurethane chemistry and its manufacturing methods, and application procedures of FX-216, the attributes of the S-243 system on different substrates will be discussed.

POLYURETHANES

Polyurethanes, and the closely related polyureas, are the products of the reaction of isocyanates with the active hydrogen containing compounds. The most common route for the synthesis of such polymers is the addition/step-growth polymerization of isocyanates by reaction with di- and polyfunctional hydroxyl compounds with di- or polyisocyanates. When difunctional reactants are used, linear polyurethanes are produced, whereas branched or crosslinked polymers are obtained when the functionality of the reactants is raised to more than two.

Chemistry of Polyurethanes

All urethane compositions can be broken into three major components: isocyanate(s), polyglycol (backbone material), and the chain extender or crosslinker portion.

Even though the main reaction product is the urethane linkage, other types of isocyanate reactions may be occurring in a system. This possibility is most pronounced at elevated temperatures and in the presence of certain catalysts. The reactions taking place with isocyanates can, in general, be divided into two main classes:

(1) The reaction of isocyanates with compounds containing reactive hydrogen

These compounds include water as well as hydroxyl and/or amino functional compounds to give additional products. These reactions can further be classified into two groups:

(a) The primary addition of isocyanates to active hydrogen compounds (Figure 1).

(b) The secondary addition reactions involving active hydrogen in the primary products and the isocyanates (Figure 2).

Although these secondary reactions occur to a relatively small extent by comparison with primary reactions, the importance of these reactions should not be underestimated. For example, formation of allophanates and particularly biurets, is responsible

for introducing crosslinks and branches in the polymer network which could influence the properties of polyurethane liners and/or coatings considerably.

(2) Polymerization of isocyanates (self-addition)

Isocyanates, especially aromatic based constituents, can react with themselves to form dimers, or "uretidine diones". They can also be polymerized to form carbodiimide, and 1-nylon structures. Both aliphatic isocyanates and aromatic isocyanates form trimers, or "isocyanurates". These reactions are presented in Figure 3. These types of reactions may also give rise to branching and crosslinking in the polymer chains and thereby influence the properties of the resulting polyurethanes.

Morphology of Polyurethanes

Thermoplastic polyurethanes are randomly segmented copolymers consisting of essentially linear polymer chains. The structure of these chains is predominately comprised of relatively long, flexible, "soft" chain segments which have been joined end-to-end by rigid "hard" chain segments through covalent chemical bonds. The "soft" segments are diisocyanate-coupled, low melting macroglycols generally with a molecular weight of between 600 and 3000. The "hard" segments include single diurethane bridges resulting when a diisocyanate molecule reacts with a macroglycol, but more particularly they are the longer, high melting polyurethane chain segments formed by the reaction of diisocyanate with the small glycol chain extender component such as 1,4-butanediol.

The polar nature of the rigid, hard, urethane chain segments results in their strong mutual attraction through the hydrogen bonding as shown in Figure 4. This interaction induces aggregation and ordering into crystalline domains in the polymer matrix (Figure 5). The net effect of hydrogen bonding is to tie together or "virtually crosslink" the linear polyurethane chains. That is, the primary polyurethane chains are crosslinked in effect, but not in fact. These types of crosslink units can be disassociated at higher temperatures to reform the material's thermoplastic character. Due to this effect, polyurethanes display the properties of strong, vulcanized rubbers over a practical range of use temperatures.

Hydrogen bonding is a relatively labile chemical bond which can be broken reversibly with heat and, depending upon polymer composition, with solvation (Figure 5). This phenomenon offers many processing alternatives for thermoplastic polyurethanes. Thermal energy great enough to (reversibly) break hydrogen bonds, but too low to appreciably disrupt the strong covalent chemical bonds, can be applied to extrude or mold the polymers. Dissolving the polymer in a solvent which reversibly breaks the hydrogen bonds makes the polymer suitable for coating applications.

The hard segment imparts rigidity, hardness, and strength to the polymer through hydrogen bonding. The concentration of the hard segment is controlled by the total NCO content. The soft segments increase the flexibility and reduce the modulus of the elastomer. They also have a significant affect on the glass transition temperature (Tg), environmental resistance, chemical resistance, and low temperature properties. This segment is controlled by the type and the molecular weight of the macroglycol. Hydrogen bonding restricts the mobility of the macroglycol's long chains and reduce their ability to organize into crystalline lattices. As a consequence, the soft segments remain as amorphous or semi-ordered regions in the polymer network. The hard and soft segments are generally incompatible due to insolubility of the hard segment in the soft segment at room temperature. This insolubility is known as phase separation. The degree of phase separation displays a great influence on the properties of elastomers.

With incorporation of crosslinkers, such as multifunctional hydroxyl compounds and multifunctional isocyanates, thermoset polyurethanes are formed. Increasing the degree of crosslinking of an amorphous polyurethane generally increases the rigidity, softening point, and modulus of the polyurethanes and reduces elongation. Largely crystalline polymers may be affected differently by small increases in crosslinking. A few crosslinks may reduce crystallinity by reducing chain orientation, changing a high-melting, hard, dense crystalline polymer into a more elastic, softer, amorphous material with improved solvent resistance.

Typical Isocyanates and Macroglycols

A fairly wide range of diisocyanates are available which offer a broad range of properties in a host of applications. They may be categorized as difunctional vs. polyfunctional, aromatic vs. aliphatic and/or monomeric vs. oligomeric. Aliphatic isocyanates have superior hydrolytic and UV resistance. Examples of aliphatic di- and multifunctional isocyanates are shown in Figure 6.

Different classes of hydroxyl containing macroglycols are available commercially. They include polyether, polyester, polycarbonate, polydimethylsiloxane, and polythioether based glycols as presented in Figure 7. Each macroglycol imparts its own specific physical properties such as abrasion and solvent resistance, hydrolytic stability, resistance to UV light, and high/low temperature performance in the polyurethane systems.

Typical Chain Extenders/Crosslinkers

Variations in the structure of the chain extender alter the characteristics of the hard segment and thus has a considerable effect upon the overall properties of the polymer matrix. The symmetrical and nonbranched aliphatic diol extenders, such as 1,4-butanediol, provide efficient hydrogen bonding and chain alignment resulting in rigid structures in polyurethanes. Typical chain extenders and crosslinkers are shown in Figure 8.

Catalysts and Additives

Catalysts play an important role in the preparation of polyurethanes. They not only shorten the reaction time, but also direct the course of various isocyanate reactions during the preparation.

UV absorbers/stabilizers are added to absorb the UVA and UVB radiations from the sun and to increase the resistance of urethane coatings to yellowing/degradation and, in general, to improve outdoor durability.

Antioxidant compounds are basically free radical scavengers which are added to the urethane systems to improve resistance to thermal oxidations.

Flow control agents such as silicones and cellulose acetate butyrates are used to enhance leveling and to avoid cratering and formation of other coating defects in urethane coatings.

Stabilizers have been used to complex with the catalyst and increase the shelf-life of the formulations by retarding the isocyanate and hydroxyl reaction.

Solvents are typically required for the flow-coatable compositions. The film integrity, appearance and application are significantly affected by the nature of the solvent mixture even though the solvents are not a permanent component of these compositions. The evaporation rate of the solvent mixture must be adjusted so that evaporation takes place quickly during the initial drying (after flow coating application) to prevent excessive flow, but slowly enough to give sufficient leveling and adhesion. Solvents used should be free from alcohol, acid, water, and components which react with isocyanates. Solvents also influence the reaction rate of isocyanate-alcohol reactions which are dependent on the hydrogen-bonding and dipole moment character of the solvents.

Polyurethane Manufacture Methods

There are three major methods for producing polyurethanes; one shot, quasi prepolymer and prepolymer.

One Shot Method - In this method macroglycol, diisocyanate, and crosslinking components are mixed and reacted at one time. The chain extending and crosslinking reactions proceed simultaneously.

Quasi Prepolymer System - The isocyanate and part of the macroglycol are combined. The extender and the rest of the macroglycol material are then mixed and reacted.

Prepolymer Method - The macroglycol is prereacted with the diisocyanate to give a prepolymer which is then combined with the extender and/or crosslinker.

All these manufacturing methods have been used to produce production polyurethanes at Sierracin/Sylmar.

APPLICATION PROCEDURES OF S-243

Three different techniques have been used at Sierracin/Sylmar to apply the S-243 system onto the substrates.

(1) Cast in Place (C.I.P.) - FX-216 (100% solids) is directly cast onto the primed substrate to give a protective liner with a controlled thickness of between 0.001 and 0.025 inches. This application technique offers excellent adhesion due to the chemical reaction between FX-216 and FX-217 primer. However, applying the liner onto compound curved parts is difficult due to the necessity for curved optical buffers.

(2) Casting and Subsequent Lamination - FX-216 is first cast into sheet at a desirable thickness of between 0.001 to 0.025 inches and subsequently laminated to FX-217 primed substrate. This time-consuming process gives inferior adhesion compared to the C.I.P. technique. The liner laminated to a flat part is subsequently formed into the desired shape if needed.

(3) Flow Coating - This technique is by far the easiest and cheapest application procedure. Solvents are used to decrease viscosity and to improve surface flow. With this technique, S-243 can be applied to any type of part with minimum difficultly and excellent adhesion and optics.

Variation in the concentration of the components of FX-216 composition results in a wide range of physical properties. However, all the reported results in this paper are based on one specific formulation for FX-216 which is applied by flow coating technique, unless specified.

PROPERTIES OF S-243 ON ACRYLIC AND POLYCARBONATE SUBSTRATES

The S-243 coating system has been applied to as-cast acrylic (PMMA), stretched acrylic (S-1000), and polycarbonate (PC) substrates and thoroughly tested for abrasion, environmental, chemical/solvent, and craze resistance. The results are documented in the following sections.

Crosshatch Adhesion

To determine the adhesion of the coating, 100 uniform squares were scribed into curing film using a scribe with 11 equally spaced (1 mm) parallel blades. 3M's #600 pressure-sensitive tape was then applied to the scribed area and rapidly pulled away from the substrate. The tape pull was repeated 10 times using a fresh tape for each pull. The adhesion was recorded as the approximate percentage of squares remaining after tape removal from the surface. This method is similar to one described in ASTM D 3359.

87. The coating system demonstrates 100% adhesion with 3M's #600 tape after 10 tape pulls (Table 1).

Bayer Abrasion Resistance (ASTM F735-83)

The Bayer abrader test has been used to evaluate the ability of a surface to resist scratching and rubbing erosion. The test has been adopted by ASTM as a standard abrasion test method for abrasion resistance of transparent plastics used in aircraft glazing. The test consists of quartz silica sand oscillating over the test specimen surface. The amount of surface haze induced in the panel is a measure of abrasion resistance. The comparative data after 600 strokes (300 cycles) for as-cast acrylic (PMMA), stretched acrylic (S-1000), polycarbonate (PC), and these substrates coated with S-243 is shown in Table 1. The results indicate that the Bayer abrasion resistance of S-243 is markedly superior to the plastic substrates.

Taber Abrasion Resistance (ASTM D-1044-90)

The Taber abrader test is a widely used method for plastics. This test consists of two grit-filled rubber wheels to which a predetermined weight is applied. The wheels abrade the substrate surface as it rotates on a table. Increase in haze is used as the criteria for measuring the severity of abrasion. The test results after 100 cycles using CS-10F wheels, 500 g load, are provided in Table 1. The data indicates that S-243 has excellent Taber abrasion resistance.

UV/Humidity Resistance

The coating was subjected to UV radiation exposure in a Q-panel QUV chamber in the form of UV/condensing humidity cycling from 104°F (40°C)/dry/313 nm peak/5-10 suns to 122°F (50°C)/95-100% RH by a method modelled after that found in ASTM G53-84. A full cycle consisted of 16 hours/104°F (40°C)/dry/313 nm peak/5-10 suns plus 8 hours/122°F (50°C)/95-100% RH. The total time per cycle equals 24 hours. After 2000 hours exposure (equivalent to 12 years service), there is no evidence of coating degradation such as yellowing, crazing, haze and loss of adhesion and/or abrasion resistance. The coating system has excellent resistance to UV light degradation.

Temperature/Humidity Resistance

The coating was exposed to condensing humidity (100% RH) at 140°F (60°C) for a period of 2000 hours according to Mil-Std-810, Method 507.3. After exposure, the specimens were inspected for evidence of degradation, adhesion loss, and change in light transmittance or haze. The coating had 100% crosshatch adhesion to PMMA (Poly II), S-1000, and PC substrates with no optical and/or physical damage (Table 1). Bayer abrasion and solvent resistance (30 minutes exposure to methyl ethyl ketone and isopropanol) showed no significant sign of deterioration.

Steam Resistance

A condensing steam chamber has been developed at Sierracin/Sylmar to test the steam resistance of various coatings. Basically, the chamber serves as an accelerated condensing humidity test at 212°F (100°C). Steam exposure is apparently more aggressive on coatings than boiling water exposure. Based on our observation, the damage caused by 6 hours exposure to steam roughly equals to 6 weeks of exposure to condensing humidity at 140°F (60°C). The steam test results in this paper have been reported after 6 hours exposure. After the test, there was no sign of physical and/or chemical damage. The coating adhesion and abrasion resistance remained intact indicating its superior hydrolytic stability.

Elongation

The elongation test was performed per ASTM D412. The elongation at break of cast FX-216 polyurethane material is 78%.

Thermal Cycling Resistance

The test consists of thermally cycling the panels in the range of -65°F (-54°C) to 220°F (104°C). After 100 cycles, the coating passed luminous transmittance, haze, chemical craze resistance, and crosshatch adhesion requirements.

Salt Atmosphere

This test is performed in accordance with Mil-std-810, Method 509, Procedure I to verify the resistance of the transparency to corrosive effects of salt spray. The S-243 coating system successfully passed the test on both acrylic and polycarbonate substrates.

Rain Erosion Resistance

Rain erosion tests were performed on S-243 coated acrylic and polycarbonate substrates at University of Dayton Research Institute (UDRI) according to ASTM G-73. As a control, bare acrylic and PC substrates were tested and were severely pitted after 10 minutes exposure to one inch per hour rain at a speed of 500 mph at an angle of 30 degrees. There was no damage, formation of haze or loss of adhesion of S-243 after a 60 minute exposure on both acrylic or polycarbonate substrates (Table 1).

Chemical/Solvent Resistance

The coating was continuously exposed to the chemical/solvents for a period of 30 minutes and then examined for any detrimental effect such as loss of adhesion, haze, blistering, crazing, swelling or any other defects. This test is similar to the one described in ASTM D-1308-81. The resistance of S-243 to the chemicals normally encountered by aircraft transparencies in service is shown in Table 2. The resistance to aggressive chemical/solvents is shown in

Table 3. The resistance to simulates of the chemical warfare agents (CWAs) is summarized in Table 4. Four of the more well known CWAs whose chemical structures are shown in Figure 9 are:

Lewisite: dichloro (2-chlorovinyl) arsine

Mustard Gas: 2,2' - (dichloroethyl) sulfide

Soman: fluoromethylpinacolyloxyphosphineoxide

Sarin: fluoromethylisopropyloxyphosphineoxide

Resistance of S-243 to actual chemical warfare agents (CWAs) was not tested. Instead, liquids which contained only one of the two or more chemical functionalities of the CWAs, their hydrolysis decomposition products or the decontaminants used to neutralize them were chosen for testing.

The data presented in Tables 2, 3, and 4 shows that S-243 offers outstanding resistance to a wide range of aggressive chemicals and solvents including alcohols, ketones, acetates, aromatics, chlorinated hydrocarbons, acids and bases.

Chemical Stress Craze Resistance

A number of as-cast and stretched acrylic materials along with CR-39 and S-243 coated Plex 55 were subjected to 75% sulfuric acid in accordance with Federal Test Method Std. No. 406A, Method 6053. The results of 30 minute exposure to maximum stress of 4000 psi are summarized in Table 5. The data indicates that the stretched acrylics are superior in craze resistance as compared to their corresponding as-cast materials. Also, Plex 55 showed the poorest result among the as-cast acrylic materials. CR-39 performed very poorly and actually broke after a short exposure to sulfuric acid.

The results of 2 hour exposure are presented in Table 6. Based on this data, the craze resistance of the wet (immersed in water for a period of 24 hours at 120°F/49°C) stretched acrylics is significantly reduced as compared to the corresponding dry substrates. The wet Plex 55 practically crazed at zero stress level showing absolutely no resistance to the aggressive acid after 2 hours. However, the wet S-243 coated Plex 55 showed no sign of crazing after 2 hours exposure. These results demonstrate the effectiveness of S-243 coating system in protection of materials with inferior craze resistance against the aggressive chemicals.

All commercial aircraft passenger and cockpit windows are currently undergoing premature crazing, especially those flying long distance routes at high altitudes in high northern latitudes. The crazing is largely attributed to the presence of unusually high concentrations of sulfuric acid in the stratosphere due to the volcanic eruptions, particularly Mt. Pinatubo in the Philippines.

The craze problem in passenger windows is drastically more severe than on cockpit side windows. This problem is thought to be due to the lower operating stress levels of the cockpit side windows. However, the passenger windows exhibit uniform crazing across the entire daylight opening up to and around the very edge of the rabbet cut, despite the reduced pressurization stress level present at the edge. This crazing suggests outright chemical attack, as opposed to a stress-craze phenomenon, against which stretched acrylic is particularly resistant among the transparent plastics. After exposure to sulfuric acid, the surface of the acrylic window contracts due to the desiccant action of sulfuric acid and subsequent loss of water. Surface stresses are then created which cause fine cracks or crazes to appear on the material. Other suggested possibilities are plasticization of the acrylic by atmospheric moisture and reduction of its critical crazing stress, and the influence of the moisture absorption/surface drying mode which creates uniform surface tensile stresses, resulting in uniform crazing over the entire surface.

To our knowledge, no coating evaluated to date has proven entirely effective in commercial airline passenger window service-life testing. The S-243 coating system combines the best properties of the prior generation of both hard and soft coatings. The polysiloxanes have usually provided superior rubbing abrasion resistance with excellent solvent resistance, but were brittle and tended to crack under tensile or pressurization loading. Prior soft coatings/liners, typically of urethane formulation, were excellent in extensibility due to their elastomeric nature and had adequate, but inferior, resistance to rubbing abrasion, but provided excellent particle (sand) abrasion resistance. A major limitation was reduced resistance to some types of solvent and chemical exposure. S-243 overcomes the limitations of both types of coating systems, hard and soft, combining the desirable properties of both systems without the drawbacks of either. This coating is expected to be particularly effective in the present commercial environment of increased exposure to airborne solvent and chemical contaminants, and yet still have the required service and maintenance exposure resistance to function effectively in daily operation.

Based on the results above, S-243 has great potential to provide excellent protection for the outer acrylic and/or polycarbonate surfaces of all aircraft transparencies.

PROPERTIES OF S-243 ON ELASTOMERIC URETHANE LINERS

Thermoplastic polyurethane elastomeric liner materials have been used to protect the exterior surface of aircraft transparencies. These liner materials are strong, tough, extensible and abrasion resistant, but they have a major shortcoming. Their solvent/chemical resistance is less than desirable. This shortcoming is alleviated by the application of S-243 system. A number of extruded liner materials, such as S-125, have been

developed at Sierracin/Sylmar. The S-125 material coated with S-243 has been designated as S-239. Besides providing a great degree of substrate protection, the liner-faced transparencies, as compared to acrylic and/or polycarbonate-faced transparencies, offer superior resistance to impingement type damage and, most importantly, they are completely resistant to crazing. A comparative resistance of different materials/coatings to sandblast abrasion is presented in Table 7. The sandblast abrasion test was performed with a sandblaster for a period of 10 seconds at a distance of 3 feet from the substrate surface with an impingement angle of 90 degrees at 50 psi. The results indicate that S-239 liner has superior sandblast abrasion resistance compared to other materials and coatings.

The liner can also be used to protect a conductive coating on the exterior surface of the transparency. The electrostatic discharge properties of the conductive coating is not adversely affected with the application of the liner as thick as 45 mils as tested with a Van de Graaff generator[1]. The properties listed in Table 8 demonstrate the suitability of S-239 as a protective liner for aircraft transparencies.

PROPERTIES OF S-243 ON TRANSPARENT CONDUCTIVE COATINGS

There is an ever increasing demand for transparent conductive coatings on the exterior surface of high performance aircraft windshields and canopies. These conductive coatings require protection from corrosion, abrasion and environmental exposure. The exterior durability of gold, ITO, and a Sierracin proprietary ductile conductive coating system (S-417), all topcoated with S-243, is being tested on acrylic and polycarbonate substrates. Accelerated environmental testing is still in progress and the complete data is not yet available.

Gold/S-243 System

Evaporated gold is the first generation conductive coating that has been used on aircraft glazing for more than 35 years. This nobel metal has met the demand for de-fogging, de-icing, IR reflection, and radar attenuation on commercial and military aircraft transparencies[2]. Despite objections to its color and high reflectance, gold is still a viable conductive coating for many aerospace glazings due to its ease of deposition, ductility, and chemical inertness. As shown in Table 9, the S-243 system has excellent adhesion to hard-to-adhere-to gold layers without any adverse affect on the adhesion of the overall system. This adhesion is maintained after 6 hours of the aggressive steam testing and 3 weeks of QUV exposure (further testing is in progress). The chemical/solvent resistance of this system is essentially the same as shown in Tables 2, 3 and 4 for S-243 on acrylic and polycarbonate substrates.

ITO/S-243 System

As mentioned above, there are some objections to gold coating systems based on their high luminous reflectance (resulting in low transmittance) and color, particularly for low sheet resistance coatings. An alternate transparent material, indium tin oxide (ITO), has less intrinsic color and offers superior luminous transmittance as compared to a gold-based coating (Table 9). Being a semiconductor, however, ITO requires a significantly thicker coating in order to give a resistance comparable to that of the gold system. The data demonstrated in Table 9 indicates the excellent durability of a typical ITO system topcoated with S-243.

A major drawback for ITO is the lack of flexibility and strain tolerance due to its ceramic nature and brittleness. ITO readily cracks in bending tests at strain levels as low as 0.5 to 0.7%. This brittleness limits its application on high performance aircraft transparencies where higher strain levels are expected during flight due to increased thermal expansion.

Sierracin/Sylmar's production ITO sheet resistance properties versus temperature is presented in Figure 10 while its RF attenuation properties versus temperature is illustrated in Figure 11. Sierracin's ITO system on polycarbonate substrate demonstrates good resistance to microcracking during this rather severe thermal exposure.

Two F-16 'C' forward canopies with different designs successfully passed the impact of a 4-pound bird at more than 550 knots. Both designs had ITO coating inside for solar reflection and a ITO Electro-Static Drain (ESD) coating on the exterior with S-243 topcoat. The cross section was impacted at 548 knots with less than 1.5" deflection. The additional cross sections were impacted at 559 knots with just 1" deflection. The S-243 exterior coating has shown no adverse effects from the bird impact.

S-417/S-243 System

Sierracin/Sylmar has developed a conductive, ductile coating system to overcome the shortcomings of gold and ITO systems. The new system, designated as S-417, exhibits superior light transmittance as compared to gold coating, and significantly greater ductility as compared to ITO. The system has been subjected to a range of environmental stresses to assess its suitability for use in advanced glazing. The strain versus resistance properties of S-417 topcoated with S-243 (Figure 12) indicates that there is no significant change in the resistance of the system strained to about 2%. And, that S-417/S-243 avoids the microcracking due to strain found in ITO systems. Optical performance and environmental testing (Table 9) demonstrate the excellent durability characteristics (some tests are still in progress) of S-417. This newly emerging coating system appears to be suitable for high performance aircraft transparency applications.

CONCLUSIONS

Sierracin/Sylmar Corporation has developed a new, versatile and universal coating system, S-243, which has outstanding adhesion to acrylics, polycarbonate, and polyurethanes, as well as to transparent conductive coating systems, including gold, ITO, and S-417 which have been applied to plastic substrates. In addition to outstanding resistance to rubbing and impingement abrasion, S-243 also demonstrates excellent chemical/solvent and craze resistance, as well as excellent hydrolytic and ultra-violet light stability. The outstanding weatherability of S-243 system extends the durability of various conductive coatings on the exterior surfaces of advanced aircraft windshields and canopies. Application of S-243 to acrylic or polycarbonate outerplies is expected to extend the service life and craze resistance of these materials. This system has been scaled up for application to production parts and is now ready for flight evaluation on existing glazings and advanced designs.

ACKNOWLEDGEMENTS

The authors wish to acknowledge the efforts of Alexi Faynerman and David Han who performed most of the lab testing and formulations.

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FIGURE 1

PRIMARY ADDITION REACTIONS OF ISOCYANATES

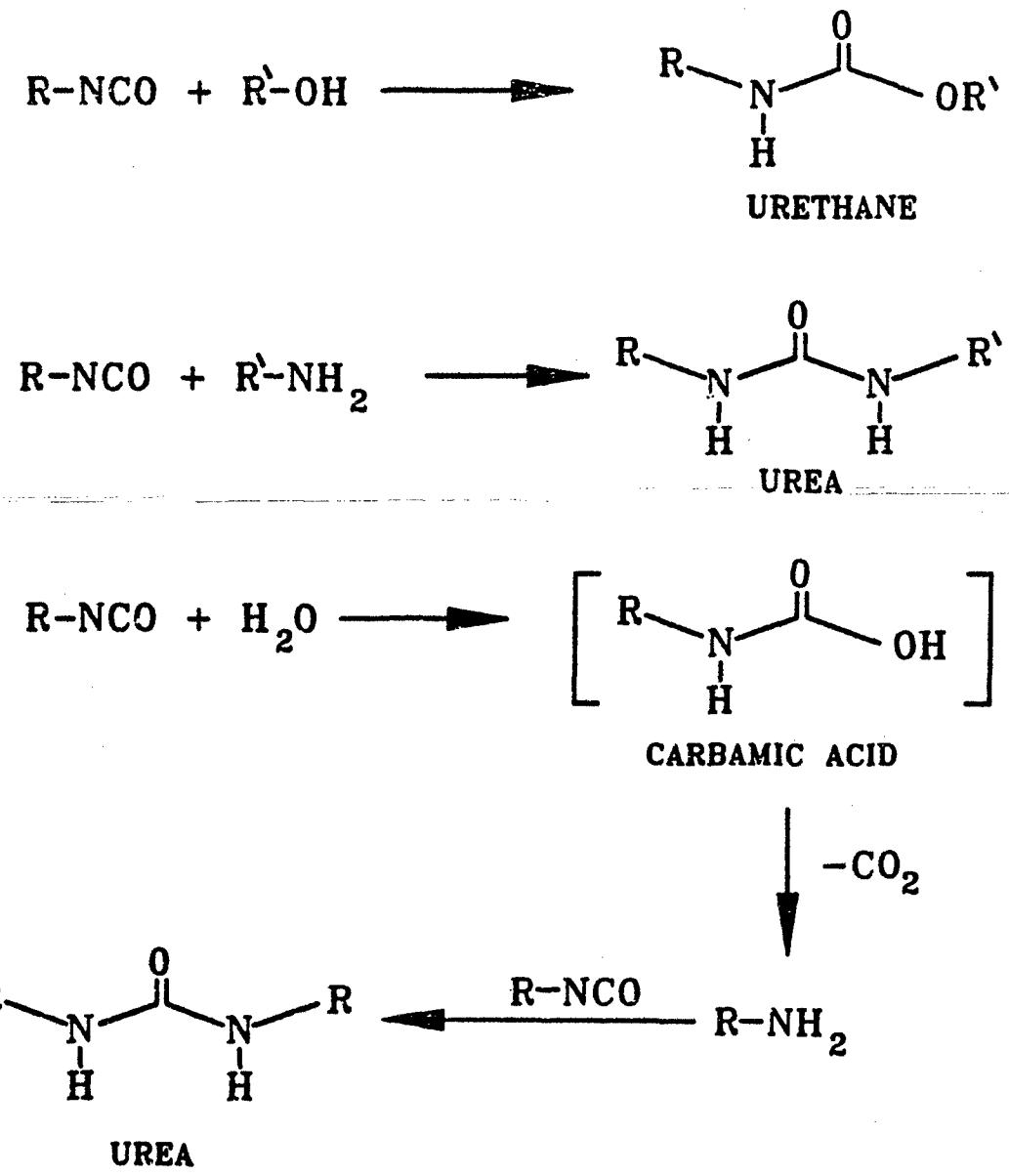
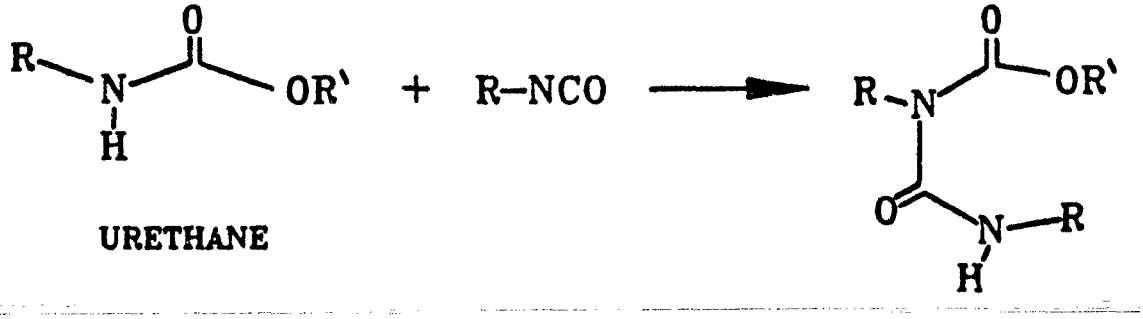


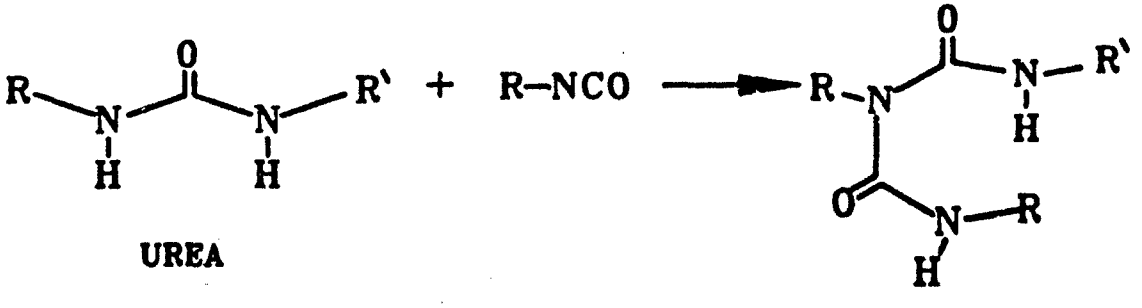
FIGURE 2

SECONDARY ADDITION REACTIONS OF ISOCYANATES



URETHANE

ALLOPHANATE



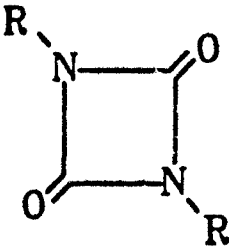
UREA

BIURET

FIGURE 3

POLYMERIZATION (SELF-ADDITION) REACTIONS OF ISOCYANATES

2 R-NCO ⟶

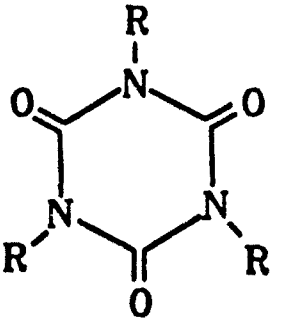


URETIDINE DIONE

2 R-NCO $\xrightarrow{-CO_2}$ R-N=C=N-R

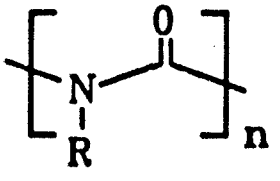
CARBODIIMIDE

3 R-NCO ⟶



ISOCYANURATE

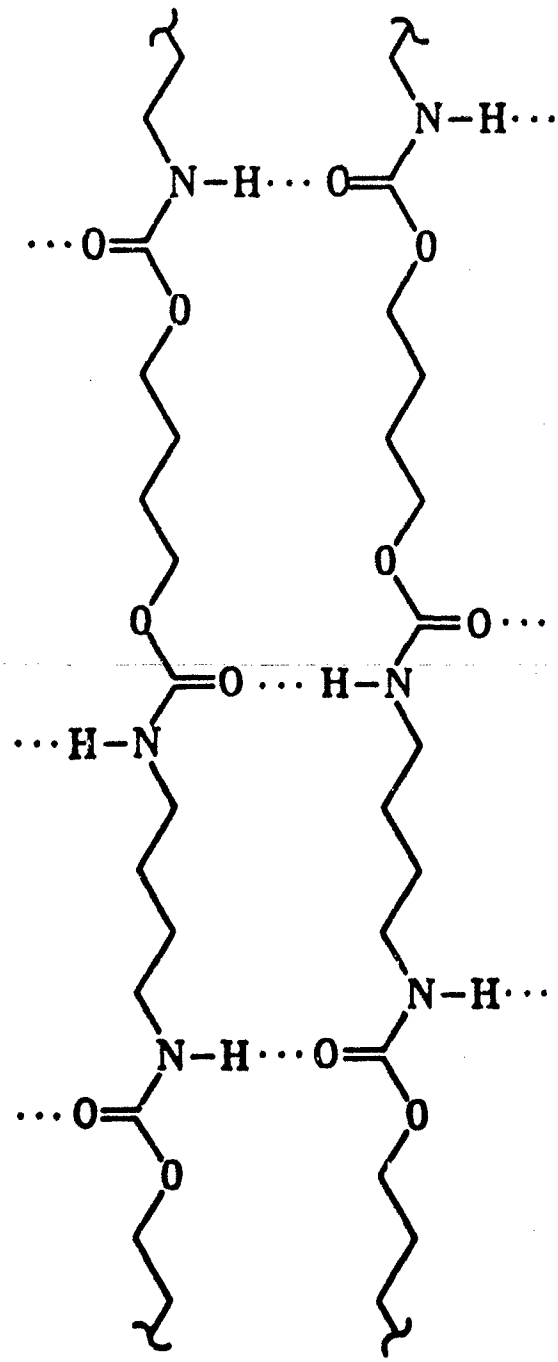
n R-NCO ⟶



1-NYLON

FIGURE 4

URETHANE GROUP HYDROGEN BONDING



PRESENTED AS:

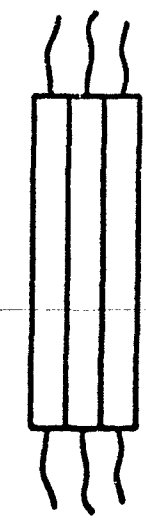
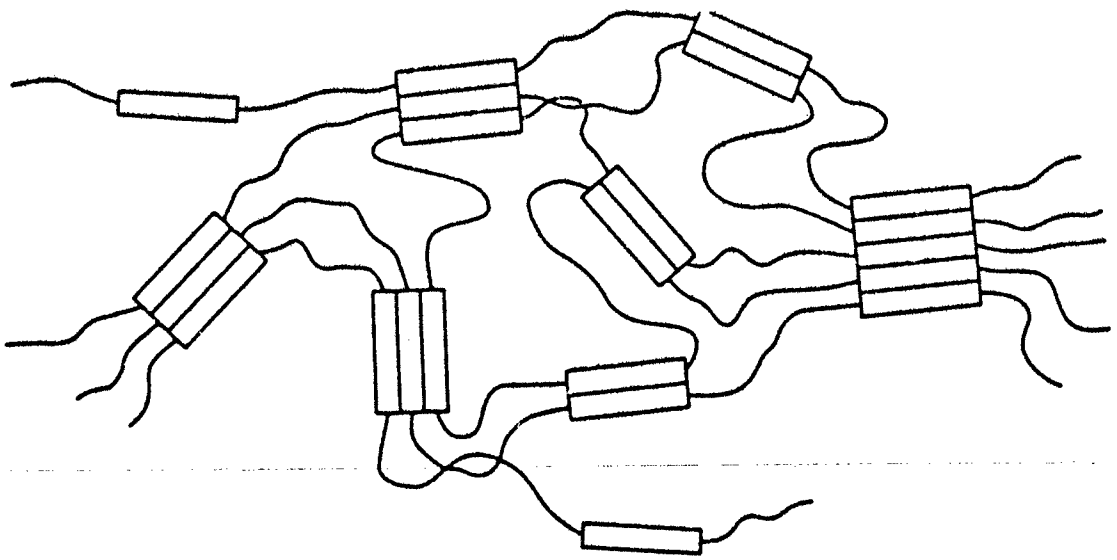
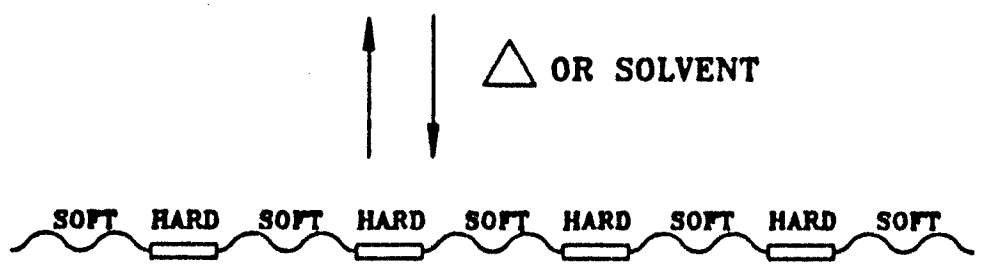


FIGURE 5

MORPHOLOGY OF URETHANES

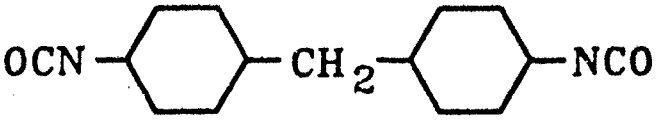




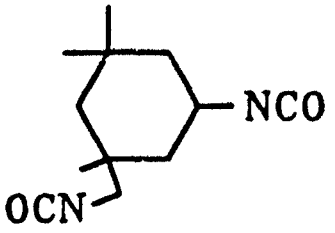
POLYMER CHAIN

FIGURE 6

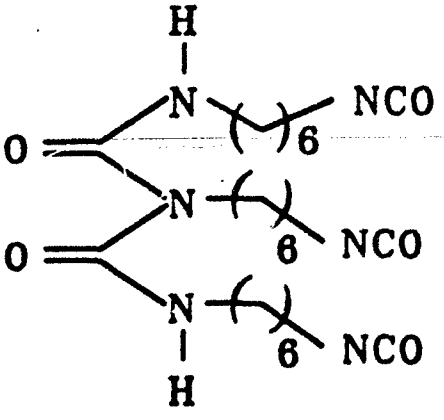
TYPICAL ALIPHATIC ISOCYANATES



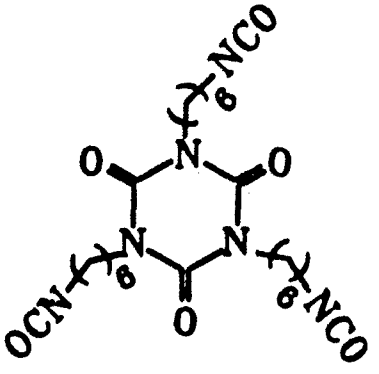
H_{12}MDI



IPDI



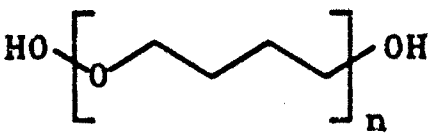
HMDI BIURET



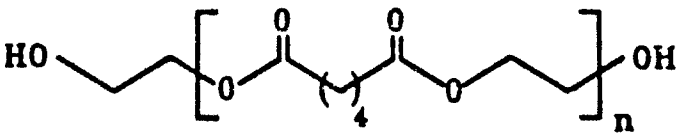
HMDI ISOCYANURATE

FIGURE 7

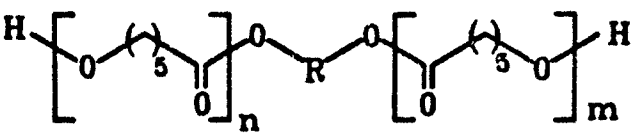
TYPICAL MACROGLYCOLS



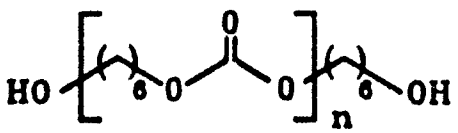
Poly (tetramethylene ether) glycol (PTMEG)



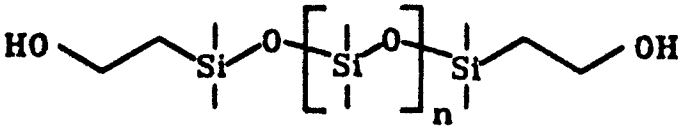
Poly (ethylene adipate) glycol (PEAG)



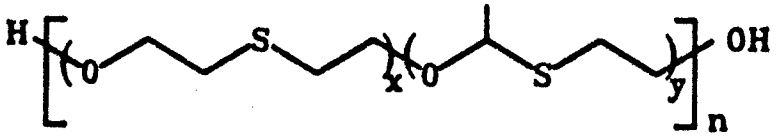
Polycaprolactone glycol (PCLG)



Poly (1,6-hexanediol carbonate) glycol (PHDCG)



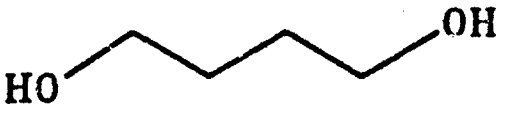
Polydimethylsiloxane glycol (PDMSG)



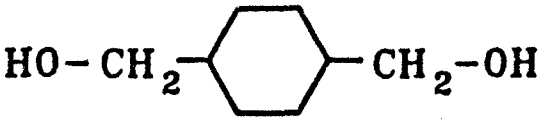
Polythioether glycol

FIGURE 8

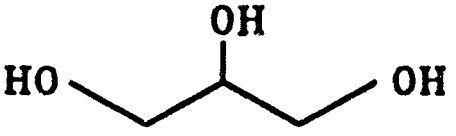
TYPICAL CHAIN EXTENDERS AND CROSSLINKERS



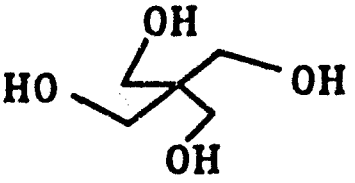
1,4-Butanediol (BDO)



1,4-Cyclohexanedimethylol (CHDM)



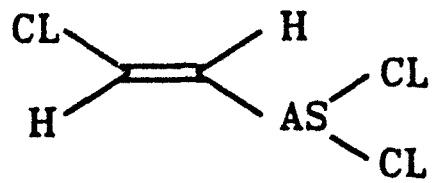
Glycerine



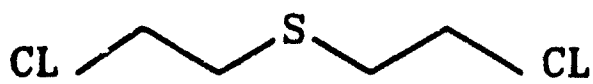
Pentaerythritol

FIGURE 9

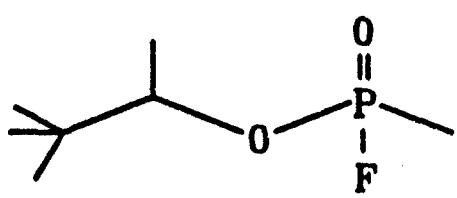
CHEMICAL WARFARE AGENTS



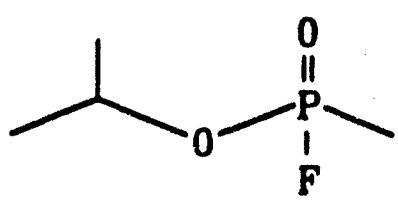
LEWISITE



MUSTARD GAS



SOMAN



SARIN

FIGURE 10

SHEET RESISTANCE OF ITO/S-243
ON POLYCARBONATE AT DIFFERENT TEMPERATURES

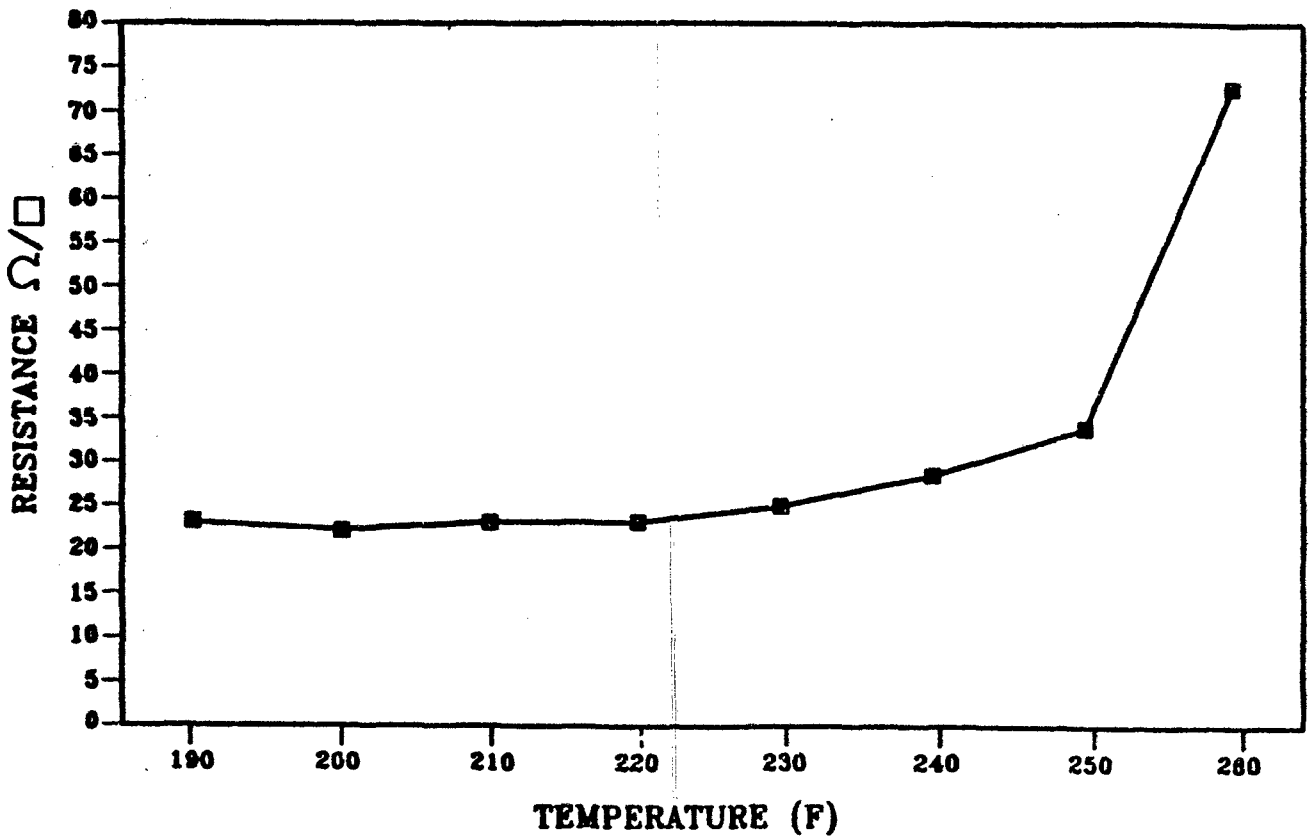


FIGURE 11

RF ATTENUATION OF ITO/S-243
ON POLYCARBONATE AT DIFFERENT TEMPERATURES

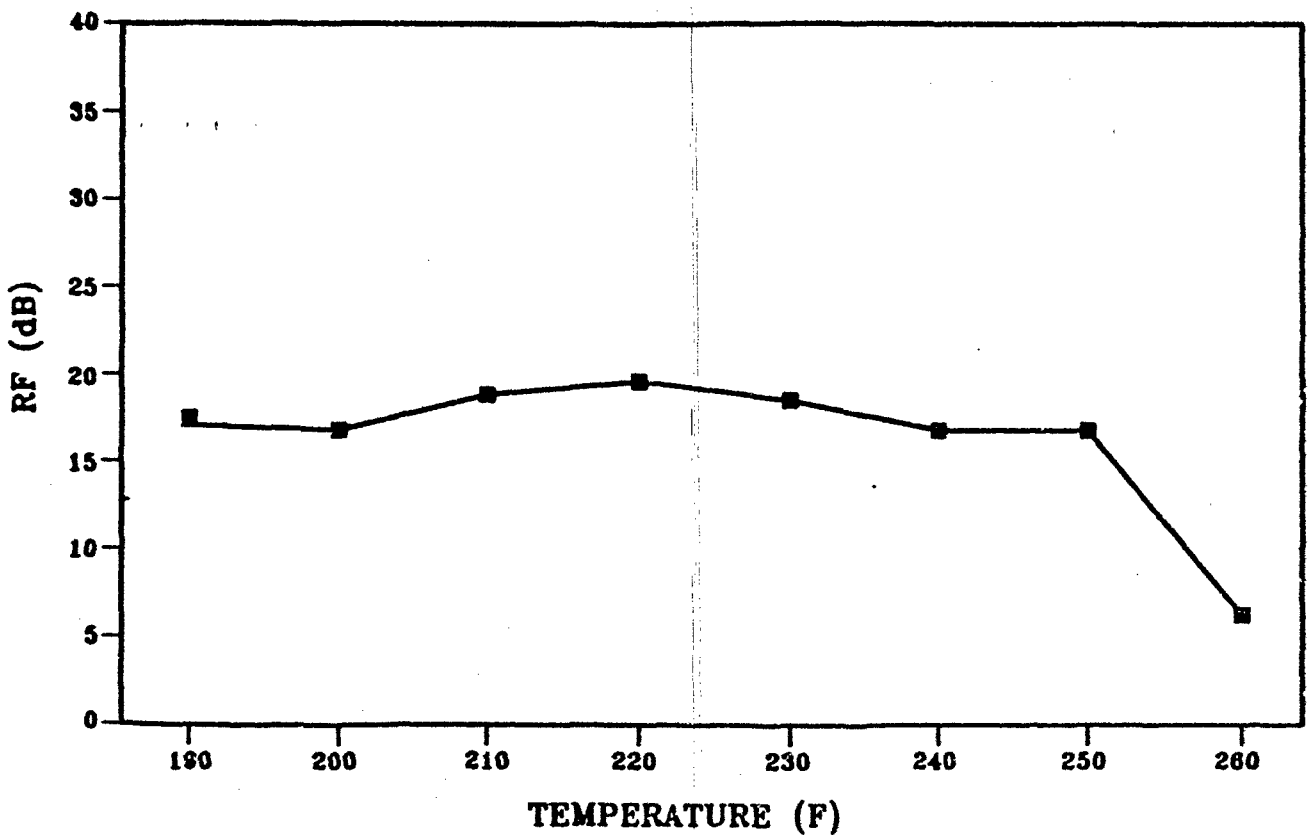


FIGURE 12

RESISTANCE CHANGE OF S-417/S-243 IN BENDING ON POLYCARBONATE SUBSTRATE AT ROOM TEMPERATURE

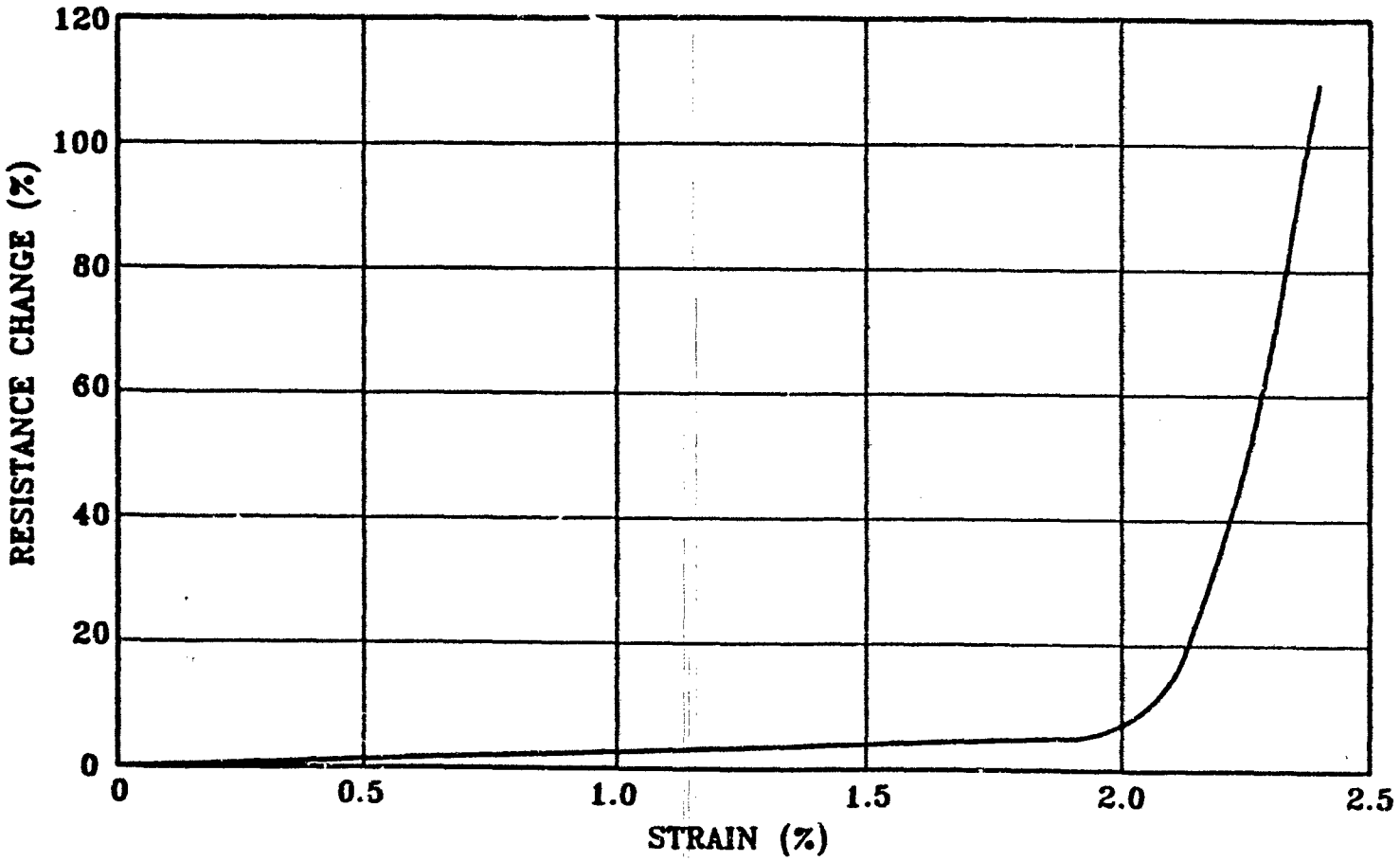


TABLE 1

SELECTED PROPERTIES OF BARE AND S-243 COATED AS-CAST ACRYLIC, STRETCHED ACRYLIC & POLYCARBONATE

PROPERTY	AS-CAST ACRYLIC	STRETCHED ACRYLIC	PC	S-243
ADHESION (%)	—	—	—	100
BAYER ABRASION △ HAZE (%) 600 STROKES	35	48	65	3
TABER ABRASION △ HAZE (%) 100/500 CYCLES	22/-	31/-	35/-	2/8
QUV 2000 HRS	PASS	PASS	FAIL	PASS
HUMIDITY 2000 HRS, 140° F	PASS	PASS	PASS	PASS
STEAM 6 HRS	PASS	EMBRITTLES	PASS	PASS
ELONGATION (%)	5	3	120	78
RAIN EROSION 500 MPH	FAIL/10 MIN.	FAIL/10 MIN.	—	PASS/60 MIN.

TABLE 2

RESISTANCE OF S-243 TO CHEMICAL/SOLVENTS ENCOUNTERED IN SERVICE

CHEMICAL/SOLVENT	RESULT
MILD SOAP & WATER	GOOD
AIRPLANE WASH CLEANER	GOOD
ALODINE	GOOD
RAIN REPELLANT (REPCON)	GOOD
NAPHTHA	GOOD
JET FUEL (JP-4)	GOOD
ISOPROPYL ALCOHOL	GOOD
DE-ICING FLUID (Ethylene glycol)	GOOD
PHOSPHORIC ACID CLEANER	GOOD

TABLE 3

RESISTANCE OF S-243 TO AGGRESSIVE CHEMICAL/SOLVENTS

CHEMICAL/SOLVENT	RESULT
METHYL ETHYL KETONE	LIGHT SWELL, NO HAZE
BUTYL ACETATE	GOOD
TOLUENE	GOOD
METHANOL	GOOD
ACETIC ACID, 20%	GOOD
SULFURIC ACID, 50%	GOOD
SODIUM HYDROXIDE, 50%	GOOD

TABLE 4

RESISTANCE OF S-243 TO SIMULATES, DECOMPOSITION PRODUCTS AND DECONTAMINANTS OF CWAS

CHEMICAL	SIMULATED OR RELATED CWA	RESULT
TRICHLOROETHYLENE	LEWISITE	GOOD
1,2-DICHLOROETHANE	MUSTARD GAS	GOOD
ETHYLENE GLYCOL DIMETHYL ETHER	MUSTARD GAS, SOMAN OR SARIN	GOOD
DIMETHYL BENZYLAMINE	MUSTARD GAS DECONTAMINANT	GOOD
BLEACH	MUSTARD GAS/LEWISITE DECONTAMINANT	GOOD
H_3PO_4 (CONC.)	SOMAN OR SARIN & THEIR HYDROLYSIS PRODUCTS	GOOD
KOH, 50%	LEWISITE DECONTAMINANT	GOOD
HCL, 5%	LEWISITE HYDROLYSIS PRODUCT	GOOD
HF, 5%	SOMAN OR SARIN HYDROLYSIS PRODUCT	GOOD

TABLE 5

CRAZE TESTING WITH 75% SULFURIC ACID
30 MINUTES EXPOSURE/4000 PSI

MATERIAL	AS-CAST/STRETCHED	STRESS TO CRAZE, PSI DRY/WET *
POLY 76	AS-CAST	1824/-
POLY 84	AS-CAST	2684/-
POLY 55	AS-CAST	1216/-
CR-39	AS-CAST	BROKE/-
S-1000	STRETCHED POLY 76	>4000/>4000
S-2000	STRETCHED POLY 84	>4000/>4000

* 24 HOURS IMMERSION IN WATER AT 120° F

TABLE 6

CRAZE TESTING WITH 75% SULFURIC ACID

2 HOURS EXPOSURE/4000 PSI

MATERIAL	AS-CAST/STRETCHED	STRESS TO CRAZE, PSI DRY/WET *
S-1000	STRETCHED POLY 76	3331/1403
S-2000	STRETCHED POLY 84	3303/1914
POLY 55	AS-CAST	-/0
S-243/PLEX 55	AS-CAST	-/>4000

* 24 HOURS IMMERSION IN WATER AT 120° F

TABLE 7

COMPARATIVE SAND IMPACT RESISTANCE

MATERIAL	HAZE % AFTER TEST
STRETCHED ACRYLIC (MIL-P-25690)	24.0
POLYCARBONATE	47.0
GLASS	32.0
STRETCHED ACRYLIC/S-243	4.6
STRETCHED ACRYLIC/S-239	2.6

SANDBLAST TEST: 90 DEGRESS IMPACT/50 PSI/ 10 SECONDS AT 3 FEET

TABLE 8

SELECTED PROPERTIES OF S-243 SYSTEM ON S-125 URETHANE LINER

PROPERTY	RESULT
ADHESION (%)	100
BAYER ABRASION Δ HAZE (%) 600 STROKES	0.8
QUV 1000 HRS	PASS no haze, degradation, yellowing or adhesion loss
HUMIDITY 1000 HRS, 140° F	PASS no haze, degradation or adhesion loss
STEAM, 6 HRS	PASS no haze, degradation or adhesion loss
CHEMICAL/SOLVENT	GOOD *
CRAZE RESISTANCE (IPA AT 4500 PSI)	EXCELLENT **
P-STATIC DISSIPATION	EXCELLENT (ON ITO) no arcing or puncture
RAIN EROSION 610 MPH/30 DEG	PASS 100 MIN.

* SAME AS S-1000/S-243 - SEE TABLES 2 & 3

** BARE S-1000 FAILS IN THE SAME CRAZE TEST

TABLE 9

SELECTED PROPERTIES OF DIFFERENT COATING SYSTEMS ON STRETCHED ACRYLIC TOPCOATED WITH S-243

PROPERTIES	Au	ITO	S-417
LT (%)	60.8	81.2	78.5
SHEET RESISTANCE Ω/SQ	8.6	8.6	9.1
COLOR	TYPICAL GOLD	NEUTRAL	NEUTRAL
HAZE (%)	0.3	1.0	1.0
ADHESION (%)	100	100	100
BAYER ABRASION Δ HAZE (%) 600 STROKES	3.0	2.6	2.8
QUV/WEEKS	PASS/3*	PASS/5 *	PASS/4 *
STEAM, 6 HRS	PASS	PASS	PASS
SOLVENT/CHEMICALS	GOOD **	GOOD **	GOOD **

* TEST IN PROGRESS

** SAME AS S-1000/S-243 - SEE TABLES 2, 3 & 4

BATTLE DAMAGE REPAIR OF AIRCRAFT TRANSPARENCIES

D. W. Carter
Flight Dynamics Directorate
Wright Laboratory

BATTLE DAMAGE REPAIR OF AIRCRAFT TRANSPARENCIES

D. W. Carter
Wright Laboratory
Wright-Patterson AFB, Ohio

Abstract

Recent combat experiences showed that aircraft will return from battle with ballistic damage to their transparencies. Canopy replacement may not be practical in an intense combat situation due to the lack of spare parts and time and resource criticalities. Therefore, a rapid, usually temporary, repair may be applied to fly additional sorties or for a one-time flight back to a repair depot. An Aircraft Battle Damage Repair (ABDR) program has been established to identify training and resource requirements for the US Air Force wartime repair needs and development of an ABDR Technical Order recommending simple, fast, and effective repair procedures. The current ABDR concept for transparencies consists of bolting a metallic patch and applying a fuel tank sealant over the damaged area. This simple repair has been used in recent conflicts such as the Falkland Islands War and Desert Storm. Although effective, this repair concept obstructs the pilots vision. An optically clear transparency repair procedure, developed by the French Air Force, is being investigated by the US Air Force. This repair consists of a two-part acrylic clear adhesive, which is mixed and poured to fill the hole in the damaged transparency. After a two hour cure, the finish is wet-sanded and polished to obtain optical clarity. The new repair procedure has been performed to damages up to two-inch-diameter holes on several US Air Force aircraft transparency types, including monolithic stretched acrylic, laminated polycarbonate, and laminated polycarbonate/acrylic canopies. The repairs have been tested (temperature and pressure parameters) using simple test procedures for ABDR. Test results showed no leaks at the repair locations demonstrating successful repairs for all canopy types.

I. Introduction

As seen from past and recent conflicts, many military aircraft will survive ballistic damage during combat. Combat experiences have shown that for every aircraft lost, three to five aircraft will return from combat with battle damage. In the event the US should become involved in armed conflict, it must be assumed the air war will start suddenly and will require maximum aircraft availability in the early phases. To meet this demand, the effect battle damage will have on aircraft availability must be minimized by developing the capability to rapidly restore operational capability to these aircraft. Also, with the current downsizing of

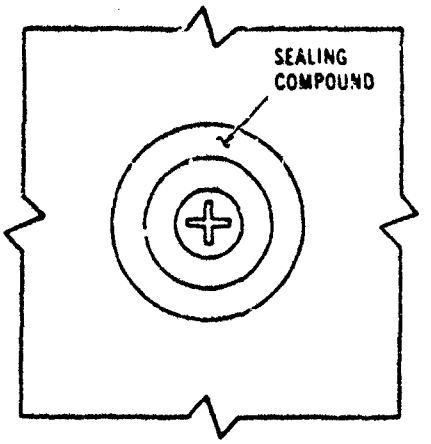
US Air Force aircraft inventory, repairing battle damaged aircraft becomes even more critical in future conflicts.

Recognizing the importance of aircraft battle damage repair (ABDR), the US Air Force has developed a program to initiate and standardize the preparation of ABDR. The primary goal of ABDR is to restore sufficient structural strength and systems serviceability to permit damaged aircraft to continue combat operations, with at least partial mission capability, within time to contribute to the outcome of the ongoing battle. The secondary objective is to perform the necessary maintenance actions to allow extensively damaged aircraft to make a one-time flight to its home station, rear base, or major repair facility. Meeting these objectives requires simple and expedient repair techniques which may eliminate most of the fatigue- and corrosion-conscious methods used in peacetime.

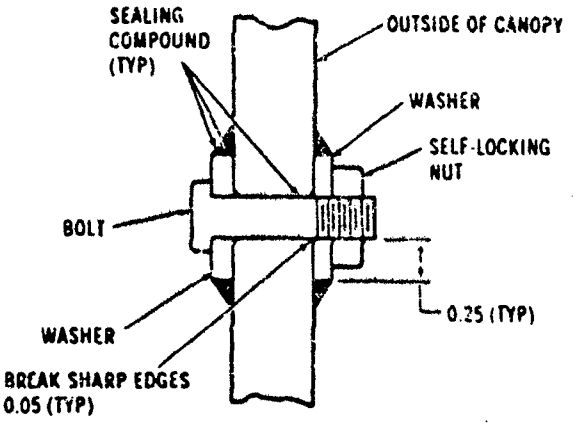
The ABDR program has established requirements to facilitate ABDR training, tools and equipment, and technical information. Technicians are trained to repair and assess battle damaged aircraft during an initial two week course, and they take refresher training at least once a year. Units have developed tool kits, usually stored on mobile ABDR trailers, ready for deployment to the combat area. These trailers are stand-alone kits with tools and materials needed to perform ABDR including a generator and a compressor. Technical information is provided in an ABDR technical order (TO 1-1H-39). The TO 1-1H-39 is a technical manual used as guidance for damage assessment and repair during wartime conditions. In general, the repairs prescribed in the TO 1-1H-39 are quick temporary effective fixes often neglecting fatigue, corrosion, aerodynamic smoothness, and other factors deemed important during peacetime, but are not essential during wartime. To obtain specific requirements for each weapon system, aircraft specific ABDR TO's are utilized to develop the repair.

The accepted ABDR concept, as described in the ABDR TO's, for aircraft transparencies includes bolting on a metal patch and applying any available fuel tank sealant to seal the damaged area. For a small hole (less than one-half inch diameter) in the transparency, a simple bolt, nut, and washer assembly with the sealant is prescribed as shown in Figure 1. For damages larger than one-half inch in diameter, but smaller than 1.5 inches, require the hole to be filled using a rubber plug (Figure 2). For larger holes (up to four inches), holes should be drilled around the damaged area through the metal patch and transparency, bolts with nuts tightened, and the fuel tank sealant applied (Figure 3).

Aircraft transparencies are not immune to battle damage. Combat experiences have shown that many aircraft have encountered and survived damage to their transparencies. During the conflict in Southeast Asia (SEA), several aircraft returned from combat with



PLAN VIEW



SECTION THROUGH REPAIR

NOTE:

NO BOLT THREADS SHALL REMAIN IN BEARING SURFACE OF TRANSPARENCY

Procedures:

1. Clean out damage maintaining minimum hole size.
2. Drill repair hole to nearest standard bolt diameter.
3. Smooth edges of repair hole.
4. Clean and rinse repair area.
5. Apply thin layer of sealing compound to faying surfaces of repair washers and bolt shank.
6. Install repair parts.
7. Remove excess sealing compound immediately.

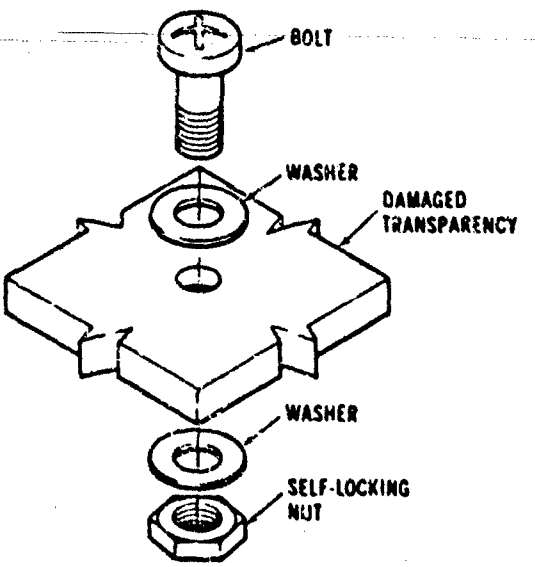
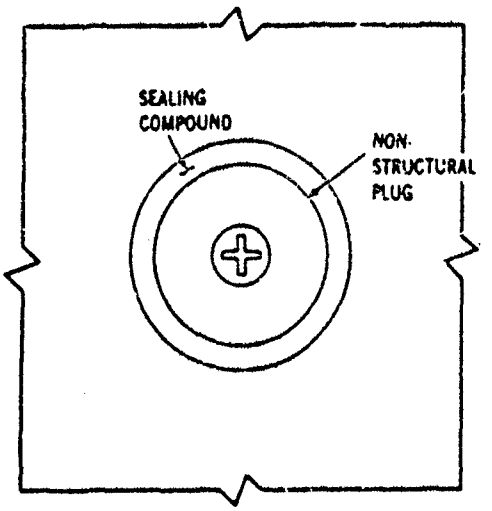
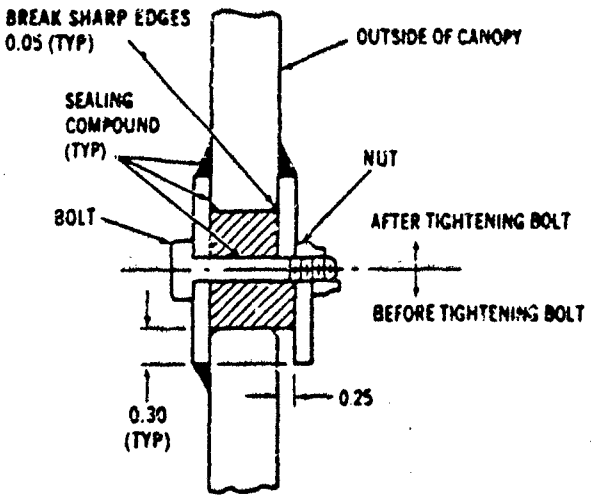


Figure 1. Transparency Repair (Up to 0.5" Diameter)



PLAN VIEW



SECTION THROUGH REPAIR

Procedures:

1. Clean out damage maintaining minimum hole size
2. Smooth edges of repair hole
3. Clean and rinse repair area.
4. Fabricate repair parts.
5. Apply thin layer of sealing compound to faying surfaces of repair washers and bolt shank.
6. Install repair parts.
7. Remove excess sealing compound immediately

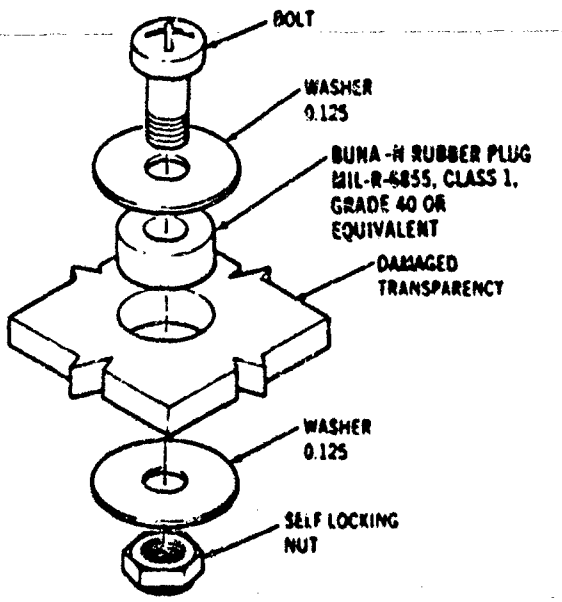
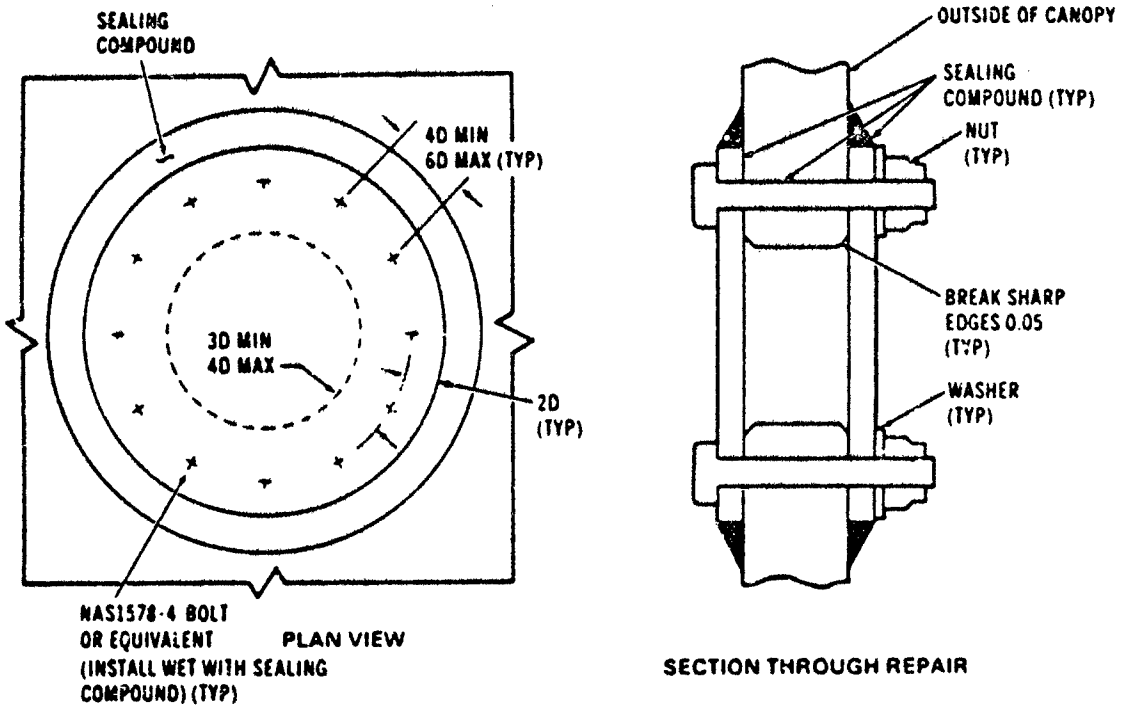


Figure 2. Transparency Repair (0.5" to 1.5" Diameter)



Procedures:

1. Clean out damage maintaining minimum hole size
2. Smooth edges of repair hole
3. Clean and rinse repair area
4. Fabricate repair parts
5. Install repair parts
6. Remove excess sealing compound immediately

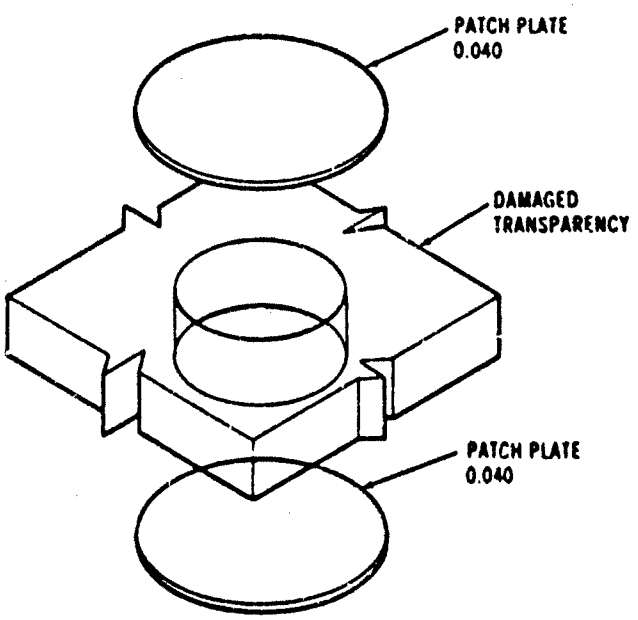


Figure 3. Transparency Repair (1.5" to 4.0" Diameter)

ballistic damage to their transparencies. In some cases, such as with the B-52 shown in Figure 4, the damage was unrepairable. When time and resources were available, the canopy was replaced such as with the F-4 shown in Figure 5. However, several repairs were made during the SEA conflict to aircraft and their transparencies in typical ABDR fashion such as the canopy repair shown in Figure 6.

The war situation the Royal Air Force (RAF) had in the Falkland Islands exemplifies the ABDR capability need. Limited resources and the lack of spare parts, because of a 4000-mile pipeline from the warzone back to their home country and a greatly escalated conflict, created a need for innovative quick repair schemes. One example of battle damage and repair to aircraft transparencies involved an RAF Harrier. The Harrier received ballistic damage to its front and left side windscreens. A fragment penetrated the front windscreen causing a hole and severe cracking around the hole (Figure 7). The side windscreen received two small gouges. Locations of the damages are illustrated in Figure 8. The front windscreen was repaired by bolting an aluminum sheet over the damaged area. This covered approximately the top one-third portion of the front windscreen (Figure 9). The two small gouges in the side windscreen were repaired by bonding a sheet of clear plastic over the holes using an acrylic adhesive (Figure 10). This aircraft, with repairs, successfully flew several sorties with limitations to vision before replacement windscreens were available.

Several aircraft involved in Desert Storm received battle damage. As example, two aircraft received ballistic damage to their canopies. In both cases, the damage was limited. Time allowed for the replacement of one of the damaged canopies. The other canopy was repaired using the bolt and washer repair scheme as shown in Figure 11.

II. Problem

The current ABDR concept for transparencies is a simple, fast, and effective procedure to permit an aircraft to continue combat operations. However, the metal patch restricts the pilots field of vision which may partially limit its mission capability. An improved transparency repair method is needed that can be performed quickly and easily in an uncontrolled combat environment which will allow at least some optical clarity to the pilot.

III. New Transparency Repair Procedure

The French Air Force (FAF) has developed a battle damage repair method for aircraft transparencies that provides a good level of optical clarity. The repair involves use of a two-part

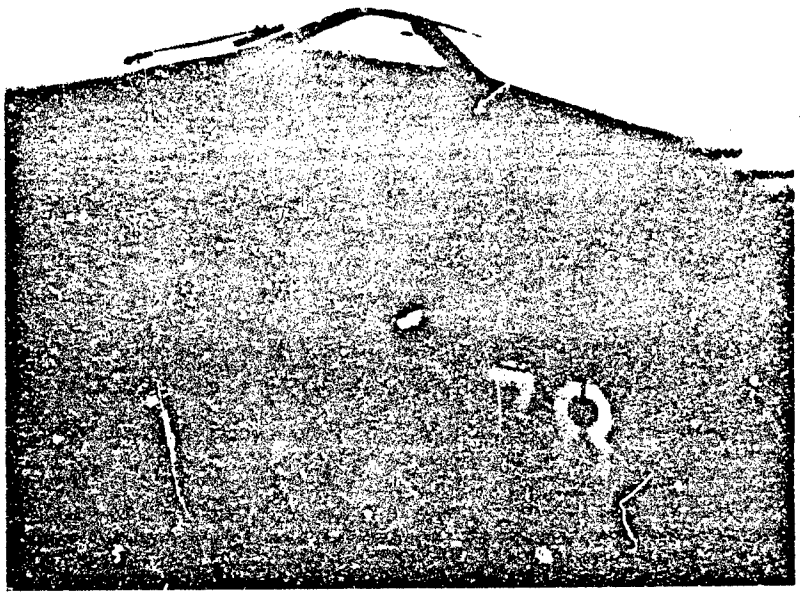


Figure 4. B-52 Damaged During SEA Conflict



Figure 5. F-4 Canopy Replacement During SEA Conflict

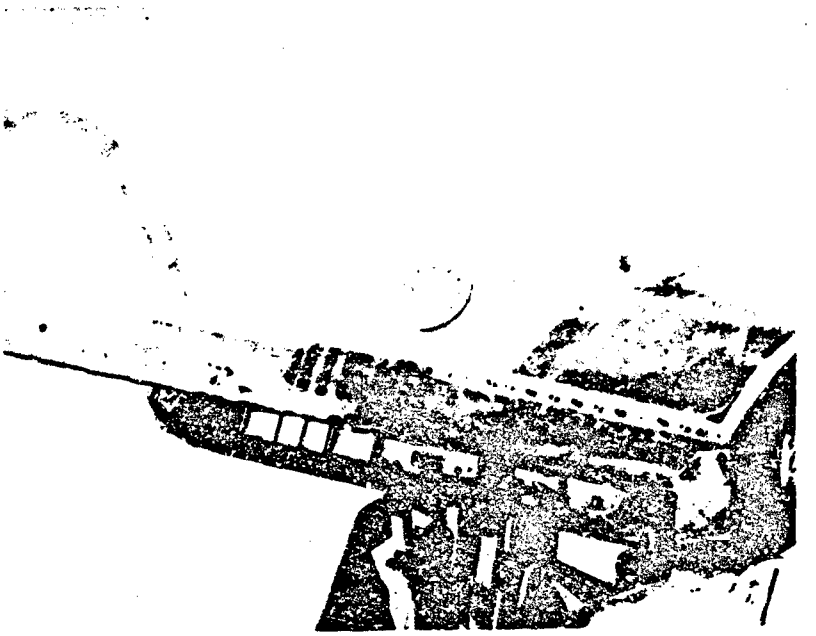


Figure 6. Canopy Repair During SEA Conflict

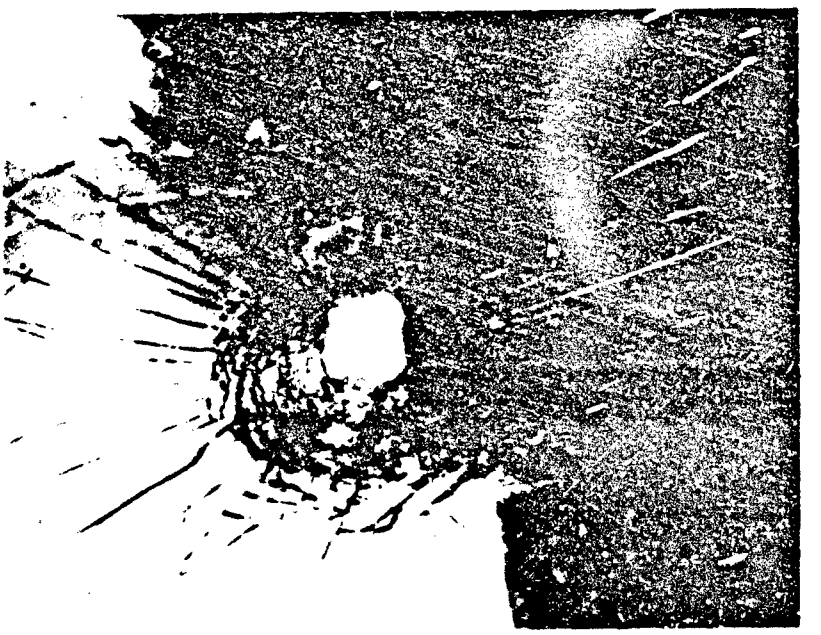


Figure 7. Harrier Front Windscreen Damage

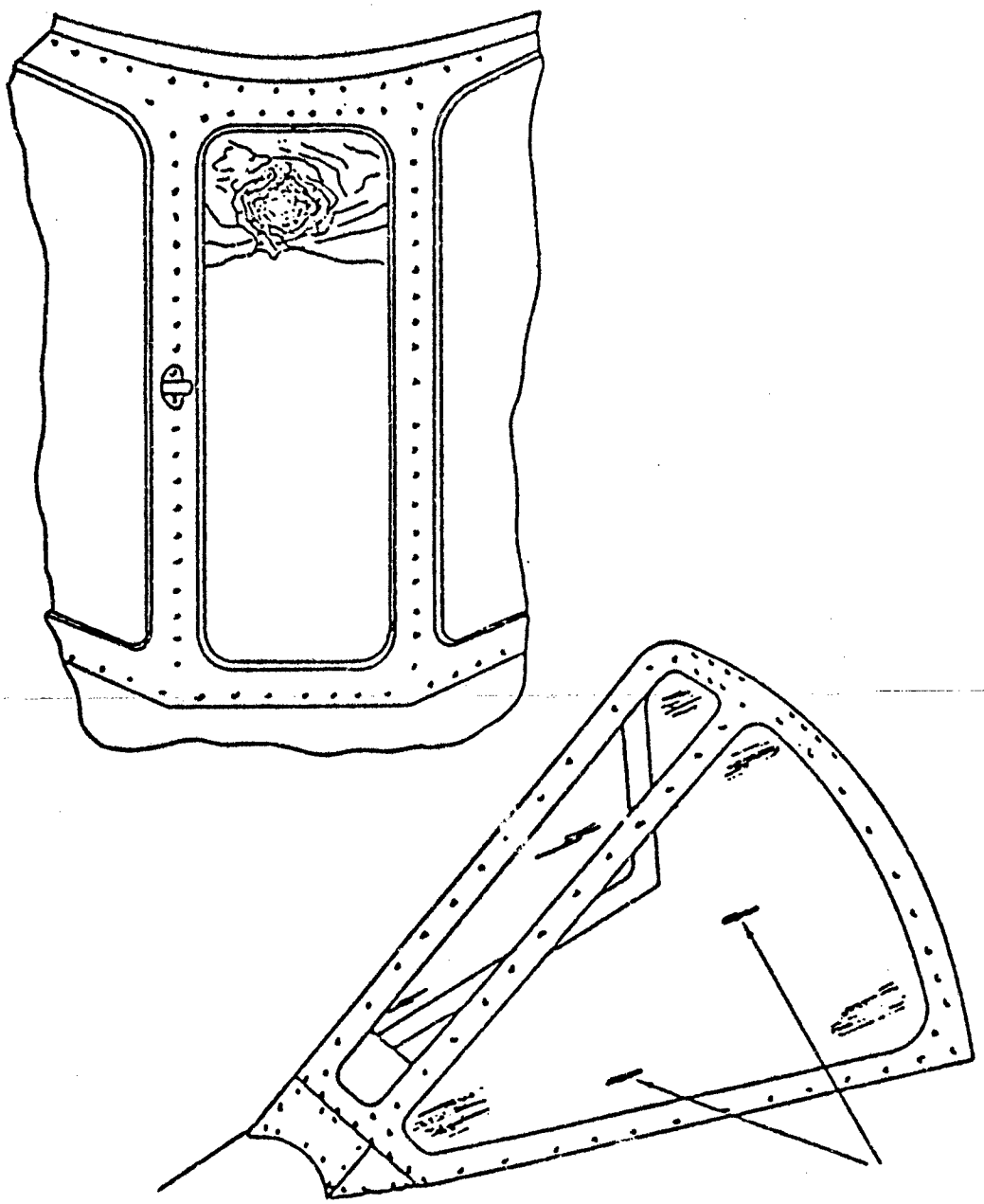


Figure 8. Harrier Damage During Falkland Islands Conflict

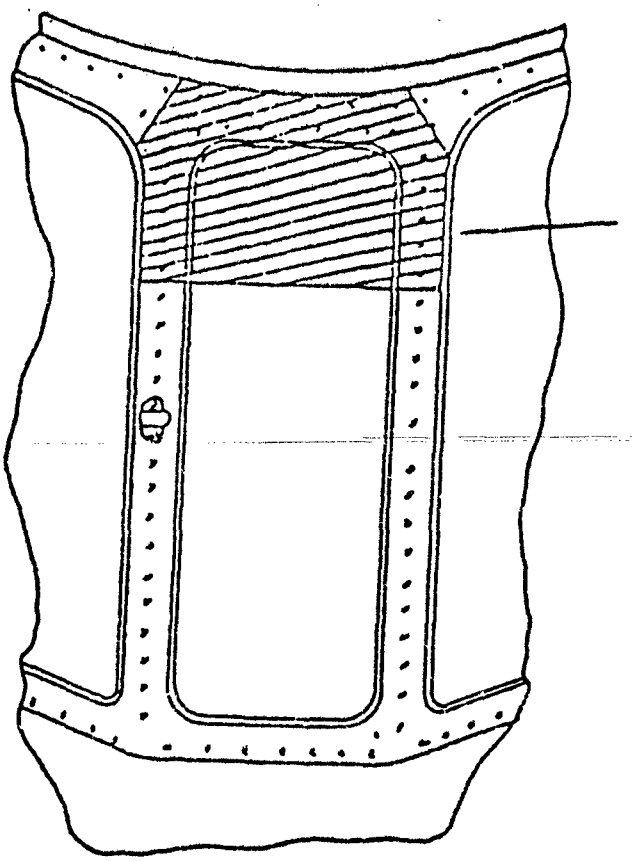


Figure 9. Harrier Front Windscreen Repair

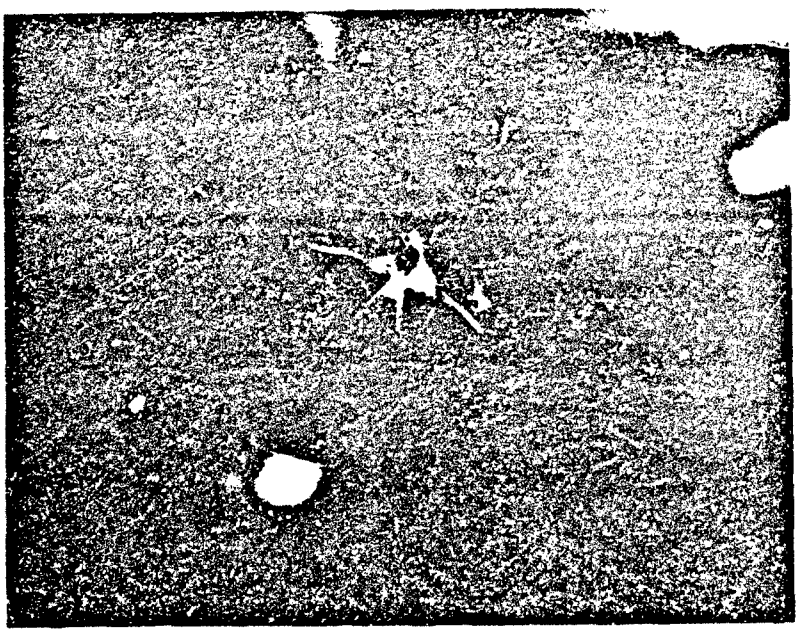


Figure 10. Harrier Side Windscreen Repair



Figure 11. Canopy Repair During Desert Storm

methacrylic filler material, called Altuglas adhesive P 10, made by Elf Atochem, a French company. The repair can be performed on all canopy types and takes approximately three hours to complete.

The new FAF developed transparency repair procedure steps follows. The damaged area must first be cleaned-up by removing all protruding transparency pieces and cracks. This is usually accomplished either by cutting out the damaged area using a hole saw or by removing the damage with a rotary file. Also, all cracks are removed or stop drilled. Next, make a taper of approximately 45 degrees around the edges of the hole on the inside and outside transparency surfaces. To contain the adhesive material in the hole before cure, use a piece of metallic foil tape, usually referred to as "speed tape", to cover the hole on the inside surface of the transparency. Next, build a dam of masking tape around the hole on the outside surface.

Once the damage is ready, the adhesive material can be prepared. Mix the two parts of the methacrylate polymer, 25 parts A to one part B. The working life of the mixture is approximately 20 minutes. Pour the polymer into the hole slightly above the level of the outside transparency surface. The methacrylate polymer will require approximately two hours to cure at normal room temperatures. However, the cure time can be greatly accelerated by raising the temperature to 150 degrees Fahrenheit.

After cure of the repair material, remove all tape and begin smoothing the repaired area. First file the repair material down to a level slightly above the transparency surfaces. Next wet sand the area using progressively finer grit papers as follows: 150, 220, 280, 320, 360, 400, 500, and 600. Finally, polish the repair using a common aircraft transparency polishing material.

The final repair allows a fair level of optically clarity. Some skill is required to obtain better levels of optical clarity. Repairs have been made by the FAF to three centimeter diameter holes which show little or no distortion. Because the repair fills the hole and does not require overlapping of a patch, the area of visual restriction is at least reduced, and therefore, is an improvement over current ABDR procedures for transparencies.

IV. Test

The FAF developed transparency repair was tested by the US Air Force at Wright Laboratories. Repairs were made to two-inch-diameter holes in three canopies, F-15, F-16, and F-111. These three types of canopies were chosen because of their differences in construction (monolithic stretched acrylic, laminated polycarbonate, and laminated polycarbonate/acrylic). It was concluded that by evaluating the repair on these three canopy

constructions, the repair was being validated as a generic repair concept acceptable for any aircraft transparency.

Pressure and temperature were the test parameters. The canopy test specimens were instrumented with thermocouples near the repair area. The test pressures were obtained by vacuum bagging the area around the repair on the free-standing canopies. A plywood ring was placed around the repair before installation of the vacuum bag to ensure the bagging material was not against the repair during the test. Therefore, generating a vacuum on one side of the canopy creates a pressure differential between the outside and inside cockpit areas.

The canopy test specimen with vacuum bag attached was then placed inside an enclosed insulated box. For high temperatures, a forced-air diesel heater was used to blow hot air into an opening in the top of the box over the outside surface of the canopy. Two openings in the ends of the box allow the hot air to exhaust. The thermocouples mounted on the canopy were monitored as the canopy temperature ramped up. The heater has a temperature control valve, which was used to maintain canopy temperatures. A maximum canopy temperature of 250 degrees Fahrenheit could be obtained in approximately one hour.

For low temperatures, dry ice was placed in the bottom of the insulated box before placing the canopy in the box. A small electric fan was used to circulate the air within the box. A minimum canopy temperature of -50 degrees Fahrenheit could be obtained in approximately two hours.

Examples of the test matrices are shown in Figure 12. Once the desired canopy temperature is obtained, vacuum was applied to the canopy using a pneumatic vacuum pump. A pressure gage was monitored while vacuum was applied to the repaired area.

V. Results

The repairs survived all tests with no leaks during testing. Additionally, the repairs were pressure tested to 12.5 psig using a vacuum bag after completion of the tests and again showed no leaks. However, some distortion with the repairs was evident after high temperature tests of 250 degrees Fahrenheit. It was noted by the test technician that the repair material exhibited softening at high temperatures.

VI. Summary

Wartime experiences show that aircraft are likely to be damaged during combat including their transparencies. The

MONOLITHIC STRETCHED ACRYLIC

ULTIMATE

① TEMP = 200°F
PRES = 12.0 PSIG
TIME = 5 SEC

② TEMP = -50°F
PRES = 7.0 PSIG
TIME = 5 SEC

③ TEMP = 200°F
PRES = -5.8 PSIG
TIME = 5 SEC

CYCLIC

① TEMP = 175°F
PRES = 0↔9 PSIG
CYCLES = 10
TIME = 6 MIN/CYC

② TEMP = -20°F
PRES = 0↔7.0 PSIG
CYCLES = 10
TIME = 6 MIN/CYC

③ TEMP = 70°F
PRES = 0↔9 PSIG
CYCLES = 30
TIME = 6 MIN/CYC

LAMINATED POLYCARBONATE

ULTIMATE

① TEMP = 250°F
PRES = 12.5 PSIG
TIME = 5 SEC

② TEMP = -50°F
PRES = 8.5 PSIG
TIME = 5 SEC

③ TEMP = 225°F
PRES = -8.5 PSIG
TIME = 5 SEC

CYCLIC

① TEMP = 175°F
PRES = 0↔8 PSIG
CYCLES = 10
TIME = 6 MIN/CYC

② TEMP = -20°F
PRES = 0↔5.7 PSIG
CYCLES = 10
TIME = 6 MIN/CYC

③ TEMP = 70°F
PRES = 0↔8 PSIG
CYCLES = 30
TIME = 6 MIN/CYC

LAMINATED POLYCARB/ACRYLIC

ULTIMATE

① TEMP = 250°F
PRES = 12.0 PSIG
TIME = 5 SEC

② TEMP = -50°F
PRES = 7.0 PSIG
TIME = 5 SEC

③ TEMP = 200°F
PRES = -5.8 PSIG
TIME = 5 SEC

CYCLIC

① TEMP = 200°F
PRES = 0↔9 PSIG
CYCLES = 10
TIME = 6 MIN/CYC

② TEMP = -20°F
PRES = 0↔7.0 PSIG
CYCLES = 10
TIME = 6 MIN/CYC

③ TEMP = 70°F
PRES = 0↔9 PSIG
CYCLES = 30
TIME = 6 MIN/CYC

Figure 12. Transparency Repair Test Matrices

capability to rapidly repair battle damage is crucial to sustaining high sortie rates. The current repair concept of bolting metal patches to damaged aircraft transparencies is quick and effective. However, because this repair obstructs the pilots vision, there is a desire for an improved repair. The French Air Force has developed a clear repair for damaged aircraft transparencies, which is simple enough to have potential use during wartime. The US Air Force has tested the repair, up to a two-inch-diameter hole, on three basic canopy constructions (monolithic stretched acrylic, laminated polycarbonate, and laminated polycarbonate/acrylic). The repairs showed no leakage during the tests.

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TRANSPARENCY RECOR DER FOR OBTAINING IN-FLIGHT ENVIRONMENTAL LIFE HISTORY

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Transparency Recorder for Obtaining In-flight Environmental Life History

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Abstract

An embedded data recording system is being developed to obtain in-flight environmental data for aircraft canopies and transparencies. This information is needed for failure analysis and reliability predictions. The data recorder is unique in that it is embedded in the transparency and operates totally independently of all other aircraft systems. It has the ability to measure and record environmental factors such as solar radiation, aerodynamic heating, pressure, hail impact, humidity, bird strikes, and vibration. The information is stored in a nonvolatile data module using a new memory technology called Magnetoresistive Random Access Memory (MRAM). This technology is ideally suited to this application which requires a fast write speed to record burst events. The storage module is attached directly to the transparency to eliminates the possibility of any data becoming separated or lost. Since the MRAM does not wear out or lose data with the loss of power, it can store manufacturing information and maintenance data for the life of the transparency. Special "smart sensors" are used to send digitized data to a control computer for storage in the data module. Digitizing the data with the "smart sensors" reduces the effects of electromagnetic interference (EMI). This system will be embedded and tested in an F16 canopy.

This work was sponsered by the U.S. Air Force Grant No. DOD SBIR91.1

Introduction

Real time data on the life experience of windshield, canopy, and window systems for Air Force aircraft is needed since millions of dollars are lost annually due to lack of this in-flight data resulting in premature replacements or catastrophic failures from lack of end-of-life data. This data must be realistic, actual, and complete and should contain the fabrication history and tne operational experience of the transparency. Present data is scattered among many sources and, when acquired and interpreted, has proven to be incomplete for use in failure analysis, reliability predictions, and durability specifications. An embedded real time data acquisition system would solve the problems posed above as well as the scattered data problem if it contained a reliable and permanent method of storing the information even during a loss of power.

The results of the lack of transparency life experience data can be seen dramatically by the problems of Celestial window blowouts that occurred in the KC-135's. This is an old aircraft which was first built in 1956 and was patterned after the Boeing 707. Of the original 732 KC-135's built, approximately 650 are still operational today. They have been upgraded with new engines, so there is no immediate plan to retire them. No complete records of the transparencies were maintained over the 37 years of service and as a result, the celestial windows started to fail with resulting fatalities.

Another example of premature failure due to lack of complete data was demonstrated in the B1 bomber during it's early service. Operational life of the transparency was only one year or 250 operational hours. Initially the total cost for the windshields was $240,000. These had to be replaced every year because of failures. With an in-flight data recording system, this could have been reduced to 3 or 4 years by changing the design of the transparency and thus saving upwards of 3/4 million dollars per plane. Evan at the current windshield cost of $140,000 a tracking system would result in even more savings.

Discussion

A study was under taken by Nonvolatile Electronics Inc.(NVE) to determine what sensors would be needed to record the life history of a transparency, the operational environment of the devices, the method of storage and packaging considerations[1]. To accurately describe the life history of the transparency certain events would have to be monitored

during the lifetime of the transparency. These are listed as follows along with the type of sensor needed:

Event	Sensor
• Solar Radiation	Ultra Violet
• Aerodynamic Heating	Temperature
• Tarmac Heating & Cooling	Temperature
• Pressurization	Pressure
• Hail Impact	Pressure, Acceleration, & Strain
• Humidity	Humidity
• Bird Strike	Strain
• Projectile Strike (Combat)	Strain
• Vibration (Flutter, Engine)	Accelerometer

These sensors along with the associated electronics in the data recording system must operate over the same environment as the aircraft. This includes operating in the heat of a desert while parked on a tarmac, in the humidity of a tropical location, and in the cold of an Arctic outpost. In order to meet all these requirements, the data acquisition system will have to operate over the military temperature range of -55^0C to $+125^0C$, humidity from 1% RH to 99% RH, vibration meeting MIL-STD-1530, and altitude from -1,000 to +55,000 FT^2.

To insure accuracy and completeness, the information obtained from these sensors would have to be contained in a data base which would be attached to the transparency. This data base must be permanent in that it must not lose data over time or with loss of power. Permanent information such as date of manufacture, fabrication data, and maintenance information would be contained in the data base or "storage module" for the life of the transparency and would be available for immediate access when needed. Temporary information such as raw sensor data would be down-loaded by the crew chief on a regular basis.

The transparency data recording system must have a minimal impact on the aircraft systems for ease of canopy installation and removal. It must be capable of gathering data even if the canopy is not attached to the aircraft but stored in a hostile environment. In order to meet these requirements, the system must be packaged within the transparency and frame. Only power can be obtained from the aircraft.

The Transparency Data Acquisition System

A system is being designed and developed to meet all of the requirement which were previously discussed. A diagram of this system is shown in Figure 1.

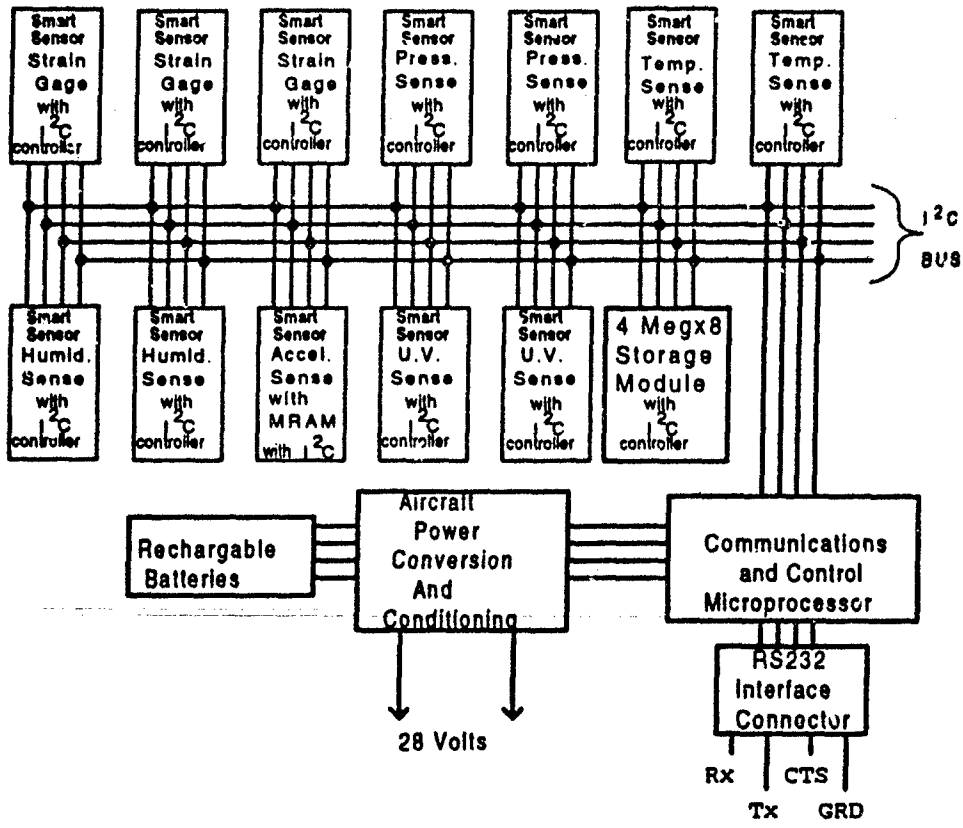


Figure 1 Transparency Data Acquisition System

Six basic sensors are used in this system to fully characterize the life experience of the transparency. The sensors are "smart" since they contain a microprocessor to condition and convert the analog signals from the sensors into digital signals for transmission to the control computer via the I^2C bus. An example "Smart Sensor" is shown in Figure 2. Each module is about 1 inch square depending on the function that it performs. Effects of noise and electromagnetic interference (EMI) are reduced by using an all digital transmission system. Only 4 wires are used in the I^2C bussing system (SDA - serial data, SCL - serial clock,

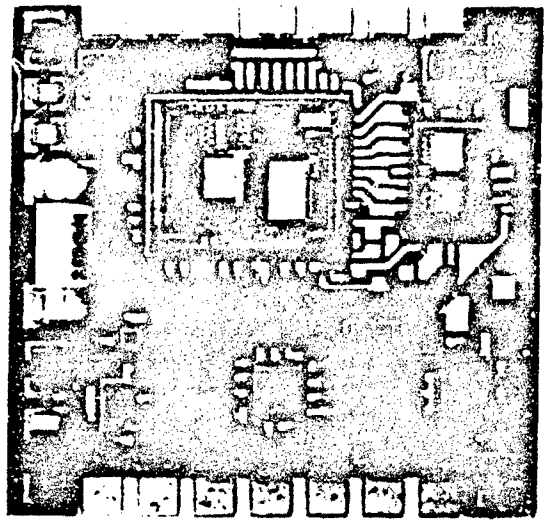


Figure 2. Example Smart Sensor

Vcc and Ground). These wires can be very small since the power requirements are low. This allows the bus to be embedded into the transparency or the frame of the canopy.

The storage module used must have enough memory to contain manufacturing data, sensor information, and burst event data. It was determined[1] that a 4 Meg x 8 storage module would be needed to perform this task. Various memory technologies were evaluated and magnetoresistive random access memory (MRAM) was selected for the storage module because of its nonvolatility, no wear-out, fast write speed (100 nanoseconds), and wide temperature range (up to 300^0C). Other memory technologies such as Electrically Erasable Programmable Read-Only Memory (EEPROM), Ferroelectric Random Access Memory (FRAM), Flash Electrically Erasable Programmable Read-Only Memory (Flash EEPROM), and battery backed Static Random Access Memory (SRAM) have limitations such as limited life (100,000), slow write speed (100 microseconds), temperature limitations (-40^0C to $+85^0$C), and potential battery failure. MRAM has none of these problems because of it's basic design. The storage module uses 32 1 megabit chips designed by NVE which are packaged in a standard credit card size module shown in Figure 3.

The control computer communicates with the various smart sensors and stores the data from them in the storage module. The control computer also communicates with an external computer such as a hand held device to down-load the information when required. This communication is

accomplished using an RS232 interface with a small 4 pin connector located on the canopy.

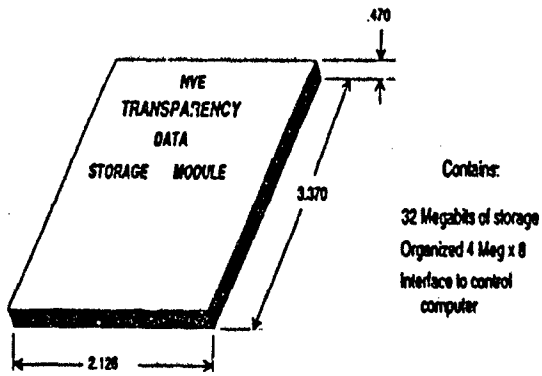


Figure 3. Storage Module

The power conversion module contains the circuitry to condition the aircraft power for battery charging and system power. It also contains the battery management circuit which conditions the battery when charging and also provides maintenance data to the storage module. This data gives charging information, battery useful life, history of the battery, and whether it is faulty or has been removed or replaced.

F16 Transparency Data Acquisition

The system described in this paper is being integrated into an F16 canopy shown schematically in Figure 4. Care has to be taken in the placement of the sensors since most areas in the transparency must be free of visual obstructions. Most sensors can be mounted along the base of the transparency above the mounting holes since they do not require light to operate. However, the Ultra Violet (UV) sensor has to be mounted where it is exposed to UV radiation. Two locations are available which will not restrict the pilot's vision, one in front of the heads-up display and the other at the back on top of the canopy. To insure that the storage module with the transparency life history cannot be separated from the transparency, it is attached to it near the rear left side on the back of an identification label. This location does not cause a vision problem as shown in Figure 4.

The sensors and storage module have to be attached to the transparency for the reasons stated above. However, the power control circuit, control computer, battery, and wiring can be mounted in the frame of the canopy. If the frame and the transparency are separated for repair, the MRAM in the data module will preserve the life history since it does not require back up power.

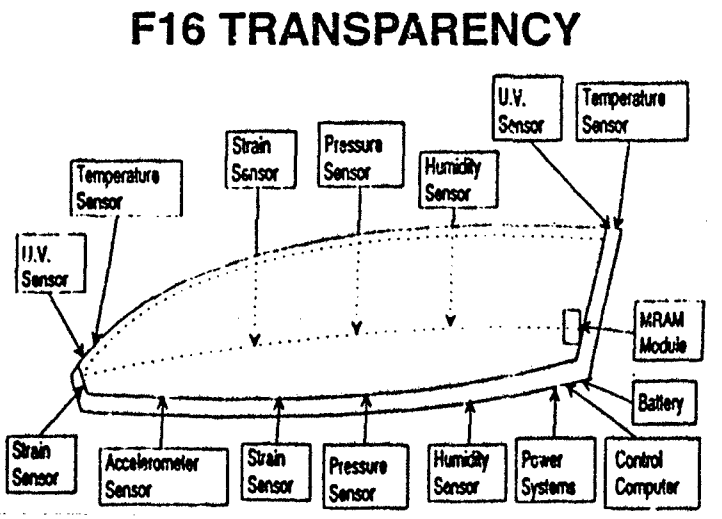


Figure 4. F16 Transparency System Integration

Summary and Conclusions

A data acquisition system is being developed for transparencies that will solve the problem of incomplete and missing environmental and life history data. This system is modular which allows it to be applied to many different transparency configurations. It contains 6 basic "smart sensors" which will measure all of the environmental effects that a transparency can experience.

After completion of the laboratory acceptance test, the Transparency Recorder will be operated on an F16 aircraft to obtain further information on its performance. Various operational aspects of the system will be tested such as sensor operation and accuracy, data compression and storage module capacity, environmental operation, power conversion and stability, ESD concerns, and power loss and recovery. Results of the operational test will be used to develop a production system.

The design and refinements of transparencies will be enhanced with the Transparency Recorder since in-flight data will be available for analysis along with complete life history. This system can be incorporated into old as well as new transparency designs giving the designer a powerful new tool. It will also give the user a more accurate replacement criteria thus reducing costs.

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SESSION IV

EMERGING CAPABILITIES - PART B

Chairman: G. J. Stenger
University of Dayton

Co-Chairman: R. E. Colclough
Flight Dynamics Directorate
Wright Laboratory

Coordinator: K. Alexander
Flight Dynamics Directorate
Wright Laboratory

TEXSTAR DESIGNED QUICK SEAL PROGRAM FOR AIRCRAFT TRANSPARENCIES

J. V. Irion
R. M. Webb
Texstar, Inc.

TEXSTAR DESIGNED QUICK SEAL PROGRAM FOR AIRCRAFT TRANSPARENCIES

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ABSTRACT

In the late 1980's, Texstar Engineers embarked on a quick seal design to reduce the amount of time necessary to install and/or replace an aircraft transparency. The system Texstar developed includes a one piece silicone foam pressure seal, along with an electrically conductive fluorosilicone fairing seal, and associated support items.

The qualification test requirements as well as the economics, specific flight test results and birdstrike test information will be discussed. The use of a seal system similar to this will help the Air Force reach the goal of 4 men installing a transparency in 4 hours.

TEXSTAR DESIGNED QUICK SEAL PROGRAM FOR AIRCRAFT TRANSPARENCIES

INTRODUCTION

In the early 1980's, Texstar personnel developed a program to salvage damaged transparencies used on the F-16 program. The damaged transparencies were returned from the field with various cuts and abrasions caused by normal flight, handling, and maintenance. It quickly became apparent that while the original damage to the transparency might have been repairable, the damage caused during removal sometimes caused structural damage which was not acceptable.

Various devices used to remove the transparencies included hammers, chisels, crowbars, razor knives and pocket knives. Evidence of broken knife blades embedded in the thermoplastic structural plies, as well as crowbar indentions, indicated that certain maintenance crews were becoming frustrated trying to remove transparencies for routine change out. See Figure 1.

It turned out that during installation of a transparency, various amounts of PR-1425 liquid sealant was used between the unit and the frame, depending mostly on the techniques used by the various maintenance or installation personnel. This sealant would harden over time and tended to bond the transparency to the frame. This bond joint was difficult to reach, which added to the maintenance crew's frustrations.

For these reasons, and because Texstar's Marketing department saw dollar resources which enhanced an existing product line, a "dry" seal development project was initiated.

There are actually two different major dry seal mechanisms available for use on the F-16.

DISCUSSION

THE FAIRING SEAL AND THE PRESSURE SEAL

The fairing seal, shown in Figures 2 & 3, is a conductive fluorosilicone rubber which seals the aluminum fairings down the outside longeron areas. It is actually bonded onto the aluminum fairing using a quick cure conductive adhesive and compresses against the outside surface of the transparency when on the aircraft. It usually remains a single continuous piece down the side of the transparency, but can be cut and spliced as required.

When fairings are removed for any reason, the conductive fairing seal is reused when the fairings are reinstalled. End cap close outs complete the seal at the forward ends. These are bonded using the same adhesive and eliminate the possibility of moisture, etc. from getting inside the extruded seal.

The pressure seal, shown in Figures 4 & 5, is a silicone sponge material. It is assembled into a compound shape on a master assembly jig. This shape matches the contour of the cast and machined aluminum frame. A double back tape is applied to the frame contacting surfaces, which holds the seal in place prior to transparency installation. Holes punched through the seal are slightly smaller than the bolt diameter used during installation, thus allowing a pressure seal around the bolt as well as the frame.

A forward hoop cover and fairing seal end caps round out the quick seal system. See Figures 6 & 7. This thin (.030") thick sheet of fluorosilicone rubber eliminated the wet adhesive filler material required to protect the leading edge of the canopy. The injection molded end caps seal the forward opening of the fairing seal.

The use of "dry" type seals in aircraft windshield/canopy applications was not a new idea. Several aircraft (both military and commercial) have used seals for many years. Nevertheless, the dry seal system required extensive testing prior to being accepted as qualified for flight on an F-16 aircraft.

Texstar's internal testing to determine the best type of material for use included:

1) Tensile tests
2) Long term weathering tests
3) Chemical resistance tests
4) Pressure tests
5) Rain leakage tests
6) Birdstrike tests

All testing was designed around the requirements of the General Dynamics (now Lockheed Fort Worth) transparency specification 16ZK002E.

Since the two place (B/D) series aircraft contained the largest seal surface area, it was chosen for the evaluation base on the full scale tests, specifically the pressure rain leakage and birdstrike tests. See Figure 8.

The first full scale test utilized non optical production version F-16 transparencies on a customer supplied frame.

The pressure test consisted of five sets of pressure/ temperature variations as follow in Table 1.0.

TEST	TEMPERATURE	CONDITIONS
Test 1	85-100°F	0-8 psi in 1.5 minutes, hold for 1.5 minutes, then to 0 in 1.5 minutes times 600 cycles
Test 2	170°F ± 5°	Same as above times 200 cycles.
Test 3	-20°F ± 5°	0 - 5.7 psi in 1.5 minutes, hold for 1.5 minutes, then to 0 in 1.5 minutes times 200 cycles
Test 4	-20°F ± 5°	0 - 8.5 psi in 2 minutes, hold for 5 seconds, reduce to 0 in 2 minutes times 1 cycle
Test 5	225°F ± 5°	0 - 12.5 psi in 3 minutes, hold for 5 seconds, reduce to 0 in 3 minutes times 1 cycle

TABLE 1.0

After completion of all pressure testing, the same transparencies were checked for leaks by continuously spraying water on the inside for 45 minutes. No significant leakage was found.

The rain leakage test was primarily for testing the fairing seals.

The fairing seals were bonded to the aluminum fairings using RTV167 sealant. After cure, the canopies were flooded with water for 30 minutes. Minor amounts of water leakage was found upon fairing removal; however, this was attributed to a mismatch of the fairings used in the test. Due to the extremely minor amount of water, the seal was considered acceptable.

Birdstrike testing using a dry seal was demonstrated on several occasions. Both Sierracin/Sylmar manufactured units and Texstar manufactured units were fit with seals prior to birdstrike testing. Texstar recently fitted a next generation version of the F-16 transparency with a dry pressure seal and successfully passed a birdstrike test with a 4 lb. bird at over 550 knots. No detrimental effects were found in any of the testing.

On December 26th, 1988, the first fairing seal was flown at Hill Air Force base in Ogden, Utah. Additional seals were later installed at Hill Air Force Base as well as Bodo Main Air Station in Bodo, Norway. All fairing seals were originally non-conductive.

The conductive additive was requested in 1992, to help drain static built up on the surface of the transparencies during flight.

The first pressure seal was installed in March of 1990, at Hill Air Force Base, after extensive installation testing at Texstar and Lockheed (then General Dynamics) in Fort Worth, Texas.

In June of 1990, the transparency on this trial "test" aircraft was replaced with another. The same dry seal was reused and is still in operation on that aircraft.

SUMMARY

Highlights of this quick seal system are as follows:

- The seal can be installed in approximately 8 manhours, compared to 32 manhours now required using the wet sealant method.
- The aircraft is ready for pressure checks and flight immediately after installation. A 72 hour cure time is not required.
- It is not necessary to re-torque the attachment bolts. Currently the T.O. requires a one-hour wait, then a re-torque of all bolts. This is because since no wet sealant is used causing a gelatinize effect on the original torque readings.
- Once the dry seal has been installed on a clean canopy frame, it will not be necessary to scrape the old sealant from the metal frame. The scraping of the old sealant from the frame can result in damage to the paint, primer and alodine coats. The loss of the protective coating has been shown to allow corrosion of the metal.
- The pressure seal can be reused. This was proven on the test aircraft at Hill.
- The transparency is less likely to be damaged during removal using the dry seal. Many potentially salvageable transparencies are damaged beyond repair because of removal damage when the wet sealant is used.
- T.O. 16W2-5-2 has many warnings that the wet sealing compound is toxic and emits harmful vapors. The dry seal is a cured silicone and does not emit harmful vapors.

- Since the fairing seal is made from silicone, it will not crack like the wet sealant now does. When the wet sealant cracks between the fairing and the transparency, water can be trapped in the joint of the longeron sill on the canopy frame, potentially causing corrosion.

- Estimated average weight savings of the dry seal system versus the wet sealant is 4.0 lbs. on single seat version and 8.6 lbs. on twin seat.

CONCLUSION

The Texstar designed quick seal program for aircraft transparencies has successfully demonstrated that a dry sealing system can be utilized in today's high speed, highly technical transparency environments.

Savings to the user includes monetary savings, installation time savings, weight savings, downtime savings and repair savings.

The U.S. Government has evaluated these savings and has purchased the data rights for the F-16 program.

This data enables other manufacturers to build on the success of this dry seal program, and should mean a savings for us the taxpayer, on existing and future transparency programs.

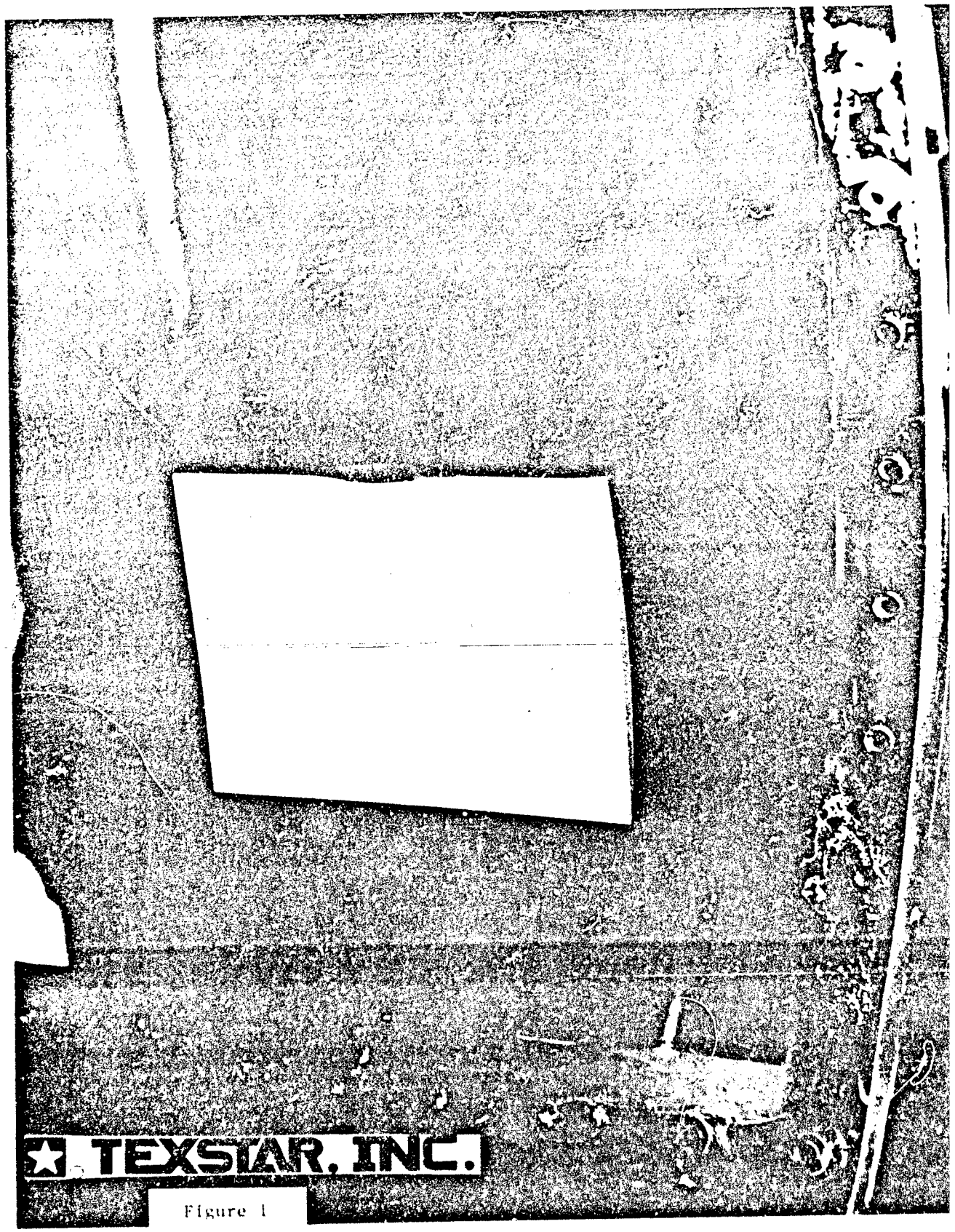
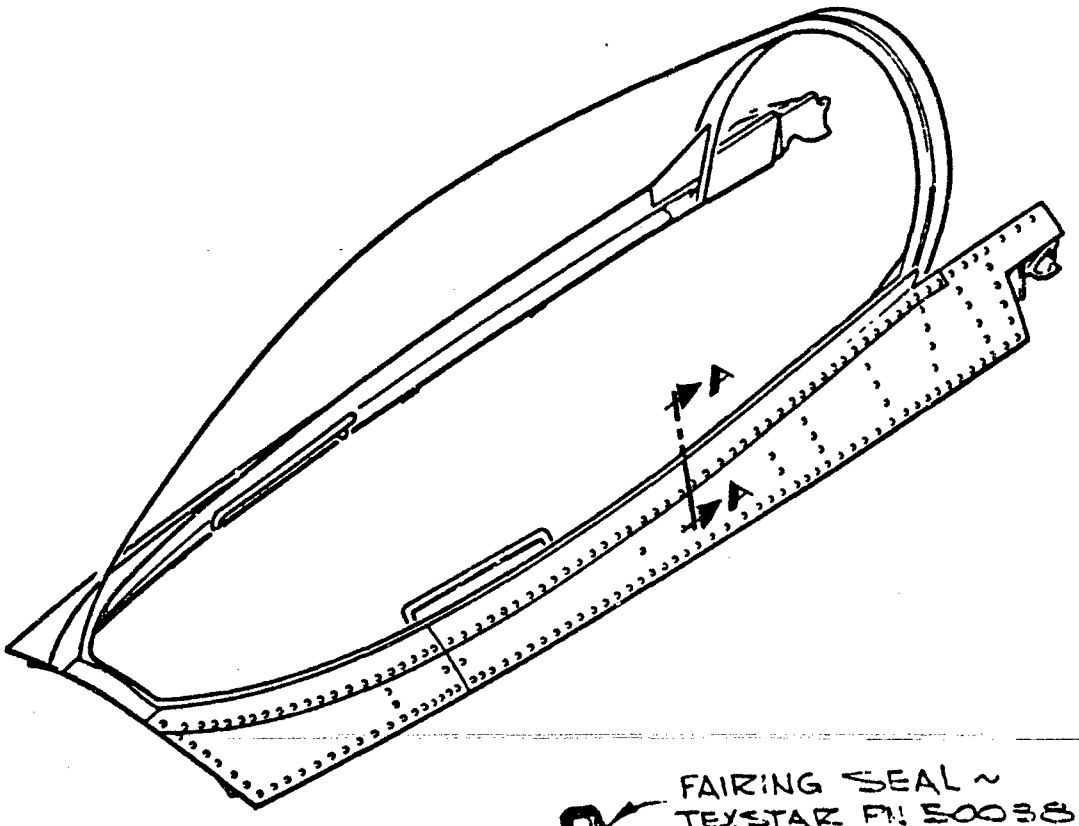
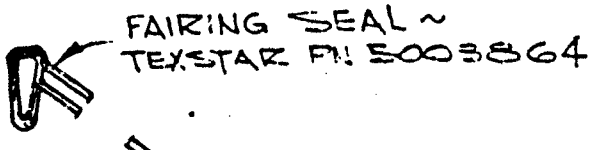
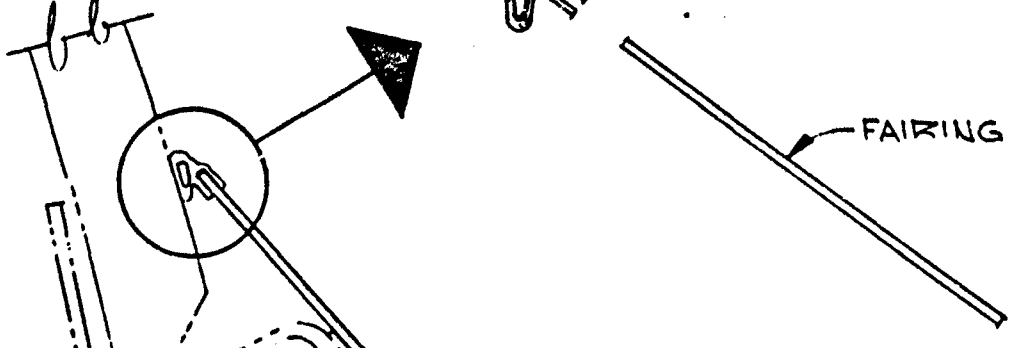
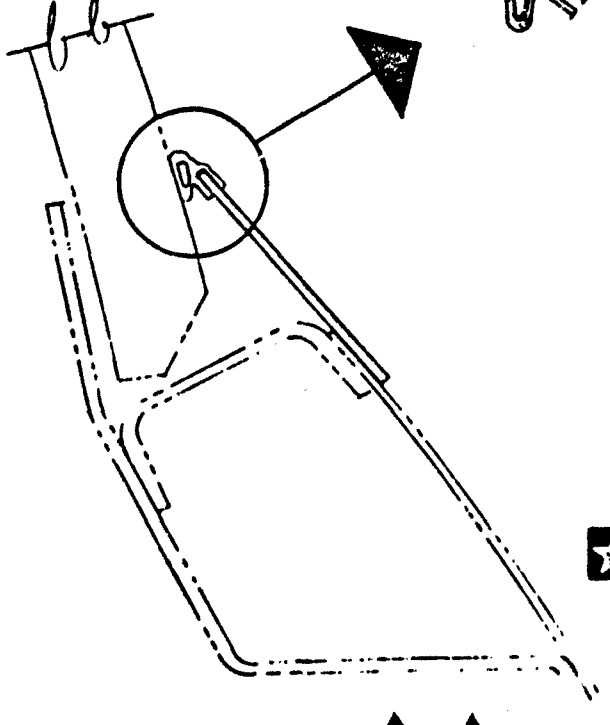


Figure 1





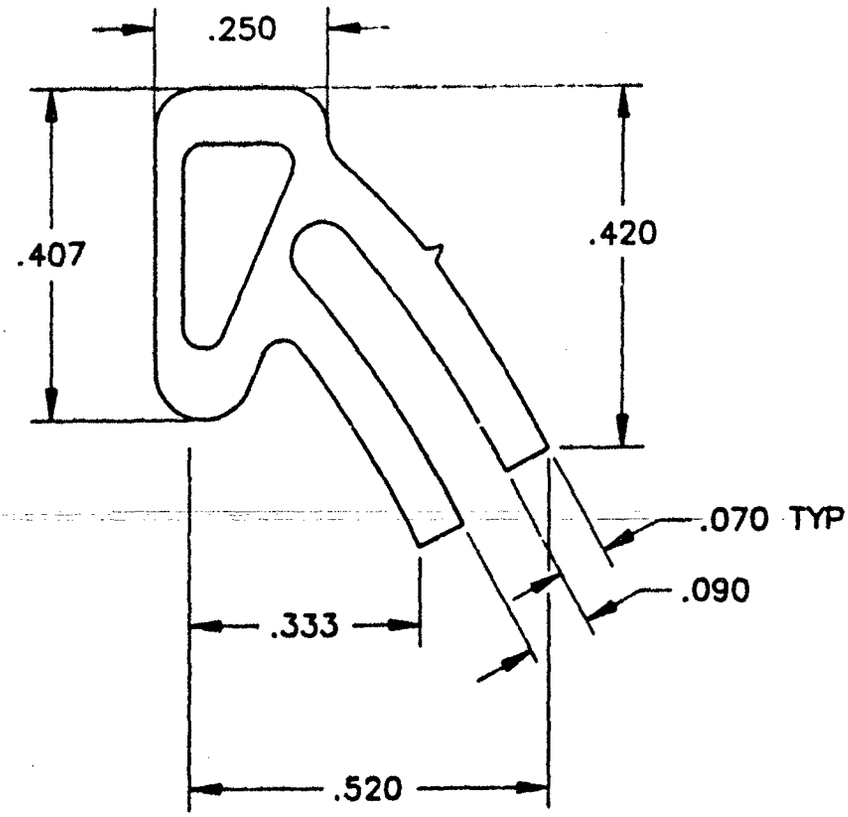




SECTION A A

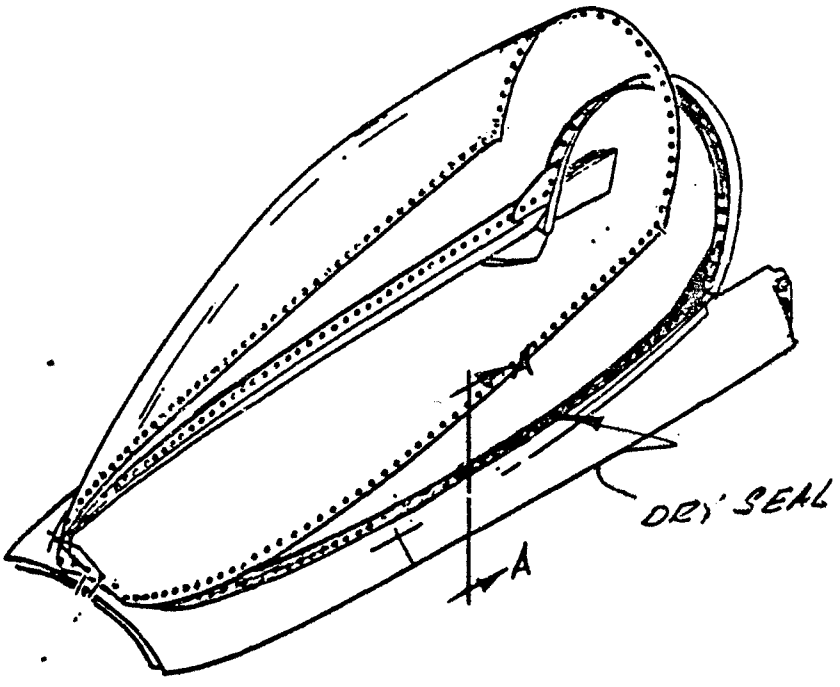
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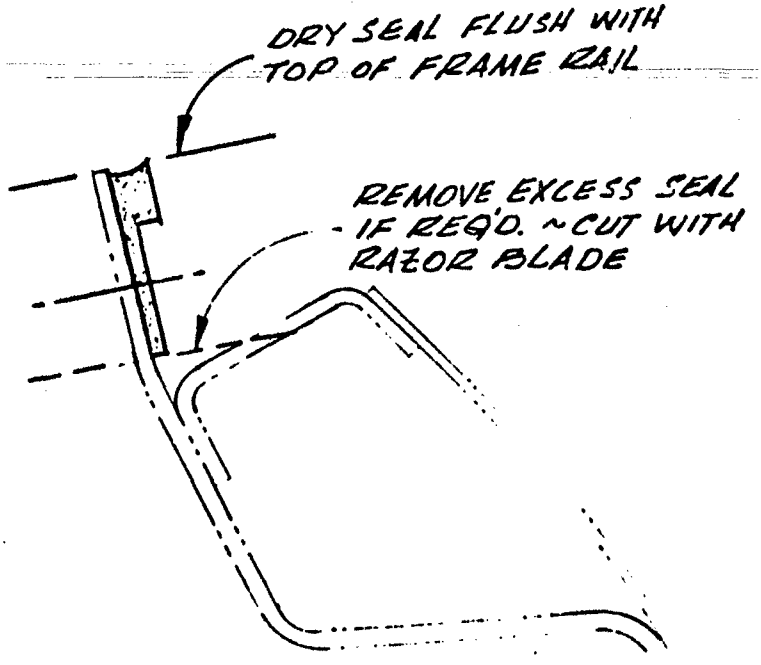
Figure 2



TEXSTAR, INC.

Figure 3

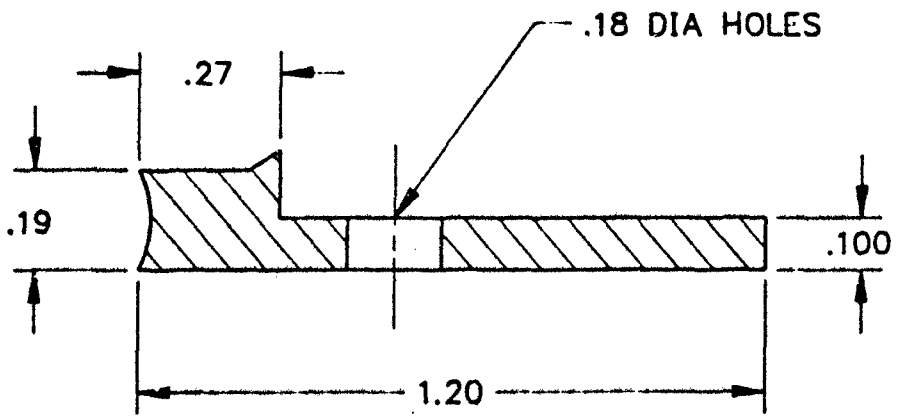




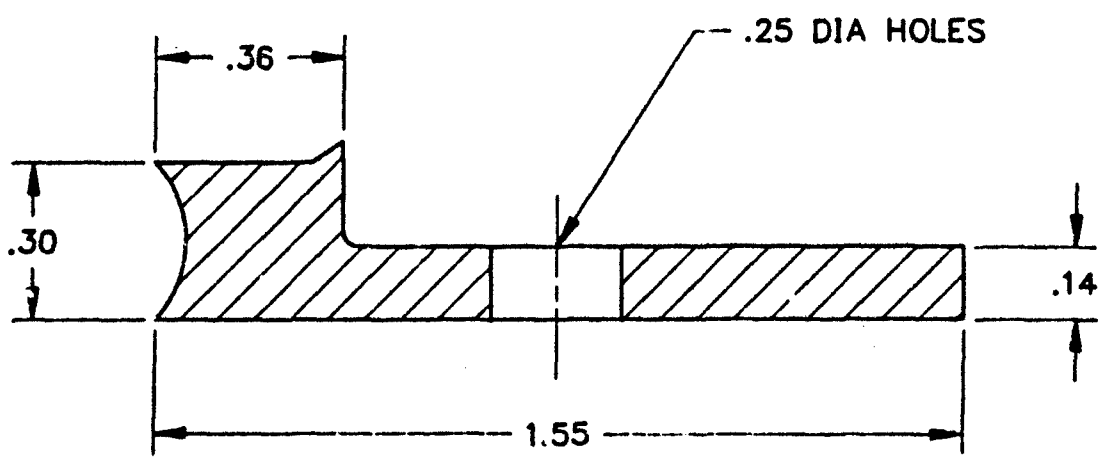
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Figure 4



HOOP



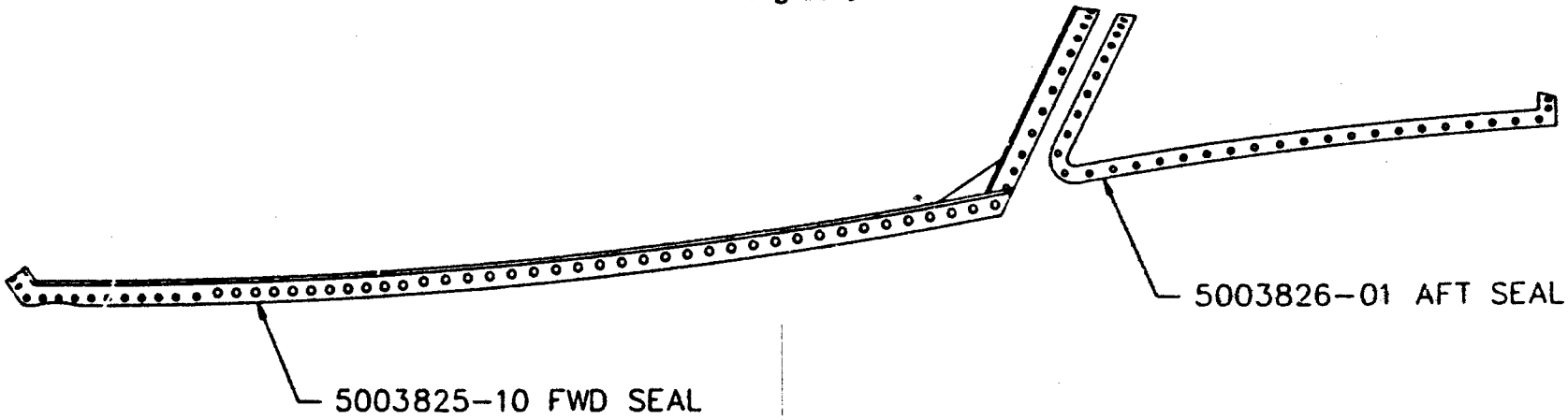
LONGERON

TEXSTAR, INC.

Figure 5

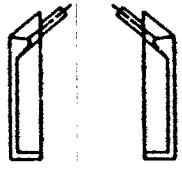
TEXSTAR, INC.

Figure 6





5003939 FWD COVER



5003957 LH END CAP

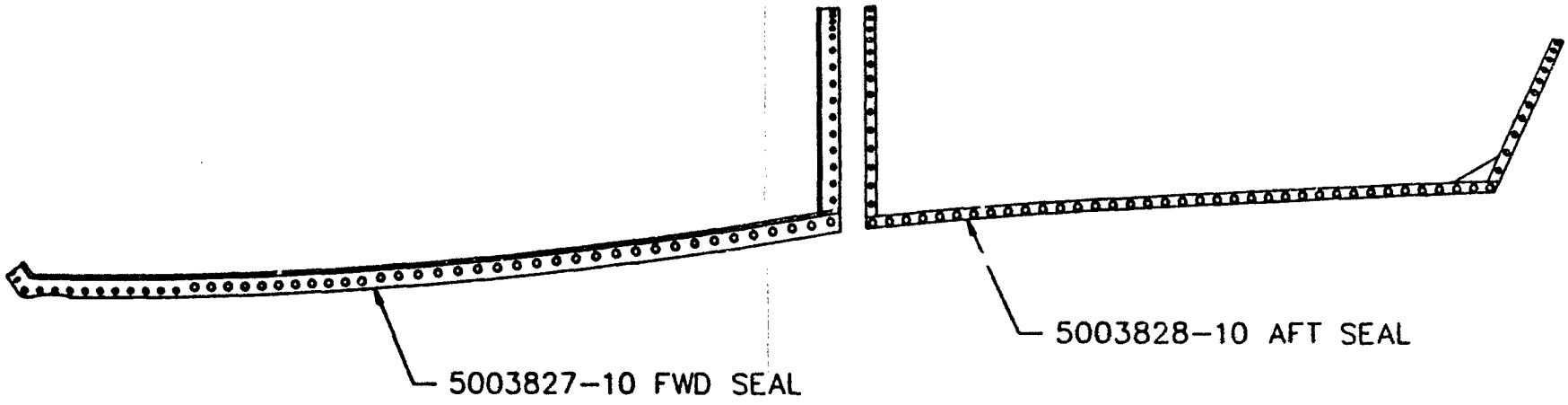
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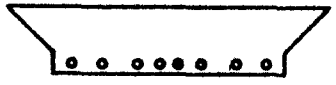
5003864-03 FAIRING SEAL

5003950-01 SEAL SYSTEM KIT

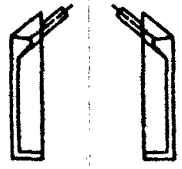
TEXSTAR, INC.



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5003939 FWD COVER



5003957 LH END CAP

5003958 RH END CAP



5003864-07 FAIRING SEAL

5003864-09 FAIRING SEAL

5003950-02 SEAL SYSTEM KIT

Figure 7

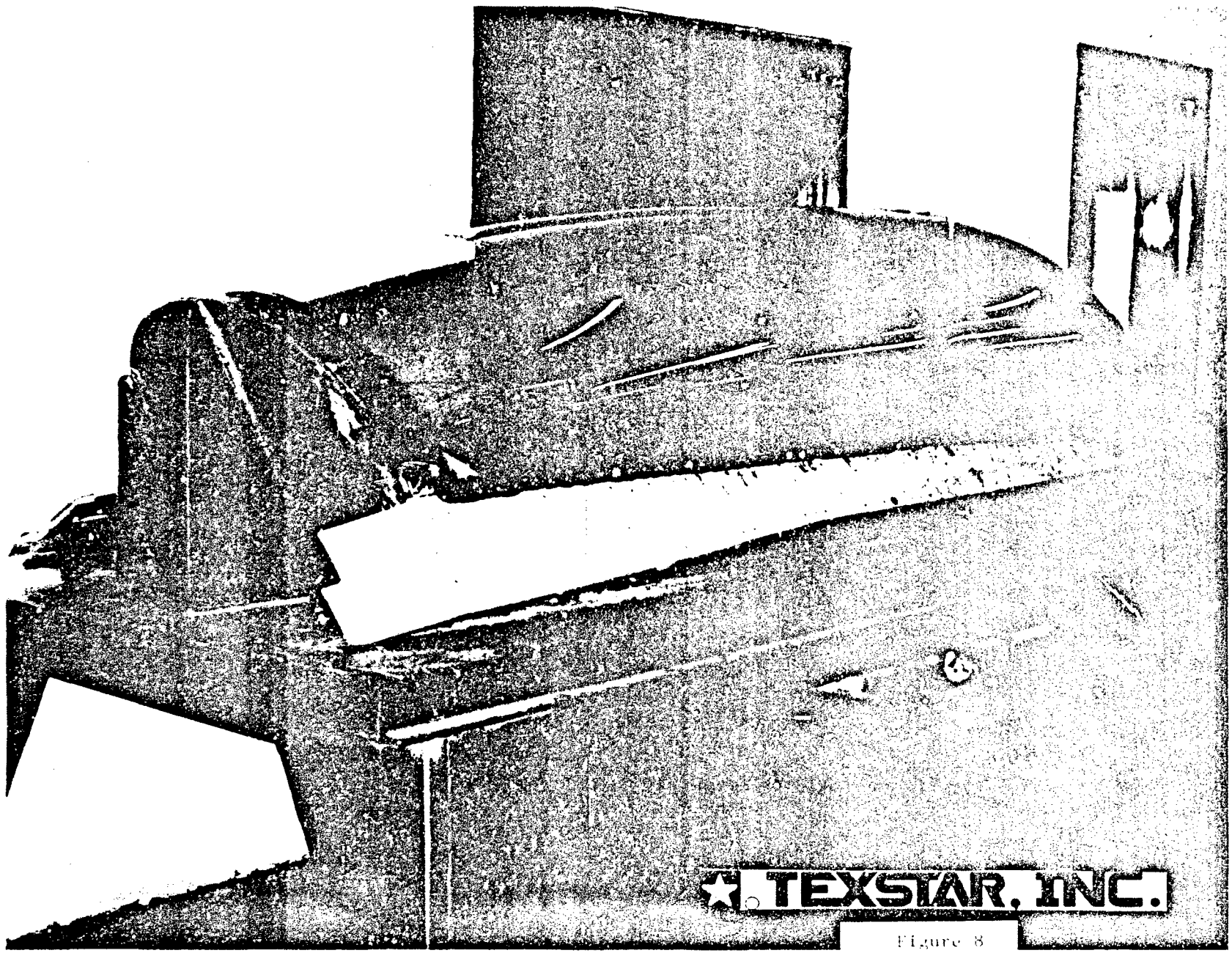


Figure 8

QUICK FIX AND QUICK CHANGE OF TRANSPARENCIES

M. E. Kelley
Flight Dynamics Laboratory
Wright Laboratory

QUICK FIX AND QUICK CHANGE OF AIRCRAFT TRANSPARENCIES

1. Aircraft windshields and canopies are items that must be present and serviceable for the aircraft to fly. Units have a strong incentive to have operational aircraft for various peacetime missions (training, etc), and an even stronger incentive to have aircraft available for combat sorties during a conflict.

2. Aircraft transparencies are subject to gradual damage and deterioration (fair wear and tear) to their optical characteristics, and when they get bad enough they are either repaired, or scheduled for replacement at some convenient time. There are many factors that help determine the "convenient time,", to include the types of missions the aircraft must fly and which ones can be flown by other aircraft instead (cancel night flights, for example), when the aircraft is off the flying schedule for other reasons, and when there is a replacement part available. If the transparency is not repairable, there is still a need to keep them usable until they can be replaced.

3. One of the most important considerations for operational units is to have usable aircraft. There is a need for both "quick fix" techniques to make permanent and temporary repairs to keep deteriorated transparencies usable, and "quick change" techniques for transparency replacements so that aircraft are returned to operational status in minimum time. This paper will address both "quick fix" and "quick change" techniques applicable to aircraft transparencies.

4. A brief word on the scope of this paper should help the reader. For this paper, battle damage repair techniques will not be addressed since many of those emergency repairs would not be used in peacetime. Some transparencies that cannot be repaired at unit level could be repaired by commercial companies or depots, but those repair/restoration techniques will not be considered in this paper. In addition, the "quick fix" techniques that will be addressed have been developed for acrylic transparencies. The permanent repair type of "quick fix" would not be applicable to either glass or polycarbonate transparencies. The temporary improvement "quick fix" techniques would help with any transparency, but would be most applicable to acrylic. The "quick change" techniques apply to all types of transparency.

5. Windshields and canopies can suffer many different types of damage and deterioration. Various coatings exist which could be applied in a factory environment and greatly reduce vulnerabilities to in-flight abrasion. Other papers by other authors will address such coatings. Prevention is a preferred approach, but very few parts presently flying have such coatings. Operational units must therefore have ways to deal with the problems that presently installed transparencies will experience.

6. QUICK FIX: Some (but certainly not all) of these problems can be effectively treated with some quick fix techniques. Common problems and candidate solutions include the following:

a. Severe structural damage is uncommon, but when it occurs there is no repair option (excluding battle damage repair techniques) and the transparency must be replaced before the next flight. Quick change techniques will come into play instead.

b. Gouges and scratches cannot be effectively repaired at unit level, but such defects can usually be tolerated for several days (or longer) until there is a convenient time to replace the transparency. The defect(s) might cause a limitation in the missions the aircraft can fly, such as no night flights. The "put up with it" approach is not a temporary "quick fix," but it sometimes has a similar effect on aircraft availability.

c. Crazing is a condition that gradually develops over many months, and these is no effective repair technique. Transparency replacement is the only solution, but the replacement action can be scheduled for a convenient time.

d. Severe abrasion can occur in one flight, creating haze that makes the transparency unusable. Abrasion is a common and serious problem for aircraft transparencies when aircraft fly at low altitudes in desert environments. Airborne dust and dirt sand-blast the surface, causing severe haze. Such conditions must be dealt with before the next flight since in-flight abrasion typically effects the entire windshield. Fortunately, abrasion is also a defect that can be effectively dealt with using "quick fix" techniques. Desert Shield aircraft experienced some abrasion problems, and quick fix solutions were devised to help the operational units.

7. Abrasion "quick fixes" fit into two categories, permanent repairs and temporary improvements.

a. Permanent repairs consist of light sanding and polishing to remove the outer abraded surface, leaving a smooth surface.

(1) The standard technique used for decades relies upon skill and elbow grease, with an individual using the sand paper/polishing materials by hand. This treatment is typically attempted when there is mild abrasion, and severely abraded windshields are discarded. The repair takes about 25 hours of labor, which means it will take several days (or several people working in shifts for one day) before the aircraft is again available. This is not very quick.

(2) The new "quick fix" technique uses the same concept, with the addition of hand held powered sanding/polishing equipment. Significant skill is still needed, but a windshield can be restored to a low haze/good optical clarity condition in about 6 hours. In addition, windshields with very severe haze can be (and have been)

successfully restored. The powered equipment technique increases the number of windshields that can be repaired instead of discarded, and reduces the time needed for each repair. The F-15E SPO deserves the lion share of the credit for developing the powered polishing kit approach, with our office providing assistance. Our office probably did not provide enough assistance to earn large credit for the success, but we did provide enough assistance to receive large blame if the concept hadn't worked.

(3) The kits were successfully used by Desert Shield/Desert Storm units to restore optical clarity to F-15 windshields. The kits have also been successfully used for several years by other F-15 units flying in CONUS desert environments. Over 50 F-15 windshields have been repaired with the new procedure and have remained in service.

(4) The process of removing very thin layers of material cannot be repeated indefinitely, whether or not the powered equipment is used. The process can be performed several times before the transparency would no longer be serviceable. Thickness reductions causing strength reductions would be a concern, and there should also be some thought given to possible strength loss due to very long service lives. Pass/fail criteria will have to be different for thin acrylic canopies versus thick acrylic windshields, and should include both strength and optics. For windshields in abrasive environments the creation of distortion would probably be the factor that would determine when the windshield had to be replaced. How many times could the process be repeated? Skill of the maintenance personnel and severity of the abrasion would be main factors. Three times? Probably. Six times? Perhaps. Ten times? Very unlikely.

8. QUICK CHANGE: Aircraft being flown by operational units (or commercial companies) are valuable assets when they are operational.

a. A military organization must be able to fight with the bad guys, while a commercial organization must be able to earn money by moving people and cargo. An aircraft that is out of commission contributes little to an organizations's ability to "meet the mission." While the missions are very different, some of the same factors effect both types of organization.

b. Manhours needed to replace a part such as a windshield are always important, and design details of windshield and canopy systems have a large effect on the number of manhours needed for each transparency replacement.

c. For military organizations and commercial companies an even more important consideration is the total time the aircraft is out of commission. Design details are important here also, especially the type of seal/sealant system used to install the transparency into the frame. The seal/sealant systems used fit into one of the following four categories:

(1) Wet sealant;

(2) Dry seals that are included as part of the manufactured transparency;

(3) Dry seal systems that are retrofitted into existing aircraft and helicopters; and

(4) Non-curing sealants.

9. There are advantages and disadvantages to each of the candidate approaches that are used.

a. Wet sealants have been the most common sealant concepts used when installing DoD transparencies. Their advantages and disadvantages include the following:

(1) Advantages:

(a) Compensate for irregularities and gaps in frames.

(b) When designing and making aircraft and canopy frames, can be less exact. Wet sealants use would allow tolerances to be less demanding.

(c) Since time needed for sealant cure has little or no effect on the time needed to manufacture aircraft or to perform depot level maintenance, the disadvantages of wet sealant use have little effect on either manufacturing or depot maintenance functions. In addition, the aircraft manufacturing function would not be removing installed parts and consequently would not be effected by any increased difficulty in removing parts which were installed with wet sealant.

(2) Disadvantages:

(a) Aircraft out of commission 1 to 4 days (or longer).

(b) Wet sealant is messy and time consuming to remove.

(c) Cure time 1 to 3 days at 70 degrees F.

(d) In winter, aircraft must stay in heated building until the sealant has completed it's cure. Cure time is a function of temperature, and cold temperatures can stop the curing process.

(e) Increasing temperatures can accelerate sealant cure, but intentionally raising the temperature with heat lamps (or the standard practice of parking the aircraft outside in hot summer) might cause other problems. Some transparencies have been damaged from improper use of heat lamps. Elevated temperatures also make it easier for hostile chemicals to attack plastics and cause hidden structural damage.

(f) Sealant is a two part material that must be mixed together. This requires significant physical effort if a machine is not available to do the mixing. Occasionally, some sealant material is not mixed, and that unmixed material will never cure.

(g) Pressure checks are performed when sealant cure is completed. Aircraft fail pressure checks 5%-10% of the time (F-111 results), requiring use of more wet sealant and additional cure time. In some cases the transparency must be removed and reinstalled with new sealant, especially if some of the sealant had not mixed well and was not curing.

(h) Wet sealant usually has a 6 month shelf life, assuming good storage temperatures. The supply system often delivers sealant with expired shelf life to operational units, who use it. Expired sealant is typically harder to mix and use, but usually seems to works OK.

(i) It is not uncommon for pressure leaks to develop with transparencies that have been installed for lengthy periods, due to deterioration of cured wet sealant. (Could use of expired sealant sometimes be a factor? Perhaps.) Such leaks are almost never severe enough to cause in flight emergencies, but must be plugged (using more wet sealant with cure time) before the next flight.

(j) Once mixed, wet sealant starts to cure. "Working time" is the time before the curing process proceeds so far that the sealant becomes stiff. The transparency must be installed and the bolts tightened before the working time is up. If there is a problem or delay, the partially cured sealant must be removed and the transparency installation process begun again. For large transparencies with many bolts, sealant with long working time is needed. Sealants with working times of two hours typically have cure times of three days.

(k) Most wet sealants available for use have aggressive chemicals that can cause structural damage to unprotected plastics. If the wet sealant chemicals can reach the plastic (example: bolt holes), cracks and strength loss can occur.

(l) Sealant cost varies widely, up to $50 per tube. More than one tube is often needed for a transparency installation.

b. Dry seals being included as an integral part of the transparency. This is the PREFERRED approach, and designers of new aircraft are urged to use this approach. This is the typical approach used for decades with commercial passenger aircraft. Advantages and disadvantages are given below.

(1) Advantages:

(a) Minimizes aircraft down time.

(b) Zero extra time and effort needed by maintenance personnel.

(c) Cleaner and neater; no mess.

(d) No delay before performing pressure checks and flying aircraft.

(e) Proven approach; decades of experience with commercial aircraft.

(2) Disadvantages:

(a) Cannot compensate for large gaps, irregularities in frames of aircraft and canopies.

(b) Retrofits to older aircraft/helicopter may be difficult, if aircraft/helicopter was designed with wet sealants in mind.

(c) Request for Purchase (RFP) must be revised to ask for dry seal with new procurement. This requires detailed knowledge of aircraft frame details, candidate dry seal systems, and knowing what to ask for and how to test the new system to determine if it will work properly. This ain't easy! Those who would benefit (folks in operational units) aren't the ones that would expend the time and effort, so it should not be surprising when the status quo (wet sealant) tends to remain in use.

c. Dry seal retrofit, typically onto the aircraft or canopy frame. Retrofitting a dry seal onto existing transparencies that do not have dry seals is also a possibility, but having new transparencies manufactured with dry seals would almost always be preferred to a retrofit to the transparencies.

(1) Advantages:

(a) If the dry seal is reusable (regains shape after crushing load removed), the retrofit dry seal approach has similar advantages to the preferred approach (dry seal on transparency). These include minimize maintenance effort, and minimize aircraft down time.

(b) Gaps and irregularities in aircraft and canopy frames could be permanently plugged/smoothed during initial installation of dry seal.

(c) Some dry seals are very strong and very resistant to chemical attack.

(2) Disadvantages:

(a) Significant design and testing effort might be needed.

(b) Dry seal is a separate item; installation techniques, inspection procedures, and training of personnel will be needed.

(c) Dry seal may not be reusable if installed for lengthy period, or if damaged during transparency removal. Removal and replacement of dry seal might be time consuming.

(d) Cost varies widely; $100 or more per transparency.

d. Non-curing sealant; "Tacky Tape."

(1) Advantages:

(a) Proven concept: F-16 (about 1980), F-111 windshields and canopy hatch transparencies (1989), F-4 one piece windshield (1989), F-15 windshield (end 1991), A-4 windshield (1993). We have been told of successful (but unauthorized) transparency installations on other aircraft using Tacky Tape.

(b) Simple installation.

(c) Easier, faster cleanup than wet sealant.

(d) Minimize maintenance manhours and aircraft downtime. No cure time needed; conduct pressure checks immediately. When adopted for the F-111 transparency, reduced maintenance manhours by 40% and aircraft downtime by 90%. This was equivalent to adding 9 F-111s to the fleet in terms of aircraft availability.

(e) Compensates for gaps, irregularities in frames.

(f) Pressure leaks very rare; much fewer than wet sealant.

(g) Easy to retrofit onto many existing aircraft.

(1) One week was needed to develop and conduct initial tests with the Tacky Tape sealant procedures for the F-111. Several F-111 Wings adopted the new Tacky Tape procedures before they received a copy of the procedures. They found that the message describing the test effort was all they needed. (Note: The draft procedures authored by WL/FIVR in 1989 are still in use. A message authorized the use of the FIVR document until the formal Tech Order change was published, and units are content with the status quo and the draft document.)

(2) A-4 windshield procedures were developed in about 1 day by US Navy personnel at depot. The Navy personnel were given some rolls of Tacky Tape and a couple page letter on "how to create new windshield install & remove procedures," and they created the new procedures without difficulty.

(h) Indefinite shelf life.

(i) Cost very low. About $5.00 of Tacky Tape needed to install F-111 windshield.

(j) Fewer transparency failures due to damage caused by "removal to facilitate other maintenance." Some F-111 transparency designs had a high failure rate during removals, with the inner edge member tearing off due to adhesion with the wet sealant. Similar failures still occur when Tacky Tape is used, but not as frequently.

(2) Disadvantages:

(a) Not a true "dry seal;" has some install and cleanup time. Much better than wet (curing) sealant, but not as good as the zero time with a dry seal that is part of the transparency.

(b) Some transparency failures during "removal to facilitate other maintenance." Significant improvement over wet sealant, but not as good as dry seals which would eliminate most of these failures.

(c) Might not be appropriate for thin acrylic transparencies. Tacky Tape is fairly tough stuff, and will not "flow" as quickly and easily as uncured wet sealant. Load concentrations might be created at the bolt holes in the thin acrylic transparency when bolts are tightened.

(d) So easy to use there is a temptation to use it where it is inappropriate. (We do not advocate making Tacky Tape harder to use as the "fix" for this "disadvantage.")

(1) Not recommended for panels that must be blown off or cut out during emergency escape, unless specific formal tests prove the Tacky Tape will not interfere with the emergency escape process.

(2) Most thin panels are probably not good candidates for Tacky Tape use as a "ripple" effect could result along the edges, with the Tacky Tape holding the panel out slightly between each bolt hole.

(3) It would not be difficult to determine which panels and other items could use Tacky Tape. However, the approach used to find out what works and what doesn't work will largely determine when (or if) successes will be transitioned to wide operational use.

(a) On many occasions, maintenance personnel in operational units have quickly developed successful solutions to operational problems. Sometimes the solutions have been devised in hours or days, especially in combat situations. (Necessity is the

mother of invention.) Maintenance personnel in operational units could probably find many items where Tacky Tape use would be beneficial if they did some experimenting. However, it is difficult (and risky) to officially report successful but unauthorized do-it-yourself solutions so that they can be widely adopted. It can also be risky to careers to ask for help when the unauthorized actions are not successful. (Who can you ask? Who do you trust?) And, if catastrophic failures occur folks can be in BIG trouble!

(b) When tests are official and authorized, successful applications can be formally adopted and widely used, unsuccessful applications can be identified and publicized, and other experts can be consulted whenever unexpected problems are encountered. Catastrophic failures in the laboratory environment produce interesting and helpful information, and usually no one gets very excited or upset when such events occur. Authorized testing efforts usually take a while to set up, but lab efforts have created solutions in less than a month when the problem was very urgent. A well-conceived research effort should involve those who need the problem solved, and maintenance personnel would be valuable participants in planning and conducting such an effort.

(c) Engineers in laboratory organizations and personnel in operational units might try the same things when attempting to devise an answer to an operational problem, and both might develop the same solution. One would be called "research," while the other might be called "failure to follow tech data." Research results are much easier to document, validate, and transition to wide operational use.

10. Aircraft transparency systems can be used as examples for why various slogans and sayings are valid. In our various transparency efforts we have occasionally shown that "Murphy's Law" is valid, but we've managed to overcome whatever Murphy put in our way. We have avoided using the slogan "'Tis better to have tried and failed..." with any of our efforts, and hope that others can also avoid this saying. Perhaps every present and future transparency system can eventually be a success story, showing the validity of slogans such as "plan ahead," "do it right the first time," "a stitch in time saves nine," and "where there's a will, there's a way."

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COCKPIT SOLAR SHIELDS FOR DOD AIRCRAFT

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COCKPIT SOLAR SHIELDS FOR DOD AIRCRAFT
...A Not-So-Short History of the Program

1. Hot cockpits caused by solar heating are an annoying and costly problem. Temperatures inside aircraft and helicopter cockpits have been measured at over 215 degrees F at some CONUS locations, and some overseas locations can generate even higher temperatures. These temperatures are not good for cockpit equipment and materials, and are not good for maintenance and aircrew personnel.

2. There are various approaches that can be used to try and overcome the hot cockpit problems. This paper will focus on one of these approaches: reflective cockpit solar shields. Another paper will address exterior covers. However, one should be aware that there are many different approaches that can be tried, and many of these other approaches are being used. They vary greatly in effectiveness, cost, and practicality. Candidate approaches that have been used or that one might consider using include:

a. Redesign cockpits so that avionics equipment, emergency escape systems, canopies, fabrics, plastics, and all other cockpit materials will survive temperature extremes. Advantages: If successful, redesign of one piece of equipment will increase the service life/reliability of that one item. Disadvantages: Costly and only partially effective; only helpful for specific items that are redesigned. Does little or nothing to help personnel who must sit in the cockpits.

b. Change work schedules for maintenance and aircrew personnel so that cockpits are avoided during the hottest periods. Advantages: Reduces illness and injury from heat. Disadvantages: Harmful for training, aircraft readiness, and could not be used during combat situations (unless the enemy was very cooperative). Does nothing to help prevent deterioration of cockpit equipment.

c. Open doors and canopies on aircraft and helicopters during hot weather. This is a common practice at many bases, and is REQUIRED by a T-38 technical order whenever the temperature reaches 90 degrees F. Advantages: Dumps hottest air, reduces maximum temperatures reached. Disadvantages: Cockpit surfaces still get hot, and personnel must be available at all times (another T-38 tech order requirement) to close doors/canopies when high winds or rain occur (to include weekend shifts for just that task). Open canopies have been blown off aircraft, and aircraft/helicopters with open canopies/doors have had their cockpits flooded with water during sudden storms. (Soaking electronic equipment is usually considered to be undesirable.) Cockpits can get filled with dirt/sand in windy environments, and aircrew members have been known to notice and complain about dirty cockpits during negative "g" maneuvers. Some aircraft do not have enough operable doors/windows to create an effective air flow rate (example: B-1B), so this option is not available for them.

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d. Run one or more air conditioning units for each aircraft that is parked outside. This is effective, but costly and not very practical DoD-wide. Operational units would usually not have enough air conditioning units to use this approach, and assigning maintenance personnel to monitor the air conditioning unit would be an unproductive use of available manpower. Mobility kits would not include one air conditioning unit per aircraft, and it would not be reasonable to expect such equipment would be available at a deployment location.

e. Start aircraft engines, turn on aircraft air conditioners, and abandon the aircraft until the cockpit cools off. This technique has been used for some aircraft, but that does not mean it is either recommended or authorized. Let your imagination run free, and consider: If someone removed the wheel chocks,....

f. Take off with systems that either have not been checked on the ground, or are not usable. Climb to altitude, cool the cockpit, and see if the equipment works. Examples: (1) Some fighter aircraft have LCDs (liquid crystal displays) that are too faint to read on the ground when hot, but become usable/readable about 10 minutes after takeoff when the display cools off. (2) Some "preflight" checks of avionics equipment were performed at 10,000 feet during Desert Shield/Desert Storm. Reason: The equipment would fail due to high temperatures if turned on while the aircraft was on the ground.

g. Direct approach: Attack the basic cause of the problem. Sunlight into the cockpit causes the problem, so block the sunlight. This can be done by various means.

(1) World-wide solution: Block sunlight with atmospheric debris.

(a) Massive volcanic eruptions in recent years have effected climates slightly and created colorful sunsets, but have not blocked enough sunlight to prevent hot cockpits.

(b) The Asteroid Answer: Massive asteroid impact to create continuous night and years of winter. Very effective (just ask the dinosaurs), but has bad side effects.

(c) The Armageddon Approach: Detonate thousands of nuclear warheads and create nuclear winter. Very effective, but has disadvantages.

(2) Down-size the world-wide solution to a local approach: The Saddam Solution. Torch oil fields, blocking sunlight with the resulting smoke. Advantages: Effective in blocking sunlight in local areas. Disadvantages: Extremely costly, harmful to the environment, and might damage the "Mr. Nice Guy" reputation of whomever used this approach.

(3) Use building/aircraft shelter to block sunlight one aircraft/helicopter at a time. HIGHLY RECOMMENDED! A building is the best "solar shield" one can have, and failure rates of many items are reduced. For example, F-111 windshields and canopy transparencies had much longer service life for aircraft inside shelters in the UK than for aircraft parked outside in CONUS. F-16 canopy transparencies lasted much longer indoors in Germany than outdoors in the US. Disadvantages: Shelters are costly to build, most CONUS bases do not have buildings for their aircraft/helicopters, and aircraft shelters are difficult (but not impossible) to send with deploying units.

(4) Put exterior cover over aircraft/helicopter. Covers could cover just the transparencies, or could cover other parts of the vehicle as well. (Some exterior cover efforts are underway, and will be covered in a different paper/presentation.)

(a) Advantages: Effective in reducing cockpit temperatures, can be inexpensive if they are "do it yourself" items (white sheets from the barracks, etc), could be designed to provide protection against chemical warfare exposures, could be designed to provide camouflage, and could be designed to provide protection against other hazards such as hail, rain, or dropped tools. Exterior cover concepts are worth pursuing, but it is not a simple task.

(b) Disadvantages: Exterior covers can cause scratches, crazing, and other damage to transparencies. Inappropriate covers (dark and thin covers for example) used in hot climates have caused windshields to become unserviceable in one day. It is difficult (but not impossible) to design an exterior cover that is easy to use and does not cause damage. Exterior covers can be heavy, expensive (over $5,000 each), difficult to install and remove, and may be impossible to safely install or remove in windy conditions. A truck is required to move some covers to the aircraft/helicopter. Some exterior cover designs prevent cockpit access when installed.

(5) Use reflective solar shields that install inside the cockpit against the transparencies. (These shields will be the primary focus of this paper/presentation.)

(a) Advantages: Very effective, easy to install and remove in any weather, causes no harm even when installed incorrectly, low initial cost (10% of comparable exterior covers), low life cycle cost, durable (2+ year service life expected), light weight, and can be stowed on/carried on many of the aircraft and helicopters during deployments. Solar shields also solve a variety of other problems, to include preventing humidity extremes (condensation and baking/drying) and helping prevent very cold cockpits during winter weather. Maintenance and aircrew personnel have found shields cause large improvements in the cockpit environment, and one base installs their prototype solar shield in whichever aircraft is undergoing lengthy maintenance checks because

it makes the cockpit more comfortable. Shields are environmentally friendly, and can be recycled as shoe inserts for both hot and cold weather. Savings to DoD from solar shield use have been estimated at $440 million per year due to decreased failure rates of cockpit equipment. This is an 80,000% "return on investment" from buying and using a solar shield.

(b) Disadvantages: None. Limitations: During actual combat when one is trying to hide from the enemy, do not use reflective shields which are easy to see. If chemical warfare exposures are involved, some other protective measures will be needed. Shields help with many different problems, but there is no truth to the rumors that shields prevent flu, baldness, or tooth decay, and shields are only a slight help in solving world hunger (some shields used in Somalia).

3. The cockpit solar shield effort was created in response to various reports of hot cockpit problems. It started with one brief flash of inspiration, but all later achievements resulted from perspiration.

a. The overall solar shield effort began as a flash of inspiration in September, a month after the invasion of Kuwait. However, the flash was NOT provoked by Desert Shield news stories. It was provoked by a "Tiger Team" report from one B-1B Wing which stated that 190 degree cockpit temperatures were causing many problems, but no suitable solar shield/cover had been created. The Tiger Team report was read on a Wednesday afternoon, the flash occurred, access to a B-1 was arranged, a crude but usable do-it-yourself shield was designed and build, and on the next Monday the shield was demonstrated at a meeting of all B-1 Wings. It successfully served two roles, one planned and one unplanned.

(1) The shield was intended as a cheap ($40) proof-of-concept that would be a one-time effort by us. This was the planned role for the shield. Operational personnel could then copy and improve upon the concept, and it was very favorably received by B-1 personnel at the meeting. At the end of the demonstration it was thought that the one-time effort had concluded.

(2) At the same time that the B-1 shield was being demonstrated in Oklahoma, our office at Wright-Patterson was getting a phone call. Contents of the call: "Units in Desert Shield are having a lot of problems due to hot cockpits. Do you think you can come up with any ideas to help them?" A "yes" answer was provided, based largely upon the crude B-1 shield. This was the unplanned and unexpected role that the B-1 shield played.

4. The solar shield program was created, and we were off and running. On second thought, it might be more accurate to say we were off and walking. "Running" didn't fit; the brief inspiration phase was behind us and we now had to rely upon perspiration and hard work. The concept created for the B-1 was a decent partial

solution for the B-1, but it was not applicable even as a partial solution for most aircraft supporting Desert Shield.

c. Much information was sought and acquired, many ideas and concepts were thought of, and many materials were acquired and tried. The needed performance characteristics and other requirements for the shields were determined, such as effective, easy to use, and cause no damage or other problems when used. The trial and error process then began, seeking usable concepts.

(1) An unserviceable F-16 canopy (with frame) was acquired and placed upside down on the floor to allow easy access to the interior. (Note: We recommend this "upside down on floor" approach not be used with serviceable canopies, especially if they are installed on an aircraft.)

(2) A roll of cheap translucent plastic film was acquired and laid into the canopy. Multiple "tucks" were needed to make the film conform to the compound curvature shape of the canopy. This film use made it apparent that compound curvature would pose challenges for any design. It also demonstrated that supplemental support/stiffeners/attachment techniques of some sort would be needed to hold the film up against the transparency.

(3) Possible sources for highly reflective materials were investigated. NASA provided a roll of aluminized, thin Mylar film. This film was the leftover material from the ECHO satellite. (We did not devise a solar shield concept that used the thin film, but we have not thrown the material away.)

(4) Many versions/variations on the theme were tried to devise some configuration that would make practical solar shields.

(a) A "backbone" stiffener for the film that would run down the center of the canopy could be used to create a usable solar shield, but the resulting shield would not be easy to use.

(b) The possibility of attaching/gluing/sticking something to the canopy frame and/or transparency was considered.

(1) Permanently attaching pieces of velcro (or anything else) to the canopy frame would require a great deal of testing, justification, and formal approvals, and the administrative workload involved was very intimidating. Velcro on the frame did not seem to offer a solution, anyway.

(2) Permanently attaching a piece of velcro to the top of the canopy transparency would have helped keep a reflective film in place, but the known and possible problems that such velcro would cause were very significant. Reasons for immediately rejecting this velcro-on-canopy concept included optical distraction to the aircrew, possible structural harm to the plastic canopy transparency from the velcro glue attachment, the need to use white or reflective glue and white velcro to prevent creating

a "hot spot" on the plastic, and the possibility that the white or reflective spot would make it easier for the bad guys to spot the aircraft in flight.

(c) Some other material was needed instead of the thin films to use as solar shields. The other material needed to have enough structural stiffness/strength to prevent it from flapping in the wind, or falling out while the person installing the shield reached for a supplemental support.

(5) Various concepts for supplemental supports were tried.

(a) Strips of cardboard were cut out and laid in to the canopy to determine lengths and widths that would be needed.

(b) A scrap venetian blind was obtained and disassembled. The long slats could be made to work as supplemental supports, but they would not work well. Something better was needed.

(c) A 1/8 inch thick acrylic sheet was obtained and cut into 2 inch wide strips. These strips worked well as supplemental supports.

(d) To determine long term effects on the acrylic from being in the bent (installed) position, two simple tests were conducted. Acrylic strips were positioned in the F-16 canopy and left there. Other strips were placed in a bent position and kept in the office area where they could be examined daily for two months. The office test used partitions/wall dividers to anchor the ends of the curved acrylic. The strips thereby served a secondary role, forming distinctive and unattractive archways leading to the desk of the solar shield project officer.

(e) The acrylic did not break when bent to the needed shape, or when bent well beyond that point. However, it seemed certain that someone, sometime would bend the acrylic strips enough to break them. Question: What would happen? A simple test was conducted to answer the question. Result: The acrylic broke into multiple pieces, and chunks of acrylic impacted a wall with considerable force. Such an event would pose both an injury risk and a FOD hazard.

(f) A sheet of 1/8 inch polycarbonate was then acquired and cut into strips. Polycarbonate looks like acrylic, supplemental supports of polycarbonate would work as well as acrylic, and polycarbonate will not break. Concentrated effort (folding the strip back and forth) is needed to intentionally break it, and then the "breaking" consists of tearing the fatigued and thinned material. Polycarbonate costs a bit more than acrylic, but the effect on the total cost of a solar shield is insignificant. The decision was made to only use polycarbonate, and acrylic was not used with any solar shield.

(6) A General's private aircraft was examined that had solar shields installed. The name of the shield manufacturer was noted. The next day an office member brought in an Aero Club magazine with an advertisement for cockpit solar shields for small aircraft. Same company. OK, we can take a hint. The company (Kennon Protective Coverings) was then called.

(7) At the start of Desert Shield, the company had offered to create and deliver (at no cost) some solar shields to help Desert Shield units. However, after repeated efforts and multiple phone calls they had been unable to locate anyone who was able to accept the gift...or understand solar shields. When we called them, they decided we were the ones they had been seeking and they extended the "free shields" offer to us. We accepted.

(8) The company had the solar shield material, and expertise and experience with solar shields for private aircraft. However, they had no experience with making shields with compound curvatures, or how to hold large, overhead shields (example: canopy shields) in place. They had what we lacked, and we could provide what they lacked. It looked like together, we could do it!

(9) The initial attempt to create a prototype solar shield used the same approach that is used for private aircraft. The Air Force Museum provided access to several aircraft, and we traced the shape and size of the transparencies on large pieces of paper. The paper tracings were sent to the company.

(10) The company determined that this approach was not workable with the large, compound curvature transparencies that were of interest. We then arranged for their Chief Designer to gain access to F-15 aircraft during his Christmas vacation in Alaska.

(11) Three days were spent trying to develop solar shields for the one and two seat version of the F-15. The designer did not have materials for supplemental supports available, and did not produce a usable design. Operational personnel at the base requested an exterior cover (seems hot cockpits aren't a big problem during Alaskan winter), and the requested cover was created.

(12) After some discussions, it was agreed that the pieces of the unsuccessful F-15A solar shield would be sent to Wright-Patterson AFB, and we would try and create a usable design.

(13) The pieces arrived in late January, and access to the Air Force Museum's F-15A was arranged for the morning of 1 Feb 91. Cockpit access was provided at 1100, the needed length of supplemental supports was determined, and the task of creating the supplemental supports was handed to a student who was assisting the project. The solar shield project officer then went into the Air Force Museum to witness a briefing to news media by the F-15E SPO.

(14) There were a few last minute changes to the briefing. The solar shield project officer found that he would also brief the media (Who, ME? Gulp!). Since the media was already there, they would also have the F-15A solar shield demonstrated (Gulp! Sure hope the supplemental supports the student is making are being made in the correct length!)

b. Fortunately, the supplemental supports had been made in the correct lengths and the initial installation of the prototype cockpit solar shield went smoothly. This was documented by TV crews from 3 stations plus reporters from several newspapers. (Talk about pressure! If the shield hadn't gone in easily the first time, would the attempt have been on network Bloopers shows?)

c. This initial F-15A solar shield design needed some significant changes to improve ease of installation, but the concept had been shown to be practical. We now entered the next phase: Refine/perfect the F-15A design, and then create comparable solar shield designs for other DoD aircraft.

d. Edwards AFB CA was chosen as the location for the next phase since they are a testing organization that has many different types of aircraft and helicopters. Kennon Products and the Windshield Program Office (WL/FIVR) would each provide individuals to perform the tasks in the next phase. The task description: "We'll meet at Edwards AFB and start inventing." Costs for design creation were rather low from the DoD point of view. Kennon Products agreed to provide people, equipment, and solar shield material at no cost. WL/FIVR would bring strips of polycarbonate for making prototype "flexible bow frame" supports. Kennon would be paid for production solar shields, with no assurance that orders for solar shields would be forthcoming. They would be paid nothing for design creation.

e. Edwards AFB was visited in late Feb 91, and the efforts to create practical solar shield designs were very successful. In the 28 months since then, designs have been created for over 70 different DoD aircraft and helicopters. Limited numbers of most of these designs have been purchased and distributed to various organizations for testing and operational evaluations. Numerous foreign governments, CONUS companies, and overseas companies have also gotten into the act, but non-DoD organizations have to purchase their own shields directly from the manufacturer. Solar shields produced by the solar shield effort are now being tested or used by someone on every continent except Antarctica. One DoD operational unit used solar shields in Alaska during the winter and found them very helpful in preventing cold cockpits, so solar shields might eventually be used on every continent.

(1) Tests and operational evaluations to date have included fit checks, use of solar shields by operational units for a summer or longer, and tests using instrumented aircraft and helicopters to measure temperature and humidity. Various tests and computer analyses have also been conducted to determine the

strength, durability, safety, and effectiveness of solar shields and solar shield material. Most tests were formal using standard testing techniques, but some additional destructive tests were also conducted that were rather informal.

(2) Tests with instrumented helicopters and aircraft have shown solar shields reduce cockpit temperatures by 60 to 80 degrees F, eliminate large daily temperature swings, and prevent humidity extremes.

(a) Shields not only prevent high daytime temperatures, but also prevent large temperature drops at night. One formal test program conducted by a foreign Air Force used two instrumented aircraft, one with solar shields and one without shields. The unprotected aircraft had a daily temperature change of 90 degrees, which could help explain why problems occur such as jammed canopies, canopies that cannot be closed, and canopies and canopy frames that crack. The solar shield aircraft had much lower daytime maximum temperatures, and slightly higher nighttime minimum temperatures. The daily temperature swing was 40 degrees.

(b) The above test program also included humidity measurements, the only formal test effort that has tracked humidity to date. In the test the unprotected aircraft had humidity extremes that indicated all moisture was being baked out of the cockpit materials during the day, creating a dew point of well over 100 degrees F. At night, falling temperatures would cause significant amounts of moisture to condense on the coolest surfaces (windshield and canopy probably). The solar shield aircraft had much higher minimum humidity, and a much lower maximum dew point. Little or no condensation would have formed.

(c) US Army personnel had previously advised us that nighttime condensation in helicopters was a problem, creating puddles of water on the glare shield and instruments. They suggested solar shields might help solve that problem. Their inputs alerted us to the humidity issue, and caused us to analyze the humidity data generated by the foreign test program. The analysis indicates the Army personnel were correct when they thought shields would also help solve humidity problems.

(d) Maximum cockpit temperatures of 215 degrees were measured in one US Army test, with the 215 degree readings being reached on five consecutive days.

(e) One Marine Corps test showed that in some situations the solar shields can cause the cockpit interior temperature to actually be lower than the outside ambient air temperature.

(f) An Air Force operational unit installed shields in a T-37 which was not on the flying schedule. Several days later they measured the cockpit temperature as 98 degrees when the outside temperature was 94 degrees.

(g) One DoD unit evaluated C-141 solar shields and found they were very effective in preventing hot cockpits during summer weather at a southern CONUS location. The same unit also used the solar shields in Alaska during the winter and found shields were good window insulation and made it easier to heat the cockpit and retain heat longer.

(h) Several informal destructive tests were also conducted, to include the following:

(1) The "stand on it" test. Solar shield material was stepped on. Result: No damage.

(2) The "run over it" tests, phase I and Phase II. Phase I had a tug run over the solar shield material on a smooth cement floor. Result: Once the dirt from the tire was wiped off, there was no indication that the test had been conducted. Phase II had a step van run over it on an asphalt surface. The rough asphalt made permanent dimple marks in the solar shield material, but the shield material was not punctured, did not deflate, and would have still been a serviceable solar shield.

(3) The "cigarette lighter" test had a piece of solar shield material held in the flame of a cigarette lighter for about 20 seconds, outdoors in somewhat windy conditions. The result: no noticeable effect on the solar shield material. (Note: Solar shields weren't designed with flame in mind, and we don't suggest that shields have to be able to endure flame exposures. We were surprised when it did not burn. Larger flame, less wind, and longer exposure might get a different result...but since passing the test isn't needed in the first place.....)

(4) The "crew chief challenge" test, Phase I and Phase II. A piece of solar shield material with unfinished edges was handed to a crew chief and he was challenged to tear it without using sharp objects/tools. It was difficult, but he managed to do it. Phase II repeated the challenge using solar shield material with finished edges. The solar shield material won this time and was unaffected by the crew chief's efforts.

(i) Fill-in-the-blank evaluation forms include sections for ease of use when the shield was installed the first time, and ease of use after personnel knew how they were supposed to be installed. Even though we suggest "easy or difficult" as the evaluation criteria, close to half the responses give "very easy" as their evaluation. (Is that a 12 one a 1 to 10 scale?) One evaluator might have gotten a little carried away when he said shields were so easy to use he didn't need people; he could teach monkeys to do it. (Animal trainers: please do not submit unsolicited proposals to pursue this; monkeys would not pass egress safety training courses needed for cockpit access.)

(3) Various Air Force, Army, Navy, Marine Corps, and Coast Guard units have evaluated solar shields, and evaluations have

almost always been very favorable. Problems have been uncommon, almost all problems have been very minor (label locations, etc), and all problems have been promptly solved.

(4) One good way to help assess a product is to look at a list of good things/achievements/benefits, compare it to a list of bad things/problems/defects, and see which list has the most important items. Good things are mentioned throughout this paper, but a complete list is too long to include. However, a complete list of all bad things/problems fits easily. Here is a list of all the problems experienced with solar shields at some time during the effort. (Not included: Paperwork and administrative problems and resulting headaches internal to DoD; which forms to use, how to process them, etc.) Those who might be horrified by the number and seriousness of the problems should remember that the problems have been solved.

(a) The first fit tests with a production solar shield used four different designs, the F-16A/C, F-16B/D, F-15A/C, and F-15B/D/E. Fit checks were performed, and shield effectiveness was evaluated for one day. Shields were very effective, but this is a "problems" list. Two problems were noted.

(1) One F-15 flexible bow frame was 1/2 inch too long to fit properly in the selected aircraft. It took about 10 minutes to trim it to the optimum length, cutting off 1/8 inch at a time until the best length was reached. Polycarbonate is easily trimmed with a hacksaw. It was eventually learned that the problem was due to tolerances in the CANOPY transparency. The original flexible bow frame length fits perfectly in about 90% of the aircraft, but manufacturing tolerances allow some transparencies to have less curvature and a lower "crown" height. This reduces the side-to-side circumference by up to 1/2 inch. The polycarbonate bow frames will have to be trimmed to fit those particular aircraft. Operational units typically assign ground equipment to each aircraft and stencil the tail number on the item, so the need for a one-time bow frame trimming for 10% of shields is no big deal.

(2) The storage bag for the F-15B/D/E shield shipset was a bit too small. This has been the MOST SERIOUS PROBLEM experienced to date, as it is the only problem that has caused lingering harm to the program. (BAG SIZE is the biggest problem???? Explanation needed.)

(a) During shield removal, the shield pieces could not be stuffed back in the bag when loosely rolled. The pieces had to be carried down the ladder and rolled against a hard surface before they were small enough diameter to fit in the bag. This created the possibility of dropping one of the pieces, and one of the pieces was dropped. The dropped piece was immediately retrieved, the shield pieces were tightly rolled and inserted in the bag, and a design change was made so that ALL future bags for ALL solar shields are oversized. All solar shields can now be

loosely rolled and inserted into the storage bag before leaving the cockpit.

(b) What's the big deal, you ask? The solar shield test was described to other F-15 Wings by someone who knew of this first test. The key part of the test that was remembered by members of other Wings is that shields can get dropped, which means they can blow around the flight line and become FOD hazards. Personnel in one Wing refused to see the solar shield demonstrated, stating they already knew about it and knew they didn't want it. They also referred to the shield as "flimsy," relying upon what they remembered being told. (Light weight, yes; but the "drive over it" test doesn't suggest it's flimsy.) This situation shows the truth of the saying, "you only get one chance to make a good first impression." (Perhaps this is one reason Ford and GM don't let anyone see prototypes until they've worked out all the bugs.)

(3) In a couple cases, some of the shield pieces in the first prototype didn't fit well. The design was redone in each case.

(4) In one case, the first prototype was missing one piece. The master shield pattern for that one window had been lost. The pattern was recreated and the shield piece provided. (Total number of individual shield patterns created for DoD aircraft/helicopters to date: about 850.)

(5) Similar nicknames for two helicopters resulted in the shield shipsets being switched: The Jolly Green Giant (HH3) shield shipset was labeled as being the CH-53 shield, and the CH-53 shield was labeled as being for the H-3. A nickname for the H-53 is the Super Jolly. OOPS! Got the "Jollies" reversed! These glitches were caught by us, so operational units didn't get the wrong items.

(6) The shield piece for the small, fixed transparency behind the canopy of the F-16A/C was the right size and shape, and worked OK. However, the way it was cut from the roll of solar shield material meant that it rolled up from front to back. It was easier to use if it rolled from side to side. The master pattern was changed.

(7) In about 30% of the shield shipsets, one or two of the pieces have a label in the wrong place. Most common error: "left" and "right" decals reversed. This causes about 10-20 seconds of confusion during initial installations. To date, only one operational evaluation has bothered to mention this very minor defect. (Understandable. It takes longer to mention it than to correct it, and it doesn't hurt anything to just ignore it.) The decals can be easily peeled off and reapplied on the proper pieces, or simply ignored. We prefer perfection (especially since it costs the same), and have tasked ourselves with performing a fit check on every design to look for such minor defects before we consider our job complete. However, we recognize these defects are "nits" we

wish to pick and we will not delay assigning NSNs until our fit check has been performed. Once operational units have evaluated the shields and indicated they are ready for general operational use, we will press on with requesting NSNs.

(8) That completes the problem list. What has it cost to correct the problems? No dollar cost to DoD for additional parts, and the administrative burden has been minuscule. Usually a phone call identifying the problem was all that was needed. Sometimes a hand-written note and/or drawing was faxed when that was a better way to communicate the problem.

(9) It is hoped that no one considers the above problem list as too intimidating, especially since they are former rather than current problems. Since each solar shield design is created and evaluated as a separate entity, the above problem list is the compiled total for 70 different efforts.

(5) Transparency manufacturers, aircraft and helicopter manufacturers, and foreign services have all conducted tests with solar shields.

(a) Several foreign governments have completed their testing and have ordered shields in large numbers. A couple aircraft manufacturers have decided to have solar shields as standard equipment with new aircraft they deliver.

(b) We have been told that the test reports are very favorable, and the follow-on orders for large numbers of shields certainly seems to indicate the evaluations are very favorable indeed. We are attempting to get copies of the various test reports from foreign countries, but to date have gotten only the RAAF test report from Australia. Since we didn't provide any of the shields to foreign countries or companies (they had to buy them directly from the shield manufacturer), none of them "owe" us a report. The RAAF formal test report described the shields as "incredible," "invaluable," "remarkable," and "fool-proof."

(5) Two different transparency manufacturers conducted tests using solar shield material to determine if any coatings or other transparency materials would be adversely effected by solar shields. They determined solar shields could be used with all transparencies, with one exception. One use limitation was found that applies only to DoD F-16s.

(a) Most F-16s operated by the DoD have a gold film solar coating on the canopy inside surface. The gold film is rather delicate and can be marred by lengthy contact with solar shields (or any other material). This would be an optical defect, and shields should not be used with these gold film parts.

(b) Some DoD F-16s have clear (no gold film) canopy transparencies, and shields can be used with those aircraft. There are also some prototype canopy transparencies that have metal films

on the outside surface, and aircraft with those parts could use solar shields. However, these aircraft are a small percentage of the total DoD F-16 fleet.

(c) Solar shields can be used with all F-16s in foreign Air Forces. Almost all foreign F-16s have clear transparencies, and can use solar shields. A few foreign F-16s have transparencies that use ITO on the inside surface instead of gold, and solar shields can be used with those aircraft as well. One foreign country that tested one F-16 solar shield ordered 22 more. We haven't seen their test results, but perhaps a purchase order can be counted as proof of a very positive evaluation.

5. So, where does the solar shield effort go from here?

a. The selected approach has proven to be very successful, most of the solar shield designs have been evaluated by someone, most of the tested designs were successes on the first try, and all designs with problems had the problems fixed and the designs became successes. So far, so good, but we aren't done yet!

b. We will continue with tests and operational evaluations of designs that have not yet been sufficiently tested, to include performing fit checks to look for small errors/defects (label locations, etc) that units usually do not bother to mention.

c. We will create some additional designs for additional aircraft when good "targets of opportunity" present themselves, but that is becoming an increasingly small portion of the overall effort. We are running out of DoD aircraft and helicopters that don't have solar shield designs already.

d. Our primary emphasis will shift toward those actions that will make it easier for operational units to acquire solar shields via the normal supply channels.

(1) The big payoffs do not come from developing and proving the worth of new products, although that is the "fun" part of any Research and Development effort. The largest payoffs come from transitioning the results so that operational units can use the new products. Some organizations (example: several foreign countries) have already acquired large number of shields and are experiencing the benefits from shield use, but they had to take special actions to get the shields. Standard, routine supply procedures were not sufficient.

(2) Creating and processing unexciting and unglamorous paperwork will make those big payoffs attainable for any operational unit that wanted solar shields. These paperwork activities include assigning National Stock Numbers to designs that have completed testing, and getting a Military Specification published to aid those that will procure future solar shields.

(3) As the saying goes, the job isn't done until the paperwork is completed. We'll get the job done.

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EXTERIOR TRANSPARENCY COVERS & HAIL TESTING

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EXTERIOR TRANSPARENCY COVERS & HAIL TESTING

1. There are a variety of hostile environmental exposures that parked aircraft transparencies are exposed to, both natural and man-made. Natural exposures include solar heating, rain, condensation, frost, snow, freezing rain, acid rain, blowing sand/dirt, bird droppings, very hard water dripping from ceilings, and hail. Man made exposures include dropped objects, paint overspray, aircraft wash rack chemicals, grease and chemicals on hands and cloths of maintenance personnel, and possible combat hazard exposures such as chemical agents and decontamination chemicals/temperatures. Other papers have discussed other approaches to prevent or deal with problems caused by some of these exposures, to include cockpit solar shields, quick fix/quick change techniques, and transparency coatings.

2. This paper will address two related topics:

a. Exterior covers to protect transparencies from hostile exposures, to include solar heating, liquid precipitation, and severe hail.

b. New concepts and techniques being pursued for conducting hail impact tests on unprotected items (to determine baseline vulnerability and need for protection), protective covers, and items (windshields, etc) with protective covers installed.

3. It is fairly easy to make an exterior cover that fits over an aircraft windshield or canopy. Many have done this, to include individuals as do-it-yourself covers, private companies, and DoD organizations (Wings, depots, laboratories, etc). The difficult part is developing a practical cover that does not cause significant harm to transparencies. Even when an exterior cover is a partial or complete success when used in some environments or situations, it may be completely unsuitable for use in other environments/situations.

4. How does one develop an exterior cover design?

a. One way would be to determine all the situations and environments where one would want to use exterior covers, consider all of the exposures and user needs that are involved, and create a comprehensive list. This list would then evolve into the list of requirements the exterior cover would be asked to meet. Candidate cover designs would then be created, prototypes made, and evaluations conducted.

(1) A similar conceptual approach with air vehicles would produce a list of missions that included strategic bomber, air defense fighter, close air support, reconnaissance, transport cargo and passengers for thousands of miles, refuel other aircraft, land and take off from small clearings, and hover while one or more individuals ascends or descends via ropes/cables.

(2) If this philosophical approach produced exterior cover designs that worked well, Great!

b. If the above approach did not produce exterior covers that met all requirements, the next step should NOT be to give up. Instead, a little re-thinking of the situation would be in order. Consider the list of missions of air vehicles. No air vehicle performs all of those missions, yet there are many aircraft and helicopter designs which are considered successful.

(1) All of the requirements for exterior covers may be valid for one or more exposures and situations, but all of the requirements would not be applicable at the same time. For example, freezing rain would not be combined with solar heating and blowing sand, and chemical warfare exposures would not be encountered in peacetime or in CONUS during a conflict.

(2) Perhaps a better, more general approach should be to consider which exposures/conditions would occur in the same climate and/or situation. Develop cover designs that would work well and meet all requirements which would apply to that climate/situation.

(a) For example, let us presume winter weather was causing severe transparency problems for a particular aircraft/helicopter, a cover design was created that solved those problems, and no other approach provided the needed winter protection. The decision might be made to purchase that cover for use in cold climates, even if it did not solve problems experienced in hot weather. If the cover was not suitable for use in hot, windy climates that fact should be recognized. However, it should not be a reason for rejecting the cover for winter use.

(b) This philosophical approach is actually used for many items. Tests have conclusively shown that personnel can be injured/harmed when they wear heavy parkas in Arizona in August, or short sleeve shirts in Alaska in January. Despite this, parkas and short sleeve uniform shirts have been purchased and used anyway.

(c) The above "cold climate" cover would compete with other options. If another cover design solved both cold weather and hot weather problems (and all other considerations were equal), then the multi-use cover should be chosen instead. The best multi-use cover would be one that worked in every climate and situation...the approach given above in paragraph 4a.

5. Before one sets out to create exterior covers, it would be wise to learn how users would use covers, what climates and conditions the covers would be used in, what materials and design concepts for exterior covers could be used, what covers have been used in the past, what good and bad things could result (or have resulted) from cover use, and other approaches/options to solve problems that could be used instead of or in addition to exterior covers.

a. Some examples are provided in Attachment 1 of exterior cover designs which illustrate problems and challenges that exterior cover development efforts must consider.

b. One alternative approach is to use reflective cockpit solar shields that fit inside the cockpit and prevent hot cockpit problems. While there is an obvious overlap in function between interior solar shields and exterior covers, it is a mistake to think one must select one approach and reject the other.

(1) They have different advantages and capabilities, and there are some situations where one or the other would be the clearly superior product. For example, reflective cockpit solar shields are much lighter than semi-rigid exterior covers, and fit in a much smaller space. They can be stowed on helicopters and on some fighter aircraft and are available for use at their destination, which is not possible with semi-rigid exterior cover designs. Since both products can be valuable (assuming a good exterior cover is available), and since both are fairly inexpensive, some operational units might decide to use them both.

(2) An analogy might be to try and decide which vehicle was superior-a pickup truck or a motorcycle. Both vehicles have superior capabilities for some missions/taskings. There is no requirement that individuals who like pickup trucks must also dislike motorcycles.

6. Climates and situations that might call for use of exterior covers would include the following:

a. Hot and dry weather, hot and wet/humid weather, hot summer weather, cold winter weather, nighttime condensation, large temperature cycles (daytime high to nighttime low), severe hail, snow, frost, freezing rain, rain, air pollution, blowing sand, wind, inside buildings for maintenance at Wing or Depot level, inside humid shelters that drip water from ceilings, daily install & remove use for aircraft on flying schedule, long term installations on stored aircraft or "hanger queens," aircraft wash rack, painting of aircraft, missions requiring rapid preflight and launch of non-alert aircraft, aircraft on alert, flightline maintenance requiring several hours in cockpit in either very cold or very hot weather, aircrew in fighter cockpit in hot weather (aircraft not running) waiting for up to an hour for word to launch, normal peacetime operations, wartime situations where camouflage of aircraft/helicopter was desired, and wartime with possible chemical warfare exposures.

b. There are many other wartime exposures that could also be encountered, but we will NOT make any special attempts to provide protection against those threats. Threats we will not try to counter (but might accidentally provide some protection against) include ballistic projectiles (bullets and shrapnel), nuclear weapon effects, and lasers.

c. Other requirements for the covers would include ease of use, as foolproof as possible (example: if installed upside down, have distinctive colors or other markings visible which make it obvious from far away that the cover was incorrectly installed), appropriate information and/or markings that identify climates/situations where the covers should and should not be used (could be on the cover, or in a technical order), all the factors that would make the cover affordable (reasonable initial cost, low life cycle cost, durable, repairable, and maintainable), factors that make the cover obtainable/available (able to purchase/acquire using normal supply channels, National Stock Numbers assigned, etc), cause no damage or injury during installation, removal, or while installed, cause no damage or injury when improperly used (if improper use is possible), and provide adequate protection against all applicable environmental exposures.

7. The examples in Attachment 1 include some designs that work well (or would probably work well) in some climates/situations, but were used in climates where they caused more problems than they solved. Some of the "lessons learned" include avoid having thin heat-absorbing covers in contact with transparency outer surfaces during hot weather; avoid the combination of three exposures: sand/dirt on transparency, cover touching surface, and cover moving due to wind; and do not use highly reflective covers if there is a wartime situation and you are trying to hide from the bad guys.

a. It would not be correct to conclude that exterior covers should not be pursued because they cause problems. Instead, one should realize that there are many requirements that must be satisfied by an exterior cover in each particular environment/situation.

b. While it would be nice if a single cover satisfied all requirements for all environments, a cover should be considered a viable candidate for procurement and use if it satisfied all requirements in one (or more) specific environment(s) or situation(s).

(1) Users would have to understand that the covers should be used in some environments/situations, and should NOT be used in others.

(2) A mistake to be avoided is settling for a compromise cover that met some of the essential requirements in every environment/situation, but failed to meet all essential requirements in any particular environment/situation. Going back to the summer/winter clothing example used earlier: Short sleeve parkas and fur lined short sleeve shirts would be compromises that would partially work in both environmental extremes, but would be unsatisfactory in almost every desired application.

8. OK, that's enough of "doom and gloom" on what can and has gone wrong, and why good exterior cover designs are difficult (impossibly difficult?) to develop. How about something

encouraging, some ideas on what could be pursued that might be feasible. What could be done to create a successful design?

9. Good news! There is ALREADY an exterior cover design for the F-15 windshield that is a big success, a design that meets all requirements in all environments and situations. Utopia!!! Folks at McClellan AFB (SM-ALC/TIEC) made the covers.

a. The first prototype F-15 windshield covers got rave early reviews from users, but some easily solved glitches eventually became apparent. The current (revised) covers have been undergoing operational evaluations (still in progress) and have received very high marks from users. The semi-rigid covers weigh about 25 pounds and are durable, effective, and easy to install and remove even in windy conditions. They protect the windshield against almost anything that would drop on, form on, or radiate on the transparency. This includes exposures such as summer sun, winter frost, snow, freezing rain, blowing sand, hail, rain, acid rain, air pollution, bird droppings, dropped objects/maintenance errors, wash rack chemicals, and chemical warfare exposures.

b. There appears to be no reason that similar semi-rigid windshield covers would not also be successful with most or all other fighter, attack, and trainer aircraft. If there is an operable canopy behind the windshield, the above conceptual approach for exterior windshield covers should work well.

c. Reason for needing an operable canopy: The individual installing (or removing) the cover stands inside the cockpit, which gives him a safe and stable place to stand. He/she can brace himself against wind gusts, and safely install/remove the cover. There also needs to be somewhere to hook/secure the cover in place. The rear arch of the F-15 windshield works great for that function, and we anticipate the rear arch of other aircraft windshields would also perform this function. If there is no operable canopy, then it's back to the drawing boards to find a way to anchor it/position it in the needed location.

d. The cover is made in a mold. Start-up costs include creating the mold, which would be in the 10K to 20K range. It also takes some time to create the mold. Neither cost nor time are excessive, but they are more than zero. Candidate users that decide they need an exterior cover for a particular aircraft should not expect covers to be designed, created, and delivered immediately. Fast response is always possible to fill urgent needs, but candidate users who allow time for the cover to be created are more likely to have the covers available when needed.

e. Does the above success mean that nothing else needs doing? Not by a long shot.

(1) Canopies still need to be considered. Some prototype F-15 canopy covers are being made using the above approach, and those will probably be good covers to protect transparencies from

mishaps/damage during indoor maintenance at Wings or Depots. The covers will be much larger than the windshield covers, and personnel must stand on ladders on the outside of the aircraft to install/remove the covers. It could be hazardous to people and transparencies to try and install/remove these canopy covers on the flight line in windy conditions.

(2) Some aircraft such as the B-1 do not have canopies or other operable cockpit windows, and windshield covers could not be installed by someone in the cockpit. A semi-rigid cover would have to be placed over the windshield from a maintenance stand. A very good attaching/securing/positioning system would have to be devised, probably relying heavily on long straps run to various attachment points on the bottom side of the aircraft (nose landing gear and access ladder). There would have to be some good design work done, to include devising ways for one person to safely remove the cover in windy conditions.

(3) Some aircraft such as the C-130 have operable side windows in the cockpit. Perhaps an approach could be devised where an exterior semi-rigid cover would be pulled or placed into position using either straps/ropes from the two side windows, or maintenance stands on the outside of the aircraft. The frame of the operable window looks like a convenient and tempting place to help tie down/secure the cover in position, until one thinks of the possible results from putting large loads on the window frame. The window frame was not designed for those loads. It would be wise to rely upon more sturdy attach points to tie down the exterior cover.

(4) There are some situations where the aircraft or helicopter will take off from one location and land in another. It would be highly desirable to have protective covers that were light enough and small enough to be stowed somewhere in the aircraft so they would be available for use at the new location. However, covers that could not be stowed on the aircraft/helicopter would still be viable designs, just not utopian designs.

10. The semi-rigid exterior cover concept works very well indeed for windshields on aircraft with operable canopies. Are there other exterior cover concepts that could provide protection against hostile exposures in applicable environments? Yes, definitely. Some concepts have already been tested, while other concepts are still on the drawing board. Others could undoubtedly come up with other ideas that might work well.

a. At the start of Desert Shield there was concern that our forces might be subjected to attack with chemical weapons. Problems due to heat and blowing sand were on the news on a daily basis. Several operational units created and used do-it-yourself exterior covers, but found that they caused transparency damage. WL/FIVR initiated several efforts to address these concerns. One of these efforts produced an exterior cover for the F-16.

b. A prototype exterior cover for the F-16 was created to provide protection against possible Desert Shield exposures, to include wind, blowing sand, solar heating, and chemical warfare exposures.

(1) The cover weighs about 25 pounds, does not touch the transparency, and would not prevent cockpit access while installed. The design features a framework that folds into a "U"-shaped arch. It is placed on the canopy by two people while in the folded/stowed position. It is then extended/unfolded forward and back to cover the entire canopy. Installation time is about 3 minutes.

(2) The cover could be safely removed in windy conditions by removing the outer cloth from the frame (attached with velcro to the frame), then removing the frame. It could be removed and thrown aside in seconds during an alert-type response situation (with some chance of damaging the cover).

(3) The cover design was developed when there was great concern that Desert Shield units could be attacked with chemical weapons, delivered via either aircraft or SCUD missile warheads. Heat and blowing sand were other important design drivers.

(a) In a chemical warfare scenario, the covers could literally have been life-savers. Protecting the transparency from "sand-blasting" of wind-blown sand was another large payoff.

(b) For normal, day-to-day use in peacetime we feel the cover may be more trouble than it's worth. Two of the largest advantages are chemical warfare & blowing sand protection. One is not applicable during peacetime, and the other is applicable in only a small percentage of operating locations. Subtract those from the "advantages" list and a compilation of advantages vs disadvantages would lead most to conclude that the covers were not a good choice for peacetime use....at least for use on the flightline. (They might work well as indoor maintenance covers.)

(c) When chem war and blowing sand are not considerations other options become more appealing for flightline use.

(1) Leaving canopies slightly open to reduce cockpit heating is feasible if blowing sand is not a problem.

(2) Internal cockpit solar shields (discussed in another paper) work well in preventing hot cockpits, and can be used with transparencies that do not have gold solar films on the inside surface. The F-16 is the only aircraft with a gold film in a location that could be marred by solar shields. Most DoD F-16 units have gold film transparencies now and should not use the solar shields (sorry!). None of the foreign services with the F-16 have gold solar film transparencies. Several countries in the Desert Shield area have tested and adopted these reflective shields for the F-16 and several other types of aircraft and helicopter.

11. How about some other concepts/approaches that could either solve one or more problems by itself, or be a specialized feature/layer in a multi-ply (or multi-function) cover concept?

12. We need to develop a light-weight cover to effectively deal with conditions that include wind, large hail, and solar heating for peacetime, plus possible chemical warfare exposures and the need for camouflage in combat situations. We also need some effective, realistic, yet inexpensive testing techniques that will represent severe hail storms.

13. Let's expand our mental horizons and look at other industries for technologies, products, concepts, and materials. Let us seek items and ideas that could be adapted for and adopted into an exterior cover effort, or that provoke ideas that could be used. We will also have to have some way of testing the resulting covers versus severe hail storms to determine if adequate protection is provided, so testing equipment/procedures/ideas must also be sought.

14. Life is too short for any individual or organization to learn everything he/she/it needs to know via personal experiences, mistakes, and discoveries. It is much better to make use of the discoveries and mistakes made by others, and build from there. Some of the areas that have contributed ideas, products, concepts, information, and/or prompted ideas include the following:

a. Protective armor vs ballistic impact; more specifically, the principle of spreading impact energy over as wide an area as possible to dissipate and absorb the energy and defeat the threat.

b. Bullet-proof glass windows for cars that met all defined requirements, yet allowed the assassins to shoot the vehicle's occupants.

(1) The requirement was to stop a bullet. The glass window did that, and (as expected) was damaged. More bullets from rapid-fire weapon, more damage, and the window collapsed.

(2) Perhaps the assassin should be reprimanded for using a machine gun type weapon rather than a single bullet as per the assumptions in requirements. Standard test methods use one bullet, and testing requirements rely upon standard test methods. An alternative philosophical approach would be to rethink the requirements, even if this results in developing and using something other than standard test methods.

(3) We suspect that severe hail storms will not obey instructions to only deliver one hail stone (as per standard, approved test methods), so we plan to adjust requirements to match hail storm conditions and devise appropriate testing procedures. The cover will be designed for (and tested against) multiple rapid-fire impacts...Mother Nature's version of the machine gun.

c. Do-it-yourself window solar heat collectors; ideas on how to intercept solar heat in an outer air space and continually dump the heated air so there is no temperature build-up. Cold air enters at the bottom, is heated by solar radiation, the warmed air rises, and exits out the top of the collector as slightly warmer air.

d. Dirt repelling concepts for the inner surface of the material that will touch the transparency. These concepts include no-stick Teflon fry pans, smooth and soft fabric surfaces (silk, etc), stain and dirt resistant treatments for fabrics, and dust repelling properties on furniture from using new furniture polishes. The ultimate product: dirt repellent "Ever-Clean" clothing (in a sci fi story) that never needs washing; just shake and it's clean. (If someone creates an "Ever-Clean" product, please give me a call.)

e. Waterproofing concepts for shedding rain, and small design detail (weep hole) in some storm windows. Avoid materials that will absorb water or deteriorate when exposed to conditions such as water. Recognize that water will get inside the cover eventually (as rain or condensation), and provide means for water to drain out if the air inlet holes do not prevent water accumulation.

f. Repairability, maintainability, redundancy, and fail-safe concepts. The cover will become damaged eventually. The damaged cover should be repairable, and it should continue to provide protection until it is repaired.

(1) If the cover had two separate inflatable layers, and each layer had replaceable subdivisions/strips, then damage to one part of one layer would degrade but not eliminate the cover's ability to protect the transparency.

(2) The damaged subsection could be repaired or replaced rather than replace the whole cover.

g. Judo; more specifically, the concept of using your opponent's strength, momentum, etc to your own advantage. Perhaps a cover design could use Judo concepts against wind also.

h. One way valves, especially flapper valves in furnace exhaust pipes.

i. Air filled mattresses for floating in swimming pools, to include design details such as long parallel tubes and a simple valve with a removable snap-in plug to close it.

j. Air bags in cars for crash protection.

k. Self-inflating sleeping mattresses for hikers and climbers to carry in their back packs.

l. Trampolines.

m. Safety nets for high wire circus performers.

n. Driving nets for golf balls.

o. Static cling films used on food leftovers in refrigerators.

p. All information on hail storms we could find, to include size and weight of actual hail stones, and tests with artificial hail stones to determine impact velocities that would be involved in severe hail stones. Worst case situations: golf ball to baseball size hail, impact velocities approaching 100 MPH. If a cover (that meets all other requirements) would protect versus multiple strikes of 100 MPH baseballs, we'll have really gotten something!

q. Form-fitting clothing, to include "W'Underwear" (underwear for horses), people underwear (with elastic), and shorts worn by bicycle enthusiasts.

r. Baseball; more specifically, baseball pitchers, pitching machines, and hitting balls with baseball bats.

s. Golf; especially hitting golf balls with drivers through 5 irons.

t. Ice cube trays, to include custom designed trays to produce optimum hail stone sizes and shapes.

u. The KISS principle (keep it simple, Sammy) as far as the user/installer is concerned. It's OK to add additional features if they increase reliability and ease of use. It's not OK if the user needs any special knowledge or training to use the cover. One does not have to be a mechanic or automotive engineer to use a car, and one should not have to know details about the cover design or why it is effective to be able to use it.

v. Ice machines in hotels, and 10 pound bags of ice cubes sold in stores.

w. Tennis balls, filled with water and frozen.

x. Jello and gelatin; no particular flavors. More specifically; tennis balls filled with gelatin.

y. Jai Alai; especially the large curved basket strapped to the hand and used to catch and throw balls.

z. Lacrosse sticks (we tried them, but they didn't work well for our effort).

aa. Sling shots. Huge designs were drafted that included long, wide elastic straps/webbing and a large framework that the sling shot user would sit/lie upon. (Half a year later, saw TV ads for a comparable product (minus the framework) intended for use at

the beach. The super sized sling shot launches water balloons 200 yards. Why didn't I think of THAT use for the concept, which promises to make money for the inventor? Oh well, a day late...)

bb. Radar guns used to measure speed of pitches in baseball games. Suitable model found for a little over $1000.00, will measure velocity of golf ball size objects.

cc. Very large fans and wind machines used in movies (to create high winds usually present in severe thunderstorms and hail storms).

dd. Small vacuum pumps, or evacuated containers that could be used as a vacuum source in the absence of active vacuum pumps.

ee. Sitting under an apple tree and getting hit on the head with an apple.

ff. Mathematical formulas for calculating impact velocity "x" for an object dropped from "y" height.

gg. Tallest building on Wright-Patterson AFB, which also has balconies and railings on the top floors. (Perhaps our reputation will get even better if word spreads that we are "able to reach (the top of) tall buildings in a single climb.")

hh. Commercial advertising campaigns. Once we have a successful design, we will have to let others know about it. TV, radio, newspapers, magazines, etc have many advertisements to make customers remember products and want to get them. Perhaps we should follow those examples and use some catchy slogans and advertising gimmicks for the covers. Let's see...if the design includes some air inflation features for impact resistance, perhaps phrases/ideas could be used such as "Not just another bag of wind," "Cover up your windows with our old wind bag, and smile, smile, smile...," a cartoon drawing of a happy airplane wearing a cover and singing "Hail, Hail, you can't get me, What the heck do I care,...", the cartoon aircraft bounding off the ground while singing "I love what you do for me, Hail cover!", and "Brought to you by the blow-hards at Wright-Patt." Another idea: To emphasize "ease of use," show a dog tugging on one of the cover straps with the caption "Very easy to install and remove; even Fido can do it!" Hmmm...needs work. Perhaps we'll forget the slogans and theme songs for now and concentrate on the technical part of the effort.

15. Let us focus primarily upon the wind part of the environment for the moment. Let us also remember that sand on the transparency plus movement of the cover in contact with the transparency produces scratches. We will also keep in mind that solar heating and hail are other concerns. Some ideas for addressing the problems include the following:

a. One conceptual approach is to not touch the transparency with the cover; no contact means no scratches. This is the

approach taken with both the semi-rigid F-15 windshield cover and the folding F-16 canopy cover. This is also the approach used with one of the most effective cover concepts yet devised...a building. Buildings have a few disadvantages (cost; hard to take along during deployments), so there is still a need for external covers.

b. Another conceptual approach is to allow contact, recognize that there will be some movement, and somehow guarantee there will never be any dirt present between the cover and the transparency. A soft, high-tech, and expensive exterior cover used on numerous RAAF aircraft in Desert Shield tried to use this approach, and we have heard that about 100 canopies became unserviceable.

c. Another conceptual approach would be to develop a reasonable cover, and then state it can only be used with transparency surfaces that are almost immune to scratches. If the cover was usable with glass transparencies, the cover concept would be a worthwhile (but disappointing) result from a developmental effort. However, if transparencies with special coatings would have to be retrofit to the fleet before the cover could be used, the cover design would have to be considered a failure. If some aircraft adopted such coatings for other reasons, the cover might eventually change from a failure to a limited success.

d. Another conceptual approach is to allow contact, but eliminate all movement. If successful, this conceptual approach could be widely applicable and could have very large payoffs. What design concepts might work?

(1) Static cling would be one property that might be utilized, at least as a partial solution for some applications and situations. Several variations on the theme are described in Atch 2, to include using a static cling film/layer by itself (a do-it-yourself approach for operational units) and using the static cling film/layer in conjunction with another exterior cover. Some areas where some testing should be done are also discussed.

(2) A somewhat messy but simple do-it-yourself approach would be to only focus on the wind blown sand problem and settle for protecting the outer surface from that exposure.

(a) Rather than use a transparent static cling film, get the same protective effect with a transparent gel-like coating that was safe to use. (A highly reflective gel coating might also work.) The coating should be easily removed with water, must not attack or otherwise harm the transparency, and must not make it easier for other environmental exposures to damage the transparency.

(b) Perhaps covering the gel with a thin film (the static cling film concept again) would protect the gel from rain, prevent it from becoming impregnated with dirt/dust, and/or keep the gel from absorbing harmful chemicals. A dirty film in summer

would become a hot film, and transparency damage would probably occur.

(c) This would be cheap (assuming the gel was cheap) and simple, but would be messy and labor intensive.

(1) If the situation were bad enough (frequent sandstorms) units might wish to use this approach if no other feasible approach were available.

(2) Some of the aircraft/helicopter design details that would have to be considered include what items, equipment, and areas the gel could get into when washed off. If the gel would puddle/collect somewhere and then hold dirt and/or chemicals, something bad might happen even if the gel itself was not attacking or harming the item.

(3) Perhaps a multi-layer design could be successfully developed that relied upon outer layers to somehow intercept and overcome the wind, so that wind forces would not be transferred to the inner surface and the inner surface would not move in windy conditions. We have some ideas/concepts on the drawing board, but we expect to perform several iterations of "make one; try it" before we determine the feasibility of this conceptual approach.

(a) Preventing wind-induced movement of the inner surface will be a big challenge. The wind forces on the edges and outside surface of the cover must somehow be defeated so that they do not cause the inside surface to move. This can be done, if we are clever enough.

(1) Consider the concept of having an inner cover that covers the transparency, and is securely held down with straps, etc.

(a) Elastic edges or even some elastic properties for the entire inner cover would probably be involved with this "underwear" for the transparency, producing a snug, form-hugging fit.

(b) Over the inner cover would be an outer cover (which actually would be a different layer of the overall cover). The outer cover/layer would extend past the inner cover in all directions by perhaps one foot. The outer cover would also be held down with straps, and there would probably be some elastic used at the edges. Wind that got under the outer cover would push the outer cover upwards, but would also push the inner cover downwards. If the inner cover still tended to move too much, another cover/layer could be between the outer and inner cover.

(2) A variation on the theme that might be feasible in some cases would be to have an inner cover that would be air-tight.

(a) The inner cover would probably have to have a plastic film layer over the entire cover to prevent air leaks through the cover itself. Some cover edge design features would also be needed to form an air-tight seal on the perimeter. A small vacuum pump could be used to draw air from underneath the cover. The cover would then be pressed tightly against the transparency surface by ambient air pressure.

(b) This concept would probably NOT be a preferred design in most cases. It would be difficult to get a good seal around the entire perimeter of the cover. The cover might have to use the perimeter of the transparency as the surface it would seal against rather than extend onto the metal panels adjacent to the transparency. Perhaps more importantly, the additional complications involved with either a vacuum pump or a vacuum bottle (which would need frequent replacement) would make the design much less user friendly.

(3) Perhaps the above approach (without vacuum) would be able to counteract the bad effects that wind could cause. However, we can probably do better than that.

(4) Rather than view the wind as just an enemy, try viewing the wind as something that could be used to advantage. Use the wind to help defeat the wind (Judo philosophy).

(a) Wind will get under the outer cover layer, so why not use it? Consider air bags, inflated air mattresses for swimming pools, self-inflating sleeping mattresses (contain compressible foam that springs back when pressure released), and one-way "flapper" valves. The air-filled bags/mattresses all provide impact resistant with low weight, the characteristics that are needed for an effective hail impact protective cover.

(b) As a means of harnessing and using wind, use one (or more) self-inflating layers.

(1) Each layer would be subdivided into sections/strips so that failure of one section would not cause the entire layer to deflate.

(2) Each section would have thin, strong film for the top, bottom, and sides. One continuous tube made from a piece of film might be the best approach.

(3) At each end there would be a simple "flapper valve". For example, visualize a small, square, vertical piece of fairly rigid material with a hole in the middle. On the top inside surface of the square piece, attach a flap of material that hangs down and covers the hole.

(4) When the wind blows in one end, the flap of material is blown back and the air enters the section of the cover. The same wind pushes the flap against the hole on the

other end of the section. The air cannot leave, so it inflates the section of the cover. The same process occurs in each section of the cover, inflating the entire layer. If there are two such layers in the cover, the same process occurs with the second layer.

(5) Note the effect of having one section with a large rip/puncture. The section would not remain inflated when the wind stopped. However, the sides of adjacent sections would be glued/attached together. The top of the damaged section would sag when the wind was not blowing, but the inflated sections on each side of it would hold the edges up. The top of the damaged section would droop/sag from side to side, but would still have an air gap and still offer some hail impact protection.

(6) The sections of cover will remain inflated when the wind stops only if the flaps remain pushed against the holes. The flaps will stay in position because they fall there via gravity, they are slightly spring-loaded to the closed position, and/or because there is a slight pressurization inside the inflated cover.

(a) We are not pressurizing a constant small volume (example: a diver's air tank). Instead, we are inflating the cover and creating a larger volume of air against a very slight resistance (example: a toy balloon). Elastic properties either in the film itself (probably degrade strength and durability), or in gentle elastic attachments for the top portion of each section would provide the resistance. The elastic would maintain a constant, very slight pressure on the inflated cover to keep the flapper valves closed.

(b) The length of time the cover will remain inflated will be a function of how air-tight the flapper valves are in the closed position (assuming the film is intact and air-tight in the rest of the section of cover). Perhaps the flapper design should not be a flat piece with a hole and a moveable flapper, since the flat piece might not stay upright. Instead, perhaps the flapper valve design should be a small cube with holes on two cube faces. A cube would stay properly oriented. Two moveable flappers could be used in series, one against each hole in the cube. Both flappers would have to fail before the section would fail to remain inflated.

(c) When the cover was removed from the aircraft, it would need to be folded or rolled into a compact shape for easy storage. The cover would need to be deflated, and there are several options for accomplishing this. The flapper valves could be defeated by inserting something to keep them open, but this would be labor intensive. Simple valves such as those used in air mattresses for swimming pools could be installed in each section, and when the plug was pulled from the valve the sections would deflate. Better, but still labor intensive. Instead, why not connect all the relief valves to one tube, and only have to "pull the plug" on one master valve. Since each

opened valve would have to be closed again the next time the cover was used, time saved during deflation would also be time saved during installation.

(c) One additional wind-powered feature that could be considered would be the addition of a wind-powered vacuum pump attached to the inner cover. The vacuum pump could be positioned in many different locations. Example: Attach it to a wide part of a cover attaching strap. As discussed previously, the vacuum pump approach would probably not be feasible for most aircraft/helicopters. However, a wind-powered vacuum pump would be a way of eliminating the need for either power or frequent maintenance attention for a vacuum bottle of an installed cover.

(d) Self-inflating sleeping mattresses (for hikers and climbers) are very light weight. They have foam that easily compresses into a small volume, but when the mattress is rolled out the foam expands, pulling air in through an open valve. The valve is then closed to trap the air, and the inflated mattress is ready for use.

(1) Using a large, foam-filled mattress as a hail/solar cover would probably work, but it might be too bulky. Then again, users might find it very satisfactory.

(2) Instead of a continuous piece of foam, use small pieces of foam periodically in the cover to keep the top and bottom edges of the inflatable sections spaced apart. They could also help keep the edges spaced apart so that wind would reach the flapper valves more easily.

(a) It is possible but unlikely that a severe hail storm would occur with no wind before or during the storm. In that situation, the foam would insure there were some air gaps in the cover. The deflated cover would provide some impact protection. The larger the foam pieces, the farther apart the cover layers would be, and the more protection the deflated cover would provide. Such a design would also provide some impact protection against dropped objects in an indoor maintenance environment.

(b) Taken to an extreme, the cover with foam piece spacers would start to use the energy absorbing principles found in trampolines, safety nets, and golf driving nets rather than air bags. This conceptual approach might work just fine, and it would make possible the deletion of the flapper valves should those prove to be an unreliable or costly component in the cover design. It would also probably make it faster and easier to fold, roll, and store the cover since there would not be the need for deflating the cover using one or more small valves.

16. Is a practical design possible that could be created and purchased at reasonable cost? ABSOLUTELY; we are certain it is. We believe there are several designs that would work well. Will we

have the resources (money, time, people, ideas) to properly pursue creating such a design? The effort won't cost a great deal, and we have high hopes that we will get the resources we need. Are we smart enough to create a successful design, given adequate resources? We think so, but time will tell.

17. So much for cover design ideas. There needs to be some testing techniques that can determine if a promising cover provides adequate protection against solar heating, rain, and severe hail.

a. Preliminary coupon level tests could determine which materials were suitable and safe to use as inner surfaces for covers which would contact the transparencies. Two of the most critical characteristics to verify would be that the material would not hold dirt and cause scratches, and the material does not have plasticizers that would "outgas" and cause crazing of acrylic.

b. It's not too hard to come up with realistic tests to evaluate covers exposed to solar heating and rain...operational evaluations using actual aircraft and actual maintenance personnel would be the most realistic testing technique.

c. Hail impact testing becomes a bit more difficult. One needs to consider (1) the testing equipment used to launch/fire the simulated hail stone at the target, (2) what projectiles would be suitable for impacting unprotected items, and (3) what projectiles would be suitable for impacting protective covers.

d. There are already several standard testing procedures published by the American Society for Testing and Materials (ASTM), but these procedures are intended for use when one is evaluating the effects on uncovered materials/objects when the items are hit with hail.

(1) The simulated hail stone is made by freezing water.

(2) The standard tests and testing equipment involve shooting one hail stone against the item. The testing equipment includes a single-shot gun which is not intended for multiple shots in rapid-fire fashion.

(3) The single shot approach is perfectly adequate for determining vulnerability of and damage to an unprotected item. For example, if hail impact caused dents in sheet metal, it would make little or no difference if multiple impacts occurred seconds apart, or hours apart.

(4) The single shot testing approach is NOT a realistic way to determine if a cover will provide adequate protection. For example, a properly designed inflated cover should have no trouble with stopping the first hail stone. However, will the first, second, and third hailstones cause the cover to partially deflate? If so, will the cover still provide protection against hail stone #20? #30? Severe hail storms do not last long, but they typically

last several minutes. The hail storm will subject the cover to multiple impacts in rapid succession.

e. The simulated hail stone projectile should have the same size, weight, and impact velocity as an actual hail stone.

(1) If the simulated hail stone will impact against the unprotected item, the projectile used should also have the same physical characteristics (hardness, strength, etc) as actual hail stones.

(a) Frozen water seems to be the best candidate as a simulated hail stone projectile. Different mold (ice cube tray) sizes and shapes can be used to produce whatever sizes and shapes are desired. By varying the length of time in the freezer, one can make hail stones with liquid centers. That is a possible real world hail stone, but it is not apparent what additional information would be gained by using both completely frozen and partially frozen hail stones in the testing program. We need only concern ourselves with hail stones that could do damage, and can therefore disregard small, soft hail stones.

(b) Test equipment and techniques used to accelerate the projectile to desired velocities will have to be appropriate. Equipment or techniques that shattered the simulated hail stones during the acceleration phase would not be appropriate. Techniques/equipment for launching/firing the frozen water projectiles must result in an intact hail stone impacting the target at 100 MPH. Some options for accomplishing this would include:

(a) The equipment specified in the ASTM test methods. Modifying/redesigning the equipment for faster reloading and firing could be considered. Using multiple launchers against the same target would be another way to get multiple impacts in a short length of time. However, the equipment is expensive.

(b) Drop or throw hail stones from the top of a tall building. The initial hail stone velocity could be zero (if dropped) or higher (if thrown), and gravity would increase the velocity during the descent. This would be a low cost and realistic testing approach, provided some conditions could be met. These conditions would include:

(1) The intended target zone must either have no items in the vicinity that need protecting (windows, cars, pedestrians, etc), or adequate protection must be provided to those items. The possibility of hail stones impacting the side of the building during their descent must also be considered.

(2) It must be acceptable to use large numbers of hail stones in a steady stream to produce multiple impacts in rapid succession and to allow for dispersion/inaccuracies. Most of the hail stones will miss the intended target.

(3) There must be no possibility that personnel conducting the test (such as myself) will accompany the hail stones during their descent. Such an event would be extremely harmful to the overall program even if the cover was not damaged.

(c) Giant sling-shots that would provide steady acceleration and not shatter the hail stones. To get rapid-fire impacts, use several sling-shots at the same time.

(d) The long, curved baskets or gloves used by Jai Alai players to catch and throw balls. For multiple rapid-fire impacts, have several baskets and several individuals using them.

(e) Perhaps some specially designed cross-bow could be created and used. The principle would be similar to that of the sling-shot (using bent wood or bent composite instead of stretched elastic), but the acceleration would be much higher because it was over a shorter distance. If the acceleration was too high with a traditionally sized cross-bow, a giant cross-bow could be created that used a longer distance to accelerate the simulated hail stone.

(f) Another variation on the theme that could be tried would be a giant size bow and arrow. The arrow could be tethered so that it was stopped before it impacted the target. The head of the arrow could consist of a cup into which the hail stone was placed.

(g) Another variation on the sling-shot, cross-bow, and bow and arrow concepts are to create a large, rigid rack or framework, and mount multiple copies of the hail stone launchers side by side. For example, position them at 6 inch intervals. Each launcher would be loaded and the string/elastic pulled back and engaged into a release mechanism. One individual could then go down the line, releasing each launcher in a rapid-fire manner.

(h) The number of different launching devices that could be created is large, and each could be constructed in a variety of ways. Perhaps a disciple of Mr Ruben L. Goldberg would agree to be a consultant during the design process to help insure the final result was memorable and noteworthy.

(2) If the simulated hail stone will impact against the protective cover, other simulated hail stone options become available. Size, weight, and impact velocity must still match the real world hail stones.

(a) The hail stone impact event will now consist of the hail stone impacting the cover, the cover absorbing the impact energy and momentum, and the cover distributing/transferring the impact energy/momentum to the aircraft in a way that causes no damage. It is no longer important for the simulated hail stone to match the strength and hardness of real world hail stones. It would be wise to use real hail stones during a final qualification

test, but real hail stones would not be essential during developmental tests.

(b) What are some simulated hail stones that could now become candidates for testing?

(1) Frozen water will always be a suitable material for simulated hail stones, but the use of frozen water places limitations on the type of equipment that can be used to launch/fire the projectile.

(2) Tennis balls filled with frozen water are one possibility. The tennis ball cover would keep the frozen water from shattering when the launching/firing equipment provides a very sudden acceleration to the simulated hail projectile.

(3) Tennis balls filled with congealed gelatin might work well, especially if there were sometimes delays in launching the projectile that would cause frozen water to partially melt.

(4) Golf ball size hail is a threat. Why not use an actual golf ball?

(5) Baseball size hail is the "worst-case" threat, so using a baseball would seem reasonable.

(3) When the simulated hail projectile is NOT ice, other options become available for projectile launchers.

(a) All of the equipment and techniques discussed above that could be used to launch frozen water could also be used with other projectiles. However, some additional precautions would be needed if the tall building approach was used for golf balls or baseballs. All hard surfaces (driveways, walkways, hard dirt, etc) would need to be covered with mattresses or similar to prevent the balls from bouncing long distances.

(b) Baseball bats could be used to launch baseballs. Place ball on "T" and hit it. Use several people, bats, and "T"s to get the rapid-fire effect. Caution: Be sure there is enough spacing between people to prevent injuries. Perhaps the entire testing setup should be enclosed in netting to ensure impact tests are not conducted with building windows, cars, or pedestrians passing by.

(c) Baseball pitching machines could be acquired and used. Multiple machines could be set close together, and create the rapid-fire effect.

(d) Baseball pitchers could be used who have major-league caliber fast balls of 90 MPH or faster. Possible problems: Would they do it for free in exchange for "thank you" letters to them and to their team? Will minor league pitchers with control

problems be sufficient, or would we try for major leaguers with multi-million dollar salaries? But...what happens if Rob Dibble or some other famous pitcher hurts his arm while participating in our testing effort? Can we take the chance of being mentioned on ESPN Sports Center and national news shows as the reason the Cincinnati Reds probably won't make the playoffs next year?

(e) Golf clubs work just fine as devices for launching golf balls. In addition, good golfers have fairly repeatable swings and the velocity of golf balls will not vary greatly from shot to shot. Desired velocity changes can be made by changing clubs. For example, perhaps a 5 iron will produce an 85 MPH velocity and a driver will produce 120 MPH. Rapid fire impacts could be produced by having two good golfers perform a special training routine: Line up a row of golf balls on tees, swing back and forth continuously, and hit a ball about every second. Large driving nets to keep mis-hits from getting away would probably be needed.

f. Whatever simulated hail stone are used, and whatever equipment and techniques are used to launch the simulated hail stone, some means of measuring the velocity of the projectile will have to be used.

(1) Standard test methods typically rely upon electronic devices that measure the time it takes for the projectile to pass between two points, and automatically calculates the velocity. Trip wires, X-rays, and other techniques for noticing the projectile are used. These techniques could be used with most of the launchers discussed above (except perhaps the tall building approach).

(2) An alternative would be to use a radar gun, such as those used on highways by police, and to measure the velocity of baseballs thrown by pitchers. The gun should be portable and must be capable of measuring the velocity of golf ball sized objects. Such an item was located which would cost less than $1200. As an additional benefit, the speed gun could have a second career after the cover effort was completed, being used as a baseball speed gun by the base baseball team. The radar gun could be used to measure and calibrate the velocity of any of the projectiles, launched by any of the equipment items.

(3) A less sophisticated but still usable technique would be to fire at some distant target (example: 200 feet away), use a projectile that has minimum air resistance, measure the time in flight for the projectile with a stop watch, assume velocity was constant (bad assumption), and calculate the velocity.

18. To summarize:

a. There are situations where exterior covers are needed to protect aircraft transparencies from hostile environmental exposures.

b. It's not easy to create a cover that will be of significant benefit, and one must beware of covers that cause significant harm.

c. Once a cover has been created, it must be evaluated. No standard test techniques now exist for evaluating the effects of severe hail storms.

d. We are confident we can create a worthwhile cover design, and we are also confident that we can devise and use realistic and effective testing techniques to evaluate covers versus severe hail storms.

e. Rather than just express confidence and philosophy, we have gone out on a limb by providing specific examples and details of what we have in mind for both cover designs and hail impact testing techniques. Such specific information is much more helpful and interesting to readers and candidate cover users. Some might become interested in and supportive of the effort, and others might want to check at a future date to see if our ideas and concepts actually worked. Perhaps someone who has both problems and resources might want to "press on" and make covers that use our ideas.

f. What will the future bring? Will our cover concepts and hail testing ideas prove to be feasible? Stay tuned for further developments.

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2 Atch
1. Ext cover lessons learned
2. Static cling cover concepts

a. Example #1: A private aircraft owner used a thin exterior windshield cover regularly on his aircraft, and was content with it. He then flew to a southern location on a summer day, and left his aircraft parked for a day with the usual exterior cover installed. When the cover was removed, the windshield was unusable due to severe crazing and had to be replaced before he could leave.

(1) Cause of problem: Unless they are perfect reflectors (nothing is), materials will absorb some solar energy and get warmer. If the material is thin, the inside surface temperature will be about the same as the outside surface temperature. If the item is in contact with the item underneath (i.e., the windshield) the outer surface of the item will also be about the same temperature.

(2) For aesthetic and other reasons, exterior covers would typically be neither highly reflective nor a bright white color. Even if they started as a white color, a used cover would become an off-white due to dirt and stains. Colors and patterns that do not show dirt or attractive scenes would tend to have appeal to private customers. Camouflage patterns would appeal to DoD customers, especially if the covers might be used during combat operations.

(3) The unrealistic "worst case" cover color (that hopefully nobody wants) would be a flat black. In hot sunny weather, black surfaces can reach temperatures between 200 and 250 degrees F. The temperature reached by an exterior cover would be determined largely by it's color, and covers with patterns or non-white colors might have temperatures in the 140 to 180 degree range.

(4) Crazing of acrylic is caused by combinations of factors, to include stress, chemicals, and elevated temperatures. The same stress and chemical exposures that have no noticeable effect at room temperatures can cause severe crazing at higher temperatures such as 150 degrees F.

(5) Lesson to be learned: Design the cover so that the exterior cover does not raise the temperature of the transparency's outer surface. There are a variety of design concepts that could accomplish this.

b. Example #2: It is important to avoid scratching the outer surface of the transparency. A soft material would seem to be a logical choice. A felt material was used with some helicopter covers, and a soft artificial fur was used with covers for various aircraft canopies. These covers caused scratched transparencies. Reasons:

(1) Consider the Desert Shield environment. High winds, blowing sand, and periods of rain were experienced.

(a) Dry felt will tend to pick up and hold sand and fine dirt. Felt would also tend to absorb water, and sand would stick to wet felt very well indeed. Wet felt plus sand might remind some of sandpaper. Place these covers against transparencies, add wind for the rubbing action, and severe scratching will result.

(b) Lesson learned: Don't use felt, especially if water or sand is part of the environment.

(c) Artificial fur is soft, but blown sand and dirt can penetrate the fur and be retained. Place the cover on an acrylic transparency and add wind. An excellent design might greatly reduce of even prevent additional sand/dirt from blowing under the cover and onto the transparency. However, sand/dirt that was already in the fur would be shaken out, and would have no place to go except onto the acrylic. Sand being rubbed back and forth by soft fur will cause scratches.

(2) Lesson to be learned: The combination of sand/dirt, wind, and rain is a very important (and very challenging) environmental exposure. If sand/dirt is present on the transparency (or is deposited on the transparency by the cover itself), then using a cover which is in contact with the transparency and moves with the wind will probably cause transparency damage.

c. Example #3. Let us go forward in time and use a hypothetical example from this future.

(1) An exterior cover design has been developed that works great in preventing hot cockpits, using a highly reflective outer surface. The design somehow manages to be utopian in virtually all other important categories, to include ease of use, preventing damage in wind/blowing sand, etc. The design is in use on all DoD aircraft and helicopters.

(2) A conflict breaks out and DoD units deploy to the area. USAF aircraft are parked at an air base fairly near the main conflict area, and US Army units with helicopters are positioned several miles from the conflict. The bad guys make a sudden, sneak attack on our forces during daylight, using small ground forces versus our helicopter units and aircraft versus our aircraft. The attack is very successful.

(3) After-action analysis determines that the reflective exterior covers made it easier for the bad guys to spot our aircraft and helicopters.

(4) Lesson to be learned: If you need to hide an aircraft or helicopter, then you should not cover it with something that makes it easy to see and identify.

STATIC CLING FILM CONCEPTS AS EXTERIOR COVERS

1. A thin static cling film could be applied as a sacrificial protective film layer. It would be removed and discarded before each flight.

a. Ways to apply the film to the transparency could include having the film on a roll, place it on one edge of the transparency, and unroll it across the surface. If there are two individuals, one could be on each end of a sheet of film, pull it tight in the air, and then lay it/place it on the transparency. Taping it down around the edges would probably be needed (not feasible with some aircraft surfaces).

b. To prevent heating of the transparency's outer surface when the film was used by itself, the film would either have to be a perfect reflector, or be completely transparent. Alternatively, the film could be applied to the transparency and then a separate reusable exterior cover installed.

2. The concept sounds simple, with little or nothing to worry about. However, it might be a tad premature to declare success, publish a final report documenting this achievement, and terminate further efforts in the area. Before pressing on with this "sure-fire" solution, perhaps the following items/topics should be considered.

a. In some conditions it might score poorly in the "ease of use" criteria. Installing it in windy conditions might be annoying. It might not stick at all in some situations, such as moist and/or cold surfaces.

b. Use of the approach would involve cleaning the transparency, then applying the static cling film. If the film is impervious to chemicals, whatever residual cleaning chemicals were on/in the transparency would be trapped there and unable to evaporate. Would this result in crazing developing sooner?

c. Perhaps the film would not be impervious to chemicals, and would allow them to pass through. This might be good news when one is worried about trapping chemicals against the transparency. It would not be good news if one was trying to use the film to protect the transparency from chemicals such as acid rain, paint overspray, or bird droppings.

d. Perhaps the film itself would deteriorate from natural and man-made environmental exposures. If the deterioration process resulted in loss of protection, that would be an annoyance. If the deterioration process instead caused damage to the transparency, it could be a very serious problem. For example, if chemicals attacked and partially dissolved the static cling film, the result might be film residue that would be difficult to remove. What

would the effects be of installing the film and leaving it installed for weeks at a time?

e. Could and should the static cling film be installed on a wet transparency, a cold transparency, or a dirty transparency?

f. Another worry: If the aircraft had a system such as hot air rain removal that was function checked by maintenance personnel, would the hot air on the film cause anything bad to happen?

3. Other cover design possibilities that would use variations on the static cling idea might include the following:

a. Static cling properties could be induced in various materials. One design variable would be the film thickness, but other possibilities would exist. For example, a thin three-ply cover might be worthwhile, consisting of a static cling film on the inside, a very thin cushioning and insulating layer over it, and a reflective mirror-like surface on the outside.

b. When the static cling film was used in conjunction with an outer reusable cover, it should be verified that the film would cling to ONLY the transparency, and would not cling to the cover. If it clung to the cover instead of the transparency, the required "no movement" feature would be eliminated.

c. A possible design variation would have the static cling layer against the transparency, the reusable exterior cover over it, and between the two would be a "super-slick" layer that would stick to neither one. The super-slick layer would prevent the cover's sidewards movements from producing any sidewards forces on the inner static cling layer. If the super-slick layer was reusable it would be an integral part of the cover. If it were a one-time use material it could either be applied over the static cling layer as a separate additional maintenance action, or the two films (static cling and super-slick) could be on the same roll and applied together by users. One minor technical difficulty: We don't know of a suitable super-slick film product. (Details, details.)

d. Another possible design concept would have a static cling inner surface as an integral part of the exterior cover. In that case, tests would have to show that the static cling feature was so strong that wind would not cause the inner surface of the cover to move. A static cling layer would probably not allow the cover to be pulled into position, so the cover would have to either be placed in position, or unrolled into position.

U.S. Government Printing Office 550-106