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SPACE TRANSPORTATION SYSTEM TECHNOLOGY SYMPOSIUM

III - Structures and Materials

NASA Lewis Research Center
Cleveland, Ohio
July 15-17, 1970

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16. Abstract

The Symposium encompassed seven technical areas, each published in a separate volume of NASA Technical Memorandum X-52876:

Volume I - Aerothermodynamics and Configurations
(includes aerodynamics; atmospheric operations; and aerodynamic heating)

II - Dynamics and Aeroelasticity
(includes dynamic loads and response; aeroelasticity; and flight dynamics and environment)

III - Structures and Materials
(includes structural design technology; thermal protection systems; and materials technology)

IV - Propulsion
(includes main propulsion; auxiliary propulsion; and airbreathing propulsion)

V - Operations, Maintenance, and Safety (Including Cryogenic Systems)
(includes general and cryogenics)

VI - Integrated Electronics (Including Electric Power)
(includes system integration; data management, systems monitoring, and checkout; navigation, guidance, and control; communication, instrumentation, and display; and power subsystems)

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FOREWORD

The prospect of undertaking a reusable launch vehicle development led the NASA Office of Manned Space Flight (OMSF) to request the Office of Advanced Research and Technology (OART) to organize and direct a program to develop the technology that would aid in selecting the best system alternatives and that would support the ultimate development of an earth-to-orbit shuttle. Such a Space Transportation System Technology Program has been initiated. OART, OMSF, and NASA Flight and Research Centers with the considerable inputs of Department of Defense personnel have generated the program through the efforts of several Technology Working Groups and a Technology Steering Group. Funding and management of the recommended efforts is being accomplished through the normal OART and OMSF line management channels. The work is being done in government laboratories and under contract with industry and universities. Foreign nations have been invited to participate in this work as well. Substantial funding, from both OART and OMSF, was applied during the second half of fiscal year 1970.

The Space Transportation System Technology Symposium held at the NASA Lewis Research Center, Cleveland, Ohio, July 15-17, 1970, was the first public report on that program. The Symposium goals were to consider the technology problems, their status, and the prospective program outlook for the benefit of the industry, government, university, and foreign participants considered to be contributors to the program. In addition, it offered an opportunity to identify the responsible individuals already engaged in the program. The Symposium sessions were intended to confront each presenter with his technical peers as listeners, and this, I believe, was substantially accomplished.

Because of the high interest in the material presented, and also because the people who could edit the output are already deeply involved in other important tasks, we have elected to publish the material essentially as it was presented, utilizing mainly the illustrations used by the presenters along with brief words of explanation. Those who heard the presentations, and those who are technically astute in specialty areas, can probably put this story together again. We hope that more will be gained by compiling the information in this form now than by spending the time and effort to publish a more finished compendium later.

A. O. Tischler
Chairman,
Space Transportation System
Technology Steering Group

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INTRODUCTION

Roger A. Anderson

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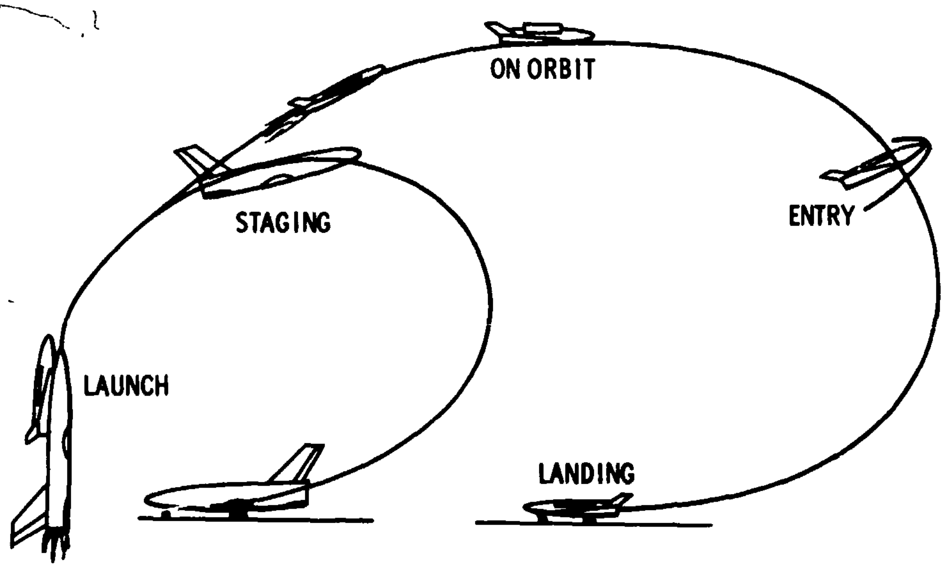
MISSION CHARACTERISTICS

The structural designer for a space shuttle sees two very large aircraft-type vehicles which undergo a spectrum of aerodynamic, acoustic, and inertia loadings, and a severe aerodynamic heating environment during a portion of the return flight. When the above mission characteristics are coupled with a requirement for repetitive use with minimum-cost maintenance between flights, he understands that he is faced with a structural design challenge of unprecedented proportions.

He has had experience with each of the principal mission characteristics singly, or in simple combination, with small test vehicles. Drawing upon this background, he has proposed preliminary design solutions appropriate to the shuttle and its complex mission profile.



- LARGE AIRCRAFT-TYPE VEHICLES
- SEVERE LOADS AND HEATING
- REUSE FOR 100 FLIGHTS

DESIGN UNCERTAINTIES

The basic complexity of the structural system for the shuttle and the current lack of detailed load definition for the many phases of the mission leads to several major design uncertainties, which are summarized here.

Prior to detail design efforts, the accuracy of weight determination is always open to question. For the shuttle, this presents a major uncertainty because of the sensitivity of gross lift-off weight and particularly payload to changes in the dry weight of the orbiter. The slope of these curves is approximately 5 times steeper than those for high performance aircraft.

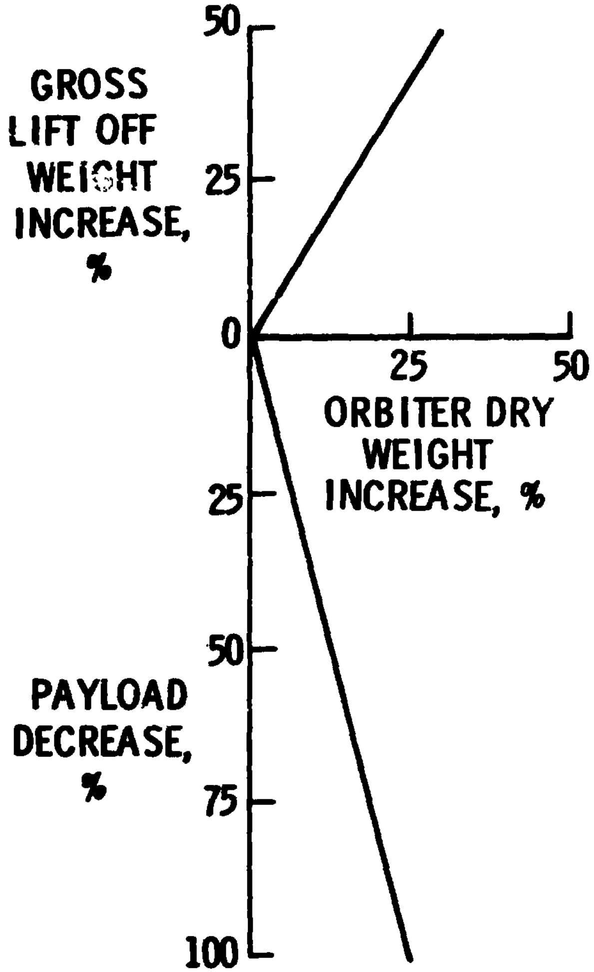
There is substantial uncertainty over the ability of several promising candidate materials for the thermal protection systems to withstand the operating environment for 100 missions. Available data suggests a more limited life with maintenance and repair at appropriate but as yet unknown intervals.

The sequence of tests necessary to establish initial flightworthiness of the structure and the maintenance and recertification procedures during mission operations are as yet undefined. Associated cost of these procedures will be determined by the quality of the initial design.

A major question is – what impact will application of advanced technology have on the system weight, reliability, and cost? The assumption made is that a technology development program will produce timely and highly beneficial results.

In view of the aforementioned major uncertainties which arise when shuttle structural design is attempted with current technology, a selective technology development program has been initiated.

DESIGN UNCERTAINTIES



ACCURACY OF WEIGHT DETERMINATION

MATERIAL PERFORMANCE FOR 100 MISSIONS

PROOF TEST & RECERTIFICATION PROCEDURES

IMPACT OF ADVANCED TECHNOLOGY ON
SYSTEM WEIGHT, RELIABILITY, AND COST

BASIC ELEMENTS OF TECHNOLOGY PROGRAM

The program content has been developed from many sources and is shown here as seven broad categories of effort. In each category, one or more major technical goals have been established for accomplishment in a 2-3 year R&D effort.

Technical highlights of each of these activities will be presented by NASA speakers in the 1st three sessions of this Conference. Each of these speakers will present his view of where we stand in solution of critical problem areas and give an indication of the kind of technical effort to be pursued in the current fiscal year.

Additional invited speakers from outside the agency will present their own views on the criticality of certain problem areas and present progress reports on their shuttle related activities.

BASIC ELEMENTS OF TECHNOLOGY PROGRAM

MISSION SIMULATION TESTING

CONSTRUCTION OF STRUCTURAL PROTOTYPES

STRUCTURAL DESIGN CRITERIA DEVELOPMENT

INTERRELATED EFFORT
IN GOVERNMENT AND
INDUSTRIAL LABORATORIES

ANALYTICAL METHODS DEVELOPMENT

THERMAL PROTECTION SYSTEM DEVELOPMENT

ADVANCED STRUCTURAL COMPONENT DEVELOPMENT

MATERIALS AND FABRICATION PROCESS DEVELOPMENT

PARTICIPANTS IN STRUCTURES AND MATERIALS

WORKING GROUP

As Mr. Tischler pointed out in his introduction to this Conference, a number of technology working groups have been established to pursue a shuttle related technology development program. This figure shows the panel organization of the Structures and Materials Group and lists the key personnel at the participating organizations.

The first three listed panels have organized formal sessions at this Conference. We will hear a report by the Chairman of the Facilities Planning and Utilization Panel in the session on TPS and Materials. The Coordination Panel works with the Chairmen of the technology panels to review, coordinate, and plan the activities of the Working Group.

STRUCTURES AND MATERIALS TECHNOLOGY GROUP

PARTICIPATION ROSTER

PANELS	NASA HDQTRS.	LRC	LeRC	MSFC	MSC	AMES	AFSC LABS WPAFB	SAMSO
STRUCTURAL DESIGN TECHNOLOGY	N. MAYER	R. LEONARD M. ANDERSON J. McNULTY L. VOSTEEN	G. SMITH J. BARBER	E. ENGLER C. LIFE[illegible]	R. VALE L. St. LEGER		W. GOESCH	
TPS AND MATERIALS		W. BROOKS B. STEIN	N. SAUNDERS	E. McKANNON	D. GREENSHIELDS	H. LARSON	N. GEYER	J. COLWELL
MATERIALS TECHNOLOGY	G. DEUTSCH		R. L. JOHNSON R. HALL R. KEMP	C. CATALDO W. RIEHL	R. JOHNSON			
FACILITIES PLANNING AND UTILIZATION		R. HOWELL	R. HALL	O. GOETZ	R. BRICKER	H. LARSON	W. GOESCH	
COORDINATION	F. DEMERRITTE G. DEUTSCH M. ROSCHE N. PEIL	R. ANDERSON	R. HALL	C. CATALDO	R. V'LE	H. LARSON	W. GOESCH	J. COLWELL

Note: Underlined personnel serve as Chairmen

STRUCTURAL DESIGN TECHNOLOGY - INTRODUCTION AND OVERVIEW

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SUMMARY

Numerous advances in structural design technology will be needed for achievement of shuttle performance goals. The eight succeeding papers summarize the state of the art and critical requirements for government and IRAD technology programs. This paper gives a brief overview of the current NASA plan for development of space shuttle structural design technology. Eight major tasks spanning design criteria, methods, advanced concepts, efficient design definition, prototype tests, and actuated structure are broken down to show the principal elements proposed for the FY-71 R&D program.

INTRODUCTION

The two-stage, fully reuseable space shuttle presents an awesome challenge for the structural designer. Its large size, multiple and severe operating environments, requirement for 100 reuses, and especially its inherent sensitivity to structural weight, together impose the need for a degree of design ingenuity, accuracy and efficiency which has no precedent in the major vehicle systems of the past. Numerous advances in structural design technology will, therefore, probably be necessary for achievement of shuttle performance goals.

The eight papers which follow will attempt to present a comprehensive picture of major structural design issues leading to new technology requirements. They will characterize the current state of the structural design art and indicate critical needs which should be met in government and IRAD technology programs. In addition, the present paper, together with three of the following papers, will present a resume of NASA's current plan for development of space shuttle structural design technology.

The first figure lists the major tasks that are currently conceived for the NASA program. Opposite each are the NASA Centers that share responsibility for each task with the Center currently having primary responsibility listed first. The column on the right gives funds, in thousands of dollars, that were expended in this program in FY-70. For FY-71, a significant increase in spending is planned on each of these tasks, except that titled Advanced Concepts Development and Evaluation. The total for FY-71 will be greatly increased over this FY-70 total.

SPACE SHUTTLE STRUCTURAL DESIGN TECHNOLOGY
PROPOSED NASA PROGRAM

MAJOR TASK	NASA CENTERS	FY 70 FUNDS, K$
DESIGN CRITERIA DEVELOPMENT	LRC, MSFC, MSC	500
DESIGN METHODS DEVELOPMENT	LRC	
ADVANCED CONCEPTS DEVELOPMENT AND EVALUATION	LRC	375
APPLICATION OF ADVANCED FILAMENTARY COMPOSITES	MSFC, LRC	854
ADVANCED TANK DESIGN TECHNOLOGY	LeRC, MSC	330
DEFN. OF EFFICIENT STRUCTURE/TPS DESIGNS	LRC	
MISSION SIMULATION TESTS OF LARGE STRUCTURAL PROTOTYPES	MSFC	1027
DESIGN OF SECONDARY STRUCTURE & MECHANISMS	MSC, LRC	
TOTALS		3086

On this and the succeeding figure, the major tasks are broken down to indicate the principal elements of the proposed FY-71 program. A series of intensive research studies will be made toward resolution of questions that are currently hindering the definition of structural design criteria and specifications; these are discussed in the following paper by Vosteen. Design methods development will be focused on the NASTRAN finite element computer program to achieve a uniform, advanced structural analysis capability for both NASA and its shuttle contractors. Some effort is also contemplated toward an alternate method of general shell analysis. Advanced concepts to be pursued are efficient panels for carrying compression and bending loads in shielded primary structure and an external gas-purged insulation system for liquid hydrogen tanks. However, the principal effort of this kind will be aimed toward application of advanced filamentary composites. Design of composite primary structure will be explored through major test programs of large subscale components; this research is summarized by Pride and Card in a subsequent paper.

PROPOSED ELEMENTS OF FY - 71 R&D PROGRAM

DESIGN CRITERIA DEVELOPMENT

LIFE CYCLE LOAD SPECTRA DETERMINATION
STRUCTURAL DESIGN CRITERIA FOR ABORT
THERMO-STRUCTURAL DESIGN FACTORS
REQUIREMENTS FOR DEMONSTRATING PROOF OF COMPLIANCE
WEIGHT AND COST SENSITIVITY TO CRITERIA

DESIGN METHODS DEVELOPMENT

IMPROVED STRUCTURAL ELEMENTS FOR NASTRAN
NASTRAN SYSTEM IMPROVEMENTS
PROGRAM FOR STRUCTURAL ANALYSIS OF GENERAL SHELLS

ADVANCED CONCEPTS DEVELOPMENT AND EVALUATION

ADVANCED STRUCTURAL PANELS WITH CURVED CROSS-SECTIONAL ELEMENTS
GAS-PURGED TPS FOR LIQUID H_2 TANKS

APPLICATION OF ADVANCED FILAMENTARY COMPOSITES

MATERIALS; FABRICATION AND NDE EQUIPMENT; TEST EQUIPMENT
DESIGN AND FABRICATION OF SHUTTLE COMPONENTS
IN-HOUSE THERMO-STRUCTURAL TESTS FOR COMPONENT EVALUATION

Some of the NASA advanced composites research and design methods research is focused on tank design. This involves filament overwrapping of both propellant and auxiliary tanks which is also discussed by Pride and Card. Fracture-control design methods will be evaluated for complex shuttle tank configurations and efforts will be made to extend these methods to apply and to generate the required fracture toughness data. The shuttle's sensitivity to inert weight lends the utmost importance to the definition of the most efficient approach to design of the structure and TPS. The program therefore provides for selected design studies to insure the achievement of this goal. Mission simulation tests of large structural prototypes will be carried out in-house. This effort is described in a following paper by Engler. Finally, design of secondary structure and mechanisms that involve shielding or accommodation of the thermal environment is expected to present major problems. Design of landing gear, control surface and docking hardware will be explored. Also considered will be doors and their actuators, windows and removable heat shield fasteners.

PROPOSED ELEMENTS OF FY - 71 R&D PROGRAM (CONTINUED)

ADVANCED TANK DESIGN TECHNOLOGY

COMPLETE DEVELOPMENT OF FILAMENT-OVERWRAPPED AUXILLIARY TANK TECHNOLOGY
DEVELOP TECHNOLOGY OF FILAMENT-OVERWRAPPED LARGE PROPELLANT TANKS
EVALUATE AND DEVELOP FRACTURE-CONTROL DESIGN METHODS FOR SHUTTLE TANKS
DATA ON DEEP FLAW EFFECTS ON STRENGTH
DATA ON COMBINED LOAD EFFECTS ON FLAW GROWTH

DEFINITION OF EFFICIENT STRUCTURE/TPS DESIGNS

MASTER CONTRACT(S) FOR SELECTED DESIGN STUDIES

MISSION SIMULATION TESTS OF LARGE STRUCTURAL PROTOTYPES

MATERIALS AND COATINGS; FABRICATION SERVICES; NDE EQUIPMENT
FACILITY MODIFICATION AND TEST EQUIPMENT
IN-HOUSE DESIGN AND FABRICATION OF STRUCTURAL PROTOTYPES

DESIGN OF SECONDARY STRUCTURE & MECHANISMS

DEVELOP ULTRA-LIGHTWEIGHT TIRE-WHEEL-BRAKING MECHANISM
SHUTTLE CONTROL SURFACE AND MECHANISM DESIGNS
SUB-SCALE MODEL OF DOCKING HARDWARE
DESIGN OF CARGO DOORS, ACTUATORS, WINDOWS, HEAT SHIELD FASTENERS

CONCLUDING REMARKS

In NASA's current plan for development of space shuttle structural design technology, eight major tasks have been defined spanning design criteria, methods, advanced structural concepts, efficient design definition, prototype tests, and actuated structure. Expenditures on these tasks in FY-70 amounted to about three million dollars. The tasks have been broken down to show the principal elements of a FY-71 R&D program which will be greatly expanded over the FY-70 program.

SPACE SHUTTLE STRUCTURAL DESIGN CRITERIA DEVELOPMENT

L. F. Vosteen

NASA-Langley Research Center
Hampton, Virginia

During the course of the Phase A space shuttle studies, it became apparent that there was at least one point of common agreement between all contractors: an early definition of a consistent structural design criteria was needed. Also, there appeared to be a number of technical questions peculiar to the space shuttle that required resolution before rational criteria statements could be made. In an effort to satisfy these needs, a program, outlined on figure 1, was initiated late last year. The purposes of the program are: to develop structural design criteria specific to the shuttle mission but independent of the vehicle configuration; to define the critical criteria related problems and effect their solution; and to provide a basis for preparing vehicle-specific criteria and specifications.

In this talk I will describe the current NASA criteria development activity and will discuss some of the special problems that require additional study before more specific criteria can be prepared.

The development of criteria for a major space flight program frequently is made more difficult because the criteria and the specifications are prepared at about the same time and hence, they become intertwined and confused. To help dispell some of this confusion, some basic definitions are shown on figure 2. Criteria are defined here as the conditions to be fulfilled to insure a flightworthy structure. Flightworthiness is defined as the capability of the vehicle to traverse the mission profile without jeopardizing mission objectives. The specifications are the contractual descriptions of the design requirements which govern the vehicle acceptance and consider such factors as performance, risk, and costs. The dictionary states that criteria and standards are synonymous. But, for the purposes of developing criteria for vehicles, it is appropriate to distinguish between them. Here we reserve the definition of standards to those measures of excellence, composition or precision against which a product is judged.

DEFINITIONS

CRITERIA - The conditions to be fulfilled to ensure flightworthy structure

FLIGHTWORTHY - The capability of the vehicle to traverse the mission profile without jeopardizing mission objectives

SPECIFICATIONS - Contractural descriptions of design requirements which govern vehicle acceptance considering performance, risk, and cost

STANDARDS - Measures of excellence, composition, and precision

To insure the usefulness of criteria, to both the designer and the program manager, certain principles should be observed in their formulation. These are shown on figure 3. Criteria should provide **an approach** for the design. That is, they should state the conditions or the ground rules for the design. Secondly, they should require positive action by the user — to show that the criteria were considered, his interpretation of the criteria was sound, and his design is adequate. On the other hand the criteria should not freeze design or impair progress. The designer should be allowed the flexibility to prove out better configurations or his own methods for compliance. Also, criteria should not preempt management's responsibility or authority to balance performance, risk, and cost.

CRITERIA PRINCIPLES

CRITERIA SHOULD:

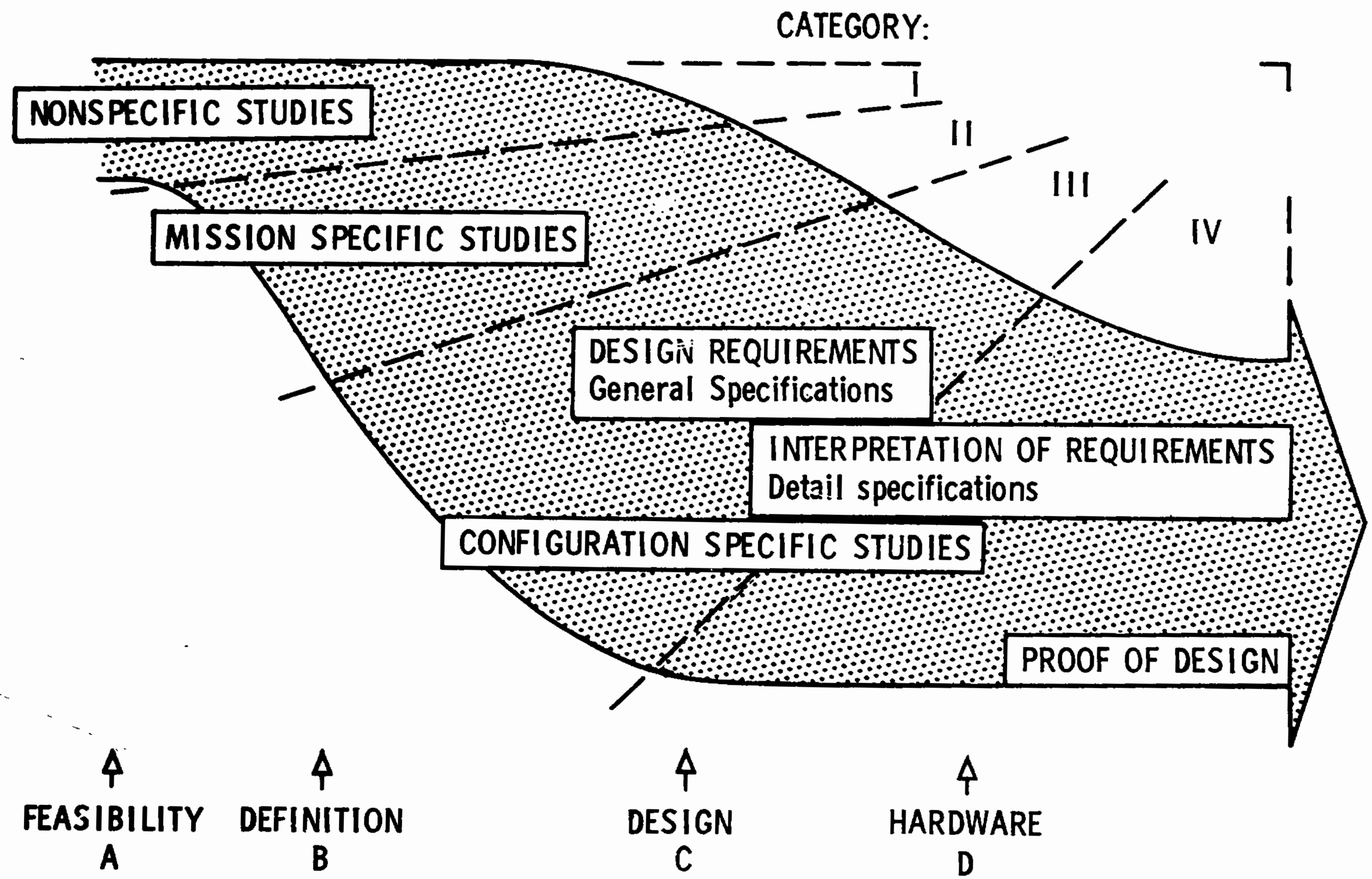
- Provide a basis for design
- Require positive action by the user to show that the criteria were considered, the interpretation was sound, and the design is adequate

CRITERIA SHOULD NOT:

- Freeze designs or impair progress
- Preempt management's responsibility and authority to balance performance, risk, and cost

Figure 4 illustrates the growth and flow of design information during the life of a program. The stream tube illustrated on the figure envelopes the information generated as a result of studies conducted before and during the program. It starts at the top left with nonspecific studies and proceeds down to the right where we generate specific knowledge needed to support hardware development and flight operations. The approximate starting points for the usual four program phases are indicated along the bottom of the figure. The left and right boundaries of the stream tube may also be considered as beginning and ending points of the bars in a program schedule — each bar being a particular study or task. The dashed lines divide the figure into sectors which serve to illustrate the changing character of the information as the program proceeds. The design requirements, or general specifications, and the interpretation of these requirements, or detailed specifications, generally come in the latter half of the program, at the same time that criteria are being developed. In fact, the specifications generally include elements of both criteria and standards.

EVOLUTION OF DESIGN INFORMATION



Our current plan is to initiate the preparation of design criteria at a point upstream of the requirements as shown on figure 5. By starting this criteria development earlier in the program, we hope to gain some important advantages. By beginning early, there should be time to do a more thorough and, hence, a better job of developing good criteria. As the program progresses, the criteria can be reviewed against the results of the definitive studies and the studies can be altered, if needed, to insure that the proper information is available to prepare meaningful requirements. As the design criteria are prepared, we expect to find instances in which our existing knowledge is not adequate to prepare a definitive criteria statement. By beginning earlier, there should be time for the additional study needed.

EVOLUTION OF DESIGN INFORMATION

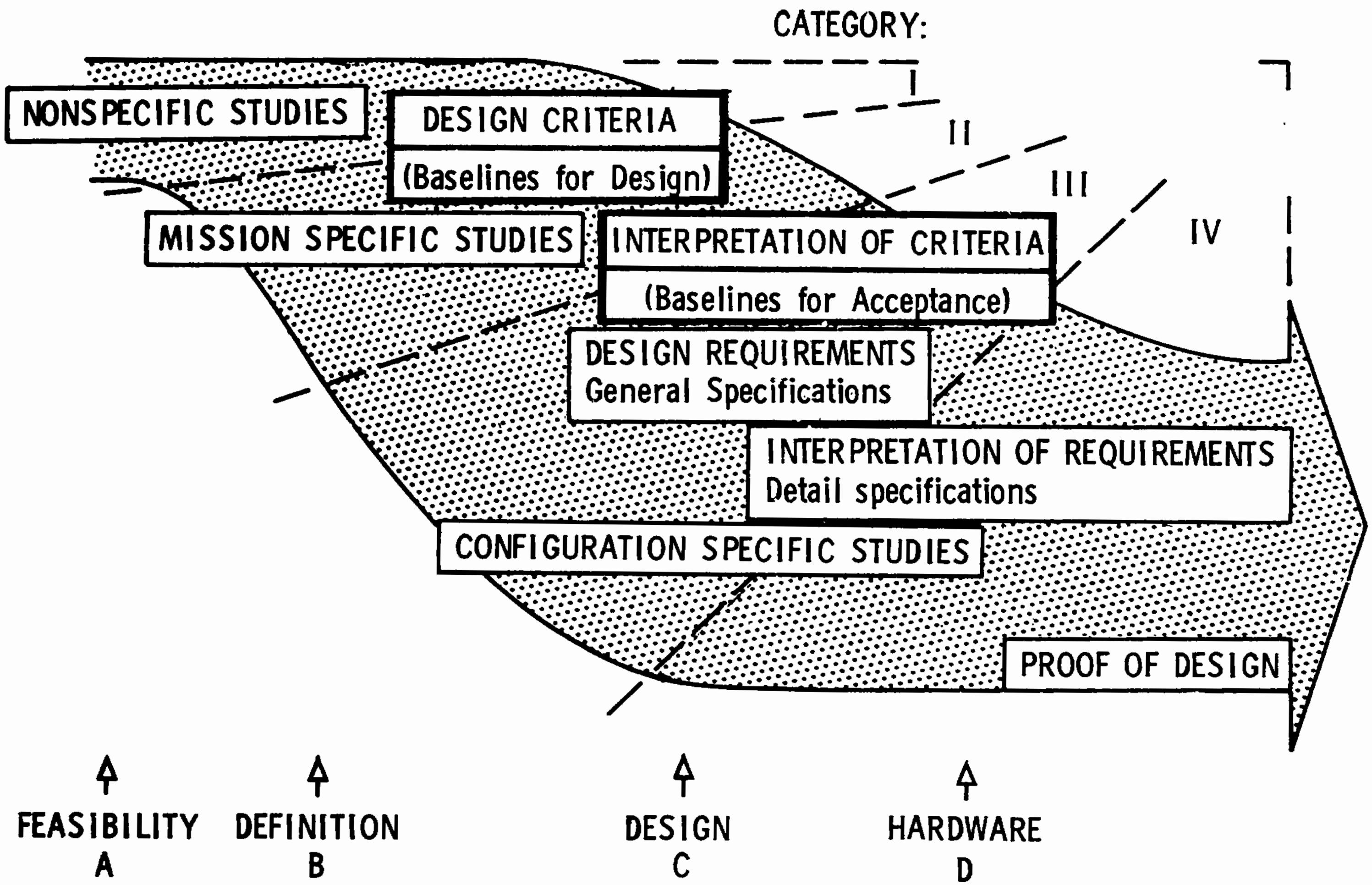
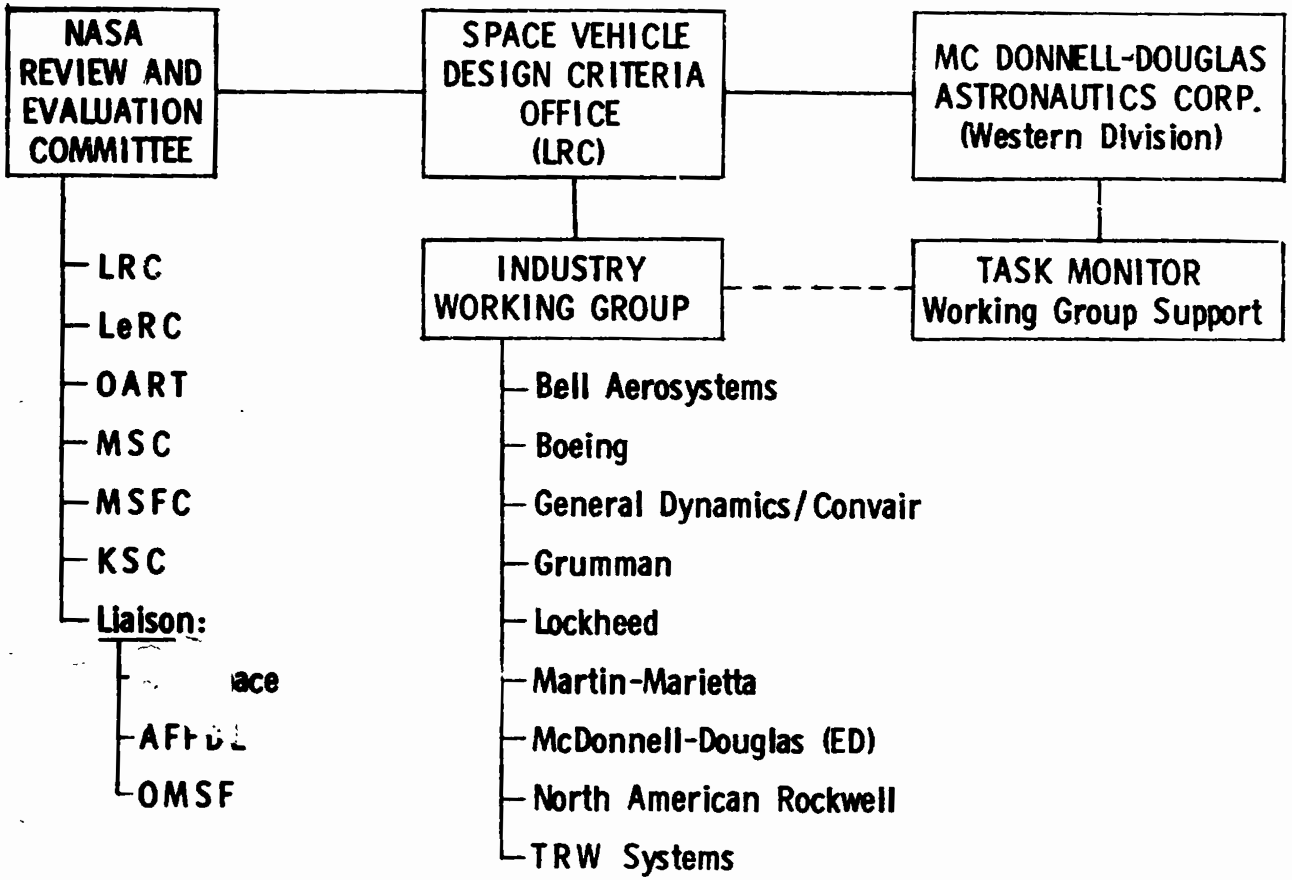


Figure 6 shows the mechanism or organization being used to prepare mission-specific, vehicle-nonspecific design criteria. For several years Langley has been engaged in the process of developing a series of non-specific criteria monographs. These monographs are general with respect to both mission and vehicle configuration and deal with various problems and aspects of structural design. In the context of the previous figure, these would be considered Category I documents. Through an existing master agreement contract, with McDonnell-Douglas Astronautics Corporation, Santa Monica, a task was initiated to enlist the services of an expert industry team to prepare structural design criteria applicable to the space shuttle. The industry working group is made up of representatives of 9 major aerospace firms and is chaired by Mr. Wilford Dukes of Bell Aerosystems. The NASA review and evaluation committee is composed of representatives from various centers and NASA Headquarters. In addition, liaison members are provided by the Aerospace Corporation, the Air Force Flight Dynamics Laboratory and the Office of Manned Space Flight.

The initial draft of the criteria document was completed in May. Extensive editing was recently completed and the document will now be sent to members of the industry working group and the NASA review and evaluation team for final comments. This review activity should be completed by about the end of July and, hopefully, comments can be incorporated before the end of August.

DEVELOPMENT OF STRUCTURAL DESIGN CRITERIA
APPLICABLE TO SPACE SHUTTLE



In the course of developing the criteria document, a number of problems arouse on which there was either disagreement as to what an appropriate criteria statement was or there was simply lack of sufficient background information to write a clear criteria statement that applied to the shuttle mission. Some of these problems are listed on figure 7. Design for abort raises several difficult questions. Fail-safe design philosophy is desirable wherever it can be applied. However, it is also necessary to have an adequate system of failure detection to determine if a major structure failure has occurred and if it warrants abort of the mission. The basic design philosophy of abort must be decided before abort criteria can be developed. The design should provide for the safe return for the crew and cargo. If the mission is aborted for nonstructural reasons, can normal design loads be exceeded on the assumption that the vehicle will require refurbishment? There is a need to examine the alternatives and determine the effect they will have on the vehicle design and cost.

There is substantial disagreement in industry on what an appropriate factor-of-safety for structural design is. The space shuttle structure is further complicated by the addition of a repeated thermal environment. Considerable study is required to determine an appropriate criterion for thermal design.

The space shuttle combines the flight environments of a launch vehicle, space vehicle, entry vehicle, and conventional airplane. The sequence and phasing of load and temperature are significant in defining how a material or structure responds. Studies are needed to determine how the various loads and temperatures resulting from trajectory dispersions should be combined to form the worst-case load conditions. The dynamic environments of the shuttle are very complex and it is not clear at this point how they should be combined. Much of the space shuttle vehicle is tankage and it is probably impractical to design tankage to be fail-safe or redundant. In some cases, the tankage may be used as primary load carrying structure, and an extension of safe-life design methods for these combined loads is needed. Finally, the space shuttle provides some unique problems for establishing proof-of-compliance. Because of the many combined environments involved, the level of simulation which must be employed in the qualification tests is not readily specified. Because the proposed vehicles are quite large and only a limited number of flight vehicles will be built, the requirements for testing full-scale hardware requires careful assessment to keep the program costs within reason.

STRUCTURAL PROBLEMS

AFFECTING CRITERIA DEVELOPMENT

ABORT

- Fail-Safe Design
- Structural Failure Detection
- Design Philosophy
  - Safe Return of Crew and Cargo
  - Flightworthy Vehicle ? Full Capability? Refurbishable?

FACTORS OF SAFETY

- Determining Appropriate Factor-of-Safety
- Application of Factors to Thermal Design

LIFE-CYCLE LOAD SPECTRA

- Sequence of Load and Temperature
- Combination Worst-Case Loads
- Combined Dynamic Environments
- Safe-Life Design for Primary Structure

PROOF OF COMPLIANCE

- Qualification Tests, Level of Simulation
- Flight Tests, Load Verification

The elements of the NASA technology program on structural criteria for the shuttle are shown on figure 8. During fiscal year 1970, the initial document for mission-specific criteria was developed. In addition, a small contractual study was initiated to help establish acceptable methods for determining limit loads. During the next fiscal year, we will initiate additional studies to help resolve some of the questions discussed on the previous figure. Some of the additional studies proposed are listed on the lower half of figure 8.

STRUCTURAL DESIGN CRITERIA STUDY PLANS

FY '70

- DEVELOPMENT OF MISSION-SPECIFIC STRUCTURAL DESIGN CRITERIA
- STUDY OF METHODS TO DETERMINE LIMIT LOADS

PROPOSED FY '71

- FACTORS-OF-SAFETY EVALUATION
- THERMAL-STRUCTURAL DESIGN FACTORS
- LIFE-CYCLE LOAD SPECTRA DETERMINATION
- WEIGHT AND COST SENSITIVITY TO CRITERIA
- REQUIREMENTS FOR DEMONSTRATING PROOF-OF-COMPLIANCE
- STRUCTURAL CRITERIA FOR ABORT
- FAIL-SAFE DESIGNS

SUMMARY REMARKS

In this talk, I have outlined the elements of the current NASA structural design criteria development effort as it applies to the shuttle mission. In the past six months, a working group comprised of structural experts from nine aerospace companies has prepared a mission-specific criteria document and defined some technical problems which require additional study. During the next year, some contractual studies will be initiated to help resolve these problems. By beginning the activity early in the shuttle development program, we will provide a sound basis for the preparation of more specific design criteria and the vehicle specifications.

SPACE SHUTTLE STRUCTURAL FATIGUE CONSIDERATIONS

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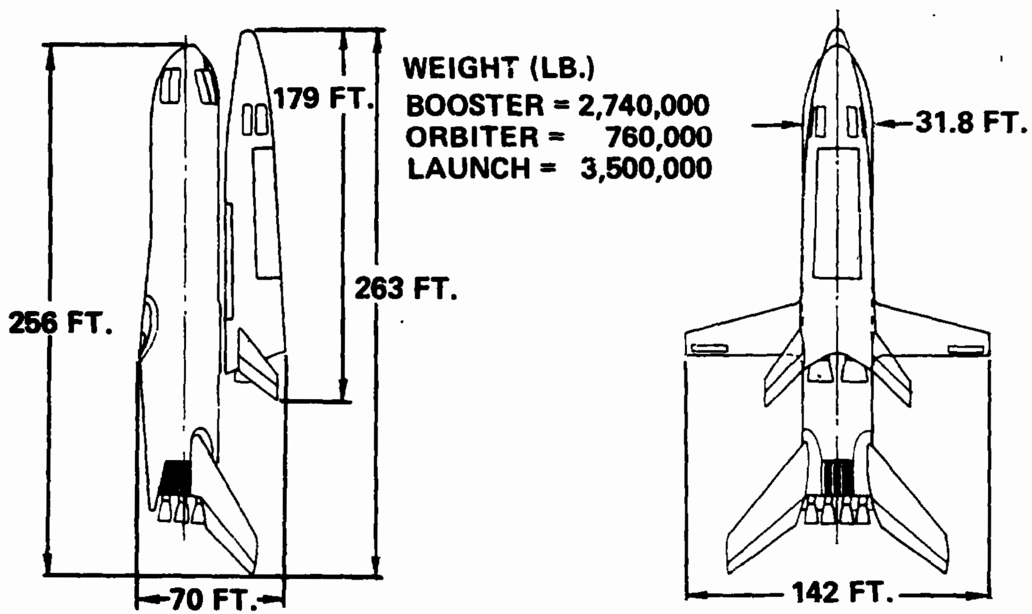
The hybrid nature of the space shuttle vehicles, Figure 1, poses some fundamental questions relative to the underlying philosophy of the structural design. Here in one system are contained all the problems related to a rocket launched space and re-entry vehicle, and also those inherent in the design of a conventional aircraft which utilizes air-breathing turbine engines. It is rapidly becoming apparent that some of the problems are not merely additive, but are also unfavorably synergistic. Analytical and experimental techniques are generally available to attack these problems, but basic data are not.



Because of the wide spectrum of environmental loads to which the system will be exposed, the cyclic nature of these loads and the requirement for a reliable vehicle with a life of 100 missions, providing a near-optimum solution to the problem of designing a light weight, fatigue-resistant structure becomes of prime importance. To assist in providing a design solution, many new materials are being considered, particularly for the high temperature environment. In the interest of minimizing weight and optimizing structural efficiency, all materials must be used to their maximum capability. It is of utmost importance, therefore, to assure that the basic material properties are developed in an orderly fashion, e.g., maximum safe stress levels, fatigue characteristics, notch sensitivity, fracture resistance, creep resistance, corrosion resistance, fabrication qualities, and others.

This discussion is primarily concerned with fatigue, not only of materials but also of fabricated and joined structures. Analytical techniques for fatigue analysis must be improved and must be verified in the laboratory on the types of materials and structures considered for the space shuttle vehicles. Mathematical over-simplifications, such as Miner's hypothesis, must be avoided at the high cycle-low stress end of the fatigue spectrum, while the science of fracture mechanics becomes a mandatory tool at the low cycle-high stress end of the spectrum.

The balance of this discussion will provide some additional detail in connection with the major areas of low and high cycle fatigue on a space shuttle configuration, some observations on cumulative fatigue damage analysis and descriptions of some materials and structural arrangements being considered, and will summarize with an outline of an overall technical approach for an optimum design solution.

SPACE SHUTTLE STRUCTURAL FATIGUE CONSIDERATIONS

LOW CYCLE FATIGUE

- LAUNCH & FLYBACK AERODYNAMIC LOADS
- THERMAL LOADS
- PRESSURIZATION LOADS
- SEPARATION LOADS
- LANDING LOADS

HIGH CYCLE FATIGUE

- ROCKET ENGINE NOISE
- FLYBACK ENGINE NOISE
- SHOCK BOUNDARY LAYER INTERACTION PSEUDO-NOISE
- PANEL INSTABILITIES & FLUTTER

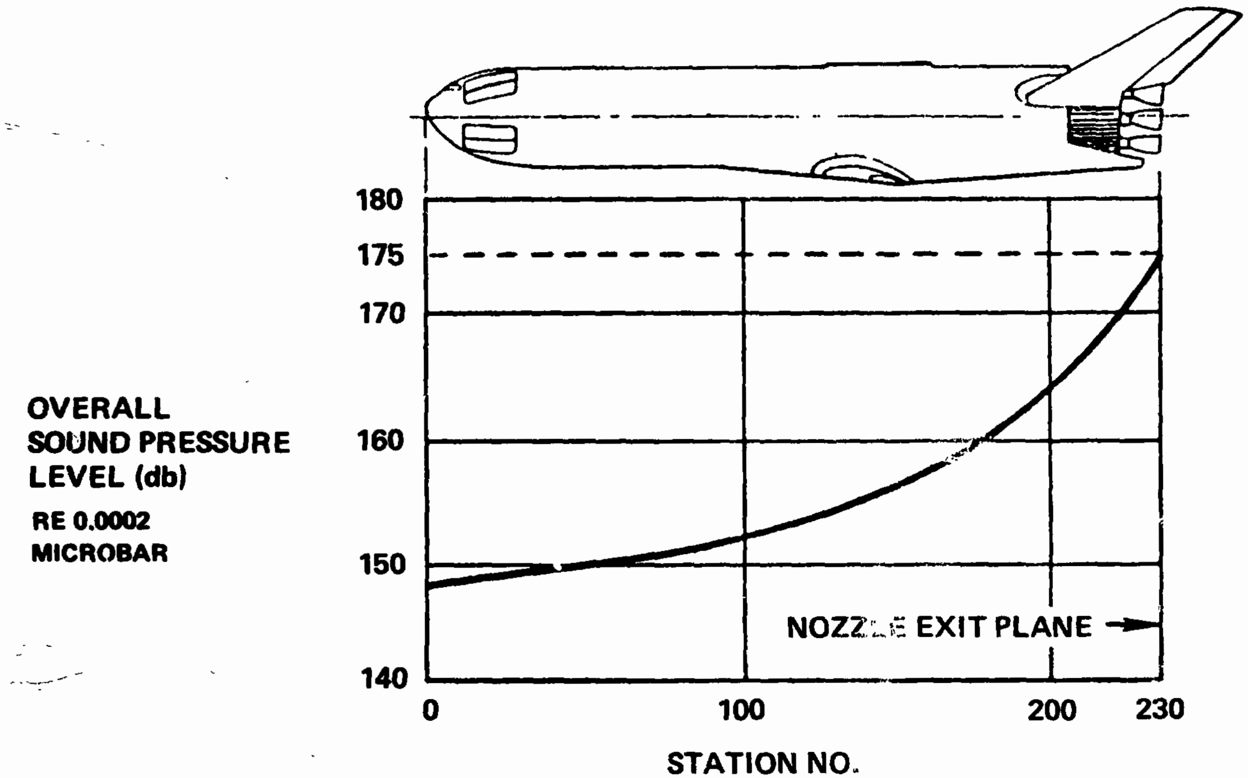
Figure 3 shows the space shuttle booster as a typical element of the shuttle system. The plot below the outline of the booster is an estimate of the acoustic level sustained over the entire length of the vehicle at launch. It shows that the minimum level is about 150 db, which builds up as an exponential function to a maximum energy level of 175 db at the plane of the nozzles. Inertia and airloads are superimposed on this environment on certain portions of the structure. Low cycle fatigue conditions are, of course, present on the pressurized tank structure at this time. Later, shock interaction effects occur due to the booster aerodynamic configuration and the presence of the orbiter.

During the flyback portion of the mission, the vehicle will experience the normal aircraft type of fatigue environment on its basic structure. In addition, however, there will be a reasonably severe local acoustic disturbance in the wake of the flyback turbofan engines which will probably determine the local panel design arrangement.

The need for careful, failure-free design of the shuttle vehicles cannot be over-emphasized. Concessions to weight for the sake of outer covering panel integrity have to be made, for example. Trade-offs have to be studied in order to arrive at the optimum means of obtaining the structural response mode for optimum fatigue resistance. To illustrate; which is the better method for obtaining the maximum fatigue resistance in a TPS cover panel - is the structural weight used more efficiently in the panel supports, or should it be used to provide additional stiffness to the panel itself? Similarly, how is crack arrestment best obtained in a cryogenically fueled integrally stiffened tank?

Coupled with the basic design considerations are the formidable problems of facilities and methods necessary to successfully verify the integrity of the structural concepts. It can conceivably require the best that both the government and industry can offer, and quite possibly, develop, to achieve this goal.

ACOUSTIC LEVEL AT LIFTOFF



During the course of a single mission, the space shuttle vehicles will be subjected to a wide variety of loads. Starting with ignition of the rocket engines at launch and ending with the final application of the brakes on the landing gear after touchdown, the loads will vary from the single event impulsive type, such as may be generated by rocket engine ignition or vehicle separation in flight, to the high frequency oscillatory forces resulting from the intense acoustic field at launch or boundary layer-shock interaction in flight.

Figure 4a shows a stress time-history for an arbitrarily selected structural element. The first major peak could be associated with launch, the second with penetration through the transonic flight regime, the third with vehicle separation, etc. The high frequency oscillations could be due to noise or could be due to "ringing" of the structure due to impulsive type loads. In general, the higher stresses are associated with relatively low numbers of occurrences while the lower stresses may be experienced for large numbers of occurrences.

Although a few specific events occur during each flight, and the resulting stresses will be roughly similar, most of the stress history will be random in nature and must be considered on a statistical basis. If the occurrences of stresses at various levels were totaled over the total mission life, a histogram would be generated which, when faired, would resemble the curve shown in Figure 4b.

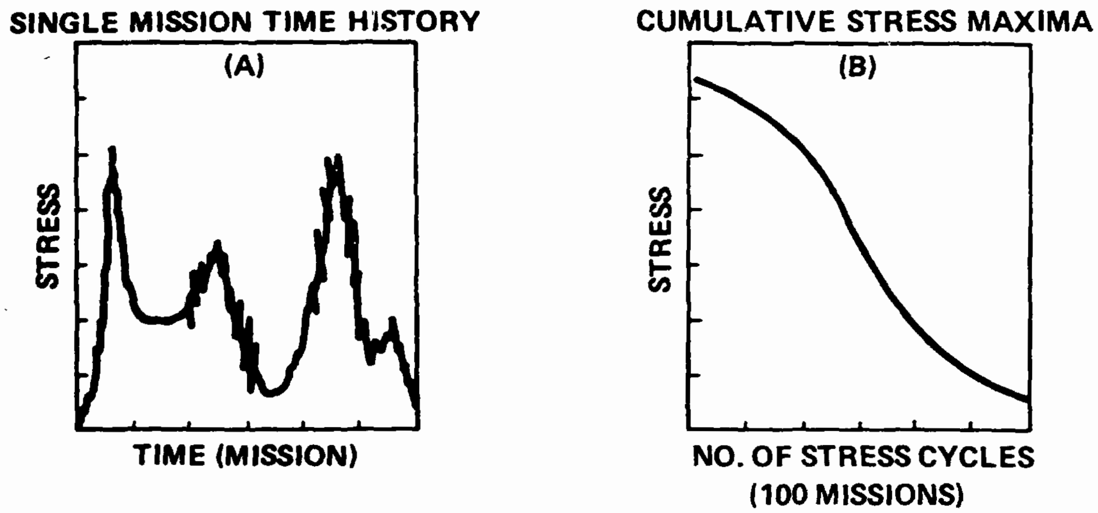
Considering the generally random nature of the stress history, the Central Limit Theorem states that regardless of the original distribution, as the number of events becomes sufficiently large, the distribution of <u>instantaneous</u> values becomes normal or Gaussian. However, since it is the distribution of <u>peak</u> stresses which is of interest, the normal distribution of instantaneous values can be converted to an integrated Rayleigh Distribution defined by

$$\left[p > s_p\right] = e^{-1/2\left(\frac{s_p}{s_o}\right)^2}$$

where $\left[p > s_p\right]$ is the probability of exceeding s_p, in percent, and $\left(\frac{s_p}{s_o}\right)$ is the ratio of peak stress/rms stress.

Not too surprisingly, the shape of the Rayleigh curve is very similar to that shown in Figure 4b. The next step is to ascertain how useful such a tool may be for predicting the fatigue life of a structural element.

STRESS VARIATION AT ARBITRARY POINT ON STRUCTURE



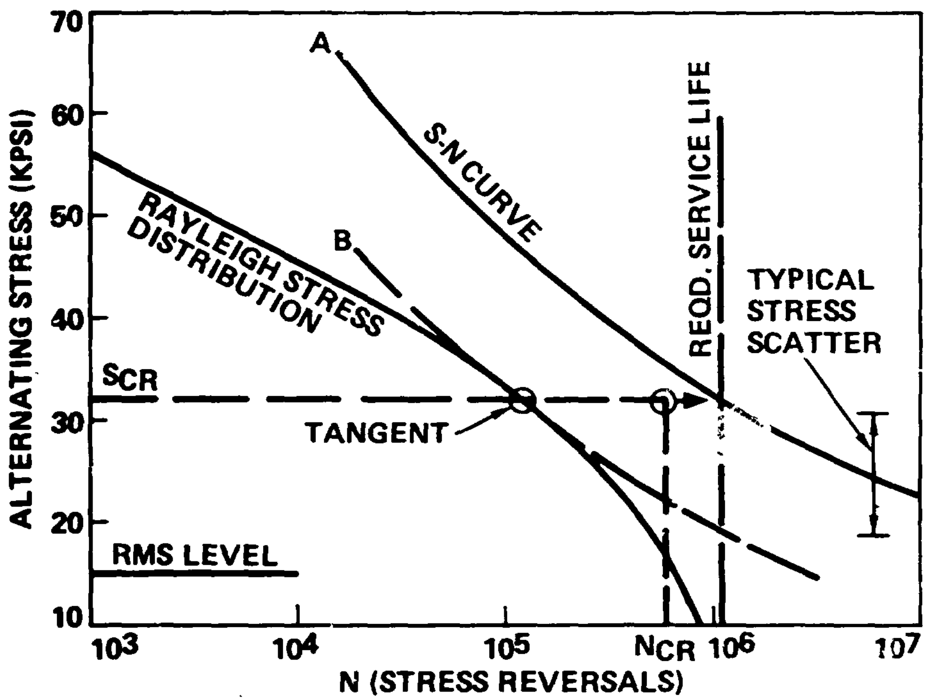
Any theory relating to fatigue damage under random loading includes a concept of cumulative, partial damage. The simplest and, hence, the most attractive theory is the linear cumulative damage concept, or Palmgren-Miner Theory. This states that the sum of the partial damage is equal to unity. The overall trend of test results indicates, however, that the linear rule is generally unconservative and over-estimates fatigue life. Hence, it is common practice to impose a scatter factor on design life requirements. A non-linear rule propounded by Freudenthal and Heller attempts to compensate for the unconservatism of the linear rule. In addition, there is also a school of thought which says that fatigue (S-N) curves must be developed on a random loading basis rather than on the classic sinusoidal loading basis.

Another cumulative damage approach which has been successfully used, particularly in sonic fatigue studies, considers the problem as one of energy absorption - or dissipation. Above its endurance limit (if one exists) a material or structure has only a finite capacity for absorbing energy before failure will occur. Where the applied loads (and structural responses) vary sinusoidally, the standard S-N curve defines the amount of work which can be accomplished before failure. With this basic concept in mind, it can be immediately extended with the idea that if there is a limit to the energy which a structure can absorb without failure then, within certain restrictions, the limiting energy should be independent of the rate, sequence or level of application.

Figure 5 shows the construction for this type of analysis. The S-N curve represents a structural element exposed to intense noise during launch of a shuttle vehicle. The stress levels include a stress raiser, $K_t = 4$, while the number of stress reversals are cumulative for a 100-mission vehicle life. S_{cr} represents a "critical" stress level defined at the point of tangency between the stress distribution curve and a parallel displacement of the S-N curve. The construction, as shown, indicates adequate service life based on the mean, S-N curve. The conservatism of this approach has been borne out by thousands of hours of airline service. (Ref. AIA/ONR Symposium-Structural Dynamics of High Speed Flight, ACR-62, Vol. 1)

However, a caveat must be stated. Any simple view of cumulative fatigue damage ignores many parameters which could modify the analytical results. For example, the sequence in which a structure is exposed to a wide range of stress levels could be very important. Another important consideration is that if the range of stress levels is such as to cause cyclic redistribution of the stress pattern, no simple theory will hold. One additional point that must not be ignored is that if stress maxima approach yield values, again no simple theory will hold. Particularly, when dealing with high strength or brittle materials, extreme caution must be used in evaluating fatigue life. These materials are extremely sensitive to small flaws or cracks, and when worked at high stress levels, fail in modes which are dissimilar to "normal" fatigue failures. This area is properly the domain of the specialists in fracture mechanics.

EQUIVALENT FATIGUE DAMAGE DIAGRAM
RANDOM LOADING



As it is now conceived, the space shuttle structure will incorporate both isotropic and composite materials. The materials will see an environment that is probably unique in its severity. Materials are called on to function in what can literally be thought of as the "worst of two worlds" - the first problem being temperatures which range from those of liquid hydrogen to reentry; the second being the extreme high and low cycle fatigue environment.

The temperature extremes dictate the candidate materials noted in Figure 6. The load carrying tank systems may be 2219 aluminum alloy, while the interconnecting structural components, [illegible] structure and parts of the heat shield may be titanium alloy reinforced in selected areas with composites. The higher temperature areas of the heat shield, fin and thrust structure will use the nickel and refractory alloys, with possibly some carbon-carbon material in selected areas. The chart is presented to point up the dearth of mechanical, physical and fatigue properties that are available for the designers using these alloys. Both low and high cycle fatigue properties are conspicuously absent and need to be obtained on a high priority basis to support the fundamental design considerations. Equally important for dynamic loading conditions are the creep properties at both room and elevated temperatures, inasmuch as structural relaxation and deformation due to creep affect the dynamic response to a large degree. Similarly, degradation of other fundamental material properties with time and temperature exposure will have a direct effect on structural response characteristics.

It is anticipated that both resin and metal matrix composites will be selectively used for the shuttle wherever their usage offers a technological advantage. In general, the composites offer attractive possibilities as fatigue resistant materials due to their highly redundant nature. Preliminary characterization studies have borne this out. However, much work needs to be accomplished. A secondary factor in the composites fatigue characterization lies in the lack of consistency of their mechanical properties. Industry-wide cooperation will be required in order to achieve an improvement in this area.

One characteristic of the space shuttle that merits close attention in design and materials are the low cycle fatigue properties. For a vehicle operating over a fixed life span of 100 missions, the most efficient structural design requires that all of the basic materials be utilized at a relatively higher stress level for a fewer number of cycles than is normal for fatigue-resistant design. Fracture toughness data, largely nonexistent for the materials considered, must therefore receive a high priority in the development of shuttle structure. It is apparent, therefore, that specialized laboratory facilities must be available to develop the required, basic, building blocks.

SPACE SHUTTLE MATERIALS UNDER CONSIDERATION

MATERIALS	MECHANICAL PROPERTIES		FATIGUE		FRACTURE TOUGHNESS		CREEP		LIFE CYCLE		PHYSICAL PROPERTIES	
	RT	±TEMP.	RT	±TEMP.	RT	±TEMP.	RT	±TEMP.	RT	±TEMP.	RT	±TEMP.
2219 Al	◒	◒	◒	◒	◒	◒			○	○	●	◒
Ti-6Al-4V	◒	◒	●	○	◒	○	●	◒	○	○	●	◒
718 Ni	◒	◒	◒	○	◒	○		◒		○	●	◒
RENE' 41	◒	◒	○	○	○	○		○		○	◒	◒
HS188	◒	◒	○	○	○	○		○		○	◒	◒
COLUMBIUM	◒	◒	○	○			○	○	◒	◒	◒	◒
CARBON-CARBON	◒	◒						○		○	◒	◒
Al-B COMPOSITE	◒	◒	◒	○	◒	○		◒		◒	◒	◒

● SUFFICIENT DATA AVAILABLE

◒ SOME DATA AVAILABLE. NEED MORE TESTS

○ DATA NOT AVAILABLE. TESTS REQUIRED

BLANK – NO REQUIREMENT

Two of the many structural areas that may be fatigue prone in the shuttle structure are shown in Figure 7, viz, the thrust redistribution structure and the high temperature thermal protection system panel that is typical of the orbiter.

The thrust redistribution structure was chosen as being typical of the fatigue problem that exists in heavy structural members that must operate in a combined elevated temperature, high acoustic disturbance area. Its proximity to the plane of the rocket engine nozzles provides up to 170 db acoustic excitation, temperatures on the order of 500°F, and the problem of redistributing the engine thrust into the hydrogen tank shell. Thrust redistribution is effected by a gridded beam arrangement. The beams will be on the order of 60 inches deep and will operate at a high stress level at the time of maximum acoustic excitation. This load environment makes the use of composites highly desirable, due to their high strength and fatigue resistance. The redistribution of the beam end loads into the tank will most likely be effected through an integrally stiffened shell. Verification of the adequacy of this arrangement requires technologically advanced methods and facilities. A similar acute fatigue problem will obviously exist on the fin surfaces which are mounted in the same region. Therefore, sandwich construction will be considered for parts of the fin structure.

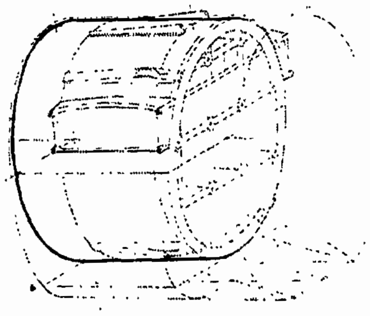
Thermal protection of the substructure, particularly on the orbiter, is planned as a series of stiffened panels mounted on a standoff arrangement from the substructure. The panels are subjected to extreme temperatures, must carry airloads and must be fatigue resistant. The right-hand picture of Figure 7 illustrates a typical panel arrangement. The panel shown was fabricated from TD Ni CR alloy and was thermally cycled to a maximum of 2200°F, and then subjected to a spectrum of acoustic excitation which culminated with 50 runs of 50 second duration at 165 db. The 165 db noise environment dictated the design of the panel and its supports. The panel shown represents only a very small section of the vehicle and only development testing has been accomplished to date. An evaluation of the suitability of the system for the shuttle vehicle will require considerably larger specimens that will permit panels, joints and supports to be tested free of side effects and test apparatus interference. This, in turn, imposes severe requirements on test facilities.

At this stage of vehicle development, similar tests must be performed on larger scale panels that are fabricated from Columbium and the superalloys in order to provide adequate design data for other temperature regimes. It can be predicted that the fatigue environment will also dictate the design of the panels in these materials.

Throughout the design of both orbiter and booster, the ABC's of good design for fatigue must be observed. The design for sonic fatigue resistance dictates the requirement for adequate panel stiffness between supports. The supporting structure must also be carefully designed to avoid all asymmetry of load paths in addition to normal good design practice which eliminates stress raisers. The design for low cycle fatigue involves a thorough knowledge of the time-load spectrum of the structure, and reliable fracture toughness information for the materials under consideration coupled with adequate criteria.

TYPICAL STRUCTURAL ARRANGEMENTS

THRUST STRUCTURE



THERMAL PROTECTION PANEL

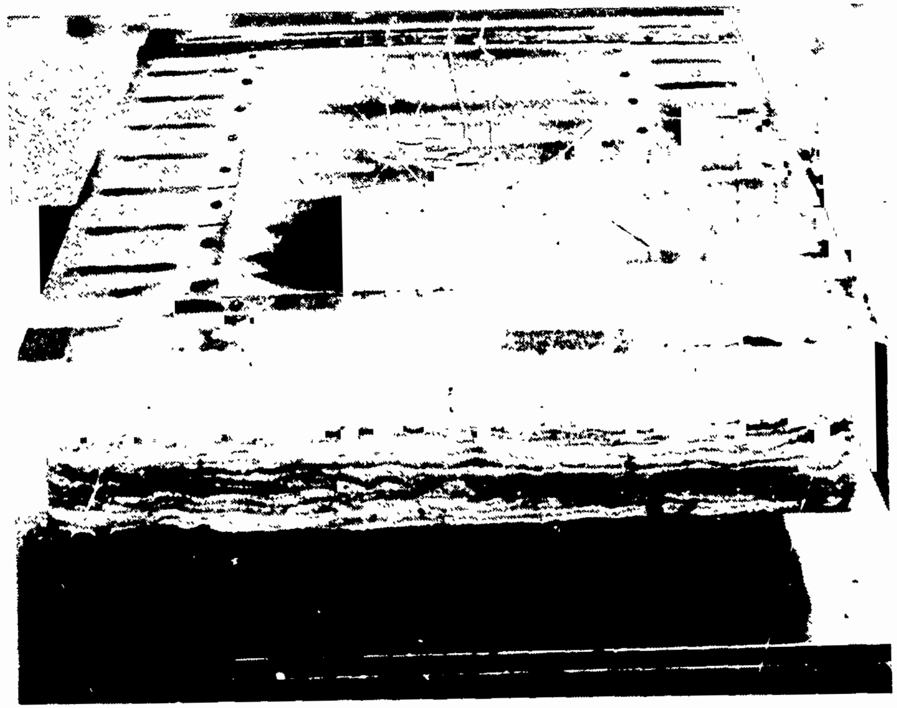


Figure 8 summarizes the requirements for a systematic technical approach to the design problem.

The dynamic environment must be defined as precisely as possible. For example, a variation in acoustic pressure level of 6 db represents a factor of 2 in applied load. Material properties must be completely defined. This means that a broad spectrum of laboratory tests must be initiated immediately so that data for the new materials will be avilable for the designers when needed.

Structural and material arrangements must be continuously re-evaluated to arrive at an optimized design solution. Finally, a comprehensive development test program for candidate structures must provide the final assurance for design integrity.

TECHNICAL APPROACH

- FULLY DEFINE THE DYNAMIC ENVIRONMENT
- FULLY CHARACTERIZE CANDIDATE MATERIALS
- CONTINUOUSLY EVALUATE STRUCTURAL ARRANGEMENTS
- TRADE OFF MATERIALS, ARRANGEMENTS, & ATTENUATION METHODS
- PROVIDE COMPREHENSIVE DEVELOPMENT TEST PROGRAM

In summary, adequate design for fatigue in a hypercritical environment requires advanced design concepts which fully utilize the desirable properties of the newer materials. In addition, analytical methods must be improved to provide more precision and special test procedures must be devised, and facilities furnished, to validate design predictions.

SUMMARY

ADEQUATE DESIGN FOR FATIGUE REQUIRES:

- ADVANCED DESIGN CONCEPTS FOR HYPERCRITICAL ENVIRONMENT
- COMPREHENSIVE CHARACTERIZATION OF NEWER MATERIALS
- REVISED ANALYTICAL METHODS
- SPECIALIZED TEST FACILITIES & METHODS

THE DEVELOPMENT OF EFFECTIVE FRACTURE CONTROL PROCEDURES FOR SPACE SHUTTLE

C. F. Tiffany

The Boeing Company
Seattle, Washington

- THE PURPOSE OF FRACTURE CONTROL PROCEDURES
- SOME MAJOR CONTRIBUTING FACTORS & SPECIFIC CAUSES OF STRUCTURAL FAILURES
- DESIGN PHILOSOPHY
  "FAIL SAFE" OR "SAFE LIFE"?
- THE ELEMENTS OF EFFECTIVE FRACTURE CONTROL
- SOME OBSERVATIONS ON CURRENT STATUS
- SOME AREAS OF RESEARCH IMPORTANT TO THE SPACE SHUTTLE

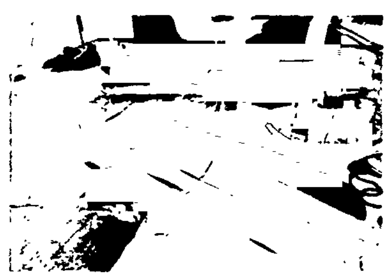
THE PURPOSE OF FRACTURE
CONTROL PROCEDURES

Fracture control procedures are required for the purpose of preventing structural failures, which in turn could result in mission abort, loss of vehicle, and/or loss of life. They are also required in order to minimize the need for expensive and time consuming repair and/or refurbishment.

During the 1950's and early 1960's numerous failures provided the incentive to develop fracture control procedures for aerospace pressure vessels. The procedures that have been developed and applied over the past decade have tended to diminish, but not completely eliminate failures such as shown in the illustration. Attempts are currently being made to apply similar procedures to obtain assurance of satisfactory life in military aircraft; however, because of added complexities in structural configuration and loading history, the task is more difficult. The introduction of the Space Shuttle Vehicle will add further complexities to the problem of fracture control. The shuttle represents a combination of the pressure vessel and airplane problems, and in addition has the added complexity of extreme environmental exposures.

- **PREVENTION OF STRUCTURAL FAILURE AND MINIMIZATION OF REPAIR (Refurbishment)**

260" Motor Case Failure



Apollo SPS Tank Failure

SOME MAJOR CONTRIBUTING FACTORS AND SPECIFIC CAUSES

In order to understand how to prevent fracture, it is necessary to first recognize factors that have contributed to many of the past failures and the specific causes for these failures. This chart itemizes a number of the most important factors and specific causes. Some of these will be discussed in more detail in subsequent charts.

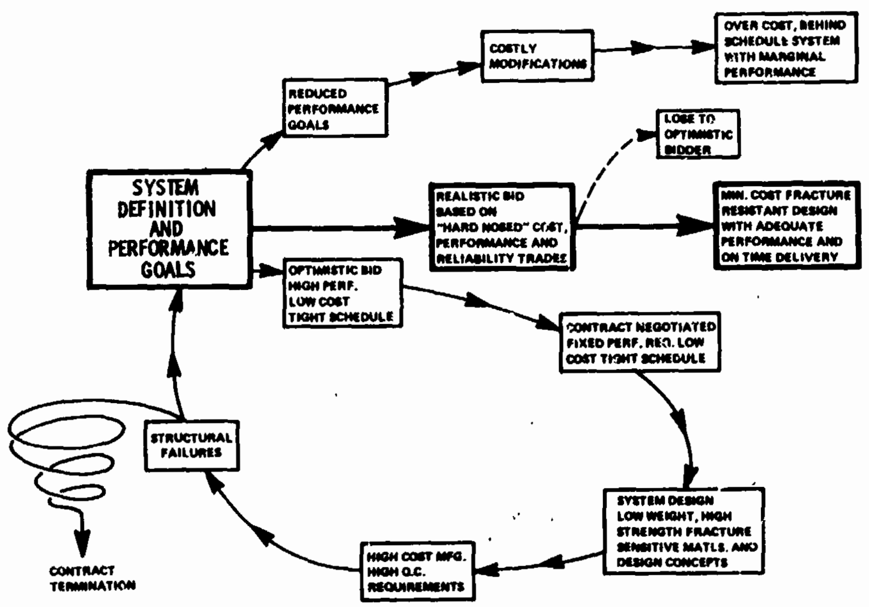
While it has been apparent for a number of years that flaws and defects are one of the primary causes of premature structural failures, it has only been during the past few years, when confronted with the evidence, that most contractors have been willing to admit that their NDI procedures were not infallible and the structures that they delivered did contain flaws. It is my opinion that most structures contain pre-existing flaws, but with the use of materials and designs that are fracture resistant, the flaws seldom degrade the structural performance. However, with increase in material strength level, increase in operating stress, increase in section size, and for most steels and titaniums, with a decrease in temperature, the probability of encountering premature brittle failure increases. The problem is further compounded if, because of complexity, the structure is not easily inspected.

- **CONTRIBUTING FACTORS**
  - SYSTEM PROCUREMENT & CONTRACTING PROCEDURES
  - LACK OF "FAILURE PREVENTION" KNOWLEDGE OF PROGRAM MANAGERS AND DESIGN ENGINEERS
  - ARBITRARY DESIGN CRITERIA & REQUIREMENTS
  - OVER CONFIDENCE IN NDI CAPABILITIES
- **SPECIFIC CAUSES**
  - PRE-EXISTING DEFECTS
  - USE OF FRACTURE SENSITIVE MATERIALS
    - Low Fracture Toughness
    - Susceptible to Stress Corrosion & Hydrogen Cracking
    - High Crack Growth Rates
  - USE OF FRACTURE SENSITIVE DESIGNS
    - High Operating Stress Levels
    - Thick Sections
    - Single Load Paths
    - Poor Inspectability

THE PROCUREMENT/DESIGN OPTIONS

(A Major Factor in Failure Prevention)

Having participated in a number of failure problems during the past decade and having seen how the government and contractors go about contracting and designing new flight systems, I have arrived at the opinion that this cycle can and indeed has contributed to premature failure problems. This chart illustrates two optional paths from system definition to hardware delivery, and in my opinion, all too often we have taken the spiral path to arrive at over-cost behind schedule systems with marginal performance. A number of specific examples can be cited.

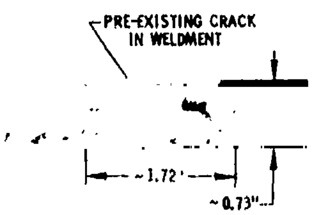


SOME EXAMPLES OF SPECIFIC CAUSES OF STRUCTURAL FAILURES

(Missile/Space Structure)

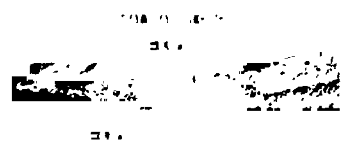
This chart shows a number of photographs of fracture origins in aerospace pressure vessels. All fractures resulted from surface or subsurface flaws that attained critical size prior to growing through the thickness of the vessel walls. Failure occurred when the flaw tip stress intensity, k, reached the critical value for the specific vessel material. With the exception of the APOLLO SPS tank, all flaws shown were pre-existing and were not detected by the NDI procedures used. The surface flaw in the titanium SPS tank was a stress corrosion crack that formed during cold flow testing with methanol.

The flaws shown in these photographs are typical of those seen in many other pressure vessel failures.



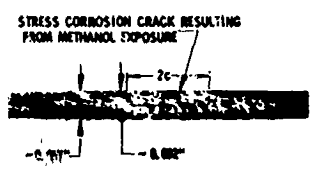
260" MOTOR CASE
FAILURE ORIGIN

BOMARC He TANK
FAILURE ORIGIN

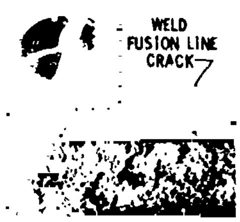


M^2 TITANIUM MOTOR CASE
FAILURE ORIGIN

M^2 TITANIUM MOTOR CASE
FAILURE ORIGIN



APOLLO SPS TANK
FAILURE ORIGIN



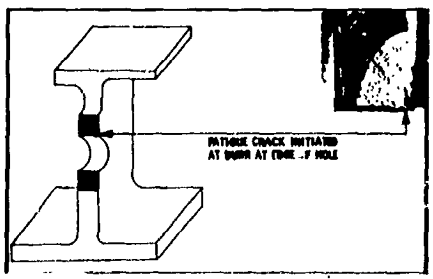
ALUMINUM
PRESSURE VESSEL
FAILURE ORIGIN

SOME EXAMPLES OF SPECIFIC CAUSES OF STRUCTURAL FAILURES

(Airplane Structure)

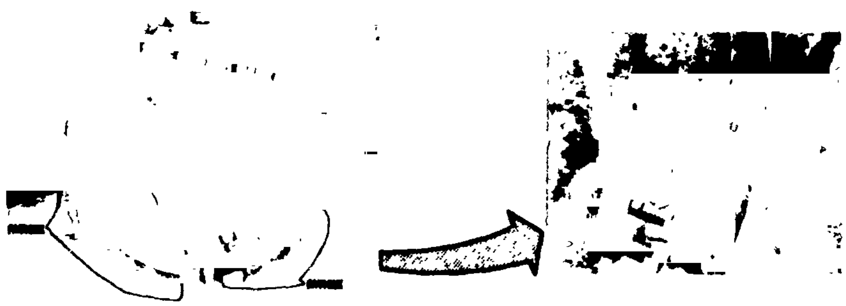
This chart shows several photographs of fracture origins in aircraft components. Two are commercial aircraft landing gear components that failed as a result of corrosion fatigue. One is a fatigue failure of a structural stiffener and the other a fatigue failure of a rotor blade socket. Probably one of the most common types of cracks observed in aircraft component failures has been the corner crack emanating from a hole like that shown in the structural stiffener.

LANDING GEAR
CYLINDER FAILURE



STRUCTURAL STIFFENER
FAILURE ORIGIN

LANDING GEAR
AXLE FAILURE

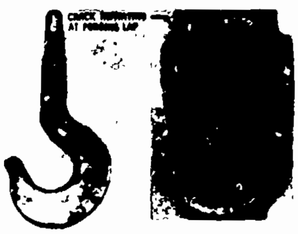


HELICOPTER ROTOR BLADE SOCKET FAILURE

SOME EXAMPLES OF SPECIFIC CAUSES OF STRUCTURAL FAILURES

(Industrial Equipment)

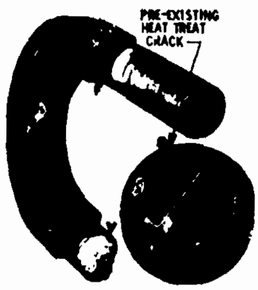
Structural failures have not been uniquely an aerospace problem. This chart illustrates several failures of industrial equipment. Again it should be noted that pre-existing flaws are the predominant cause of failure.



CRANE HOOK
FAILURE ORIGIN

CRANK SHAFT FAILURE

4350 STEEL SHAFT
FAILURE ORIGIN



4340 STEEL YOKE
FAILURE ORIGIN

CRANK PIN ON AN
800 TON
MECHANICAL PRESS

HYDROGEN BLISTERS
IN A STEEL
SULPHURIC ACID TANK

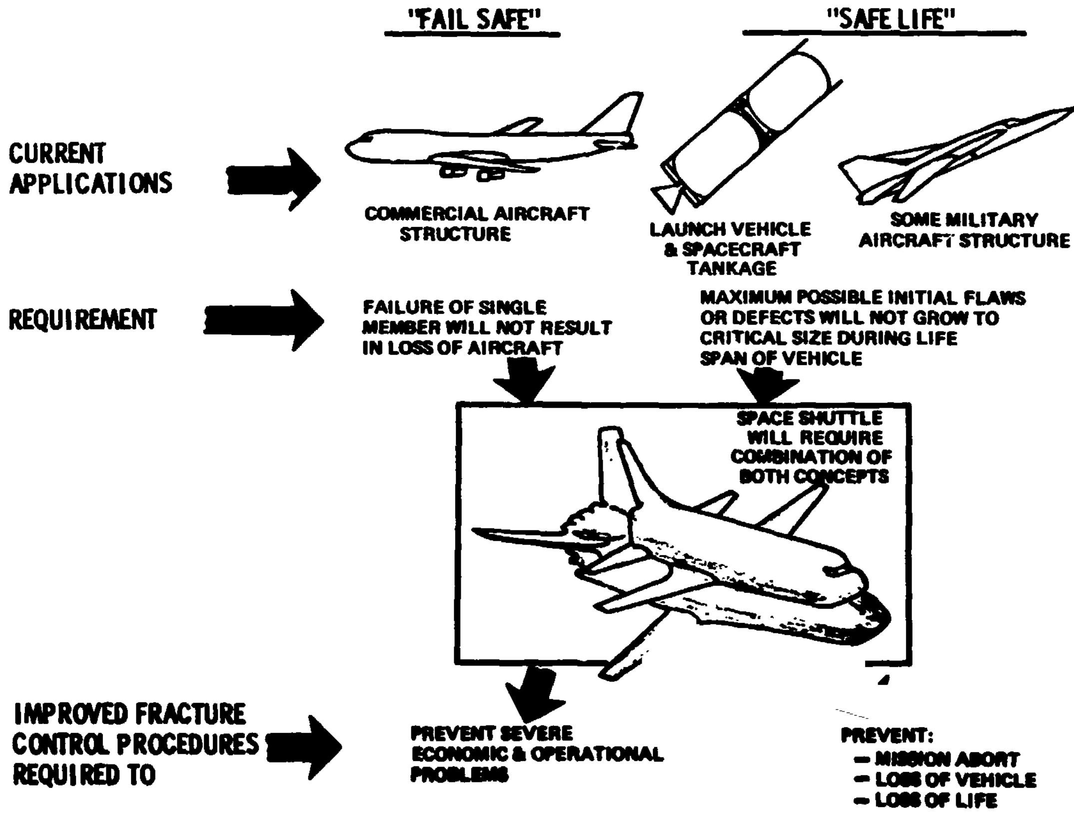
Reference: T. J. Dolan, U. of Ill. T&A.M. Report #682

DESIGN PHILOSOPHY

("Fail Safe" or "Safe Life"?)

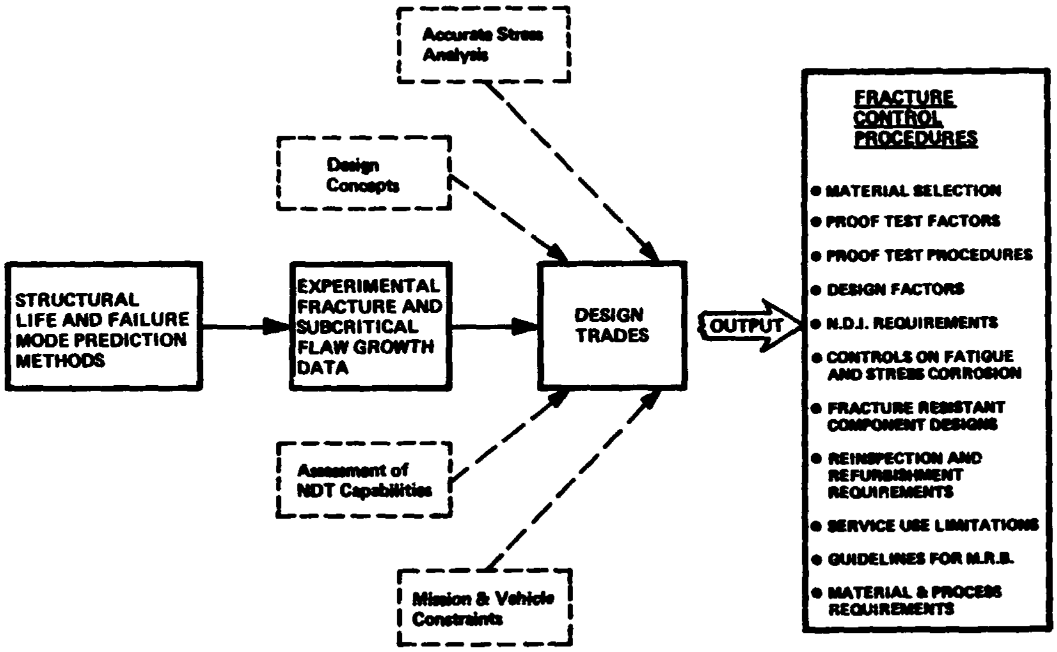
There are currently two basic design philosophies or criteria in the design of aerospace structure that are intended to provide assurance against structural failure. They are the "Fail Safe" criteria and the "Safe Life" criteria. The requirement for "Fail Safe" design is that the failure of a single structural member, for example, "a wing spar, skin panel, fuselage stringer, etc" shall not result in failure of the vehicle. The recent failure of the C-5A lower wing skin panels, main landing gear component failures, and commercial airplane upper wing skin cracking problems are examples of "Fail Safe" structure. These failures have not resulted in the loss of aircraft but obviously have imposed severe economic and operational problems. Most launch vehicles and spacecraft pressure vessels are "Safe Life" structures. They are highly stressed single load path structures where failure can result in abort of the mission, loss of the vehicle, and/or loss of life. Some major components on several military aircraft have also been designed to be "Safe Life". The F-111 wing pivot and carry-thru structures are examples of single load path structures that were intended to be "Safe Life" but have experienced some difficulties in attaining this goal.

It is my opinion that the Space Shuttle Vehicle will require a combination of both design concepts. The main and auxiliary tankage will undoubtedly have to be primarily "Safe Life" structure. However, it is likely that the wing, empenage, and perhaps part of the fuselage can be designed to be "Fail Safe". In order to achieve "Safe Life" and also prevent expensive component fractures in "Fail Safe" structures, it is necessary to assure that maximum size initial flaws that could be in the structure will not reach critical size during the required life span.



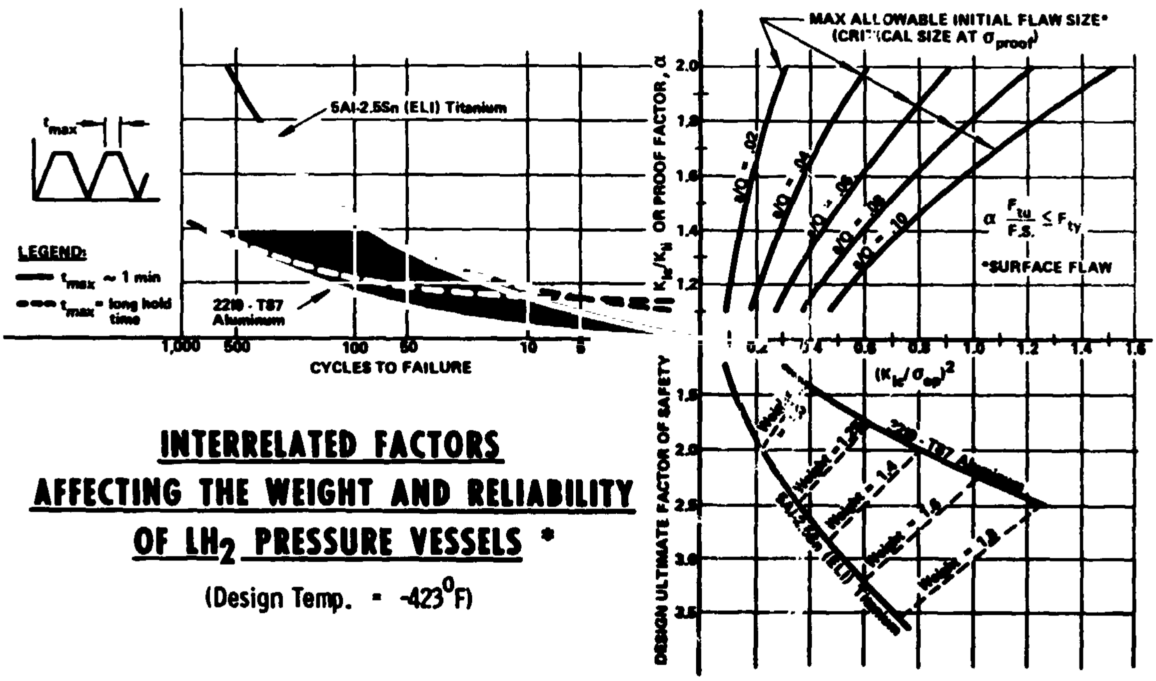
THE ELEMENTS OF EFFECTIVE FRACTURE CONTROL

The elements of failure prevention for effective fracture control are illustrated in this chart. The three main requirements (or needs) are: 1) accurate structural life and failure mode prediction methods, 2) experimental fracture and subcritical flaw growth data for use with these methods and 3) design trades which integrate the many other design considerations shown. The output of these studies or trades should be the best possible material selection, proof test factors, design factors, NDI requirements, etc.



EXAMPLE OF A DESIGN TRADE

This chart shows a relatively simple example of a LH_2 pressure vessel design trade study which I have presented several times in the past and have explained in detail in Reference (*). It shows the relationships between cyclic life, proof test factors, design ultimate factors, allowable initial flaw sizes, and relative tank weights for two candidate tankage materials. The significance of the chart can best be illustrated by selecting a specific required cyclic life and then comparing the required proof test factors, allowable flaw sizes, and relative weights of the aluminum and titanium vessels as a function of the design ultimate factor of safety. This is done in the reference.



INTERRELATED FACTORS AFFECTING THE WEIGHT AND RELIABILITY OF LH_2 PRESSURE VESSELS *

(Design Temp. = -423°F)

*THICK WALLED (i.e. $(a/Q)_{cr} < t$)

*NASA Space Vehicle design criteria monograph "Fracture Control of Metallic Pressure Vessels", May 1970

SOME OBSERVATIONS ON CURRENT STATUS

This chart presents some general observations as to the status (or the state-of-the-art) of several requirements for effective fracture control. As seen from the chart, there is a need for further improvement in the structural life and failure mode prediction methods, and a need for additional experimental fracture toughness and subcritical flaw growth data (cyclic and sustained load) for candidate structural materials, particularly at elevated temperatures. Boeing's current data on steels, aluminums and titaniums are for room temperature and lower. Current NDI procedures are not considered to be capable of reliably detecting all potentially dangerous flaws and this in turn has lead to further emphasis on the use of a properly designed proof test for high strength pressure vessels. The proof test is being used as inspection method for some high strength steel structure in the F-111 aircraft. The need for the use of the proof test on general airframe structure in the future is uncertain. Comprehensive documentation of fracture control procedures is needed.

REQUIREMENT (OR NEED)	STATUS
METHODS FOR PREDICTING FAILURE MODES AND MINIMUM STRUCTURAL LIFE	NEED IMPROVEMENT IN ORDER TO HANDLE STRUCTURES WITH COMPLEX GEOMETRIES, LOADING & ENVIRONMENT HISTORIES
RELIABLE FRACTURE TOUGHNESS AND SUBCRITICAL FLAW GROWTH DATA	CONSIDERABLE DATA AVAILABLE (More High Temp. Data Needed)
NDI CAPABILITIES	NOT ADEQUATE FOR HIGH STRENGTH FRACTURE SENSITIVE MATL'S
PROOF TESTING	USED AS NDI METHOD FOR PRESSURE VESSELS VERY LIMITED USE ON OTHER AIRFRAME STRUCTURE
DOCUMENTATION OF FRACT. CONTROL PROCEDURES & METHODS	FRAGMENTED

SOME AREAS OF FRACTURE RESEARCH IMPORTANT TO THE SPACE SHUTTLE

This chart summarizes a number of areas of fracture research that I feel are important to the Space Shuttle and is considered to be self explanatory.

- RESOLVE UNCERTAINTIES WITH REGARD TO GROWTH OF FLAWS ... PRESSURE VESSELS & UPDATE LIFE PREDICTION PROCEDURES
- DEVELOP IMPROVED LIFE PREDICTION PROCEDURES FOR GENERAL AIRFRAME STRUCTURE.
- EXPAND THE PRESENT QUANTITY OF FRACTURE TOUGHNESS AND SUBCRITICAL FLAW GROWTH DATA FOR MATERIALS BEING CONSIDERED FOR USE ON SPACE SHUTTLE
- PERFORM AN ASSESSMENT OF THE SUSCEPTIBILITY OF SPACE SHUTTLE MATERIALS TO STRESS CORROSION CRACKING IN THE ANTICIPATED TEST AND SERVICE ENVIRONMENTS
- PERFORM APPLICATION AND TRADEOFFS BETWEEN CANDIDATE TANKAGE MATERIALS, DESIGN FACTORS, PROOF FACTORS, LIFE REQUIREMENTS AND PROBABLE INSPECTION CAPABILITIES TO DETERMINE MINIMUM WEIGHT DESIGNS
- CONTINUE BASIC AND APPLIED RESEARCH IN SUCH AREAS AS:
  - Corrosion Fatigue
  - Thermal Cracking
  - Fracture Resistant Component Design
  - Failure Of Composites
  - Stress Intensity Factor Development

N70-42981

STRUCTURAL DESIGN AND ANALYSIS OF THE SPACE SHUTTLE

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ABSTRACT

Quoting from NASA sponsored ILRVS studies, characteristics of the space shuttle are discussed that make the use of advanced design and analysis methods imperative. Special features of the vehicle are outlined together with structural and thermal problem areas requiring the use of advanced methods. Existing methods and analytical tools are reviewed, and a survey is made of areas in need of improvement. The basic tool is matrix analysis based on the finite element approach. There is a need for updating available programs by including improved or refined finite elements. One example shows the advantages of such refined finite elements. A second example deals with a problem of random structural response solved with the help of finite elements.

1. NEED FOR ADVANCED METHODS

Recent NASA sponsored studies of the Integrated Launch and Reentry Vehicle have revealed three principal characteristics that make necessary the use of advanced methods in the design and analysis of the structure and thermal protection system of the space shuttle. One is the fact that weight estimates not based on experience with actual vehicles could easily be exceeded. The studies show that a mere 15 percent increase in inert weight of both stages would wipe out the entire payload capability. Further, the mission profile includes a diversity of regimes never encountered before: piggy-back vertical launch, space flight, atmospheric entry, and flight with speed range from hypersonic down to airplane-type landing. Third, the goal of using the same vehicle 100 times with minimum refurbishment implies a long life of all components; this may be difficult to achieve in view of the environmental degradation of these components.

In more detail, some of the structural problems calling for advanced methods are:

- non-cylindrical shape of both stages, requiring special design of propellant tanks for optimal utilization of volume;
- unexplored phenomena of buffeting and flutter that may be encountered during launch and reentry;
- the highly nonlinear behavior of materials due to temperature dependence of elastic and thermal properties;
- various structural problems connected with the heat shield.

—WEIGHT CRITICALITY AND SENSITIVITY

—COMPLEXITY OF MISSION AND VARIETY OF REGIMES

—CONDITION OF RE-USABILITY

—NEW SHAPE FOR RE-ENTRY ENVIRONMENT

—UNEXPLORED FLUTTER AND BUFFETING PHENOMENA

—ANTICIPATED NON-LINEAR MATERIALS BEHAVIOR

—STRUCTURAL PROBLEMS OF HEAT SHIELDING

2. SAMPLING OF PROBLEM AREAS

To be more specific, we list some of the problem areas in the thermal and structural analysis.

For adequate analysis the structure with its thermal protection system must be represented by a model sufficiently accurate to reproduce the essential features, but not so encumbered with details as to make the large-scale calculation inefficient and difficult to interpret.

The analysis must take into account nonlinearities such as inelastic material behavior, temperature-dependent elastic and thermal properties, the occurrence of creep and of buckling phenomena associated with elevated temperatures and high rates of heating.

Since the vehicles are to be used to to 100 times, the effect of repeated loading and thermal cycling on the flightworthiness of the structure must be carefully taken into consideration to obtain a satisfactory structural life.

- ADEQUATE IDEALIZATION OF COMPLEX STRUCTURE INCLUDING THERMAL PROTECTION SYSTEM

- HIGH TEMPERATURE AND HIGH BUFFET LOADS WILL REQUIRE CONSIDERATION OF NON-LINEAR BEHAVIOR INCLUDING CREEP AND STRUCTURAL INSTABILITY

- RE-USABILITY DEMANDS ASSESSMENT OF RESIDUAL DEFORMATION BUILD-UP DUE TO CYCLING

3. ANALYTICAL TOOLS AVAILABLE

The basic tool for analyzing complex structures consists in using matrix calculus along with adequate computer hardware and software. The methods of finite differences and numerical integration are suitable only for relatively simple configurations. The preferred method requires the idealization of the structure as an assemblage of finite elements. Two versions of the finite element method are in general use: the displacement (or stiffness) method, and the force method using redundancies. There is a wide choice of elements. The kind of elements used has a decisive bearing on the quality and size of the analysis. The simplest elements are not always the most suitable.

Efficient programming is facilitated by employing general programs designed for that purpose. Such as the few listed on the slide.

In addition, most of the aerospace companies have developed their own programs and would, no doubt, prefer to use them as much as possible.

Finally, there is a large number of programs designed to solve specific structural problems. The analyst should be able to take advantage of the best of them.

MATRIX METHODS OF ANALYSIS

- FINITE ELEMENT METHODS:
  - DISPLACEMENT METHOD
  - FORCE METHOD
- FINITE DIFFERENCES (REGULAR GEOMETRIES ONLY)
- NUMERICAL INTEGRATION

GENERAL COMPUTER PROGRAMS

- NASTRAN, FORMAT, SAMIS, ASKA

COMPANY HOUSE PROGRAMS

SPECIAL PURPOSE PROGRAMS

- ICES-STRUDL (FRAMED STRUCTURES)
- SABOR (SHELLS OF REVOLUTION)
- BOSOR (BUCKLING OF SHELLS OF REVOLUTION)

AND MANY OTHERS

4. NEED FOR METHOD DEVELOPMENT

In view of the advanced nature of the project and of the magnitude of the structural problems anticipated, the existing methods cannot be expected to be satisfactory. Hopefully, in most cases it may be sufficient to stretch and improve the existing methods. Let us list those areas in which such developments seem the most necessary.

In the analysis of heat transfer and of resulting transient thermal stresses in irregularly shaped bodies (possibly with temperature dependent and anistropic conductivity) the finite element method offers great promise. This is true in particular of the various details of the heat shield, such as joints, access doors and cut-outs, supports and edge members.

The prediction of random response of complex structures requires a good representation of the structure. Here too, the use of refined finite elements should be a key factor in obtaining good solutions.

A variety of optimization problems will have to be tackled, and in each case it will be necessary to choose the most appropriate method among the large number available. A few examples of such problems are listed on the slide.

Finally, in our experience with finite element methods at the N.R.C., we have found that the brute force approach using a large number of simple elements is not always the most efficient one.

FINITE - ELEMENT METHOD APPLIED TO HEAT TRANSFER AND TRANSIENT THERMAL STRESS ANALYSIS

APPLICATION OF FINITE-ELEMENT METHOD TO DETAIL DESIGN AND ANALYSIS OF DISCONTINUITIES IN HEAT SHIELD

IMPROVEMENT IN PREDICTIONS OF RANDOM RESPONSE

OPTIMIZATION STUDIES

- DESIGN MINIMIZING THERMAL STRESSES
- WEIGHT vs. FAIL-SAFE DESIGN
- TRADE-OFF BETWEEN MATERIALS
- NATURAL FREQUENCY CONSTRAINTS
- AERO-ELASTIC CONSTRAINTS

INCLUSION OF REFINED FINITE ELEMENTS IN EXISTING GENERAL PROGRAMS FOR MORE EFFICIENT ANALYSIS

5. ADVANTAGES OF REFINED FINITE ELEMENTS

The advantages of refined finite elements are three-fold:

(a) In comparison with first generation elements, equal accuracy can be obtained with fewer overall degrees of freedom. Since computing time and cost both increase with the number of overall degrees of freedom, refined elements promise economies. This is particularly important in optimization studies where repeated analysis and re-design may be required.

(b) Reliable and direct predictions of stresses are obtained. First generation elements, in contrast, generally give rather crude predictions of stress which, moreover, usually have to be doctored by semi-empirical smoothing and averaging procedures.

(c) Refined elements satisfy sufficient conditions to guarantee convergence to the mathematically "exact" solution. This not only bolsters confidence in a finite element analysis, but also elucidates the nature of the errors and makes error estimates possible. Furthermore, the rate of convergence is far superior to that for first-generation elements.

SMALLER OVERALL PROBLEM SIZE

IMPROVED PREDICTIONS OF STRESSES

RAPID CONVERGENCE ASSURED

6. REFINED FINITE ELEMENTS DEVELOPED BY N.R.C.

For the past two to three years a group within the Structures and Materials Laboratory, of the National Research Council of Canada, has pioneered the development of many refined finite elements. In most cases the development has been carried to the point where there are in being well-checked element subroutines which could be incorporated into an up-dated version of NASTRAN or other systems. Unfortunately the documentation has lagged behind the development of programs, so that only part of the work is contained in readily available references.

The elements have been applied to studies of (a) random response to jet noise of an integrally stiffened panel, (b) natural modes of tapered cantilevered fan blades, (c) supersonic flutter of rectangular panels and cantilevered delta fins, (d) stresses around cut-outs in cylindrical shells, among others.

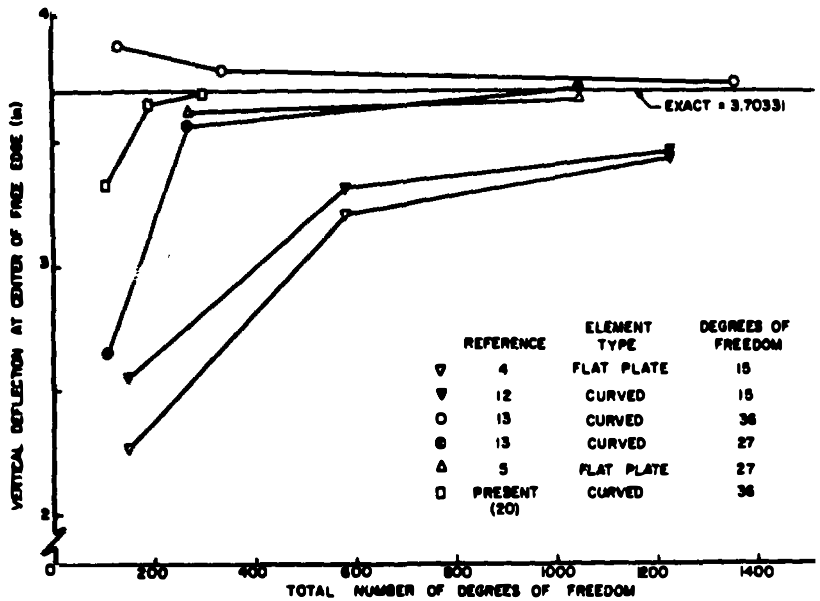
- TRIANGULAR PLATE BENDING ELEMENT
- TRIANGULAR PLANE STRESS ELEMENT
- TRIANGULAR SHALLOW SHELL ELEMENT
- TRIANGULAR CYLINDRICAL SHELL ELEMENT
- CURVILINEAR TRIANGULAR ELEMENTS
- TRIANGULAR DEEP SHELL ELEMENT (IN PROGRESS)

7. COMPARISON OF ELEMENTS

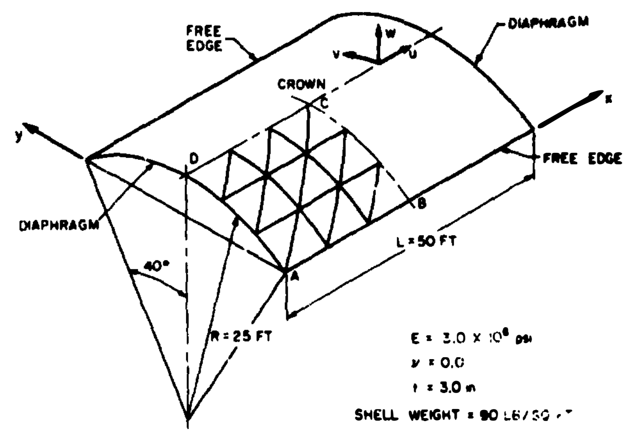
The illustration shows a civil engineering problem which has been analysed using various finite elements. The problem provides a searching test of finite elements for shells since both membrane and bending actions are important. The cited references represent previous analyses of the same problem using various single flat plate and curved shell elements. The great improvement in efficiency obtained by the use of refined elements (shown in the illustration by the square symbols) should be noted.

Ref.: Cowper, G.R., Lindberg, G.M., and Olson, B.D., "A Shallow Shell Finite Element of Triangular Shape", International Journal of Solids and Structures, in press.

STRUCTURAL DESIGN AND ANALYSIS OF SPACE SHUTTLE



CYLINDRICAL SHELL ROOF PROBLEM COMPARISONS



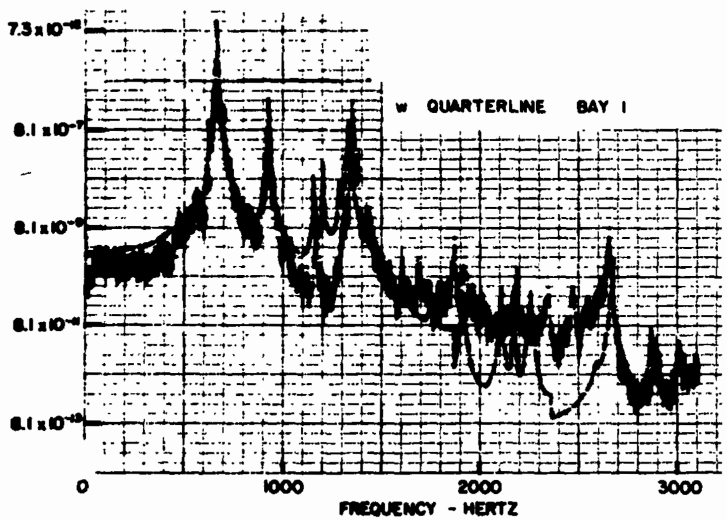
8. RANDOM RESPONSE STUDIES

The integrally stiffened panel shown is an example of built-up aerospace structure. The panel is represented by 18 degree of freedom trinagular plate bending elements*, and the stiffeners, by beam bending and torsional elements. Complete conformity is provided between all elements. The free vibration modes and frequencies are thus predicted and then used in a modal analysis of the panel's random response to jet noise. A consistent finite element method for calculating the modal force cross spectral matrix is developed. The figure shows the comparison between a predicted (dashed) and measured displacement spectra on the panel.**

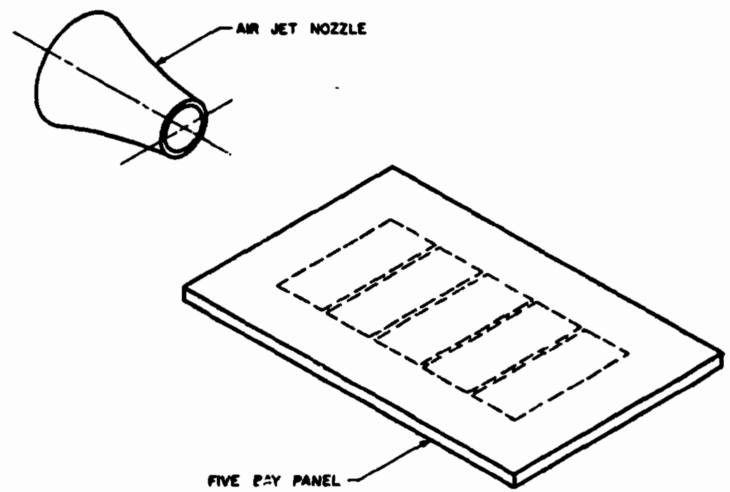
*Cowper, G.R., et al, "Static and Dynamic Applications of a High Precision Triangular Plate Bending Element", AIAA J., Vol. 7, No.10 1969, pp. 1957-1965.

**Olson, M.D., and Lindberg, G.M., "Free Vibrations and Random Response of an Integrally Stiffened Panel", Presented at the Symposium on Current Advances in Sonic Fatigue, The University, Southampton, England, July 6-9, 1970.

STRUCTURAL DESIGN AND ANALYSIS OF SPACE SHUTTLE



EXPERIMENTAL AND THEORETICAL SPECTRA



APPLICATIONS OF ADVANCED COMPOSITE MATERIALS

TO SPACE SHUTTLE STRUCTURES

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SUMMARY

The NASA programs for composite space shuttle structures are reviewed. Programs at MSFC, LeRC, LaRC are presently focused on beam, truss, stiffened plate and pressure vessel applications. The cost-weight advantages of the concept of reinforcing metals selectively with composite materials are discussed. An assessment of current composite technology is made and guidelines for composite applications are proposed.

INTRODUCTION

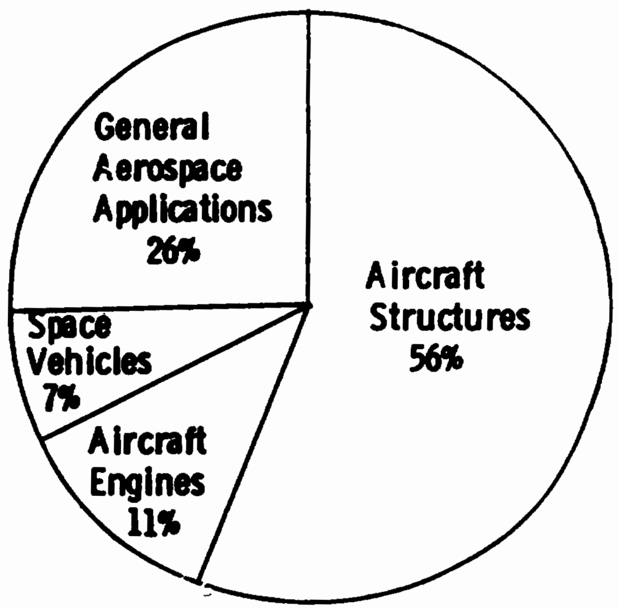
The NASA programs for composite materials applications for the space shuttle system are reviewed. Composite technology is assessed. Based on the present shuttle schedule, guidelines for composite structure development are suggested.

GOVERNMENT COMPOSITE PROGRAMS

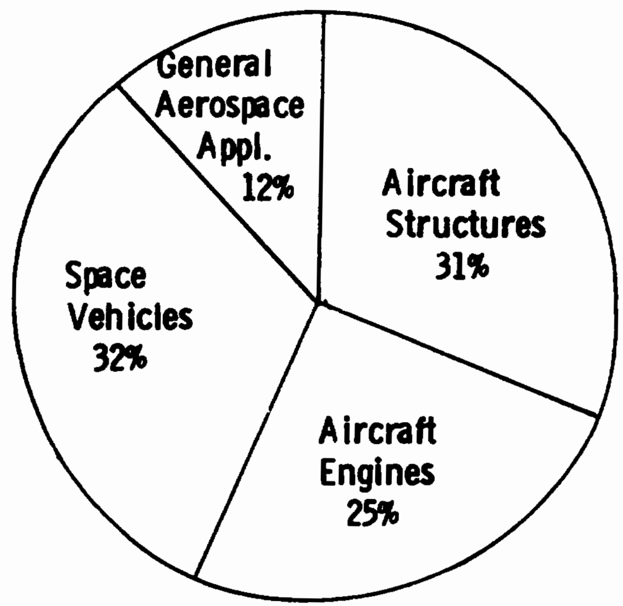
The space shuttle system offers an opportunity to obtain a return from the U.S. Government's large investment in composite materials. Since 1960, this investment has accumulated to over 400 million dollars, with FY 70 spending estimated at 22 million. The present Government effort has been keyed largely on USAF requirements and, therefore, a large portion of the effort has been directed toward military aircraft structures. The majority of design experience in Government and industry has been obtained on demonstration hardware for wing, tail and rotor structures. The information generated in the DOD programs must be exploited in composite designs for the shuttle. NASA's composite programs have been more diversified, and in FY 70 amounted to roughly 20 percent of total Government spending. The NASA programs have been research-oriented, and have not focused, up to this point, on development of hardware prototypes.

FY 1970 COMPOSITE STRUCTURES AND MATERIALS RESEARCH PROGRAMS

ALL U. S. GOVT. ($21,885K)



NASA ($3585K)



PRESENT NASA COMPOSITE PROGRAMS

Three NASA Centers are currently involved in composite programs associated with the space shuttle: Marshall, Lewis and Langley. Professional man-years of effort reflect total in-house composite programs -- not just shuttle programs, but much of this effort is directly translatable into shuttle structures. Activities at MSFC are associated with the fabrication and test of large structural components. Problems in detail design, materials development and reliability in engine support structure for the shuttle are being investigated. Activities at LeRC have been associated with metal-matrix technology for engine applications, and with filament-wound tank technology. For shuttle, the center will focus on the development of large filament-overwrapped propellant and gas storage tanks. Activities at LaRC have been associated with the evaluation of resin-matrix materials and the development of the concept of composite reinforced metal structures for both aircraft and space vehicles. The center will attempt to exploit this concept on the orbiter structure of the space shuttle system. In addition, the Manned Spacecraft Center which has not been involved previously in composite activities, will be releasing a request for proposals in the near future for lightly-loaded graphite honeycomb panels.

PRESENT NASA COMPOSITE PROGRAMS

CENTER	ACTIVITY	SHUTTLE FOCUS	PMY
• MSFC	Fabrication and NDE, polymer development, space vehicle applications	Thrust Structure	10
• LeRC	Metal and resin-matrix technology, aircraft engine and filamentary composite tank applications (filament-wound tanks, several contractors)	Tankage	22
• LaRC	Resin-matrix technology, aircraft and space vehicle design concepts (composite reinforced metal structures, Boeing)	Orbiter Structure	15

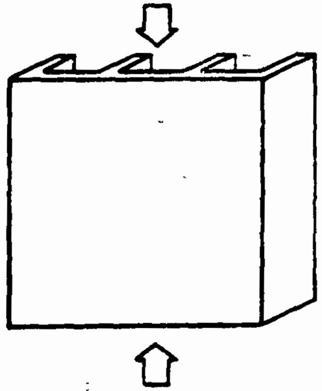
MSFC PROGRAM

The MSFC effort is focused on thrust structure. Two elements are being considered: stiffened panels for the thrust structure skirt, and the beam gridwork necessary for multiple engine supports. Materials considered include graphite-epoxy, boron-epoxy, boron-aluminum and titanium alloys. Practical fabrication and detail design problems in large structural sections will be considered. Present in-house studies are directed to design of beams to transfer large concentrated engine loads and the materials characterization of boron-aluminum, graphite epoxy and graphite polyimide systems. Two contracts were recently awarded; one to Martin-Marietta for the design and fabrication of a large stiffened graphite-epoxy panel for test at Marshall, the other to General Dynamics/Convair for development of design data for graphite-epoxy and graphite-polyimide composites. Future plans will include development of NDE standards for composite materials, and in-house designs of panels, shear webs and truss structures.

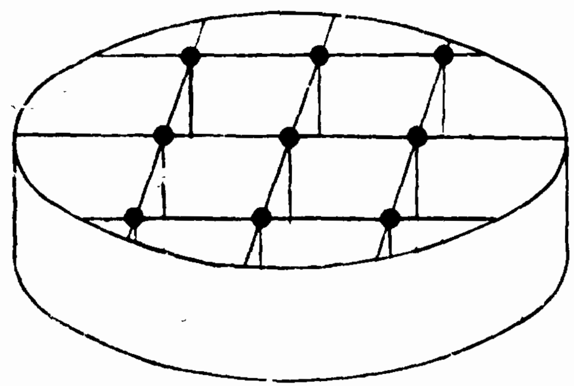
MSFC PROGRAM

ELEMENT

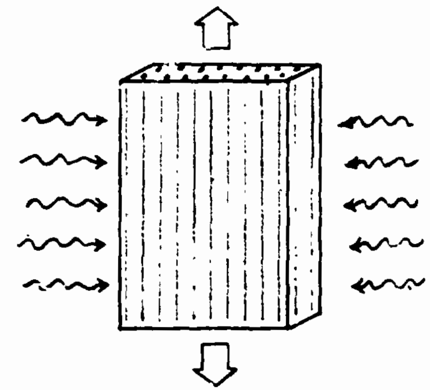
Stiffened Panel



Beam Gridwork



Composite Materials Characterization



SHUTTLE APPLICATION

Stiffened Panel	Beam Gridwork	Composite Materials Characterization
Thrust structure skirt	Engine thrust structure support	Materials selection

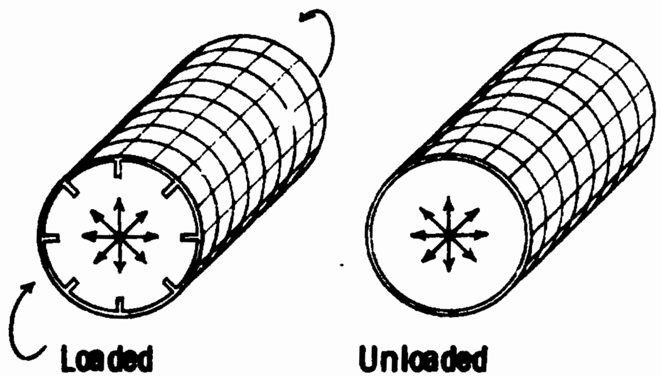
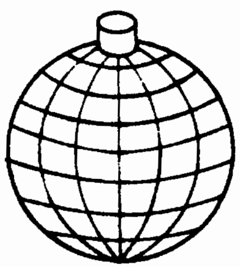
MATERIALS

Stiffened Panel	Beam Gridwork	Composite Materials Characterization
G-E, B-E, B-Al	B-E and Ti G-E, B-Al	G-E, G-PI, B-Al

LeRC Program

The program will consider two types of pressure vessels: large, low pressure vessels for use as cryogenic propellant tankage and smaller, high pressure vessels for use as pressurant gas storage tanks. Materials being considered include graphite-epoxy, boron-epoxy, glass-epoxy and aluminum, titanium and nickel-base alloys. Current efforts are to evaluate the performance of subscale filament-wound vessels with thin metal liners. Studies will be made of the effects of cyclic loading on the burst strength of vessels at cryogenic temperature. Future plans include studies of tanks with load-bearing metal substructures, and the behavior of flawed liners under cyclic loading.

LeRC PROGRAM FOR TANKAGE

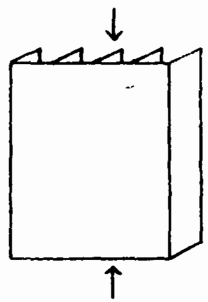
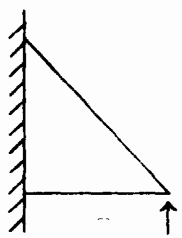
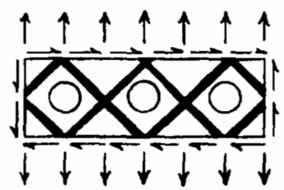
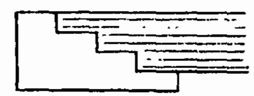
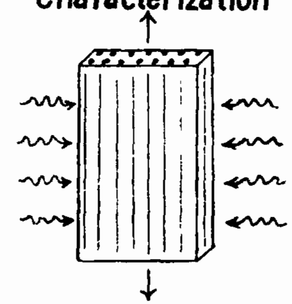
ELEMENT	SHUTTLE APPLICATION	MATERIALS
 Large, low pressure vessels	Cryogenic propellant tankage	G-E, B-E, Gl-E and Al, Ti, Ni
 Small, high pressure vessels	Storage bottles	G-E, B-E, Gl-E and Al, Ti, Ni

LaRC PROGRAM

The LaRC program is presently a broad research program in which the concept of selectively reinforcing metallic structures with composite materials is being developed. In addition, LaRC is continuing to perform in-house evaluations of most of the available composite materials systems. The selective reinforcement program includes studies of the stability of stiffened panels, the strength of tubular struts as truss members, and design concepts for stiffening cutouts. Detail design efforts have included studies of fatigue and static strength of stepped joints and the effects of residual stresses induced by laminate cures. The concept has been pursued contractually with Boeing's Commercial Aircraft Division and with parallel in-house studies. The selective reinforcement concept is believed to be applicable to shuttle structures, and some of the motivations for its use are discussed in the next two slides.

LaRC PROGRAM

ELEMENT

Stiffened Panel	Tubular Truss Work	Window Cutouts	Stepped Joints	Composite Materials Characterization
				

SHUTTLE APPLICATION

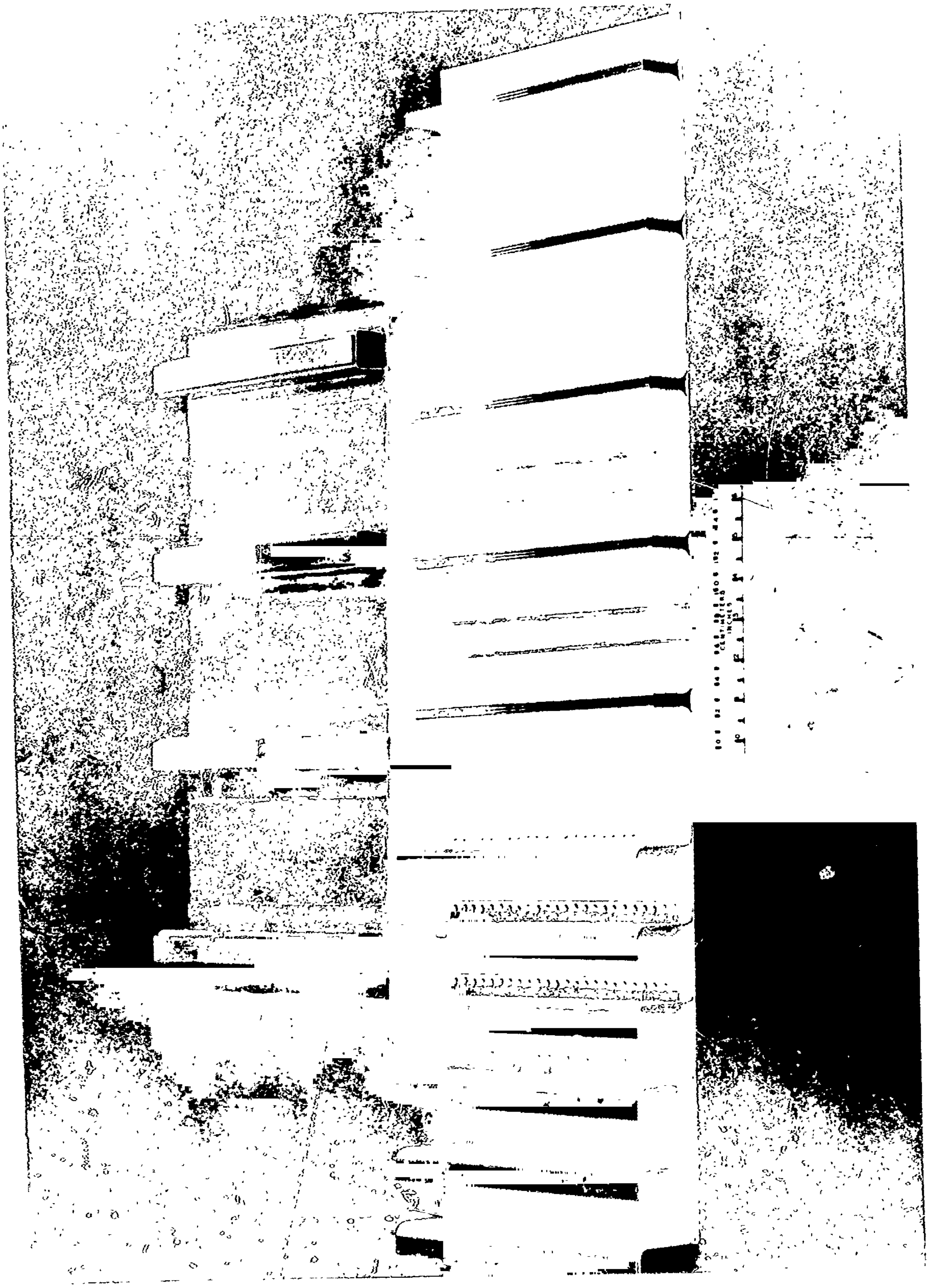
Fuselage or wing structure	Internal framework	Reinforced cutouts	Metal-to-composite joints	Materials selection

MATERIALS

B-E and Al B-E and Ti B-PI and Ti	B-E and Ti B-E and Al	B-E and Al	B-E and Al B-E and Ti B-PI and Ti	Gl-E, G-E, B-PI, B-E B-Al

STIFFENED PANEL CONCEPTS

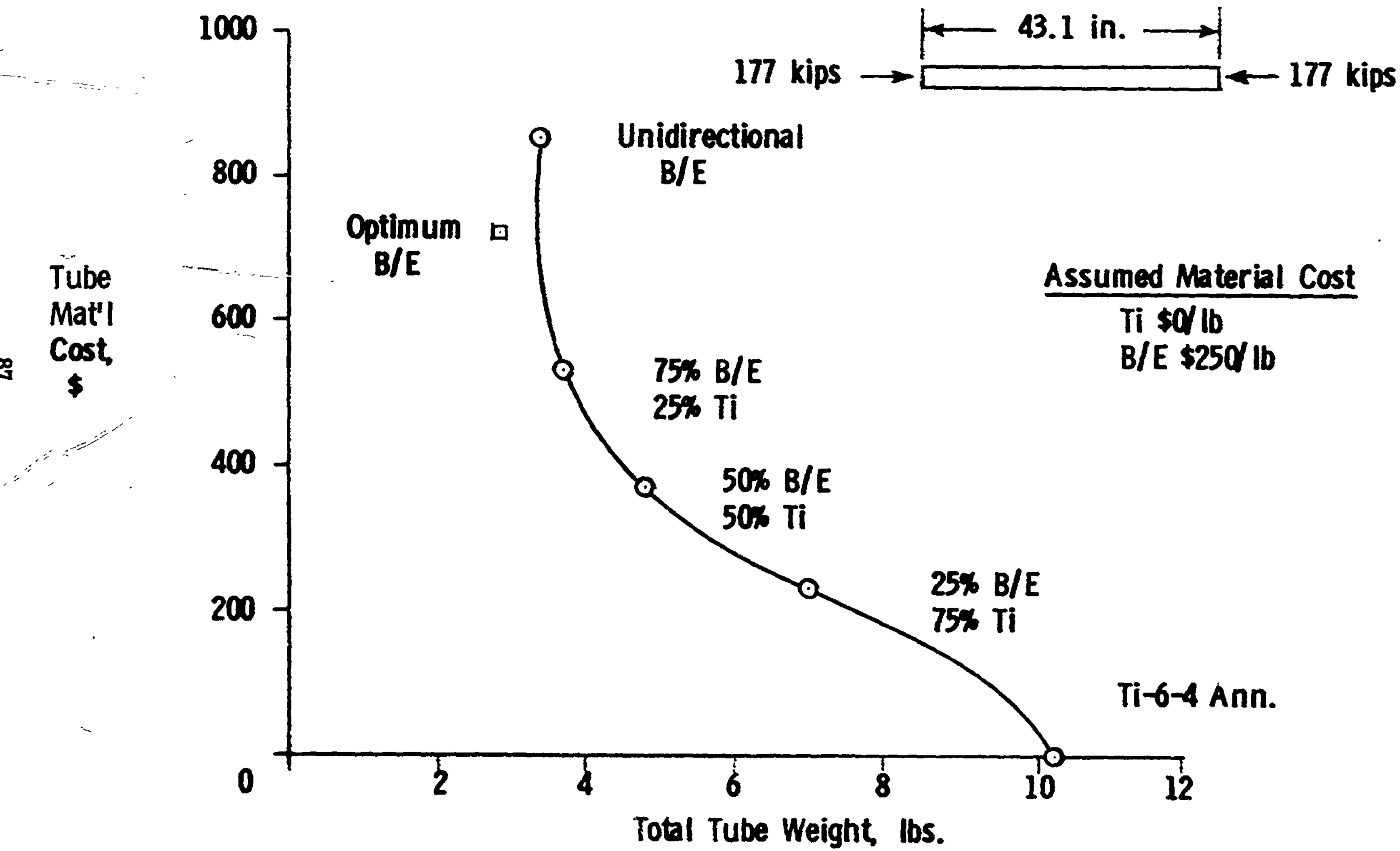
Some examples of compression panel concepts are shown in the slide. The top panel is titanium while the other two panels are aluminum. All three panels have been selectively reinforced with unidirectional boron-epoxy. The major feature of the selective reinforcement concept is that only a limited amount of composite material is required and it is used in an efficient manner. As shown in the top panel, the reinforced metal can be tapered to provide transitions between composite and metal, so that conventional joining techniques can be used for large structural sections. Studies of panels similar to those shown have suggested that weight savings of 20 to 25 percent might be realized for compressive structures for orbiter and booster fuselages.



COST EFFECTIVENESS

With the present high costs of advanced composite materials, it is apparent that cost-weight trade-offs must be performed on candidate composite material designs. A simple example of this kind of consideration for the selective reinforcement concept is shown in the slide. Large circular tubes, suitable for a 1/3-scale prototype of a thrust structure truss, have been optimized to carry the load shown. Using the assumed material costs shown and neglecting fabrication costs, the tube materal cost and weights of tubes with varying percentages of unidirectional boron-epoxy reinforcement overlayed on titanium are shown. The slope of this cost-weight curve can be used to judge at what point the $ per pound required to save weight becomes unacceptable. In the case shown, the curve becomes very steep as all-composite structures are approached. The optimum all-composite structure (a tube with 60 percent unidirectional filaments and 40 percent 45° filaments), shown as the square point, is lighter than the unidirectional column but still remains in a steeply rising cost regime. We anticipate that additional studies such as these may reveal that the selective reinforcement concept is a desirable interim method of achieving practical, cost-effective composite structures in the shuttle.

COST-WEIGHT COMPARISON OF TUBULAR COMPOSITE COLUMN



COMPOSITE TECHNOLOGY ASSESSMENT

Of the candidate high modulus composite materials for the space shuttle system, boron-epoxy appears to be the only on-the-shelf structural material; graphite-epoxy systems are emerging, but sufficient design data have not yet been generated. Experience with metal and polyimide matrices suggests that these materials are still not developed sufficiently for design. New adhesives for joining metal to composite parts are needed. Fabrication processes for manufacturing tubing and laminates from prepregged material appear to be under control. Only limited experience with winding graphite and boron has been obtained and this process needs further exploitation. Procedures for joining are still being investigated and reliable nondestructive evaluation of composite parts needs much development. The mechanis of composite structures with respect to elastic stress analysis and stability are reasonably well understood through orthotropic elasticity. Design procedures for estimating maximum strength of complex laminates are on shaky theoretical foundations and require further development. Limited experience with low cycle fatigue behavior suggests no serious problem areas for shuttle applications; creep of plies with skewed filaments will require mo[illegible]uation, however. Design concepts for struts and plates are reasonably well established; however, additional effort on detail design in beam structures and in the behavior of shell structures is needed.

COMPOSITE TECHNOLOGY ASSESSMENT

MATERIALS

- B-E, G-E
- Metal-Matrix, Polyimide*
- Adhesives*

FABRICATION TECHNIQUES

- Tubing
- Laminates
- Winding
- NDE*
- Joining*

MECHANICS

- Elasticity
- Stability
- Maximum Strength*

TIME-DEPENDENT LOADING

- Creep*
- Low Cycle Fatigue

DESIGN CONCEPTS

- Struts
- Beams*
- Plates
- Shells*

*Additional technology required

SUGGESTED GUIDELINES FOR COMPOSITE SHUTTLE STRUCTURES

Based on the present space shuttle schedule, the following guidelines are suggested. Only moderate temperatures can be accepted by available materials. Boron-epoxy, graphite-epoxy, and glass-epoxy are the only systems with which we have had design experience and, therefore, are the only candidates.

A design evaluation of these materials must consist of proper tradeoff studies between thermally protected metal and composite structures, and a large-scale design allowables program for shuttle-like structural elements. We should realize that composite structures, because of their cost and risk, are considered a backup system for the shuttle and will probably be used only if metallic structural weight is critical. However, we anticipate that weight will become critical and, therefore, composite structure studies must continue so that the technology will be ready when needed. Because of temperature limitations, cool primary structures, such as beams, trusses, and stiffeners are the most feasible applications. Any realistic application of composite materials in the shuttle will be determined by the capability of producing large structural sections. These sections must be cost effective, require realistic quantities of material and must have demonstrated reliability.

SUGGESTED GUIDELINES FOR COMPOSITE SHUTTLE STRUCTURES

1. Moderate thermal environment ($<$ 350° F)

2. Candidate materials - B-E, G-E, or Gl-E with Al or Ti

3. Evaluation:

   a. TPS-composite structures vs. TPS-metal structures tradeoff studies
   b. Comprehensive design allowables program

4. Composite structures are backup system; only used if weight is critical

5. Most feasible present applications are internal structures

6. Realistic applications will be determined by producibility of large structural sections - weight, cost, reliability

CONCLUDING REMARKS

The NASA composite programs for space shuttle structures have been reviewed. The programs are presently focused on beam, truss, stiffened plate and pressure vessel applications. Most of the available composite material systems are being evaluated. Realistic composite applications for the shuttle will depend on structurally efficient, cost-effective use of composite materials. The selective reinforcement of metal structures by composites offers an opportunity to achieve these goals. If the present stringent shuttle schedule is to be met, development must be focused on limited composite systems, and an exhaustive evaluation of shuttle-type structures must be made to establish a design technology.

N70-42983

SECONDARY STRUCTURES AND MECHANISMS – DESIGN TROUBLE AREA FOR THE SPACE SHUTTLE

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SUMMARY

The design of secondary structures and mechanisms for a reusable space shuttle can produce problems as serious as those involved in designing the primary structure and thermal protection system. Several events involving failure of secondary structures during the X-15 flight program have been selected to illustrate potential problem areas. Similar problems may be expected on future reusable vehicles until additional research provides adequate design information in these areas.

INTRODUCTION

The principal design effort for the space shuttle will rightfully concentrate on the selection of materials and on the structural concepts for the primary structure and the thermal protection system. However, secondary structure and mechanism could present as serious design problems as any encountered on the primary structure. Many of these problems are associated with the difficulties of predicting the extent and magnitude of local heating from small-scale models or from analysis. Reusable vehicles require many doors, hatches, control surfaces, and antennas that produce local heating and their own types of thermal design problems. During the X-15 program, which utilized the only vehicle to attain repeated flights to hypersonic speeds, events occurred involving mechanism and secondary structures that could have led to loss of the vehicle. Many of these events are typical of piloted reusable vehicles, and the same effects or problems can be expected on future vehicles, including the space shuttle. This paper describes some of these events and points out the types of problems that are expected to require special attention during the design of the space shuttle.

Figure 1

Deformation of door edges from local heating that is not adequately defined during analysis or simulation may result in the installation of inadequate edge seals. When seals fail to maintain closure, the hot airstream can enter compartments and damage interior structures and equipment. Thermal distortion of the nose gear door on the X-15 airplane during a flight to Mach 5.2 caused heat damage to aluminum tubing as shown in figure 1. The seal on the rear edge of the door could not prevent the hot airstream from impinging on the tubing and bulkhead. Bulkhead temperatures reached 520° F, which scorched paint and caused smoke in the cockpit.

HEAT DAMAGE TO ALUMINUM TUBING

Figure 2

Failure of door covers to close and latch after release of external equipment or vehicle separation can **result in extensive** internal damage, especially if the cover is intended to protect fuel disconnect fittings. **Figure 2 shows the damage** that resulted to the X-15 airplane when the cover door on a drop tank disconnect **fitting failed to close.** Titanium structures and instrument wiring was extensively damaged. This failure **is believed to be caused** by separation loads and local pressures in the door area at the time of tank release. **The door and surrounding** structure were coated with a thin layer of low-temperature ablative material.

DAMAGE FROM DOOR FAILURE TO CLOSE
DROP TANK DISCONNECT COVER

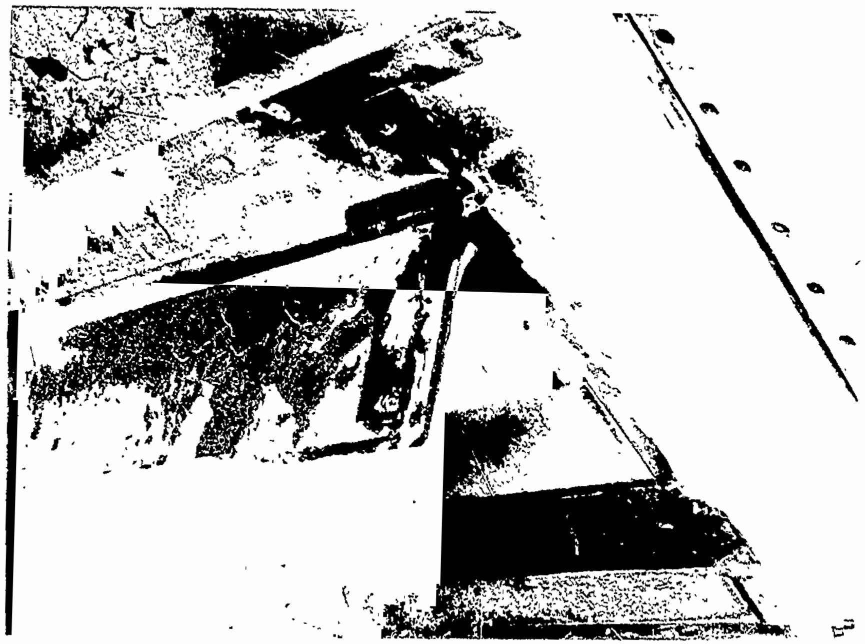

Figure 3

The lack of detailed local heating data together with insufficient analysis of local and overall thermal expansion can lead to problems with door latches and locks. Extremely high loads can be generated by differential expansion and distortion of doors and frames. Thermal deformation of the nose-gear door on the X-15 airplane which was not predicted by design analysis caused an overload of the nose gear uplock and allowed the gear to extend at Mach 4. 3, resulting in large changes in trim and drag. The open door and insulation and wiring in the compartment were damaged by the airloads and aerodynamic heating. The tires were overheated and failed on landing. Figure 3 shows the damage to the door and the blown tires. The wheels did not fail, and a safe landing runout was made. Failure of even one of the wheels would have caused loss of the aircraft.

DAMAGE TO NOSE GEAR DUE TO UPLOCK FAILURE

M = 4.3; ALTITUDE = 89,300 FEET

Figure 4

The inadequacy of small-scale models to provide the required data on local heating on the upper surface of the fuselage can present problems in the design of local structure. Figure 4 shows damage to the windshield glass following an X-15 flight to Mach 6.04. The failure started near the center of the upper edge of the glass. This damage was caused by local buckling of the glass retainer frame by thermal stresses. The buckle near the center of the upper edge produced a bending load on the glass and a local hot spot due to stagnation point heating. A change in the frame material and an increase in thickness were sufficient to prevent additional failure.

DAMAGED WINDSHIELD GLASS FOLLOWING FLIGHT TO M_{MAX} = 6.04

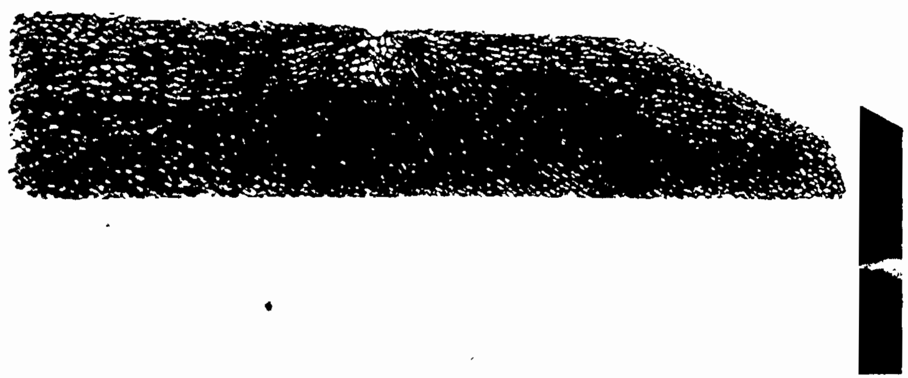

Figure 5

Although the local heating produced by shock impingement is known to be severe, the effect of this heating on local structure and fairings of full-scale vehicles cannot always be predicted. On the last flight of the X-15 airplane, a dummy ramjet package was attached to a fairing below the fuselage. During the flight, the heating caused by impingement of the bow shock from the test package resulted in severe damage to the fairing structure. Figure 5 shows, on the left, the ramjet package attached to the X-15 airplane before the flight. On the right is a photograph of the support fairing showing extensive heat damage to the leading-edge structure. This fairing was covered with ablative material designed to protect the structure from the expected heating.

DAMAGE TO SECONDARY STRUCTURE
BY SHOCK IMPINGEMENT

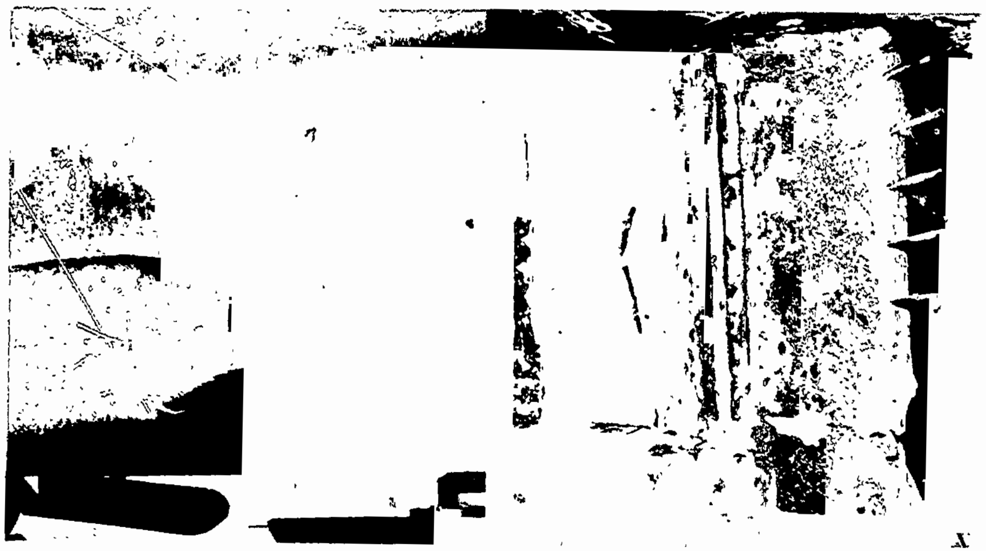

Figure 6

Reusable flight vehicles are characterized by discontinuities in the exposed surface caused by drains, access doors, antennas, and recessed fasteners. Local heating resulting from these discontinuities is difficult to predict from small models because of scale effects. The last flight of the X-15-2 airplane with ablative coatings produced visual evidence of the extent of local heating produced by several typical discontinuities. Figure 6 shows local heating from two types of discontinuities. The photograph on the left shows charred areas around and downstream from small depressions in the surface caused by fasteners around a small door and by plugged reaction control jet nozzles near the fuselage nose. On the right is a photograph of the local heating around a protruding antenna on the lower surface of the fuselage near the airplane center of gravity.

EFFECTS OF SURFACE DISCONTINUITIES
ON LOCAL HEATING

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N70-42984

APPLICABILITY OF AIR FORCE FLIGHT DYNAMICS LABORATORY

HIGH TEMPERATURE STRUCTURAL FACILITY

TO SPACE SHUTTLE COMPONENT TESTS

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Wright Patterson Air Force Base, Ohio

The necessity for thorough ground testing of manned space systems prior to flight is universally recognized. We are all well aware of the large investment in complex test facilities for Apollo hardware. None of these facilities can be used for the very vital structural integrity testing that will be required for space shuttles. Existing NASA or U.S. Air Force structural test facilities will have to be greatly expanded or totally new facilities will have to be constructed to do the job.

The test techniques for space shuttle structural testing also will require extensive study and development efforts, but many of the techniques and significant facility capability have been developed by the U.S. Air Force as a result of its long standing interest in the hypersonic flight regime. For more than a decade the Air Force Flight Dynamics Laboratory (AFFDL) and its predecessor organizations at Wright-Patterson Air Force Base have been active in furthering the technology required for the design, development and testing of hypersonic vehicles. All of this technology is directly applicable to today's space shuttle efforts.

Most of the USAF's structures test experience with lifting body, hypersonic vehicles has been obtained in the Structures Test Facility of the AFFDL (Figure No. 1).

This experience began in 1959 with the structural tests of the Boeing "Hot Structure" (Figure No. 2) and Bell "Double Wall Structure" (Figure No. 3). These were end items of a manufacturing methods demonstration program and they represented proposed X-20 (Dyna-Soar) structural concepts. From 1959 to 1961 these test programs were successfully accomplished. For the first time, programmed test temperatures simulating flight isotherms to 2000°F were applied to large structures (Figure No. 4). Predicted flight loads were simultaneously applied on a common time base and temperature and deflection measurements were successfully made in this temperature regime.

At the conclusion of these two test programs the AFFDL Structures Test Facility was committed to conduct all required structural tests for the X-20 (Dyna-Soar) Glider (Figure No. 5) and transition section. At the time, it was anticipated that test temperatures as high as 3000° might be required. The "Hot Structure" and "Double Wall Structure" test programs had indicated that extensive test techniques development would be required to push the state-of-the-art of heating from 2000°F to 3000°F. The same intensive development effort would be required for the loading and instrumentation systems and techniques. This development program was begun in 1962. The initial efforts were concentrated in the following areas:

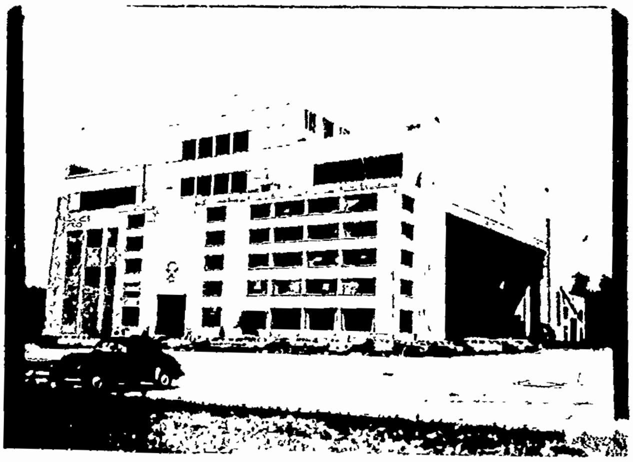

FIGURE 1

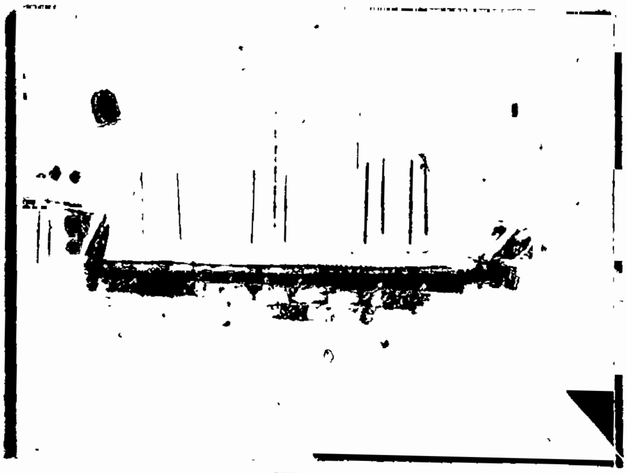

FIGURE 2

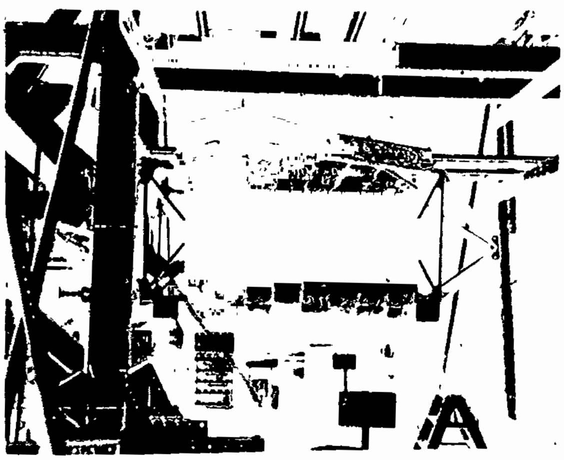

FIGURE 3

FIGURE 4

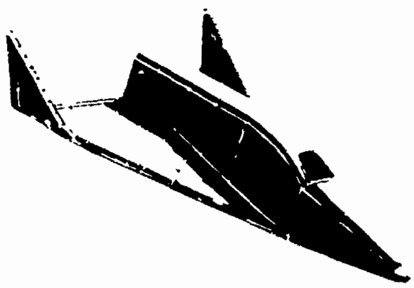

FIGURE 5

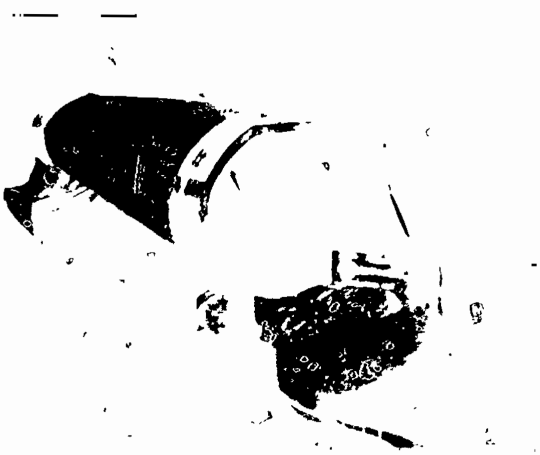

FIGURE 6

1. Instrumentation - These development efforts were primarily aimed at determining thermocouple types, sizes, and methods of attachment to coated refractory metal heat shields. High temperature strain gages for the anticipated test temperatures were not expected to be a likely development for use during these tests, but efforts to develop an 1800°F strain gage were pursued. A 3000°F deflection measuring was required.

2. Thermal Simulation

Infrared heater development capable of 3000°F simulation including heater spacing; power requirements; compatibility with instrumentation when large amounts of power are required; control thermocouple requirements per unit area, and control thermocouple paralleling techniques.

3. Load Simulation - Loads were to be applied to the cooler upper surface and would involve development of loading methods in a 1500°F environment.

4. General Problems - Such as the use of remote television coverage, support equipment problems, etc.

This development effort was conducted on a priority basis until the Dyna-Soar was cancelled in late 1963. Heater development efforts were continued in order to satisfy the 3200°F heating requirements for the Aeronca Thermantic Structure Test Program which began in December 1964 (Figure No. 6).

Flame heating techniques were considered because it was questionable if infra-heating could be used for temperatures over 3000°F. Flame heating was discarded because of poor temperature control and an unbearable noise level when heating large areas. The thermantic structure was internally cooled and was to be loaded internally during the tests. Due to problems with the structure and some of the test equipment, all of the test goals were not met. However, we did achieve test temperatures up to 3195°F on a heated area of 70 square feet (Figure No. 7). This demonstrated our capability of the accurately controlled heating of large structures to temperatures in excess of 3000°F and also taught us a great deal about the capabilities and proper handling or our large scale heating test support equipment.

Hypersonic structures testing was continued in 1965 with the tests of an X-20 Elevon (Figure No. 8) and an X-20 side window (Figure No. 9). Both of these programs were part of fall-out efforts relating to Dyna-Soar hardware already fabricated prior to Dyna-Soar program cancellation. All ascent and reentry loading and heating environments were successfully simulated using the test techniques developed for the previously described programs.

In 1966 the Structures Test Facility began its most ambitious effort in hypersonic structures testing. A cryogenic fuel carrying test article, built by the Martin Company for the Advanced Structural Concepts Experimental Program (ASCEP), was subjected to real-time ascent and reentry condition simulation. This was the largest

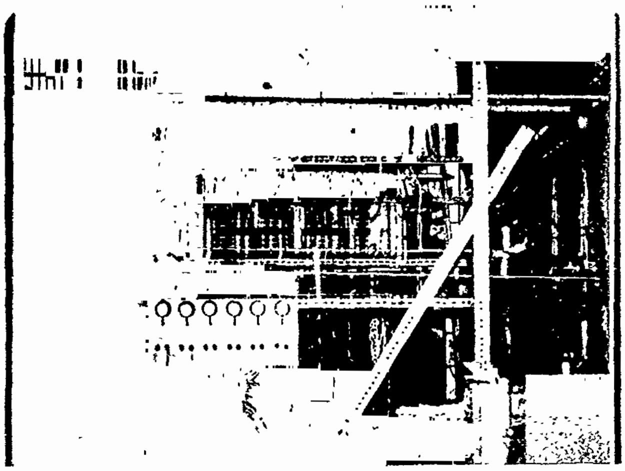

FIGURE 7

FIGURE 8

FIGURE 9

refractory metal structure ever assembled (Figure No. 10). Additional facility capability for the liquid nitrogen fuel simulant was required

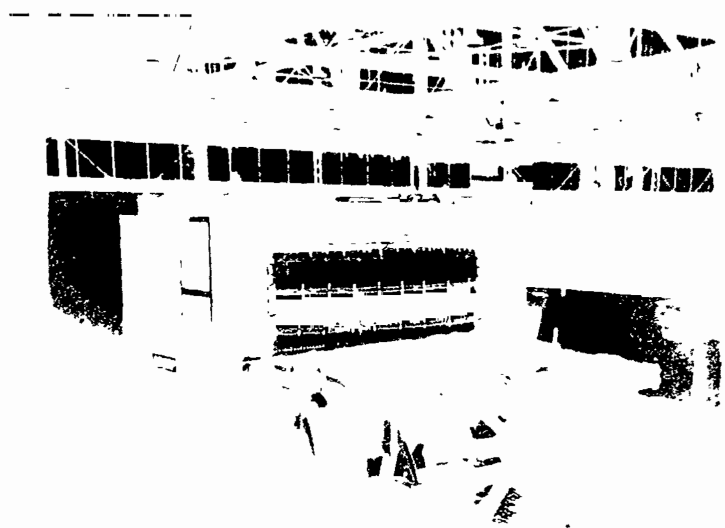

FIGURE 10

and developed for this program. The overall test requirements are shown in Figure No. 11.

- 60 HEATING CHANNELS
- 53 LOADING CHANNELS
- 5000 RADIANT HEAT LAMPS
- 19 LOAD VS. TIME PROGRAMS
- 12 HEAT VS. TIME PROGRAMS
- 800 GALLON LN_2
- 260 THERMOCOUPLES
- 31 DEFLECTION TRANSDUCERS
- 524 STRAIN GAGES (ROOM TEMP.)
- 2140°F MAX. LOWER SURFACE TEMP.
- CDC 1604B BACK-UP SAFETY

FIGURE 11

This has been a summary of the major hypersonic structure test programs conducted within the AFFDL Structures Test Facility. Reports with details of these programs are available at the AFFDL. A brief description of the Structures Test Facility and its present capability is now in order.

The AFFDL's Structural Test Facility was originally designed to conduct structural integrity qualification tests of United States Air Force aircraft. The main test floor is 251 feet in length, 170 feet in width and is constructed of 36-inch thick reinforced concrete. Load reaction fittings are built into the test floor in such a fashion that a 10,000 pound load can be reacted at each corner of a five-foot by five-foot grid covering the entire test floor.

The test jig members are standard structural steel shapes having 13/16 inch holes pre-punched or drilled on either three or six-inch centers. The jig members are bolted together.

Test jig construction and test article emplacement is by two 75-ton traveling bridge cranes which can work the entire test floor area. A third traveling bridge crane, of 150-ton capacity, covers a 110 foot by 66 foot area in the center of the test floor. The center bay area has a clear height of 121 feet, while the two end bays have clear heights of 86 feet. Test jig heights are normally limited to approximately 60 feet in order to permit travel of the 75-ton cranes over the entire test floor, although exceptions can be made to this limitation.

MAIN TEST FLOOR CAPABILITY

- UNOBSTRUCTED TEST AREA
  251 FT. LONG X 170 FT. WIDE X 86 FT. HIGH
  (121 FT. HIGH CENTER BAY)

- CRANE CAPACITY
  2 75-TON CRANES COVER ENTIRE TEST AREA
  1 150-TON CRANE COVERS ENTIRE HIGH BAY AREA

- STRUCTURAL FLOOR WITH 5 FT. X 5 FT. LOAD REACTION GRID

- "ERECTOR SET" TEST JIG MEMBERS

FIGURE 12

The size of the facility can be better appreciated when one realizes that a B-70 and B-58 test structure can simultaneously be accommodated within the test facility.

Elevated temperature testing can be conducted anywhere on the test floor unless cryogenic fuel simulation is required. In such cases, testing presently must be confined to a speci?l area on the test floor.

AFFDL STRUCTURAL TEST FACILITY MAIN TEST FLOOR

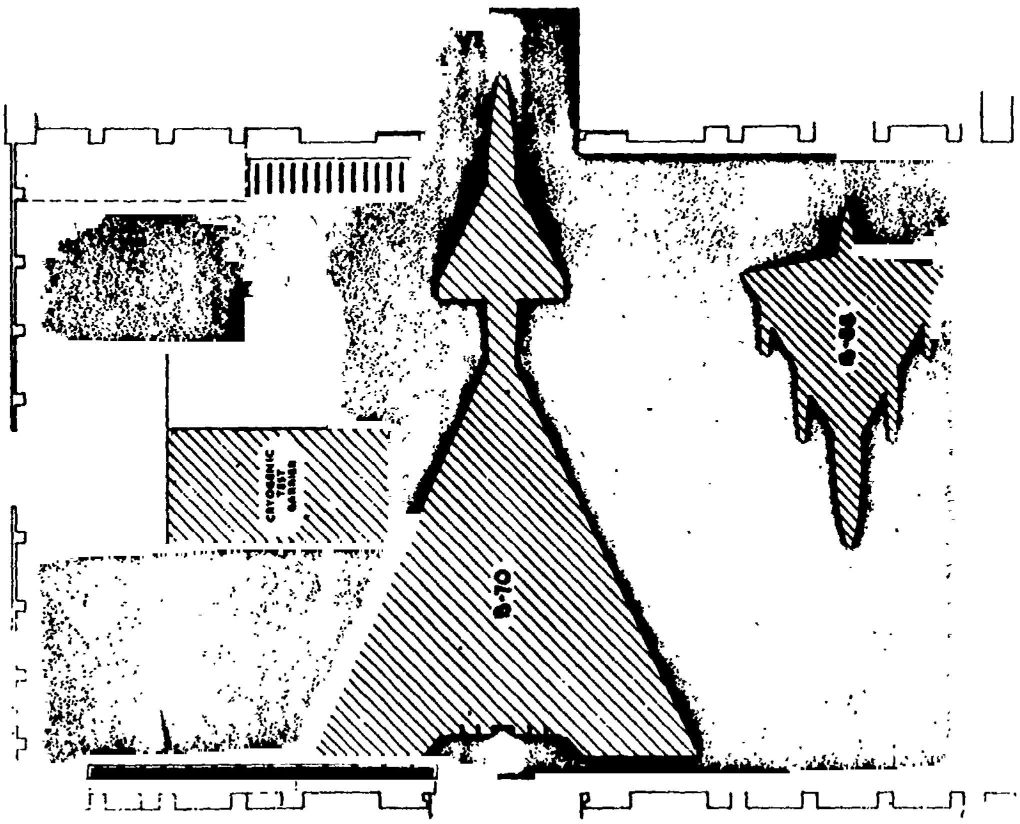

FIGURE 13

The aerodynamic and inertia load simulation systems used by the test facility are representative of present state-of-the-art within the testing community.

Whenever the Thermal Protective System (TPS) of a test structure prohibits penetration by loading hardware, and the TPS cannot be omitted, either concentrated loads must be applied to structural members extending beyond the region of test interest or the loading must be done internally. No fixed rule can be applied in such cases and each test structure must be considered individually.

If combined load and thermal simulation is a firm test requirement, it is desirable to involve test personnel very early in the design of the test structure in order to thoroughly explore all problems and arrive at compromises satisfactory to all interests.

LOAD ENVIRONMENT SIMULATION

- CENTRAL HYDRAULIC SUPPLY SYSTEM OF 200 GPM CAPACITY AT 5000 PSI
- 166 HYDRAULIC CYLINDERS RANGING IN CAPACITY FROM 2,000 LBS. TO 65,000 LBS. (TENSION) AT 5000 PSI, WITH STROKES RANGING FROM 12 INS. TO 36 INS.
- CONTROL BY PRESSURE (EDISON - OPEN LOOP) OR BY ELECTROHYDRAULIC CLOSED LOOP
- LOAD INTRODUCED BY RTV BONDED TENSION PADS (600°F) OR BY LOAD BOLTS.

FIGURE 14

Closed loop, force feedback, analog load control systems having the general characteristics shown on Figure 15 are used by the Test Facility for cyclic loading, or ascent and entry load history simulation on a real-time basis. Load magnitude and load rate limits can be set on each channel for protection of the test structure. Force feedback is supplied by one bridge of dual-bridge, resistance strain gage load cells in series with each hydraulic loading cylinder. The desired force input functions are obtained from various types of function generators to be described later.

The program input and force feedback signals are compared in a differential amplifier within the controller. Any difference is an error voltage which is amplified and applied to the servovalve associated with the loading cylinder, for directing hydraulic fluid in the proper direction to correct the load and reduce the error to zero.

LOAD ENVIRONMENT SIMULATION

- CLOSED - LOOP LOAD CONTROLLING EQUIPMENT
  - EXTERNALLY PROGRAMMED OR MANUAL CONTROL
  - FAILSAFE PROVISIONS
  - MAXIMUM CYCLING RATE - 10 CPS
  - TENSION - COMPRESSION
  - PRE-LOAD PROVISIONS
- 50 - CHANNEL SYSTEM:
  - 0 - 35,000 LBS. PER CHANNEL
- 26 - CHANNEL SOLID STATE SYSTEM:
  - 0 - 50,000 LBS. PER CHANNEL
  - INTERLOCK WITH 50 - CHANNEL SYSTEM
- 10 - CHANNEL IN-HOUSE LOAD SYSTEM

FIGURE 15

The simulation of an orbital entry thermal environment on even modest sized lifting body test structures requires a large quantity of electrical power. The power available to the test facility is shown on Figure 16.

Three phase, 60 cycle electrical energy is supplied at 69,000 volts to an electrical substation immediately adjacent to the test facility. Within the substation it is stepped-down to 6900 volts and transmitted to unit substations within the test facility. In turn, the unit substations supply 600 volt power to the 80 580KW, back-to-back ignitron power regulators.

Regulated power from the ignitron units, controlled by special purpose heat control computers, is available for radiant energy heating purposes at any of three locations on the main test floor.

If smaller power blocks are desired, the 60 KW (ignitron) or 25 KW (thyratron) power regulator/controllers may be employed. These regulator/controllers may be operated in either the temperature set-point or time-temperature control mode using external function generators (including the special purpose heat control computers).

TEMPERATURE ENVIRONMENT SIMULATION

- **ELECTRICAL POWER RATING -**
  - **20,000 KVA - CONTINUOUS**
  - **38,800 KVA - 2 HOURS**
  - **43,740 KVA - 30 MINUTES**
  - **[illegible]0,000 KVA - 5 MINUTES**

- **RADIANT HEATING POWER**
  - **80 580-KW POWER LOOPS OPERATING UNDER SPECIAL PURPOSE COMPUTER PROGRAMMING (HEAT FLUX OR TIME-TEMPERATURE) AND CONTROL**
  - **210 60-KW AND 440 25-KW POWER LOOPS OPERATING UNDER TIME-TEMPERATURE CONTROL.**

FIGURE 16

In the event that the flight vehicle carrys on-board cryogenic propellants it is important that the cryogenic heat sink and associated insulation be simulated within the test structure if the proper temperature gradients and thermal stresses are to be achieved.

At the present time the test facility uses liquid nitrogen as the simulated cryogenic fuel. Safety considerations require that such testing be conducted within a special, open top test enclosure. The cryogenic fuel simulation system is presently operational, but if the cryogenic fluid is to be withdrawn from the test structure's fuel tank (e.g., to simulate engine usage) the 750 GPM discharge pump and 8-inch line should be replaced with units specifically compatible with each test structure to avoid pump cavitation problems.

TEMPERATURE ENVIRONMENT SIMULATION

- **CRYOGENIC FUEL SIMULATION**
  - **10,000 GAL. LN_2 STORAGE**
  - **30 FT. X 60 DFT. X 20 FT. TEST ENCLOSURE**
  - **OUTSIDE LN_2 DUMP PIT**
  - **AUTOMATIC CONTROL AND SEQUENCE VALVES**
  - **250 GPM SUPPLY PUMP**
  - **750 GPM DISCHARGE PUMP (8 IN. LINE)**
  - **COMPATIBLE WITH OTHER TEST SYSTEMS**

FIGURE 17

Input functions to the load and temperature controllers within the facility are analog signals generated by either pure analog or hybrid equipment. Elevated temperature testing of lifting body entry vehicle structures generally utilize the Special Purpose Heat Control Computers for function generation.

Two 40-channel and one 30-channel units make up the 110 channels of temperature function generation/control equipment. Each unit has a special purpose analog computer which is time shared by its associated control channels at a commutation rate of ten computations per control channel per second. Although the computers may be operated in either the time-temperature, power-feedback, or time-heat flux mode, the time-temperature mode is utilized almost exclusively.

In this mode of operation the temperature feedback signal from heat control zone thermocouple(s) is compared within the computer to the programmed temperature function for that channel which has been stored within the computer. The amplified difference (error) signal is transmitted to the channel's 580KW ignitron power regulator where the electrical power to the radiant heating units of the channel is adjusted to drive the error signal to zero.

ENVIRONMENT PARAMETER FUNCTION GENERATION

- **PURE ANALOG**
  - **17 DATA TRAK CHANNELS**
  - **19 FM TAPE TRANSPORT CHANNELS**
- **HYBRID DIGITAL-ANALOG**
  - **110 CHAN. SPECIAL PURPOSE HEAT CONTROL COMPUTERS**
  - **39 CHAN. INFORMATION TECHNOLOGY INC. 4901**
  - **40 CHAN. CONTROL LOAD PROGRAMMER**

FIGURE 18

Realizing the need for valid data concerning the response of the test article to its environment a large data acquisition and processing system has been installed within the facility and has the characteristics and capabilities shown on Figure 19.

The higher sampling rate channels are usually assigned to strain, load, and deflection transducers while the slower channels are associated with temperature measurements. Sampling rates can be varied in discrete steps during the test from the maximum rate to "on demand". All signal conditioning, multiplexing, and digitizing is done on the main test floor. The digitized data samples are cable transmitted to the Data Room where recording, computer processing, and display operations are conducted.

Small scale experiments in on-line digital control of test parameters have been conducted using the Data Systems' CDC 1604B computer.

It should be obvious from the foregoing material that the AFFDL's Structural Test Facility is operational and that a significant amount of experience has been accumulated in testing developmental structures for hypersonic flight vehicles. However, like any test facility, it does have certain limitations and deficiencies. The most important of these are listed on the figure (Figure No. 20).

HIGH SPEED DATA ACQUISITION :& PROCESSING SYSTEM

- 952 CHANNELS WITH MAX. SAMPLING RATE OF 100 SAMP/CHAN/SEC.
- 972 CHANNELS WITH MAX. SAMPLING RATE OF 20 SAMP/CHAN/SEC.
- ACCURACY: $\pm$ 1.3% READING OR 0.35% FULL SCALE
- RANGE: 25 MICROVOLTS TO 100 MILLIVOLTS
- RESOULTION: 12.5 MICROVOLTS
- MAGNETIC TAPE RECORDING OF ALL SAMPLED DATA
- ON-LINE OR OFF-LINE COMPUTER PROCESSING OF DATA
- ON-LINE OR OFF-LINE DISPLAY OF PROCESSED DATA
- MICRO-FILM HARD COPY
- OFF-LINE PLOTTING OF PROCESSED DATA
- CDC 1604B COMPUTER USED AS BACK-UP SAFETY AT EXPENSE OF ON -LINE PROCESSING.

FIGURE 19

AFFDL STRUCTURAL TEST FACILITY LIMITATIONS

1. CANNOT ACHIEVE HIGH TEMPERATURES REQUIRED ON NOSE CONES AND LEADING EDGES.
2. ABLATIVE TPS CANNOT REALISTICALLY BE TESTED
3. THERMO-STRUCTURAL EFFECTS OF LH_2 PROPELLANTS NOT ADEQUATELY SIMULATED
4. NO ALTITUDE SIMULATION CAPABILITY
5. AVAILABLE ELECTRICAL POWER LIMITS TESTING TO COMPONENT-SIZED STRUCTURES.
6. MUCH OF EQUIPMENT REPRESENTS 1960 STATE-OF-ART

FIGURE 20

Solutions to these limitations exist; for example, the AFFDL's 50-MW Gas Dynamics Facility is an excellent experimental structure's research and development tool with which the effects of high temperature and mass flow can be investigated.

This facility can be used to experimentally evaluate nose cone and leading edge structural configurations and their attachment to primary structure. It can also be used to explore candidate ablative thermal protection systems, their attachment to radiatively cooled structure, and the radiatively cooled structures themselves.

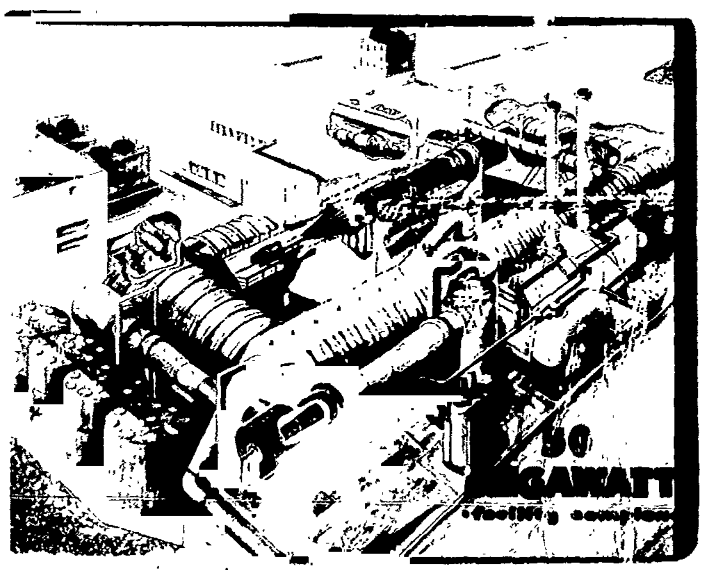

FIGURE 21

Sections of a lifting-body, orbital entry vehicle containing liquid hydrogen propellants internally could be tested with a proposed MCP addition to the AFFDL's Structures Test Facility. The addition would have the characteristics shown on Figure No. 22. Extensive studies by the AFFDL indicate that such testing can be conducted safely by surrounding the test structure with an inert atmosphere. After placing the structure and its test hardware in the test chamber, the chamber would be pumped-down and an inert gas such as nitrogen introduced at either reduced or sea-level pressure. The internal tanks of the test structure would then be filled with LH_2 and the test condition of interest accomplished.

Such a test leg would have the additional benefit that structural components whose thermal protection system efficiency is altitude dependent can be more realistically tested by first pumping down the chamber and then bleeding it back to sea-level pressure on a time-programmed basis as the test progresses.

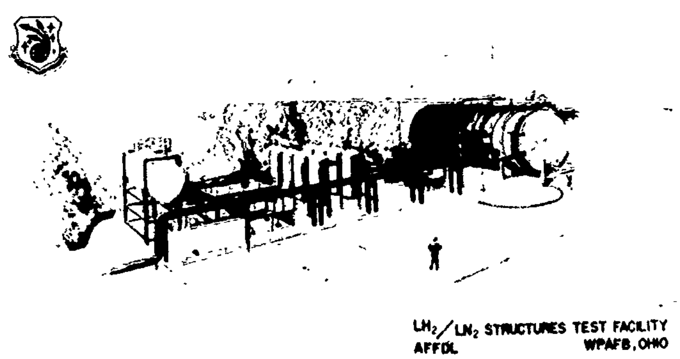

FIGURE 22

LIQUID HYDROGEN / LIQUID NITROGEN TEST LEG

- 20,000 GAL. LIQUID HYDROGEN
- 28,000 GAL. LIQUID NITROGEN
- 30 FT. DIA. X 60 FT. LONG HORIZONTAL TEST CHAMBER
- 250,000 FT. PRESSURE ALTITUDE (10^{-2} TORR)
- 129 LB / SEC GASEOUS HYDROGEN BURNING
- 30 PSIG TEST CHAMBER RELIEF PRESSURE
- INTERNAL LOAD REACTION FITTINGS
- WILL HOLD FULL SIZE OR LARGE SCALE COMPONENTS
  - 20 FT. SPAN X 12 FT. HIGH X 40 FT. LONG
  - 20 FT. DIAMETER X 40 FT. LONG
  - 7,000 LBS. LH_2 INTERNAL FUEL CAPACITY

FIGURE 23

Sufficient space has been provided within the test facility to permit doubling its power rating. Such an increase in power would very nearly double the length of the structural component that could be tested, although the exact size increase would be dependent upon test structure geometry and heating requirements.

Increasing the electrical power would require an increase in the number of control channels. Should this be done it would be wise to consider replacing all 1960 vintage control equipment with modern equipment.

Recent studies conducted by facility personnel indicate that direct-digital-control techniques offer many advantages over the hybrid-analog methods presently used, particularly in the area of temperature control. Equipment replacement, therefore, would probably be by "mini-computers" capable of either stand-alone or satellite operation.

In summary, the AFFDL's Structures Test Facility has an operational capability and extensive experience in the structural development testing of lifting body, orbital entry vehicles. The AFFDL's 50-MW Gas Dynamics Facility nicely compliments the Structures Test Facility. Specific modifications and additions to the Structures Test Facility could enhance its contribution to the successful development of this new class of flight vehicles.

N70-42985

SPACE SHUTTLE STRUCTURAL PROTOTYPE

DESIGN, FABRICATION, AND TEST

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Introduction:

In the development of large launch vehicle structures, a number of different structural tests are performed to verify selected design approaches. Depending on the program phase, these tests will be of different type and magnitude.

During conceptual design, early material selection will necessitate physical property evaluation such as strength, modulus of elasticity, corrosion resistance, specific heat and others, if these can not be found in the current standards.

Also, selected structural elements will be tested for stability, strength, fatigue resistance, acoustic response, thermal performance, and others as required.

During preliminary design, prototypes of representative structures will be designed, fabricated, and subjected to a variety of loading and environmental conditions to assess the performance under simulated operational conditions.

In the final development phase of a vehicle system, full-scale hardware is built, inspected, and tested to qualify the structure. These tests include static load, vibration and acoustic environment, thermal cycling and proof pressure, depending on the particular element. The size of test hardware, load intensity, and combination of test conditions is limited by facilities and cost.

Critical Development Parameters

The listed parameters constitute the most important ones in the development of the space shuttle structure.

Weight of the aerothermal structure and related parameters such as mass fraction and CG location are dependent on the design requirements (reuse, inspection and refurbishment).

Program cost for development and operation must be predicted accurately.

Development risk must be minimized to avoid program delays and cost increases.

MASS FRACTION (DRY WEIGHT)

TPS PERFORMANCE

WEIGHT DISTRIBUTION (CG LOCATION)

AEROTHERMAL STRUCTURE (MORE THAN 50% OF STAGE DRY WEIGHT)

STRUCTURE DESIGNED FOR REUSE, INSPECTION AND REFURBISHMENT

DEVELOPMENT AND OPERATIONAL COSTS

DEVELOPMENT RISK

Major Design Aspects

The listed design aspects are a compilation of design aspects pertinent to the space shuttle structure. Design solutions in these areas must be answered during the prototype test program to reduce development risk and provide direction for the development of an operational system.

- FAIL-SAFE TPS/PRIMARY STRUCTURE FOR 100 REUSES
- LOW WEIGHT AND COST
- TRANSITION BETWEEN DIFFERENT TPS MATERIALS
- SEALING OF TPS ELEMENTS (PURGE AND VENTING)
- LARGE DOORS IN TPS AND LOAD CARRYING STRUCTURE
- OPERATIONAL REPLACEMENT OF TPS ELEMENTS
- ACCESSIBILITY OF ALL STRUCTURAL ELEMENTS FOR INSPECTION AFTER ASSEMBLY
- TRANSITION OF HOT AND COLD PRIMARY STRUCTURE
- MINIMUM HEAT INPUT INTO PRIMARY STRUCTURE
- SIZE OF COMPOSITE STRUCTURES
- ASSEMBLY OF COMPOSITE AND METAL STRUCTURES
- ENVIRONMENTAL PROTECTION OF COMPOSITE PARTS
- REPAIR & REFURBISHMENT CONSIDERATIONS
- INTRODUCTION OF LARGE CONCENTRATED LOADS INTO COMPOSITE STRUCTURES

Test Program Objective

The planned test program will provide recommendations for the aerothermal structure development program for the operational system and a realistic weight forecast. It also will demonstrate the validity of selected materials, analysis and design methods, fabrication, inspection and refurbishment aspects for the required number of reuses.

It will also supply development and operational cost figures to be used in preparing a realistic cost picture for the operational space shuttle program.

PROVIDE SOUND TPS/PRIMARY STRUCTURE RECOMMENDATIONS

PROVIDE REALISTIC WEIGHT DATA

DEMONSTRATE VALIDITY OF APPLIED ANALYSIS AND DESIGN METHODS

DEMONSTRATE REUSABILITY

DEMONSTRATE APPLICABLE FABRICATION METHODS

DEMONSTRATE OPERATIONAL INSPECTION METHODS

DEMONSTRATE OPERATIONAL REFURBISHMENT METHODS

PROVIDE BETTER DEVELOPMENT AND OPERATIONAL COST PREDICTIONS

Program Ground Rules

The listed ground rules are established to assure similarity to a hardware program.

Materials, analysis methods, manufacturing techniques, inspection and refurbishment procedures still in research are not utilized. The simulation of flight environment is limited to the available facilities.

UTILIZE EXISTING MATERIALS AND COMPOSITE SYSTEMS

TEST STRUCTURE SIZE SUFFICIENT TO REPRESENT FLIGHT HARDWARE

TEST STRUCTURES REPRESENT DIFFERENT SECTIONS OF SHUTTLE VEHICLE

DESIGN, ANALYSIS, AND MANUFACTURING TECHNIQUES EQUIVALENT TO FLIGHT HARDWARE

TESTS WILL SIMULATE FLIGHT ENVIRONMENT

INSPECTION AND REFURBISHMENT PROCEDURES EQUIVALENT TO OPERATIONAL CONDITIONS

PROGRAM OUTLINE

a.) MISSION SIMULATION STRUCTURAL TESTS

DESIGN AND EVALUATE

4 PROTOTYPES REPRESENTING DIFFERENT ORBITER AND BOOSTER SECTIONS INCLUDING TPS

LH2 CONTAINER SECTION (BOOSTER)*

LH2 CONTAINER SECTION (ORBITER)

CARGO COMPARTMENT SECTION (ORBITER)

FUSELAGE SECTION WITH WING ROOT (ORBITER)

b.) COMPOSITES FOR PRIMARY STRUCTURES

DESIGN AND EVALUATE

STAGE THRUST STRUCTURE*

SKIN PANELS

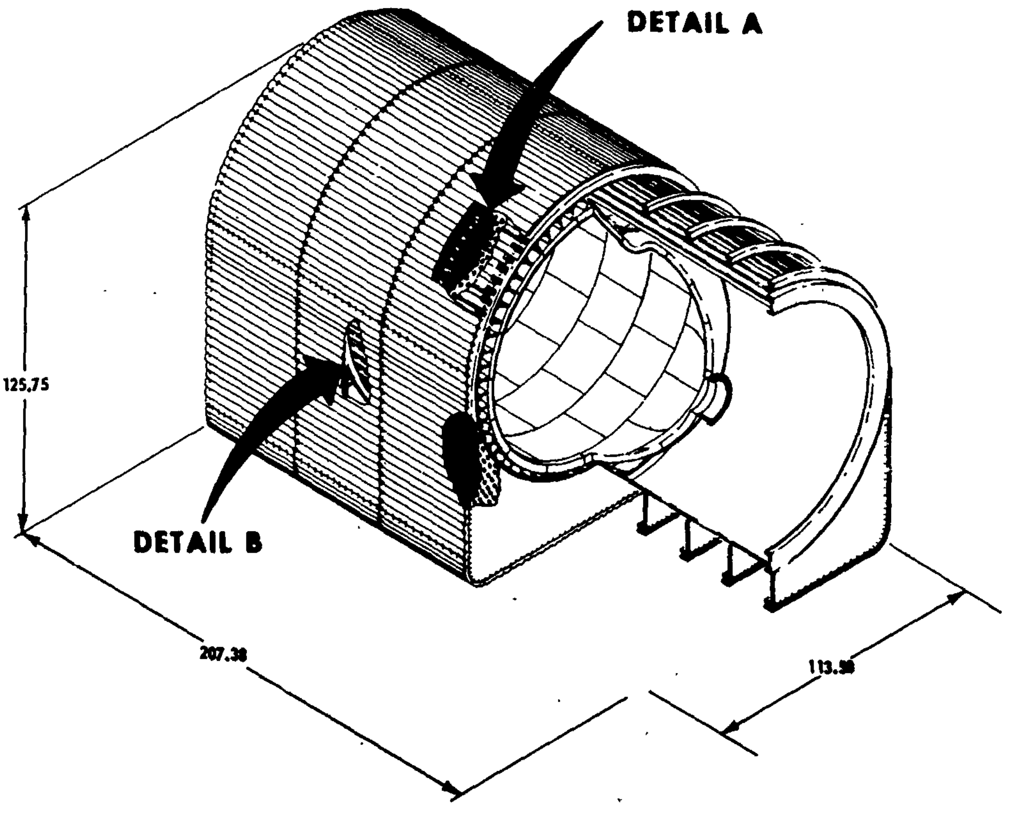
ENGINE SUPPORT BEAMS (SHEAR WEB AND TRUSS)

Prototype #1

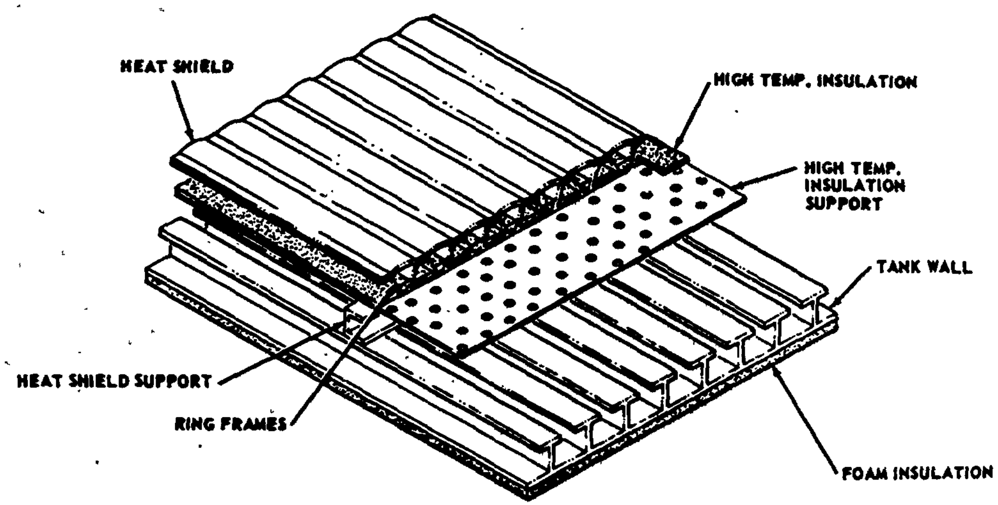
The first prototype is representing the LH_2 area of a Space Shuttle booster. The propellant container, fabricated from AL 2219-T87, has external stiffeners and rings and internal foam insulation similar to the SIVB stage of the Saturn launch vehicle. The forward and aft skirts are external stiffened cylinders, also manufactured from AL 2219-T87. The heat shield is supported by the ring frames, which extend to form the frames on the bottom of the structure.

The thermal protection system consists of corrugated skin panels with circumferential expansion joints and multiple flexible supports. The appropriate materials used in the individual temperature zones are shown later. The high temperature insulation consists of Microquartz battens, supported either by separate titanium panels or are attached to the outer heat shield panel by wire stitching.

LH2 TANK BOOSTER

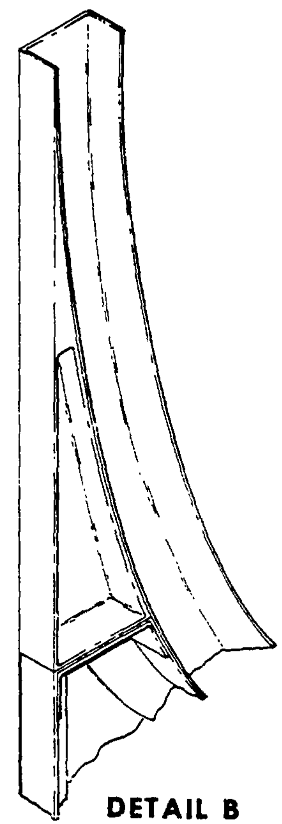


TYPICAL CROSS SECTION



DETAIL A

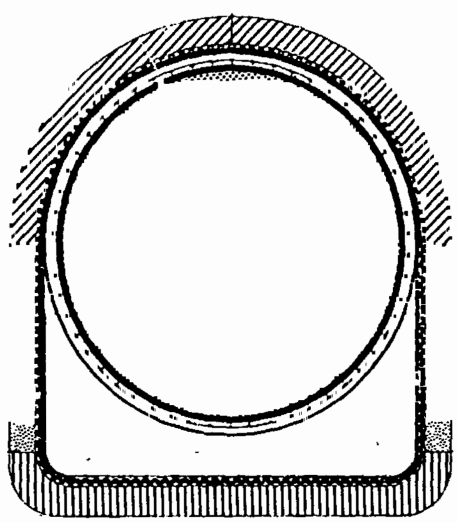
PRIMARY STRUCTURE FRAME



DETAIL B

Temperature Zones

The shown temperature zones and utilized material are typical for prototype #1. The weights given are for the respective TPS area and include high temperature insulation, support panel, heat shield, and associated multiple supports.

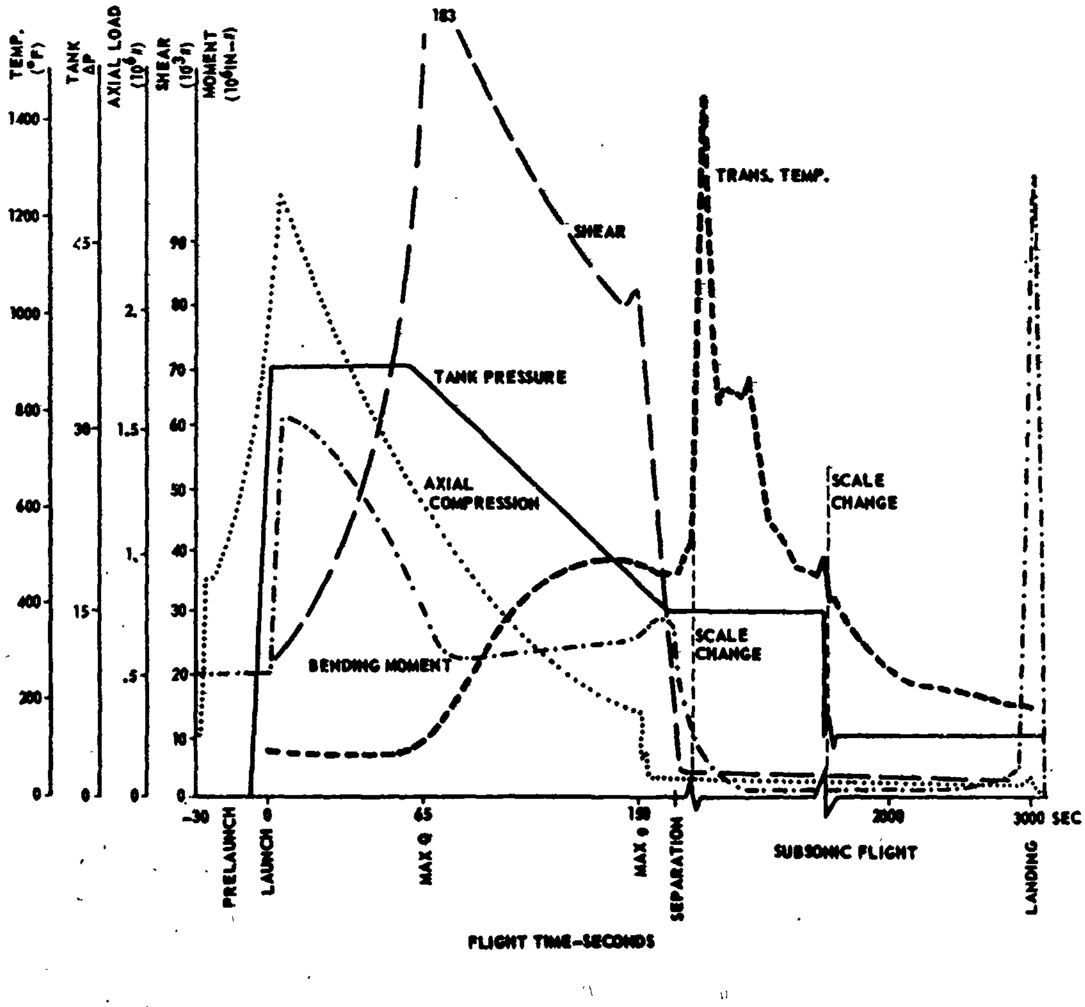


ZONE	TEMP	MATERIAL	INSULATION	WT	DESCRIPTION
	-423°F	POLYURETHANE	3-D INTERNAL	.42 LB/SQ FT	TANK
	200°F MAX	2219-T87	———	N/A	PRIMARY STRUCTURE
	700°F MAX	T1-6A-4V	.25" THICK MICROQUARTZ	1.8 LB/SQ FT	EXTERNAL SKIN
	1200°F MAX	RENE' 41	1.25 THICK MICROQUARTZ	2.8 LB/SQ FT	EXTERNAL SKIN
	1600°F MAX	L605	1.25 THICK MICROQUARTZ	3.3 LB/SQ FT	EXTERNAL SKIN

Typical Test Environment

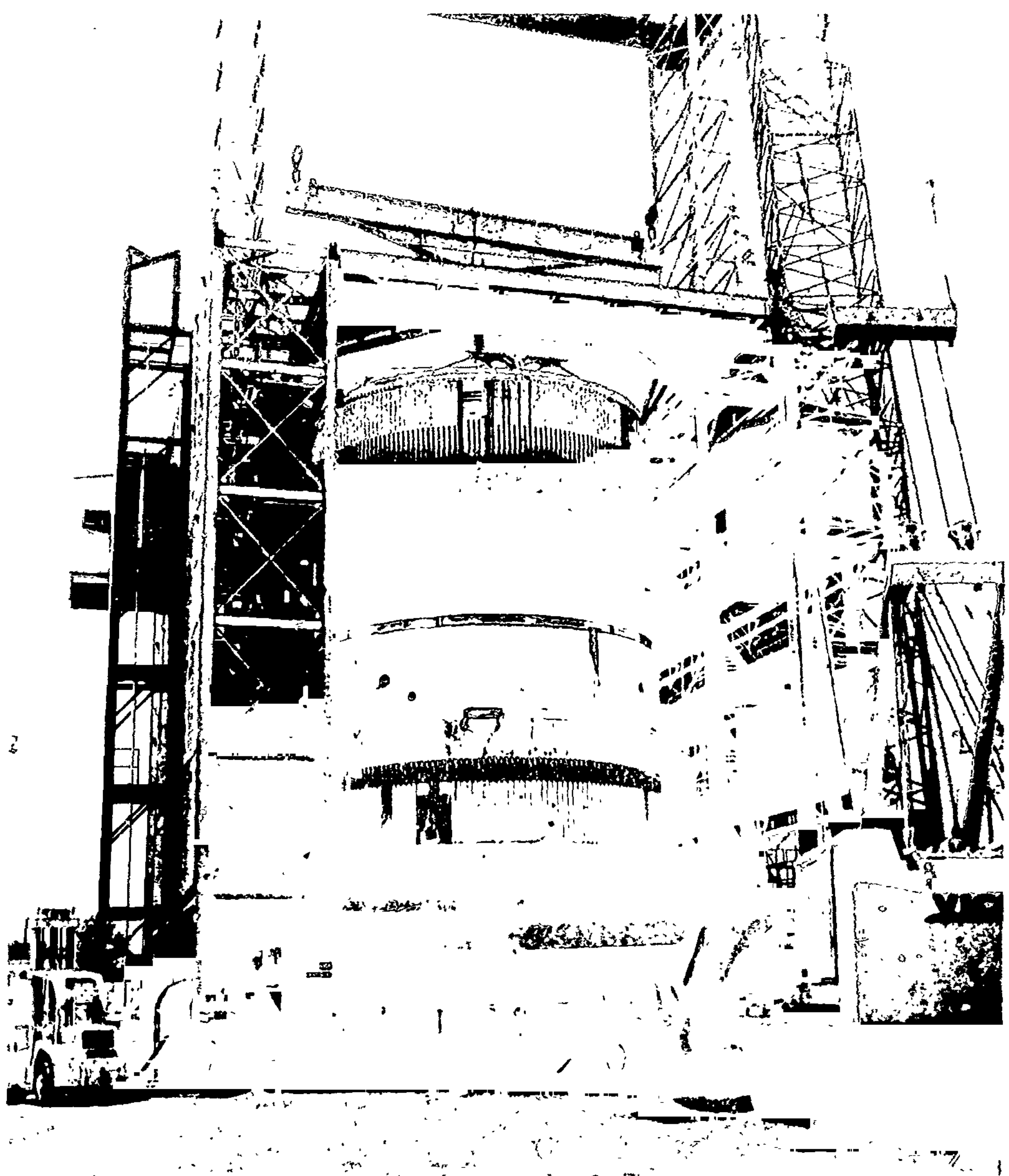
Prototype #1 will be subjected to a simulated flight profile as shown. In the prelaunch phase, the propellant tank is filled with LH_2. Vehicle loads due to ground wind and weight are applied. During the ascent phase, propellants are depleted and flight loads and heating is simulated. In the reentry phase, the structure is subjected to heat load according to the predicted temperature profile. During subsonic flight, the structure is allowed to cool according to the indicated temperature profile and body loads are applied. The bending moment peak during landing is simulated as shown.

Low frequency vibration tests in three axis are performed to verify dynamic response characteristics of the thermal protection system.



Test Facility

The mission simulation tests are performed in the shown test facility. Hydraulic load equipment, LH_2, GN_2 and LN_2 and adequate instrumentation are on site. The heater cage will be placed in the center of the facility. Location of the facility allows hazardous testing (pressure, LH_2 etc.).



Inspection and Quality Assurance

The shown sequence is the planned inspection and quality assurance effort for the various prototypes and related components. It is similar to operational requirements during flight hardware fabrication.

The post-test inspection also determines the necessary refurbishment between simulated mission cycles.

RECEIVING

- CONFIRM MATERIALS
  - A. EDDY CURRENT
  - B. THERMOELECTRIC
- ESTABLISH TRACEABILITY

FABRICATION

- PIECE PART INSPECTION
  - A. DIMENSIONAL ANALYSIS
  - B. CONFIRM MATERIALS (EDDY CURRENT/THERMOELECTRIC)
  - C. VISUAL INSPECTION
- WELD EVALUATION
  - A. DYE PENETRANT
  - B. X-RADIOGRAPHY
  - C. ULTRASONICS
- TANKAGE TESTS
  - A. LEAK TEST
  - B. HYDROSTATIC TEST

DURING TEST

- TANKAGE
  - A. ACOUSTICAL EMISSION

POST-TEST

- A. VISUAL INSPECTION
- B. ACOUSTICAL RESONANCE
- C. MISC. NDT AS REQUIRED

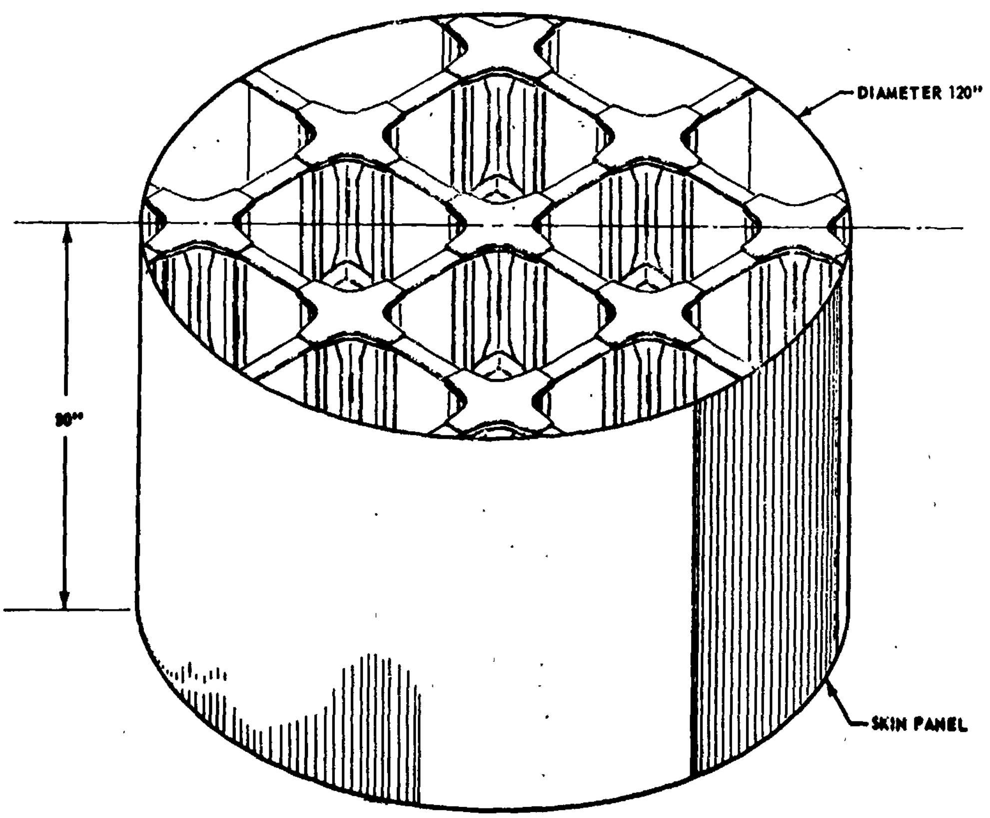
Thrust Structure

The thrust structure test section is a 1/3 scale simplified tail section of the Space Shuttle booster. Nine load points, simulating the engine gimbal points, are located on a system of beams. These cross beams are tied into the outer barrel section. Each load point is capable of supporting 80 Klb simulated engine thrust load.

The selection of the thrust structure for the application of composites is made because major elements of typical primary structures are present. These are: stiffened skin panels, ring frames and beams.

Composite systems utilized are: Boron-Epoxy, Graphite-Epoxy, and Boron-Aluminum. During preliminary design, two beam constructions, shear web and open truss are studied for each composite system. A parallel design, using titanium, is used as comparison. One of each beam is designed, fabricated and tested. The composite system selection is dependent on the results of the preliminary design study. Three skin panels are fabricated and tested, using each of the mentioned composite systems.

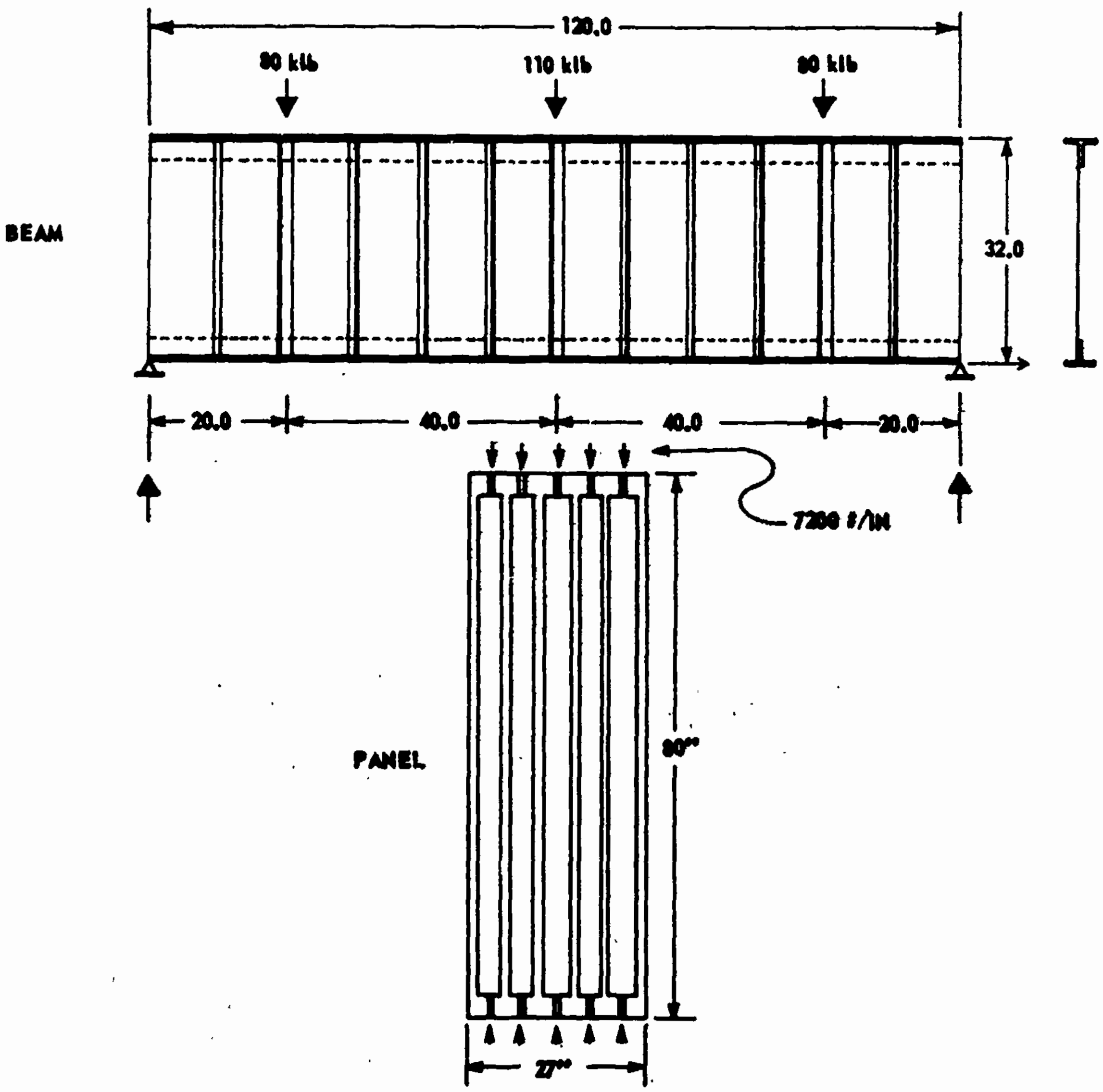
An element selection for the final thrust structure design is made after evaluation of the different elements. The thrust structure and all elements are subjected to multiple load cycles to determine the reuseability. Acoustic and low frequency vibration tests are also performed.



Typical Test Arrangement

Size and typical loading conditions for the two thrust structure elements, shear web beam and skin panel, are shown.

The loads shown are used for preliminary sizing and additional load cases are planned for the hardware tests.



Inspection and Quality Assurance

The shown sequence is the planned inspection and quality assurance effort for the different composite elements and the thrust structure. It is similar to operational requirements during flight hardware fabrication.

The post-test inspection also determines the necessary refurbishment between simulated mission cycles.

RECEIVING

- CONFIRM MATERIALS
- ESTABLISH TRACEABILITY

FABRICATION

- PIECE PART INSPECTION
  - a.) DIMENSIONAL ANALYSIS
  - b.) CONFIRM MATERIAL (ULTRASONIC, X-RAY, OTHERS NDT)
  - c.) VISUAL INSPECTION

POST TEST

- a.) VISUAL INSPECTION
- b.) CHECK INTEGRITY (MISC. NDT)

SUMMARY

PRIMARY ATTENTION GIVEN TO:

- WEIGHT AND COST OF STRUCTURE
- TPS/PRIMARY STRUCTURE INTEGRATION
- DESIGN COMPLEXITY
- DESIGN FOR 100 REUSES
- DESIGN FOR INSPECTION AND REFURBISHMENT
- FAIL-SAFE STRUCTURE

CONCLUSION

OUTLINED PROGRAM WILL ESTABLISH DESIGN TECHNIQUES AND HARDWARE EXPERIENCE NECESSARY TO MINIMIZE DEVELOPMENT RISKS OF THE SPACE SHUTTLE PROJECT.

N70-42986

THERMAL PROTECTION SYSTEMS - INTRODUCTION AND OVERVIEW

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Many assessments of the technology status for the space shuttle have concluded that the most critical technology area is that related to thermal protection systems. Although weight is important because of the shuttle's small payload margin, the most demanding requirement for thermal protection systems is an extended service life with as many as 100 entries with minimal refurbishment. These requirements of minimum weight and extended service life necessitate a vigorous program to determine materials capabilities in the shuttle environment, provide needed advances in materials and thermal protection technology, and produce light-weight systems designs with verified performance.

The following eight papers have been selected to provide information on major aspects of shuttle thermal protection technology. The status of present technology, problem areas, and needed developments will be identified. In addition, this paper and the following NASA papers constitute a brief review of the NASA Space Shuttle Thermal Protection Systems Technology Program. The paper entitled "Dispersion Strengthened Alloys" by N. Saunders describes activities which are part of the NASA Space Shuttle Materials Technology Program and is presented with the following group of papers for continuity.

A summary of the proposed NASA Space Shuttle Thermal Protection Systems Technology Program is given in figure 1 with a listing of the major task areas and the identification of the NASA Centers working each area. The summary consists of activities initiated in FY 70 (most of which continue into FY 71) and proposed new starts for FY 71.

The approximate FY 70 funds allocated to each area are shown to provide an indication of the scope of the program. The large dollar amount shown for Non-Metallic TPS Development results in part from advanced funding for planned FY 71 activities, and thus is essentially funding for two years.

The individual dollar amounts and the total are necessarily estimates as several of the initiated procurement actions involved have not been finalized and a final account of the FY 70 funding situation is not yet available. A moderate increase in total funding for this technology is expected in FY 71.

SPACE SHUTTLE THERMAL PROTECTION SYSTEM TECHNOLOGY
PROPOSED NASA PROGRAM SUMMARY

MAJOR TASK	NASA CENTERS	FUNDS, K$ FY 70
THERMAL PROTECTION SYSTEMS STUDIES	MSC, LRC	125
METALLIC HEAT SHIELD DEVELOPMENT	LRC	525
HIGH TEMPERATURE INSULATION DEVELOPMENT	MSFC	150
NON-METALLIC TPS DEVELOPMENT	MSC, LRC, ARC	2475
LOW-COST ABLATIVE TPS DEVELOPMENT	LRC, MSC	425
TEMPERATURE CONTROL SYSTEMS DEVELOPMENT	MSC	xxx
COOLING SYSTEMS DEVELOPMENT	MSFC, LRC	xxx
CONTRACT TESTING OF TPS MATERIALS	ARC	xxx
MODIFICATION & OPERATION OF NASA TEST FACILITIES	ARC, LRC	160
	APPROXIMATE TOTAL	3860

xxx - FY 71 START

Figure 1

Figures 2, 3, and 4 expand the summary by showing the program elements under each major task. Unless otherwise noted, these elements involve contractual studies with those being initiated in FY 71 indicated by an asterisk.

Under Thermal Protection Systems Studies, a major activity is the development of an efficient computer program for weight and performance analysis in design optimization studies. A detailed study of venting and joint sealing requirements is also planned. Interest in the latter area results from preliminary calculations of the heating due to air inflow through gaps in overlapping metallic heat shields. These calculations show that significant temperature increases can occur at the edge of a metal panel that is overlapped by an adjourning panel. In addition, it is planned to continue the study of the interaction of the boundary layer and oxide surfaces — a phenomenon that is particularly important in understanding the degradation of the dispersion strengthened alloy TD Ni-Cr. Finally, the results from contractual fabrication and refurbishment studies will be used in the development of a cost analysis.

NASA work related to Metallic Heat Shield Development will be amply covered by the second and third papers of this group. Briefly, the principal items are scaling-up coated columbium technology and studying the respond of the metal systems to simulated shuttle entry environments — particularly TD Ni-Cr.

In High Temperature Insulation Development, studies are in progress on the determination of the reusability and design properties of high temperature insulation. Principal problems associated with high temperature insulation are the development of methods for containing these low-density materials and attaching the insulation blankets to the heat shield or primary structures.

SPACE SHUTTLE THERMAL PROTECTION SYSTEMS TECHNOLOGY

ELEMENTS OF PROPOSED NASA FY 71 R&D PROGRAM

THERMAL PROTECTION SYSTEMS STUDIES

DEVELOP COMPUTER PROGRAM FOR WEIGHT AND PERFORMANCE ANALYSIS
*DETERMINE VENTING AND JOINT SEALING REQUIREMENTS FOR METALLIC PANELS
DETERMINE AEROTHERMO-CHEMICAL INTERACTION OF BOUNDARY LAYER AND OXIDE SURFACES (IN-HOUSE)
DEVELOP HEAT SHIELD COST ANALYSIS (IN-HOUSE)

METALLIC HEAT SHIELD DEVELOPMENT

SCALE-UP COATED COLUMBIUM HEAT SHIELD TECHNOLOGY
DETERMINE EFFECTS OF SHUTTLE ENVIRONMENTS ON TD Ni-Cr AND COATED REFRACTORY METALS (IN-HOUSE)
CONDUCT HEAT SHIELD DESIGN STUDIES (IN-HOUSE)

HIGH TEMPERATURE INSULATION DEVELOPMENT

DETERMINE DESIGN PROPERTIES AND REUSABILITY
DEVELOP INSULATION PACKAGING AND ATTACHMENT TECHNIQUES

* FY 71 START

Figure 2

Non-Metallic TPS Development will be covered in the following paper by D. Greenshield. The NASA program centers mainly on the development of light-weight rigidized ceramic for use on the shuttle exterior surfaces and the development of oxidation-inhibited carbon-reinforced carbon materials for stagnation areas.

Low-Cost Ablative TPS Development, except for refurbishment studies, is the subject of a following paper by R. Swann. The principal goal of this work is to identify and reduce the costs of ablative heat shields. Refurbishment cost studies are discussed later by R.Goldin who describes a beginning NASA contract study and presents results from a Lockheed Missiles and Space Company study.

Temperature Control of the shuttle orbiter has received very little attention in the shuttle studies thus far. The proposed program consists of examining all mission phases and systems to identify critical problem areas, to provide conceptual solutions for the critical areas, and to develop thermal control components such as heat rejection systems. A later paper by W. Neuenschwander discusses thermal control aspects.

SPACE SHUTTLE THERMAL PROTECTION SYSTEMS TECHNOLOGY

ELEMENTS OF PROPOSED NASA FY 71 R&D PROGRAM (CONT'D)

NON-METALLIC TPS DEVELOPMENT

DETERMINE PROPERTIES AND DESIGN DATA OF EXTERNAL INSULATION AND CARBON/CARBON MATERIALS
DEVELOP PROTOTYPE EXTERNAL INSULATION TPS PANELS
DEVELOP LEADING EDGE SECTIONS OF OXIDATION-INHIBITED CARBON/CARBON MATERIALS

LOW-COST ABLATIVE TPS DEVELOPMENT

DEVELOP REFURBISHABLE ABLATIVE TPS FOR STAGNATION AREAS
DEVELOP LOW-COST ABLATIVE TPS FABRICATION TECHNIQUES
IDENTIFY CRITICAL DEFECTS IN ABLATIVE TPS
DEVELOP AND VALIDATE ABLATION ANALYSIS FOR SHUTTLE ENTRY
IDENTIFY REFURBISHMENT COSTS AND PROCEDURES

TEMPERATURE CONTROL SYSTEMS DEVELOPMENT

•IDENTIFY CRITICAL THERMAL CONTROL SYSTEMS PROBLEMS
•DETERMINE OPTIMUM THERMAL CONTROL CONCEPTS
•DEVELOP THERMAL CONTROL COMPONENTS

• FY 71 START

Figure 3

Cooling Systems Development includes activities related to surface cooling concepts for stagnation areas. The objective is to reduce surface temperatures to the level where superalloys could be used. Structural cooling systems (active cooling systems, ground cooling after entry, and the heat sink afforded by phase-change materials) will be evaluated and developed as required.

Our experience to date with testing shuttle materials in 100 cycles of simulated shuttle entry environments indicates that more testing capability than that presently existing in NASA may be required. Accordingly, Contract Testing of TPS Materials will be utilized to supplement NASA capabilities when needed.

Modification of NASA Test Facilities is also related to testing TPS materials and concepts. The arc facilities at the Manned Spacecraft Center, Ames Research Center, and Langley Research Center require modifications to provide better shuttle entry simulation. The types of modifications proposed are indicated here and are presently under study. The last paper (by R. Howell) in this group concerns test facilities for TPS testing.

SPACE SHUTTLE THERMAL PROTECTION SYSTEMS TECHNOLOGY

ELEMENTS OF PROPOSED NASA FY 71 R&D PROGRAM (CONT'D)

COOLING SYSTEMS DEVELOPMENT

•DEVELOP STAGNATION AREA SURFACE COOLING CONCEPTS
•EVALUATE AND DEVELOP STRUCTURAL COOLING SYSTEMS

CONTRACT TESTING OF TPS MATERIALS

•MASTER AGREEMENT CONTRACT(S) FOR TESTING IN SIMULATED SHUTTLE ENTRY ENVIRONMENTS

MODIFICATION OF NASA TEST FACILITIES

PROVIDE LONGER RUN TIMES
PROVIDE LARGER MASS FLOWS
DEVELOP CHANNEL FLOWS DEVICES

• FY 71 START

Figure 4

Figure 5 displays a theme that is contained in several of the papers which follow. Specifically, the extent to which various materials can be used on the shuttle orbiter. The types of materials being considered for thermal protection are listed with an approximate maximum use temperature for each material. For many materials, the maximum use temperature has yet to be established. For example, can titanium be used to 1000°F and the superalloys to 2000°F? It should be noted that efforts have been initiated to develop the ceramic and carbon/carbon materials for future use at temperatures as high as 3000°F.

On the right of the figure, the percentages of the orbiter area below the corresponding maximum temperatures are given. These percentages are influenced by the vehicle configuration, the entry mode, assumptions concerning transition, and other factors. Clearly, well-developed materials such as titanium and superalloys can be used over large areas of the orbiter. Depending on the particular orbiter, those materials requiring further development must be used on 10 to 35 percent of the surface area. Subsequent papers will provide detail on the required developments.

APPLICATION OF TPS MATERIALS

MATERIAL	APPROX. MAX. USE-TEMP, °F	APPROX. ORBITER AREA BELOW MAX. USE-TEMP, %
TITANIUM	800	25-50
SUPERALLOYS	1800	65-90
TD-NiCr	2200	85-95
COATED COLUMBIUM	2400	90-98
COMPACTED CERAMIC FIBERS	2400 (fut. 3000)	90-98
COATED TANTALUM	2800	95-100
CARBON/CARBON COMPOSITES	2600 (fut. >3000)	
ABLATORS, DIBORIDES	>3000	

Figure 5

CONCLUDING REMARKS

The present Thermal Protection Systems Technology Program includes development activities related to most of the principal options being considered for use on the space shuttle. Furthermore, the philosophy used in defining the program is to retain sufficient flexibility to include promising new ideas.

As presently constituted the program is largely a contractual program. The latter part of FY 70 was devoted to defining and initiating contract studies. Initial results from these studies will be available in a period beginning about six months from now.

N70-42987

SUPERALLOY HEAT SHIELD TECHNOLOGY

FOR THE SPACE SHUTTLE

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Introduction

Aerodynamic heating effects during reentry subject the external surfaces of the Space Shuttle to a range of temperature levels varying from 500 to 3000°F.

Superalloys are a family of iron, nickel, and cobalt-based alloys having good strength in long-term exposure to temperatures in the 800 to 2000°F range. They possess good oxidation and corrosion resistance in this temperature range, usually without protective coating. Their development and use have been mostly in the area of jet engine hardware. Their chemical stability, useful strength in this high-temperature environment, and relatively broad background of design and fabrication information make the superalloy class of materials attractive candidates for a reusable thermal protection systems (TPS) for a shuttle-type vehicle.

Of all the reuseable materials suitable for the shuttle TPS the superalloy systems appear nearest to being state-of-the-art. However, there are some technological problems which remain to be solved before the superalloy TPS can be considered truly operational. These involve design problems concerning panel-to-panel expansion, sealing and attachment, and the containment of the insulation materials which must be used with these panels. Also, the true upper temperature limits of the superalloys must be established based on a more precise definition of the elevated-temperature exposure cycles and the effect on the pertinent properties of the alloys.

This paper discusses the applicability of the available superalloys, required design features, insulations systems, and problem areas which still remain. The results of a test program in progress at Grumman, which is directed toward investigating solutions to these problems, will also be presented.

Typical Superalloy Design Data

Fig. 1 lists four of the most attractive superalloys for use in the 800 to 2000°F range. They are the nickel-based alloys Inconel 718 and Rene '41 and the cobalt-based alloys Haynes-25 and 188. The two cobalt-based alloys are closely related, with the 188 being a recent improvement on the 25 alloy in oxidation resistance in the 2000°F range. The maximum-use temperature for these alloys is based on oxidation resistance, metallurgical stability, and strength characteristics. Three of the more significant strength characteristics affecting weight are tensile yield strength, creep rupture strength and stress oxidation effects. In the lowest temperature range, 800 to 1200°F, Inconel 718 has a superior strength-to-weight ratio and excellent oxidation resistance. Above 1200°F, the superior tensile yield strength and creep rupture characteristics of Rene '41 make this alloy appear advantageous. The maximum-use temperature of 1500°F was arrived at to avoid overaging and subsequent loss of strength and room-temperature ductility. Haynes 25 and 188 are utilized above 1500°F and are limited to 1800°F. This limit is a tentative one established on the basis of the poor creep strength and the anticipated accelerated effects of stress oxidation on thin-gage sheet. The stress-oxidation performance was extrapolated from data available for Hastelloy 235.

Additional data are required in the area of creep behavior and stress oxidation, and the effects of multiple cycle exposures before the exact upper temperature limit of these materials can be determined. However, it is quite probable that use above 2000°F for a 100-mission capability is not possible.

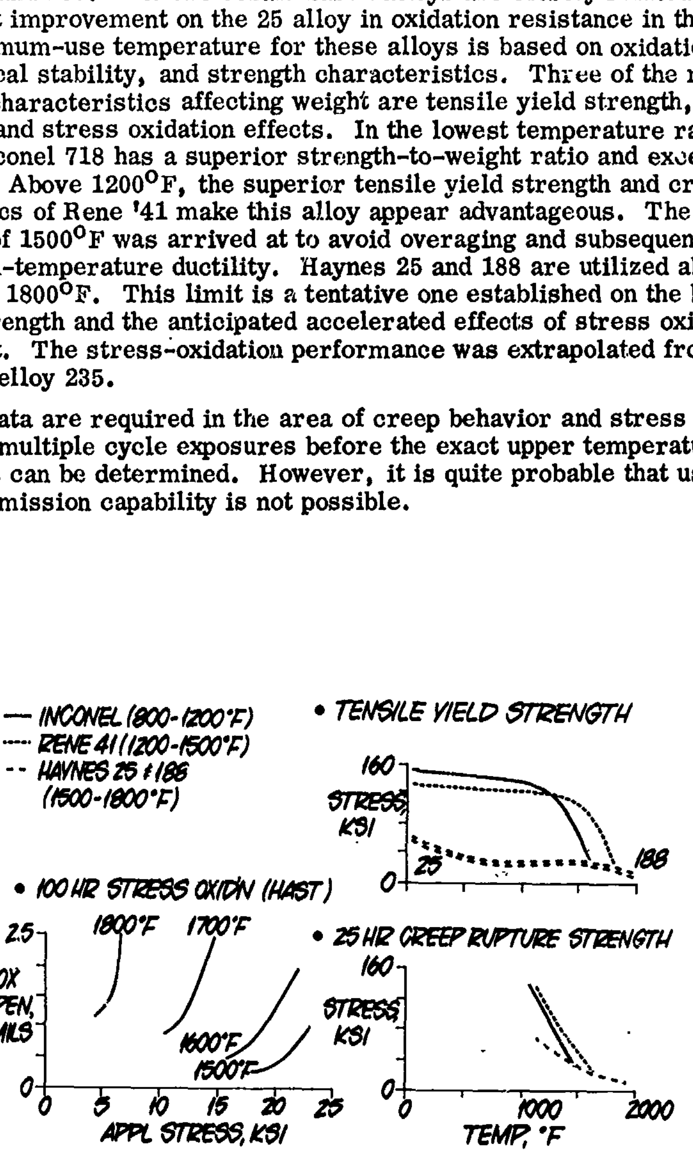

Figure 1

Shuttle Environment

The thermal and structural loading environments which must be considered in the TPS design analysis are indicated in Fig. 2. During boost these are maximum aerodynamic pressures, acoustic sound pressure levels, and mechanical vibrations. The maximum aerodynamic pressures occur at the max $q\alpha$ conditions when the vehicle speed is slightly in excess of Mach 1. Here, under the critical wind shear, dynamic pressures on the order of 500 to 700 psf are encountered. The surface TPS panels must also be flutter-free under the combined dynamic pressure and Mach number conditions.

The acoustic environment is most severe during "hold-down" while the booster engine thrust is built up. Maximum values of 170 dB near the engines have been predicted, with diminishing values as the distance from the engines increases. Mechanical vibrations occur during boost due to dynamic response of the structure to inputs from the propulsion system.

During reentry the maximum temperature conditions occur on the orbiter and, for the most part, the booster. The orbiter experiences the most severe heating with the longest duration. Temperature levels for the critical lower surfaces of the orbiter are higher and of considerably longer duration for the high crossrange maneuver than for the low crossrange. As can be seen the majority of the TPS area for the low crossrange vehicle is below the 1800°F level where superalloys find their most efficient application. Essentially, only the leading edges and nose cap areas for the low crossrange vehicle require material beyond the superalloy range.

The aerodynamic pressures during reentry are considerably lower than the maximum value during boost because of low wing loadings and load factors. Lower-surface normal pressures are expected to be on the order of 40 to 60 psf during the maximum temperature portion of the entry maneuver. Because of the reduced strength of the materials at elevated temperatures this condition can be critical for creep deformation of the panel, when evaluated.

The dynamic inputs due to boundary layer noise are considerably below the noise levels during boost. However, the effect on the TPS panel and its insulation package in the heated condition must be assessed.

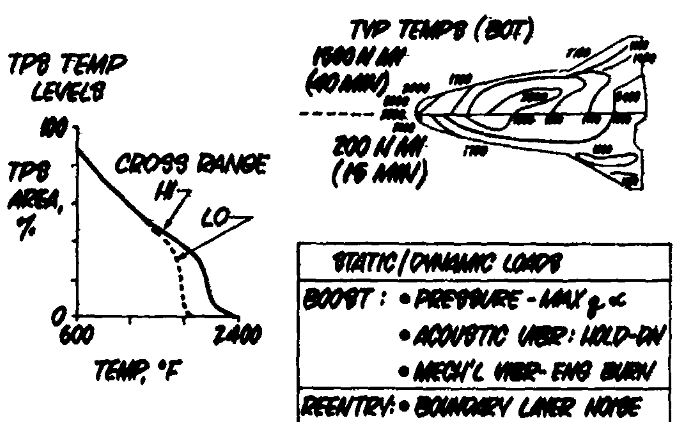

Figure 2

Typical Metallic TPS Design

A typical metallic TPS design using superalloys is shown in Fig. 3. The basic concept is an external insulative system consisting of a surface panel, insulation package, and attachments which protect the primary load-carrying structure from the high surface temperatures. This permits the use of lower-temperature materials which are more efficient for the primary structure. The external surface panel must maintain an acceptable surface contour and smoothness while subjected to aerodynamic pressure loads and thermal gradients. To support these pressure loads, the panel must be sufficiently stiffened to prevent excessive deflections. From the standpoint of aerodynamic considerations, typical allowable panel deflection in an 18-in. span are 1/2 in. for a low L/D vehicle and 1/4 in. for a high L/D vehicle. The required stiffening can be accomplished by utilizing honeycomb sandwich, corrugation stiffeners, beaded skins, or sheet stringer.

The surface panel must be designed to resist the acoustic environment and be flutter-free. In addition, the surface panel must be able to withstand the ground handling conditions which an operational flight vehicle capable of 100 reuses must be expected to sustain.

The surface must also have a high emissivity to radiate the maximum amount of incoming connective heating back out to space. Emissivities of 0.8 or greater are required for efficient performance. This high emissivity can be obtained on the nickel and cobalt-based alloys by controlled oxidation.

The panel size is critical since it determines the amount of expansion which must be accomodated at the edges and the number of supports. The panel is attached to the primary structure via supports which must be capable of transfering the aero-pressure loads and the panel inertia loads while also minimizing the heat flow from the hot panel to the cool primary structure. In addition these supports must accomodate the differential thermal expansion between the hot surface panel and the cool substructure. The panel support can be a number of individual posts or continuous supports along two or four edges, or combinations of these. The support design should also be such that the panel can be easily removed for inspection and replacement.

Since the surface panel operates at a much higher temperature than the primary structure to which it is attached, the surface panel is segmented with gaps between adjacent panels to permit growth. These gaps must be sealed to prevent leakage of the high-temperature boundary layer gases under the panel which would nullify the effectiveness of the TPS. The panel seal must allow free expansion of the panels and maintain its sealing ability despite panel distortions.

The insulation package provides the main barrier to heat flow between the hot surface panel and the substructure. The insulation material must wit[illegible] the thermal and vibration environments characteristic of the TPS. The [illegible]fficient insulating materials in the 800 to 2000°F range are the low-density fle[illegible] mats made of silica fibers. These materials are very susceptible to compaction [illegible]ater absorption. They, therefore, require packaging which protects them against moisture and also supports the insulation so that it fills the entire cavity to prevent radiation "windows." The package will also be subjected to pressure levels ranging from one atmosphere to the vacuum of space. The venting and sealing of this package, therefore, presents a significant problem. A partial solution may be to use a rigidized insulation with non-interconnected porosity.

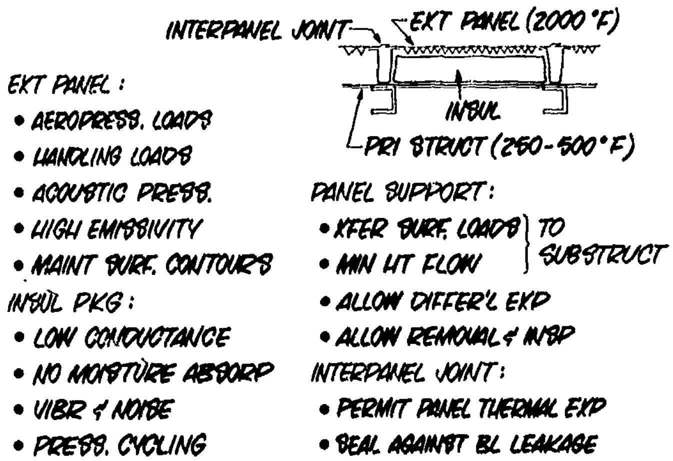
INTERPANEL JOINT
EXT PANEL (2000 °F)
INSUL
PRI STRUCT (250-500 °F)
EXT PANEL :
• AEROPRESS. LOADS
• HANDLING LOADS
• ACOUSTIC PRESS.
• HIGH EMISSIVITY
• MAINT SURF. CONTOURS
INSUL PKG :
• LOW CONDUCTANCE
• NO MOISTURE ABSORP
• VIBR & NOISE
• PRESS. CYCLING
PANEL SUPPORT :
• XFER SURF. LOADS
• MIN HT FLOW
TO SUBSTRUCT
• ALLOW DIFFER'L EXP
• ALLOW REMOVAL & INSP
INTERPANEL JOINT :
• PERMIT PANEL THERMAL EXP
• SEAL AGAINST BL LEAKAGE

Figure 3

Weights of Typical Metallic TPS

The accompanying figure shows the variation of weight of a metallic TPS for high and low crossrange orbiter reentry maneuvers and for a typical booster reentry. The temperature scale indicates the peak temperature that occurs during the heating pulse. The elements included in the weights are surface panel, panel supports, joint seals, and insulation package. The insulation is sized by assuming that an aluminum substructure weighing 2.0 psf is allowed to experience a 250°F rise during the entry maneuver.

The metallic surface panel was considered to be a corrugation-stiffened Haynes 188 beaded face sheet with a 3.5 lb/ft^3 Microquartz insulation blanket contained in a nickel-foil package, up to 1800°F. Above 1800°F the metallic surface panel was assumed to be made of a coated columbium alloy and the insulation was a composite of Zirconia felt and Microquartz. This is the reason for the sudden increase in unit weight above 1800°F.

The general variation of weight with temperature is due to the increased insulation thickness required at the higher temperatures. The effect of the length of heating on insulation thickness is evidenced by the difference in weights for the booster, which has the shortest heat pulse, and the high crossrange orbiter, which has the longest heat pulse.

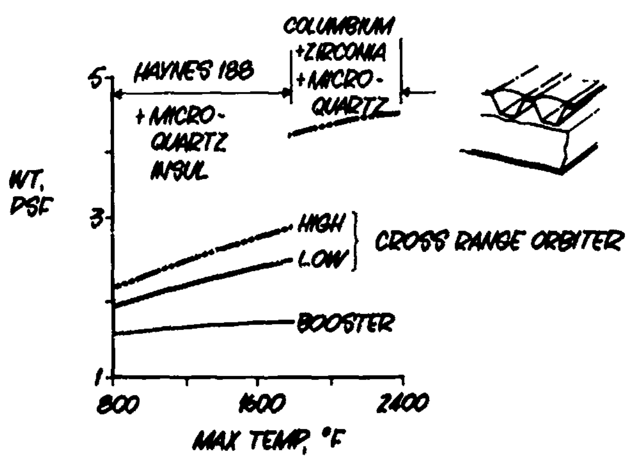

Figure 4

Design & Test Conditions for Development Program

Two TPS units were designed for (and tested under) the critical temperature, load, and acoustic conditions that realistically could be expected during 100 repeated missions of a low crossrange space shuttle vehicle.* The design conditions were established during the course of preliminary studies conducted at Grumman. For current and future work, these conditions have been reevaluated and more severe airloads are being used. Conditions were grouped so as to be consistent for particular locations on the vehicle surface. They are summarized in Fig. 5.

Critical heating occurs during reentry. An analytically derived 20-minute temperature-time profile with an 1800°F peak was chosen for design because 1800°F is a reasonable maximum temperature at which Haynes-25 and Haynes-188 superalloys could be expected to endure 100 missions. A profile with a 2100°F peak was selected to simulate an overheat condition that the superalloy TPS would have to withstand safely in one mission only. The temperature of the protected substructure should not rise above 500°F at any time during the 1800°F mission. The high heating rate is accompanied by varying air pressure normal to the surface. Considering combinations of temperature, pressure, and temperature-dependent material properties, a load of 40 psf, constant through the temperature profile, was chosen as a critical reentry design condition; it was later increased to 60 psf.

The critical room-temperature pressures for static strength occur during the launch trajectory at the maximum-q condition. The surface temperature is only negligibly above room temperature. For the initial two specimens a maximum positive pressure of 150 psf normal to the TPS surface was used. In current work the much more severe 450-psf positive and 200-psf negative pressures are the design conditions.

Critical mechanical vibrations occur during liftoff. For design of the TPS panels a maximum room-temperature acceleration of 100g was selected.

The critical acoustic vibration environment for the TPS occurs during the hold-down phase of rocket launch. The design conditions were the maximum predicted octave band levels up to 152 dB at 10,000 Hz.

In the design and analysis of the test panels the material properties of Haynes-25 were obtained from specification MIL-HDBK-5A.

At high temperatures the material allowables were reduced to stresses that create 0.4% creep strain at 25 hours of heat and load exposure.

Maximum permissible surface deflection of 0.5 in. per 18-in. span was calculated analytically on the basis of aerodynamic heating and drag penalties.

Minimum required element and panel stiffnesses were calculated analytically to prevent flutter.

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* "Development of a Reusable Metallic Thermal Protection System for Lifting Reentry Vehicles," by R.T. Ratay and W.E. Fisher, Grumman Aerospace Corp., ADR 02-04-70.1, April, 1970.

LOW CROSS-RANGE ORBITER		INITIAL	CURRENT
ROOM TEMP LAUNCH	• POS PRESSURE (PSF)	150	450
	• NEG PRESSURE (PSF)	–	200
	• MECH'L VIBR (G)	–	100
	• ACOUST VIBR (5100 SEC)	142-152 DB	
HI-TEMP REENTRY	• NORM HT CYC (20 MIN)	1800°F PEAK	
	• O'HEAT CYC (20 MIN)	2100°F PEAK	
	• POS PRESSURE (PSF)	40	60
	• SUB STRUCT TEMP	≤ 500°F	
DESIGN CRITERIA	• MATL ALLOW STRESS (PER MIL HBK-5)		
	• CREEP STRAIN (≤0.4% @ 25 HR EXPOS)		
	• SURF DEFL LTD BY AERODYN (≤ 0.5 IN.)		
	• LOCAL & O'ALL STIFF TO PREVENT FLUTTER		

Figure 5

TPS Test Panel Configuration

Several TPS configurations with metallic/radiative heat shields were considered, including corrugation-stiffened skin, double-faced corrugated panel, integrally stiffened plate, and honeycomb sandwich panel. The corrugation-stiffened beaded-skin heat shield with supports that allowed thermal deformations and flexible insulation were selected for the detailed study because it appeared to have more of the features necessary to satisfy the design requirements. (See Fig. 6.) Two full-scale TPS units were designed, fabricated, and tested. The first panel was intended primarily as a pilot specimen, the second was prepared for a more elaborate study.

The heat shield panels were made of 0.008-in. thick L-605 (Haynes-25) superalloy sheets, fabricated into 18-in. square corrugation-stiffened panels. The face sheet had 0.025-in. high by 0.6-in. chord circular cross-section beads, at 1-in. pitch, running the whole length of the panel. The face sheet was welded (resistance spot welded in panel No. 1 and resistance seam welded in panel No. 2) to a panel of 1/2-in. depth by 1-in. pitch corrugations. The longitudinal beads and corrugations provided the strength and stiffness needed to span the end supports. In the lateral direction the beads and corrugations provided the flexibility needed to minimize lateral thermal stresses by allowing local deformation of the cross-section.

The heat shield supports, running across the beads and corrugations under the ends of the shield, were fabricated by forming and welding (as for the panels above) 0.008-in. thick L-605 superalloy sheets. They were beaded and segmented to provide discontinuities for relieving thermal stresses. They were welded to the heat shield and bolted to the substructure on load-bearing insulation. The supports were designed to have sufficient strength to carry the air loads, and be sufficiently flexible to accommodate the longitudinal thermal expansion and bowing of the heat shield.

Insulation, consisting of 1-1/2-in. thick blanket of fiberous silica quartz felt (Microquartz), made up of 3/16-in. thick layers, was placed between the heat shield and the substructure. In panel No. 2, the insulation was "bagged" in 1/2-mil thick nickel foil whose edges were resistance seam welded.

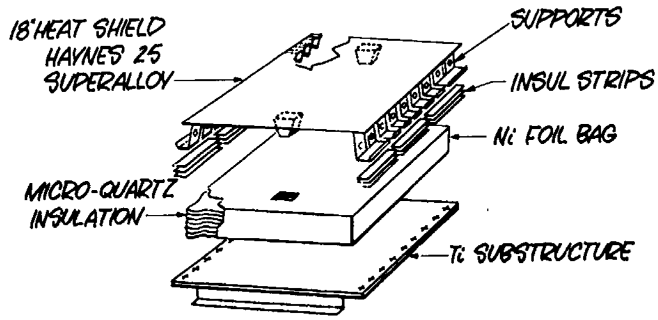
The TPS unit was attached to a titanium substructure, consisting of a 0.1-in. plate with stiffeners, that was to simulate a vehicle primary wing structure.

Temperature distribution, thermal deformation, stress, and deflection analyses were performed as part of the heat shield design. The calculated weight of the TPS panels was 1.57 psf, of which the heat shield and supports were 1.17 psf and the insulation was 0.40 psf.

Testing

The test setup consisted of an aluminum support fixture, quartz heating array, loading weights on wire hangers, movable loading platform, deflection-measuring probes, and thermocouples. For the room-temperature acoustic testing, the TPS unit was suspended in front of an exponential horn. A new test fixture is being built to accommodate more than one unit.

The specimens were subjected to repeated exposures to heating, loading, and acoustic vibrations. Panel No. 1 was subjected to 10 normal thermal cycles and 5100 seconds of acoustic disturbance; panel No. 2 was subjected to 14 combined thermal/load cycles, including two overheat cycles, 5100 seconds of acoustic disturbance, and to 150 psf loading at room-temperature. Continuous temperature and deflection measurements and many photographs were taken.



TESTING:

- REPEATED EXPOSURES TO HEATING, LOADING, ACOUSTIC VIBRATION

Figure 5

Test Results

The test results confirmed analytical predictions. Typical temperature and deflection measurements are shown in Fig. 7; a photograph of the tested panel is shown in Fig. 8.

During the 1800°F heating cycles the maximum temperature gradient between the face sheet and the bottom of the corrugation was 275°F for panel No. 1 and 350°F for panel No. 2; it occurred at 1-1/2 minutes of heating. The maximum mean temperature of approximately 1790°F occurred at peak heating; at this time the gradient across the heat shield was only about 20°F. After about 2-1/2 minutes past the peak heating the gradient reversed: the face sheet became, and remained, cooler than the corrugations. The measured temperatures on the bottom of the titanium substructure plate under the heat shield supports ranged up to 650°F, while under the middle of the panel the maximum temperature rose to 440°F. The latter temperatures were slightly higher in each consecutive test, implying a degradation of insulating effectiveness from mission to mission. Under the 2100°F heating profile the titanium substructure plate reached approximately 560°F near the middle of the panel, and approximately 1000°F under the heat shield supports.

The maximum upward bowing of the heat shield in each run occurred at approximately 1-1/2 minutes of heating, coincident with the time of highest temperature gradient across the corrugations. The maximum thermal bowing with no load was 0.165 in.; with a 40-psf load it was 0.125 in. After the time of maximum bowing, the 40-psf load induced significant downward deflections of the heat shield. The maximum in-plane longitudinal thermal expansion of 0.15 in. at the two ends of the heat shield occurred at approximately 11-1/2 minutes, which was the time of peak heating.

Permanent deflections of approximately 0.015 in. were accumulated in each temperature/load cycle. The flexural stresses in the heat shield were well below the elastic limit of Haynes-25, even at 1800°F; therefore, the permanent set was attributed to high-temperature creep rather than to plastic flow.

Following 14 temperature/load cycles, the heat shield was loaded to 150 psf at room temperature. It deflected 0.10 in., a very much greater deflection than the calculated 0.037 in. This descrepancy implies that a great deal of the heat shield stiffness was lost during the severe environments of the previous tests.

The overall width of the heat shield increased by approximately 0.2 in. during heating. The heat shield supports were expected to restrain, and the face sheet beads were expected to absorb, this thermal expansion. The lateral expansion returned to zero as the heat shield cooled off.

Clearly identifiable permanent deformations of the beads occurred in both heat shields during the first 1800°F heat cycles. These deformations did not progress noticeably with subsequent 1800°F tests. The "bulging" of the beads was more pronounced at locations where the end supports are segmented (at every other bead). The deformations were not uniform; some of the beads bulged more than others. Analytical work indicated that thermal bending stresses and strains at the crown of a bead would go far into the plastic range of the material during the first and subsequent cycles. (Less plastic deformation would occur with beads deeper than 0.025 in.) Repeated, reversed deformations of the beads at such high strains may present a potential fatigue problem. The deformations of the beads grew more severe during the 2100°F temperature profiles.

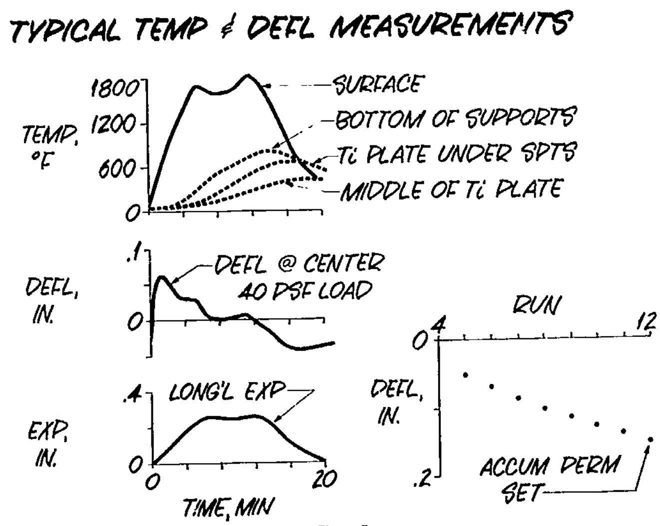


Figure 7

TPS PANEL 2 AFTER TESTS

PERMANENT DEFORMATIONS - NO FAILURE

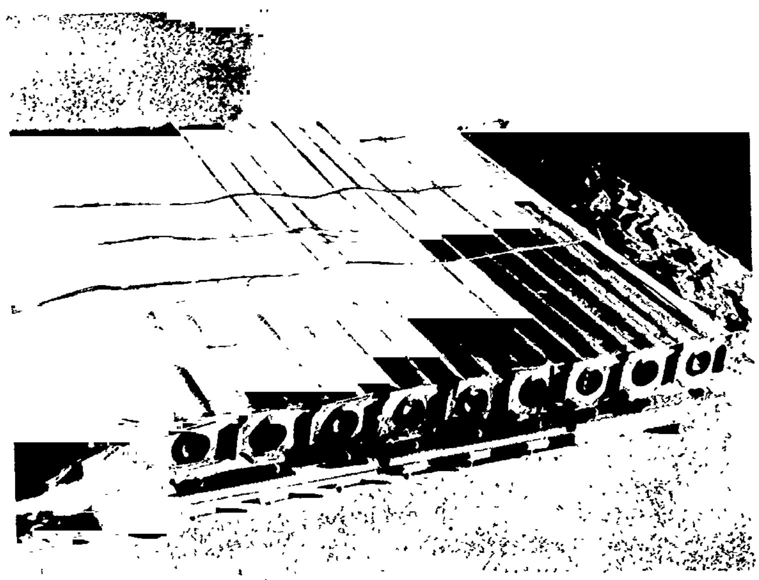


Figure 8

The end supports of the heat shields of both test panels survived the test programs with no apparent damage.

Both panels survived the 100-mission equivalent of 5100 seconds of exposure to the acoustic test levels without apparent failures. Resonant frequencies and transfer functions were obtained from the tests.

Visual examination of the heat shields showed no evidence of weld failures or material degradation. After exposure to two cycles of the 2100°F over-temperature profile in still air, the nickel foil insulation bag was severely oxidized.

The following conclusions and recommendations were based on evaluation of the quantitive and qualitative test data, and on analytical results:

- The described thermal protection system is a thermally effective and safely reusable configuration; however, further improvements are needed to perfect the system.
- The limitations of test panel No. 2 were not reached at the end of the 12 simulated, normal, reentry cycles (1800°F temperature profile and 40-psf pressure) and 100 simulated launch acoustic environments; apparently, the TPS could successfully perform many additional missions. However, it is not possible to project reliably the safe life of this TPS based solely on this program; cyclic thermal stress analyses and additional tests are needed for such a projection.
- The test panel endured the two 2100°F over-heat cycles, but the nickel foil insulation bag completely oxidized and the heat shield deformed to such extent that it would require refurbishment.
- The heat shield sustained permanent spanwise bending deflections after each simulated reentry heating and loading cycle. These deflections were attributed to high-temperature creep. The accumulated magnitude of these deflections was considered neither aerodynamically nor thermodynamically critical after 12 cycles.
- The beads of the heat shield also sustained permanent deformations, which were not the same in every bead. The beads should be deeper (more flexible) to lessen permanent sets. The end restraints should be improved to produce uniform deformations in every bead without overall lateral expansion of the heat shield.
- The heat shield supports performed satisfactorily in carrying the loads and accommodating longitudinal thermal expansion of the heat shield. However, their attachments to the beaded face sheet must be redesigned to allow each bead to deform freely and to restrain overall growth of panel width.
- Under the 1800°F temperature profile, the insulation maintained the titanium substructure below 440°F, except at local areas under the supports where the substructure temperature rose to 650°F. More effective insulation may be desirable between the supports and the substructure.
- Closures to provide continuity of the surface between adjoining heat shields were not tested.

Conclusion, Current & Future Effort

For temperatures up to 1800°F, the metallic heat shield panel utilizing nickel and cobalt-based superalloys holds promise for a practical, reliable, and reasonably efficient reuseable TPS for the shuttle system. It appears particularly applicable to the low crossrange orbiter and the booster.

A number of problem areas remain which require further investigation; they are:

- Determination of creep and stress oxidation characteristics of the cobalt-based alloys in the 1500 to 2000°F temperature range
- Resolution of design features for optimum interpanel joints and support methods
- Investigation of panel response to high-energy acoustic environments
- Flutter characteristics of metallic panels when subjected to launch and re-entry conditions
- Insulation package design which prevents excessive moisture absorption and provides adequate support to prevent insulation damage due to vibration and acoustic effects
- Flowing-gas tests of representative TPS panels to demonstrate overall performance during repeated cycling

A major advanced development program is underway at Grumman for perfecting superalloy and refractory-metal TPS components and assemblies. This includes design, thermal and stress analyses, component fabrication and assembly techniques, and testing.

Current emphasis is placed on the design of heat shield supports and closures at the ends of adjoining panels. These two components are considered the critical ones in restricting the lengths of panels. The attachment of the insulation and methods of preventing moisture absorption by the insulation are also being investigated.

A test fixture for heating and loading more than one adjoining TPS unit is being built. Specimens are planned to be tested under more of the mission-environment conditions.

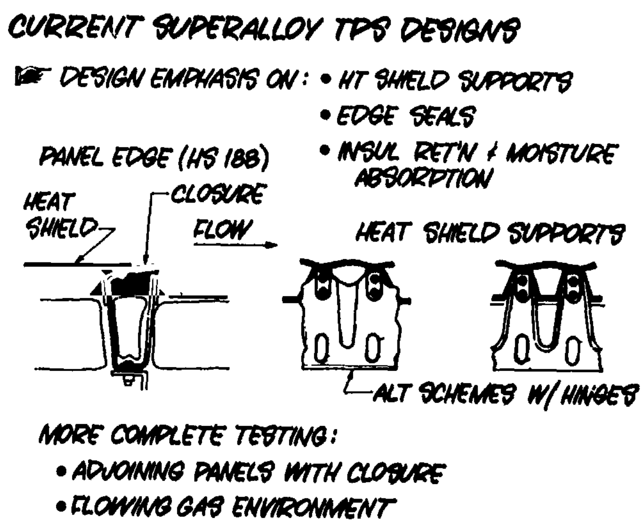


Figure 9

N70-42988

DISPERSION-STRENGTHENED ALLOYS FOR SPACE SHUTTLE HEAT SHIELDS

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INTRODUCTION

Dispersion-strengthened nickel-base alloys are currently considered as one of the prime candidates for use in the hotter regions of the heat shields for the Space Shuttle. These alloys are attractive primarily because of their good high temperature strength. As shown in figure 1, the stress-rupture strengths of conventional superalloys (such as the nickel alloy Rene 41 and cobalt alloy HS-25) decrease quite rapidly with increasing temperature. So these alloys appear to be limited to heat shield regions which will see maximum temperatures of about 1800°F. Above this temperature, the stronger refractory metal alloys could be used (for example, the columbium alloy Cb-752 or the tantalum alloy T-222). But these alloys rapidly oxidize in air and thus require protective coatings to resist oxidation under re-entry conditions. Because the reliability of these coatings is questionable for repeated use, it is desirable to minimize the use of coated refractory metals to only those areas where they are absolutely necessary.

Another class of materials that could fill a gap between superalloys and coated refractory metals are dispersion-strengthened alloys, such as TD-NiCr. These alloys have adequate strength and oxidation resistance to be considered for use in an uncoated condition over a temperature range of about 1800° to 2100°, or possibly 2200°F. This temperature range is important because it could be associated with up to 30 percent of the heat shield area, depending upon the vehicle configuration and flight path. By using dispersion-strengthened alloys in this temperature range, the need for coated refractory metals, or non-metallics, could be reduced to perhaps less than five percent of the heat shield area.

CANDIDATE ALLOYS

Several dispersion-strengthened alloys have been developed during the past decade. Prime impetus for this work has come from DOD and NASA-sponsored programs to develop alloys for use in the hottest components of advanced turbojet engines. Relatively little work has been done to evaluate the alloys for use in heat shields.

Most of the dispersion-strengthened alloys developed to date are listed in figure 2 along with potential suppliers. This listing includes mostly nickel-base alloys, but a limited amount of work has also been done on some higher strength cobalt alloys and on a more oxidation-resistant iron alloy.

In all of these alloys, their unique high temperature strength is associated with the dispersion of very small particles of a metal oxide in a metal matrix. The amount of the oxide dispersoids used is relatively small--usually in the range of one to four percent (by volume). Thorium dioxide (ThO_2) has been used in most of the alloys because of its good chemical stability in this temperature range. However, Al_2O_3 and Y_2O_3 are being used in some of the newer alloys to alleviate the handling problems associated with the toxic ThO_2 powders.

Several different organizations have been active in developing these alloys, and each of these organizations uses a different (and usually proprietary) process to produce the alloys. Most of the alloys are produced from powders, but the methods of making, blending, and compacting the powders, plus the sheet rolling processes vary greatly from one producer to another. Thus, dispersion-strengthened alloys of the same nominal composition can have appreciably different properties if made by different producers.

Only two of the dispersion-strengthened alloys have reached a status of large production quantities. These are the relatively simple Ni-2ThO_2 and Ni-20Cr-2ThO_2 alloys. They are the only ones that we have considered for wide-use within the next few years. The Ni-2ThO_2 alloy is further developed, but this alloy lacks adequate oxidation resistance for re-entry use. So the best current choice appears to be the Ni-20Cr-2ThO_2 alloy because this alloy has the best combination of strength, oxidation resistance, and availability of all the dispersion-strengthened alloys. However, we will continue to evaluate some of the newer alloys to determine if they warrant further development for later versions of the Space Shuttle. For example, the recently-announced development of International Nickel in dispersion-strengthening relatively complex superalloys offers promise of achieving both the high temperature strength of dispersion strengthening plus the better intermediate temperature strengthening of precipitation-hardened superalloys. Another process being developed in-house at the Lewis Research Center also offers promise for achieving these combined strengths in dispersion-strengthened superalloys. However, these processes are in the very early stages of development and will require several more years of development to come to fruition.

At the present, our prime emphasis for Shuttle heat shields is on the Ni-20Cr-2ThO_2 alloy. And most of this effort is on the alloy commercially-termed "TD-NiCr." This alloy was originally developed by E. I. DuPont deNemours, Inc.; but the rights and facilities to produce it were subsequently sold to Fansteel, Inc. who currently is the only commercial producer of this alloy.

TECHNOLOGY PROGRAM

Although TD-NiCr offers considerable promise for use in heat shields, it needs more development before this promise can be fulfilled. Questions have been raised about such things as adequate availability of sheet, inconsistent properties, adequate oxidation resistance, etc. Therefore, we have started an extensive technology program to seek answers to these questions and further the development of this type of alloy.

This work can be generally grouped under the following three areas:

1. Improvement of the sheet manufacturing process

2. Development of fabrication processes for making heat shield panels

3. Evaluation of design-allowable properties.

In addition, we are pursuing a "back-up" development program which involves the following studies:

1. Optimization of the alloy composition to improve properties which may be marginal for heat shield use

2. Development of an alternate sheet fabrication process to assure material availability.

These technology programs are being pursued through a combination of in-house studies at several NASA Centers and contracted studies with various industrial organizations. These studies are summarized in the following sections in terms of the current status of development and our plans for advancing the state-of-technology in each area.

TD-NiCr DEVELOPMENT PROGRAMS

Sheet Manufacturing Process

One of the concerns that has been frequently expressed about the use of TD-NiCr is the wide variations in properties that have been reported by various organizations. This property variation can be largely attributed to the fact that the material has been in a state of "advanced development" in which the processing methods have been frequently changed. One of the unique features of dispersion-strengthened alloys is that their properties are highly dependent upon the processing methods used (much more so than for most other metals). For example, figure 3 illustrates the effect of various rolling and annealing methods on the 2000°F tensile strength of TD-NiCr sheet. These DuPont results indicate that the strength can be varied by a factor of three or more by changing rolling procedures. Also, the corresponding ductility can be varied by a factor of ten or more (for example, the tensile elongation of the strongest sheet shown in figure 3 was about two percent while that of the weakest sheet was about 20 percent). Since this material has been primarily developed for jet engine applications, warm pack rolling has been used most extensively to achieve maximum strength. This strength advantage is gained primarily at the expense of lower ductility. On-the-other-hand, heat shield applications may require sheet with greater ductility, but some trade-off in strength may be tolerable. Thus, the sheet manufacturing process needs to be optimized to tailor the resultant properties for heat shield use.

To achieve this purpose, we recently awarded a contract to Fansteel to further develop TD-NiCr sheet processing methods. This one-year program will involve study of the powder consolidation process, the rolling schedules and associated heat treatments, and the effects of minor contaminants. Particular emphasis will be directed toward improvement of cold rolling techniques. Cold rolling is more desirable than warm rolling to achieve better dimensional control. Good control is particularly needed in sheet thickness where minor variations could appreciably affect the heat shield weight. As an overall result of this process improvement study, we expect to achieve sheet produced to tighter specifications with more reproducible properties than that currently available.

Another objective of this sheet processing program involves process scale-up to achieve larger sheet sizes and greater production capacity. Current size capabilities for TD-NiCr sheet are about 18 x 36 inches in thicknesses of 0.020 inches or more. However, some heat shield designs require larger panel sizes and thinner gauges. So the processing methods will be scaled-up to produce sheet sizes up to 24 x 60 inches in thicknesses as thin as 0.010 inches. Also, foil rolling procedures will be developed to produce six-inch wide foil in thicknesses of 0.005 to 0.010 inches. This foil could be used to produce honeycomb-type panels for heat shields.

In addition to the scale-up in sheet sizes, Fansteel is increasing its powder production capabilities to meet potential Space Shuttle requirements. Low powder production rates have limited sheet production capacity to about 10,000 pounds per year. But the additional powder-making facilities will result in sheet production capacity of at least 25,000 pounds per year, which should be adequate for Space Shuttle use.

Sheet resulting from this process development program will be supplied to other NASA Centers and contractors for use in Space Shuttle-related programs. This should afford more direct comparison of test results on sheet of reproducible and "pedigreed" quality. This direct comparison is imperative for a material with properties that are so highly dependent on the sheet processing methods used.

Panel Fabrication Processes

Several airframe manufacturers have conducted initial panel fabrication studies to investigate forming and joining processes applicable to making various heat shield configurations from TD-NiCr sheet. Examples of the results of these studies are shown in figure 4. The 18 x 18-inch panel on the left side of figure 4 was produced by General Dynamics-Convair, and the 12 x 18-inch panel on the right side was produced by McDonnell-Douglas. These test panels were produced from cold-formed TD-NiCr sheet. Joining processes used for panel assembly included spot-welding, brazing, and riveting. These panels illustrate that forming and joining of TD-NiCr are certainly feasible processes. However, these panels were fabricated through basically trial-and-error techniques. The fabrication procedures have not been optimized nor are they necessarily applicable to other panel configurations. Thus, there is a need for further development of the panel fabrication processes.

Forming of TD-NiCr sheet into various configurations is not too difficult because this material has moderate ductility at low temperatures (e.g., 10 to 15 percent tensile elongation at room temperatures). This ductility level is similar to that found in many superalloys currently in aircraft use. On this basis, most of the cold-forming operations used for superalloy sheet structures should be applicable to TD-NiCr. However, the forming criteria and limitations have not been fully established for this material. For example, the minimum allowable forming radius and the maximum allowable strain need to be determined. The effects of forming strains are particularly important since over-straining could appreciably change the microstructure and the resultant properties. Also, changes in the sheet processing methods can affect the subsequent formability. So it is imperative that forming operations be closely coupled with the sheet processing methods being developed to better define the inter-relations of these processes. Forming studies such as those described will be evaluated in a contracted program that we plan to initiate later this year.

Joining of TD-NiCr sheet presents more difficult problems than forming. Conventional fusion welding processes are unacceptable for this use because the ThO_2 particles vaporize in the molten weld zone. Also, the controlled microstructure is altered by fusion welding. This results in very weak welds. For this reason, most joints made to date have utilized either mechanical fasteners or brazes. These joining methods have worked reasonably well, but the joints are relatively weak. For example, brazed joints exhibit only about 50 percent of the tensile strength of TD-NiCr at elevated temperatures. This low strength requires that considerable overlapping material be used in the joint region to achieve the full load-carrying capability of the alloy. Thus, better joining methods are desired to improve the strength of joints and reduce weight of the panels.

The most promising methods for this purpose appear to be the newer solid state welding methods. These processes involve contact of the mating surfaces under high pressures and temperatures to produce solid state diffusion across the interface. Welding temperatures used are in the range of 1800° to 2400°F which is below the melting point of the alloys. An example of the type of joint that results is shown in figure 5. This joint was produced in an in-house Lewis program using a high temperature, gas-pressurized autoclave. The welding process is termed "hot isostatic pressure welding." As shown in the microstructure, the interface of the mating surfaces has been completely eliminated by diffusion, and the joint region is indistinguishable from the base material. Tests of this type of joint at 2000°F have indicated that the tensile strength of this joint is as good as those of the base material.

Thus, solid state welding processes offer considerable promise for joining TD-NiCr sheet. But the particular process used to produce the simple butt-joint shown in figure 5 would be difficult to apply to more complex joint regions expected in heat shields. So further work is needed to develop other solid state welding processes that will be applicable to more complex panel configurations. This we plan to pursue through another contracted program later this year.

Design Allowable Properties

After the sheet fabrication process for TD-NiCr has been optimized and a standard product results, a contracted program will be started to determine the design-allowable properties of this sheet. Property measurements to be included in this study are tensile strength, creep strength, modulus of elasticity, thermal fatigue resistance, emissivity, thermal expansivity, and oxidation resistance. Most of these properties will be measured as a function of sheet orientation since TD-NiCr can have anistropic properties as a result of the rolling techniques used.

Also, the effects of simulated re-entry conditions on the properties will have to be determined. These conditions could have appreciable effects on the allowable strengths of materials. For example, USAF-sponsored studies at McDonnell-Douglas and at Solar Aircraft have indicated that exposure to high temperatures and low pressures can result in a decrease in room temperature tensile strength of about 20 percent. So the combined effects of stress, temperature, and air pressure will have to be thoroughly evaluated for this material.

In lieu of the optimized sheet, we have been using currently-available TD-NiCr sheet to get a preliminary "feel" for the two properties that are of most concern to us. These are the high temperature ductility and the oxidation resistance.

High Temperature Ductility: The ductility of TD-NiCr is somewhat unique for metallic systems since it continually decreases with increasing temperature, as shown in figure 6. Most other metals (such as the nickel alloy HS25 and the columbium alloy D-43 in figure 6) go through a ductility minimum at intermediate temperatures, but they usually have higher ductility at their operating temperatures. In the case of currently-available TD-NiCr, the tensile elongation values at the desired operating temperatures are about one to two percent. This low ductility could limit the material's resistance to low-cycle fatigue.

Since adequate high temperature fatigue data were not available for TD-NiCr, flexural fatigue tests are being conducted in an in-house Lewis program. These tests involve cyclic (tension/compression) loading of sheet formed into a semi-circle. The test configuration is shown schematically in figure 7 along with the initial results of the tests. Resistively-heated strips of TD-NiCr are fixed at one end, and push-pull loading is applied to the other end to simulate panel buckling loads. These tests are being run in air at 2200°F at a cyclic frequency of 0.05 hertz.

The initial results for TD-NiCr correlate well with fatigue-life predictions based on creep and tensile-ductility measurements. In fact, the results indicate that the mode of failure in these tests is cyclic creep-rupture from the additive effects of the tensile component of the cyclic stresses applied. Although the allowable strain range for this material is relatively low (as shown in figure 7), these values appear to be tolerable for most heat shield panel designs. However, higher levels of allowable strain are desirable to allow more flexibility in panel design. This appears to be achievable in dispersion-strengthened alloys through increases in high temperature ductility. For example, figure 7 shows that the allowable strain range for TD-Ni (Ni-2ThO_2) is nearly double that of TD-NiCr. This difference is largely due to the differences in high temperature ductility of the two alloys (e.g., 2000°F tensile elongation values of about seven percent for TD-Ni and two percent for TD-NiCr). Thus, improvements in the high temperature ductility of TD-NiCr are desired. This is being pursued in both the sheet processing studies (described previously) and the alloy optimization studies (to be described later). Our goal for these studies is five percent tensile elongation at 2000°F.

Oxidation Resistance: In regard to the oxidation resistance of TD-NiCr, the picture is not quite as clear as that of ductility. Re-entry conditions impose three major parameters which affect the oxidation of TD-NiCr. These parameters are high temperature, high-velocity air flow, and low oxygen pressure. The combined effects of high temperature and air flow on TD-NiCr have been examined in burner-rig tests for jet engine applications. However, the additional effects of low pressure oxygen have not received much attention, and this effect can pose a special problem for this alloy. The oxidation resistance of TD-NiCr is associated with the chromium addition which forms a protective Cr_2O_3 surface layer when heated in air. But low pressure-high velocity conditions can cause vaporization losses and partial depletion of the chromium. This can lead to the formation of less-protective surface layers.

The severity of this effect is currently our prime concern with TD-NiCr because this will probably determine the upper use-temperature of the alloy. Therefore, oxidation tests are now being vigorously pursued in arc-jet tunnels. Because of the complexities of this type of testing, programs have recently been started at both the Langley and Ames Research Centers. The initial tests in these studies are limited to use of small TD-NiCr specimens (0.5 to 1-inch diameter disks) because of test equipment limitations. These tests involve an airstream velocity of about Mach 5, a stagnation point pressure of about 15 Torr, and a 90° orientation of the specimen (in relation to the air stream). The specimen is heated by the arc-jet air stream, and cyclic testing is used with 30 minutes hold at the test temperature for each cycle.

An early test was run at 1800°F for 50 cycles, and the specimen showed little effect of the test conditions. So other tests have been run at 2200°F to evaluate the upper-end of the use-range for this material. The initial results from tests at this temperature are plotted in figure 8. For comparison, static test results obtained at Lewis in a vacuum furnace are also shown on this plot.

These results indicate the following:

1. Comparison of the static and dynamic test results confirm that the combined effects of low pressure and high velocity appreciably increase the amount of surface loss from TD-NiCr at this temperature. Thus, simple vacuum furnace tests are not adequate to determine the true oxidizing effects of re-entry.

2. Comparison of the two sets of dynamic test results indicate relatively good correlation considering the differences in test conditions (e.g., enthalpy, material lot, specimen size and thickness, etc.). The amount of surface recession (1.5 to 2 mils in 25 hours at 2200°F) might be considered a tolerable level for Space Shuttle use. But post-test metallographic examination of the specimens indicates the formation of considerable amounts of internal porosity. Since this porosity would probably cause a substantial decrease in material strength, the material would be considered marginal for use under these severe test conditions.

Much more of this type of dynamic testing is needed to determine the true oxidation resistance of this material under re-entry conditions and to establish the upper limit on its usable temperature range. This must include a study of the effects of test variables. For example, the initial tests were run at enthalpy values of about 1000 to 3000 Btu/lb. which are considerably less than the values expected for Space Shuttle heat shields (i.e., greater than 10,000 Btu/lb.). Theoretical analysis of this effect by Langley personnel indicates that the material's lifetime is proportional to the square of the enthalpy. If this correlation is correct, these arc-jet tunnel tests are much more severe than re-entry conditions. Therefore, better re-entry simulation tests must be devised before the full capabilities of TD-NiCr can be determined.

BACK-UP DEVELOPMENT PROGRAM

Alloy Optimization

In parallel with the development and evaluation of TD-NiCr, our back-up development program includes evaluation of the effects of alloy compositional changes. This work is being done through in-house studies at Lewis and the Fansteel sheet-process development program, plus some other augmenting contracted studies. This work is primarily directed toward improving the oxidation resistance, but some of the effort will also seek improvements in high temperature ductility. Various combinations of alloying additions will be evaluated to determine if they have beneficial effects on these properties.

The feasibility of this alloying approach to improve oxidation resistance was demonstrated in recent tests on a Ni-Cr-ThO_2 alloy modified with aluminum and yttrium additions. These elements are known to have beneficial effects on the oxidation resistance of nickel and iron-base alloys. The improved behavior is thought to be due to the formation of an Al_2O_3 surface layer which is more protective than the Cr_2O_3 layer formed on TD-NiCr. A specimen of a dispersion-strengthened Ni-Cr-Al-Y alloy (produced by Fansteel for the Pratt and Whitney Aircraft Co.) was tested in an arc-jet tunnel at the Ames Research Center. Test conditions were similar to those used for the TD-NiCr tests described earlier. After forty 30-minute cycles to 2200°F, the specimen exhibited no significant surface recession nor internal porosity. Thus, this initial test indicates that this alloy has much better oxidation resistance than TD-NiCr.

However, this modified alloy does not represent the full solution to the oxidation problem with TD-NiCr. Tests on the initial samples of this alloy indicate that the alloy additions also reduce the low temperature ductility of the alloy to a level which makes sheet forming operations extremely difficult. Also, the formation of an Al_2O_3 surface layer reduces the emissivity of the sheet. Thus, more extensive study of alloying additions (and processing methods) is needed to achieve the proper balance in desired properties.

Alternate Sheet Manufacturing Process

Another part of our back-up program involves development of alternate supply sources and manufacturing processes for Ni-Cr-ThO_2 sheet. For this purpose, we recently awarded a contract to Sherritt-Gordon Mines, Ltd. for development of a much different sheet manufacturing process. This process will involve the diffusion of chromium into wrought Ni-2ThO_2 sheet which is currently available in production quantities. The process has been successfully demonstrated for small, laboratory-scale specimens, but scale-up of the process is needed to produce large sheets. The current program will involve process development to produce sheet sizes of at least 24 x 48 inches in the thicknesses of 0.010 to 0.030 inches.

The differences in the microstructures of the Ni-Cr-ThO_2 sheet produced by either Fansteel's powder blending process or Sherritt-Gordon's chromium-diffusion process are illustrated in figure 9. The TD-NiCr specimen contains more elongated grains which result in higher strength than the more equiaxed structure of the experimental material. However, the experimental product is much "cleaner" which results in better ductility and possibly better resistance to porosity formation under oxidizing conditions. The dark spots in the TD-NiCr sheet are Cr_2O_3 particles which are formed during the powder processing steps and are difficult to eliminate during subsequent processing. These oxides do not form in the alternate processing method since chromium is not introduced until the material is in a wrought-sheet form. Thus, this alternate process offers potential for possibly achieving better properties, if the process can be fully developed for large-size sheet.

(Note: We want to acknowledge the cooperation and support of the Canadian government, through the Defense Research Board and the Canadian Commercial Corporation in the award of Contract NAS3-14313 with Sherritt-Gordon.)

CONCLUSIONS

In conclusion, I have tried to show that dispersion-strengthened alloys, such as TD-NiCr, offer great potential for use in re-entry heat shields at temperatures in the range of about 1800° to 2200°F. However, these materials are relatively new and thus have their share of problems which must be resolved before their potential can be fulfilled.

Improvements are needed in the processing of TD-NiCr sheet and the fabrication of heat shield panels, but these seem to be achievable in a relatively straight-forward manner. And we expect that these needed advancements will result from the various technology programs now being started.

The most crucial problem area for this material appears to be its oxidation resistance. TD-NiCr is probably acceptable for initial flight-use, but better simulation testing is needed before we can establish its upper use-temperature for eventual use in 100 flights or more.

In short, TD-NiCr offers both great potential and some problems! But I personally feel that the problems associated with this material are much easier to solve than the problems associated with any other material considered for use in Space Shuttle thermal protection systems at temperatures above 1800°F.

Figure 1

RELATIVE STRENGTHS OF HEAT SHIELD MATERIALS

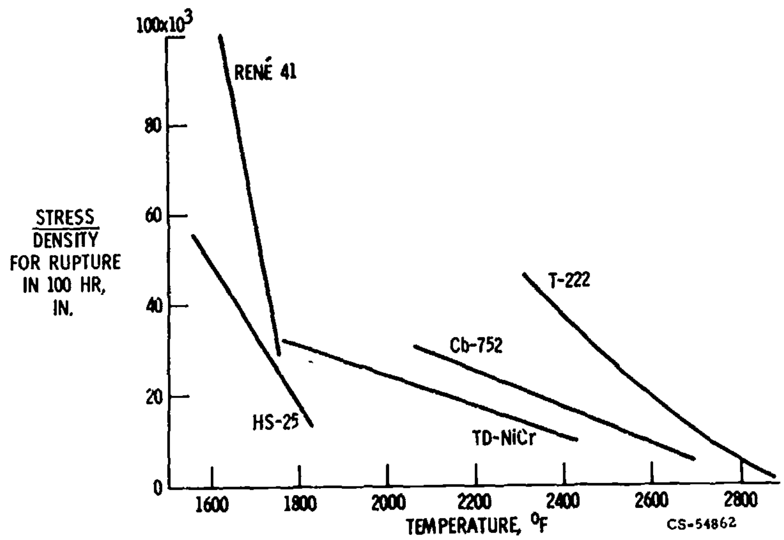

Figure 2

DISPERSION STRENGTHENED ALLOYS

ALLOY COMPOSITION	DISPERSOID	PRODUCER	STATUS
Ni	ThO_2	FANSTEEL SHERRITT-GORDON SYLVANIA CURTISS-WRIGHT	PRODUCTION PRODUCTION EXPERIMENTAL EXPERIMENTAL
Ni + 20Cr	ThO_2	FANSTEEL SHERRITT-GORDON CURTISS-WRIGHT	SEMIPRODUCTION EXPERIMENTAL EXPERIMENTAL
Ni + (5-20)Mo	ThO_2	FANSTEEL SYLVANIA	EXPERIMENTAL EXPERIMENTAL
Ni + Cr + Al + Y	ThO_2	P & W/FANSTEEL	EXPERIMENTAL
Ni + Cr + Al + Ti + C	$Y_2O_3 + Al_2O_3$	INCO	EXPERIMENTAL
Co + (20-30)Cr + (15-20)Ni	ThO_2	FANSTEEL SYLVANIA	EXPERIMENTAL EXPERIMENTAL
Co + 20Cr + 10Ni + 15W	ThO_2	CURTISS-WRIGHT	EXPERIMENTAL
Fe + 15Cr + 4Al + 1Y	Al_2O_3	GE	EXPERIMENTAL

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Figure 3

EFFECT OF PROCESSING METHODS ON THE STRENGTH OF TD-NiCr SHEET

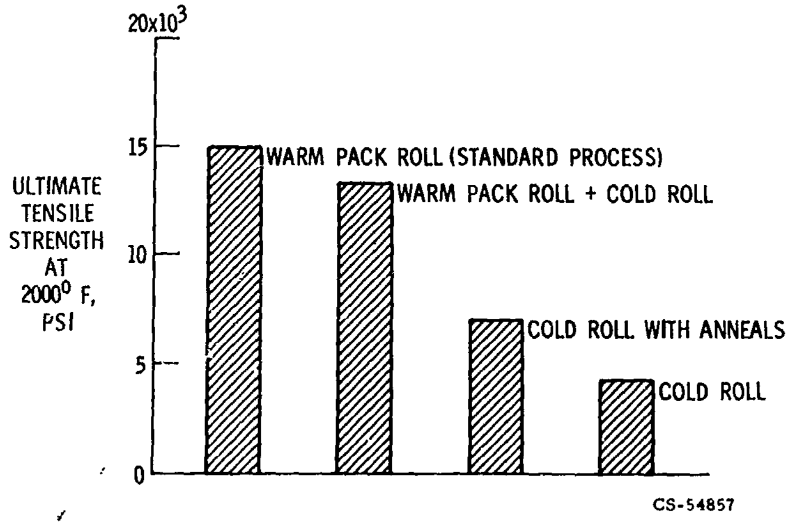

Figure 4

EXPERIMENTAL PANELS OF $Ni-Cr-ThO_2$ SHEET

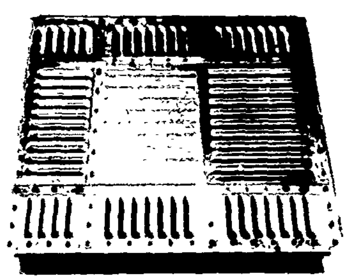

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Figure 5

SOLID STATE WELD IN $Ni-Cr-ThO_2$ SHEET

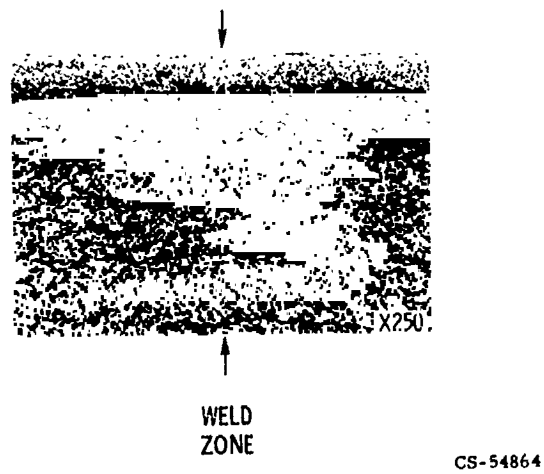

Figure 6

DUCTILITY OF HEAT SHIELD MATERIALS

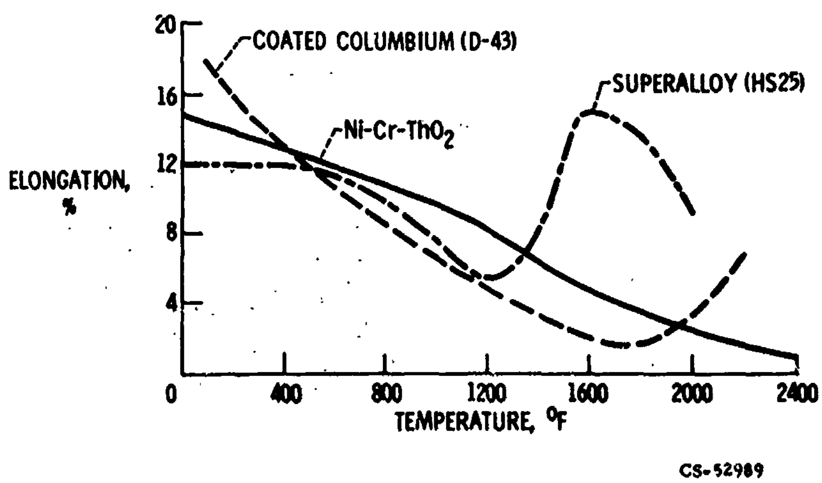

Figure 7

EFFECTS OF HIGH TEMPERATURE FLEXURAL FATIGUE ON SHEET OF DISPERSION-STRENGTHENED ALLOYS

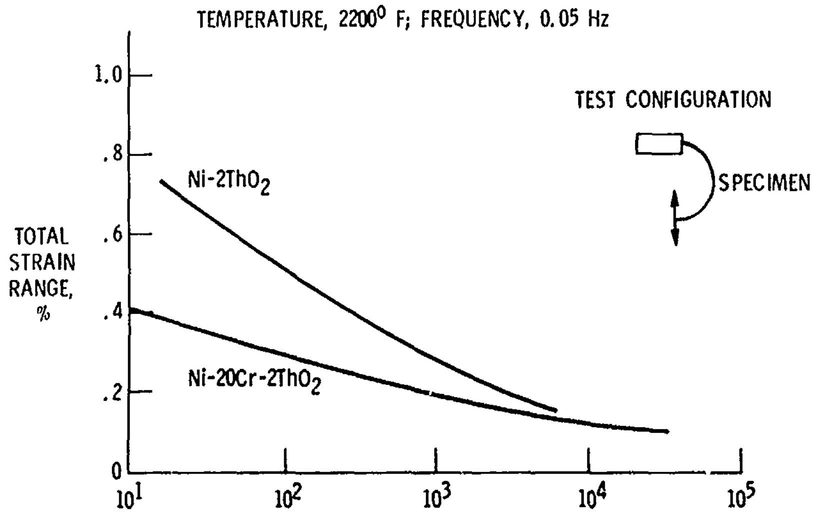

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Figure 8

LOW PRESSURE OXIDATION EFFECTS ON TD-NiCr SHEET

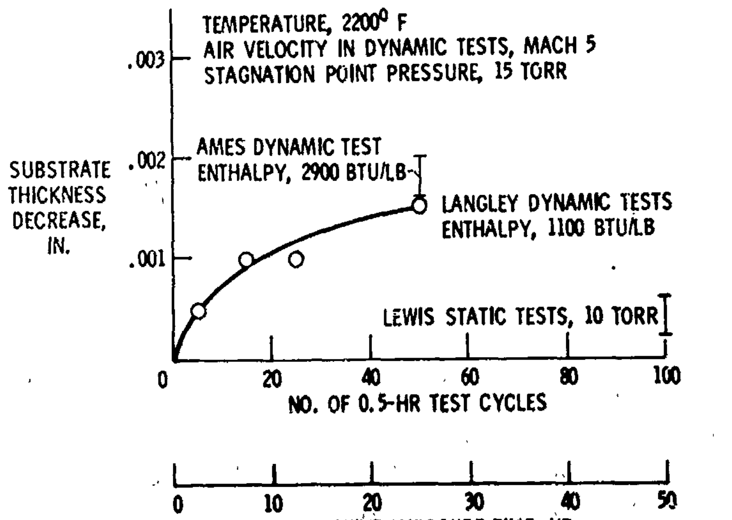

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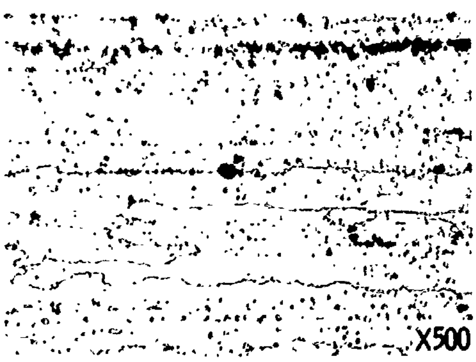
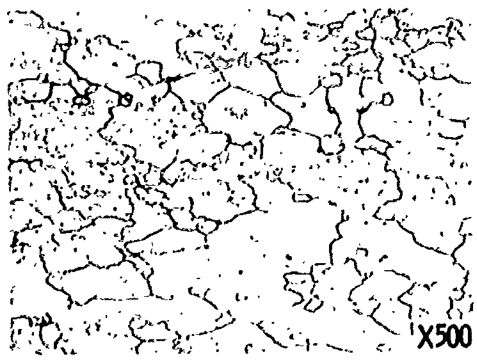
Figure 9

MICROSTRUCTURES OF Ni-20Cr-2ThO_2 SHEET



PRODUCED BY CURRENT PROCESSING METHOD (FROM PRODUCTION LOT)



PRODUCED BY ALTERNATE PROCESSING METHOD (FROM EXPERIMENTAL LOT)

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REFRACTORY METAL HEAT SHIELD TECHNOLOGY FOR SPACE SHUTTLE

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SUMMARY

In developing heat shields for thermal protection systems of space shuttle, the most important factor to be considered is reliability. A high probability that flight weight shields will survive the complex environment for 100 flights with minimum refurbishment is the goal of the NASA technology development effort. This paper first briefly reviews the status of coated refractory metals for heat shields. The important environmental factors which these shields must survive are listed. Specific heat shield configurations which have been proposed are reviewed and some of the configuration restraints which are important for coated refractory metals are discussed. A new heat shield configuration is proposed to satisfy these restraints. Finally, NASA contractual and in-house studies pertaining to coated refractory metal heat shield technology are listed.

~~Stein-Rummler-Jackson~~

INTRODUCTION

It is generally recognized that thermal protection systems (TPS), consisting of heat shields, shield supports and thermal insulation to protect the primary structure, are among those systems which require technology development to perform the shuttle mission. The most important factor to be developed for flight weight metallic heat shields of these thermal protection systems is reliability. There must be a high probability that the shields will survive the operational environment for 100 flights with minimum refurbishment. Two other important goals in the development of these systems are minimum weight and cost.

This paper reviews the status of coated refractory metals for heat shields and the environment the heat shields must survive. Refractory metal heat shield configurations which have been proposed are described along with configurational restraints which are important for refractory metal construction. A new heat shield concept is proposed to satisfy these restraints. Finally, a brief review of NASA contractual and in-house studies pertaining to coated refractory metal heat shields is given. These studies and others to come in the future make up the NASA technology development program in this area. The purpose of these studies is to advance the state-of-the-art in refractory metal heat shield thermal protection systems to the point where a reliable system is ready for the shuttle application.

APPLICATION OF TPS MATERIALS

Material usage currently anticipated for the shuttle orbiter thermal protection system is shown. For the maximum use temperatures listed, titanium alloys and nickel- or cobalt-base superalloys may cover from 65 to 90 percent of the orbiter surface, depending on vehicle configuration and reentry trajectory. Technology for titanium and superalloys appears to be pretty well in hand with only demonstration of full-size heat shields and validation of weight and cost predictions remaining prior to shuttle application.

All the other materials listed here are candidates for the hotter portions of the orbiter surface. Technology for dispersion-stabilized alloys such as TD-NiCr, compacted ceramic fibers, carbon/carbon composites, and ablators has been covered in other papers in this conference.

This paper covers technology for coated columbium and tantalum alloys; these refractory metals are prime candidates for heat shields operating at temperatures from 2200° F to about 2800° F. The latter temperature appears to be the upper limit for use of current oxidation resistant coatings in the shuttle application. Columbium and tantalum alloys are currently envisioned for 5 to 10 percent of the orbiter surface. However, it is important to realize that, should dispersion-stabilized alloys like TD-NiCr not be available in time, coated columbium alloys will be prime candidates for up to 25 percent of the thermal protection system. Thus, a large effort in coated columbium alloy technology has begun this year, as indicated subsequently.

MATERIAL	APPROXIMATE MAXIMUM USE TEMPERATURE, °F	APPROXIMATE ORBITER AREA BELOW MAXIMUM USE TEMP., Percent
Titanium	800	25-50
Superalloys	1800	65-90
TD-NiCr	2200	85-95
Coated Columbium	2400	90-98
Compacted Ceramic Fibers	2400 (future 3000)	90-98
Coated Tantalum	2800	95-100
Carbon/Carbon Composites	2600 (future >3000)	95-100
Ablators, Diborides	>3000	95-100

CURRENT CAPABILITIES OF COATED REFRACTORY METALS

Development of high strength columbium and tantalum alloys and oxidation resistant coatings for them has been in progress for the past decade, under Department of Defense and NASA sponsorship. These studies have provided a major improvement in coating technology, the fused slurry silicide coatings for columbium alloys which, in our opinion, are ready for scaleup to full-size heat shields.

This slide indicates the current status of the coated refractory metals presently considered for heat shields. For columbium the maximum use temperature for the shuttle heat shield application is about 2400° F; this limitation is due primarily to creep limits in current alloys. Based on static air environmental tests of small fabricated specimens, a 60 to 100 flight capability is possible for silicide coated Cb 752 alloy. However, this potential remains to be proven for full-size heat shields in high-speed airflow environments. Another advantage of the fused slurry silicide coatings is that defected specimen tests indicate the potential reusability of these coated alloys, even with small failure sites, for several flights with no massive heat shield damage. Again, this remains to be proven in simulated shuttle environment tests.

For tantalum, on the other hand, although alloys have been developed with acceptable strengths above 2800° F, no acceptable coatings are available for reliable reuse at the temperatures of interest.

COLUMBIUM ALLOYS

1. Nominal use temperature limited to ~2400° F by creep strength of alloys
2. 60-100 flight capability possible for silicide-coated Cb-752
3. Defected coatings possibly usable for several flights

TANTALUM ALLOYS

1. Alloys available with acceptable strength to >2800° F
2. No acceptable coatings available for reliable flight use

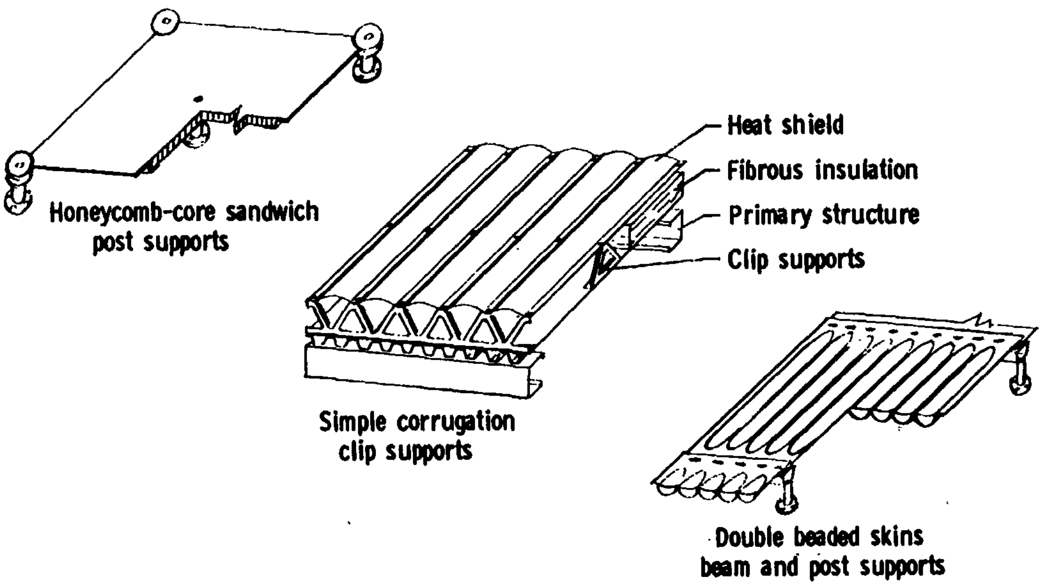
ENVIRONMENTAL FACTORS FOR COATED REFRACTORY METAL THERMAL PROTECTION SYSTEMS

The environmental factors which may affect the design of a thermal protection system for space shuttle are listed. For each phase of the orbiter mission several factors and the possible modes of damage for the thermal protection system are given. An overall assessment shows that physical coating damage by erosion and impact can come during launch, orbit, landing and ground operations while chemical damage to coatings is most likely during ascent and reentry; general corrosion may be a problem for many materials in ground operations. Noise and vibration damage to heat shields and insulation is possible during launch, ascent, reentry, and landing. The reentry subjects the system to maximum temperature with some pressure loading so that oxidation, creep damage and panel buckling become possible during that mission phase but maximum pressure loadings on heat shields will occur during either the ascent phase when the heat shield is at a moderately elevated temperature or during the subsonic cruise phase, depending on venting details in the shuttle vehicle. In general, the design, fabrication and testing of thermal protection systems to demonstrate developed technology must consider all of these environmental effects. The tests should be run to simulate the 100 flight operational life span projected for shuttle. Complex tests of this type are not possible at this time. Tests which can be run are expensive for full-size hardware and still cannot include all important test parameters simultaneously. Consequently, extreme care must be taken in planning test programs which will verify thermal protection system technology.

Mission Phase	Environmental Factors	Possible TPS Damage Mode
Launch	Rain, hail or debris Noise Vibration	Coating erosion Support and fastener fatigue Panel flutter
Ascent	Vibration Noise Surface heating Max. dynamic pressure	Flutter Sonic fatigue "Pest" failure Static strength
Orbit	Vacuum Micrometeoroid impact Solar radiation	Insulation degradation Coating erosion or pitting High internal heat loads
Reentry	Max. surface temperature Surface pressure Aerodynamic noise	Cumulative creep damage Panel buckling Coating oxidation Insulation degradation Thermal fatigue
Cruise and landing	Pressure differential Rain, hail or ground debris Noise and vibration	Static strength Coating erosion or pitting Support or fastener fatigue
Ground operations	Variable temp. and humidity Ground handling	Insulation degradation Substructure corrosion Impact damage

TYPICAL METALLIC HEAT SHIELD CONFIGURATIONS

Three heat shield configurations which are typical of those that have been suggested for shuttle applications are shown. The center sketch indicates the entire thermal protection system consisting of heat shield panel, panel supports, and fibrous insulation to protect the primary structure from the external environment. It should be noted that fibrous insulation may require packaging. The three shield configurations pictured here include a brazed honeycomb core sandwich structure with post supports, a simple corrugation with clip supports and a double beaded skin panel with beam and post supports. These configurations were developed primarily for uncoated superalloy construction and, in our opinion, may run into serious problems of reliability if constructed from coated refractory metals.



CONFIGURATION RESTRAINTS FOR COATED REFRACTORY METAL TPS DESIGN

Even with the assumption that current technology can provide a uniformly distributed coating of adequate thickness on a columbium substrate, insufficient attention to heat shield design details will still lead to premature coating failures. Nine configuration restraints are listed which should be taken into account in any coated refractory metal TPS design. Edge and corner radii should be maximized for good coating performance and minimum coating handling damage. Adequate radii may be a problem with both the simple corrugation and the beaded panel (see previous slide). Elimination of faying surfaces prior to coating appears impossible with both the honeycomb and the double beaded panel. Coated sliding surfaces supporting airloads are undesirable as well as coated supports which must withstand appreciable strains during each heating cycle; thus the support systems shown the previous slide appear to be undesirable for coated refractory metal construction.

Configuration restraints which preclude coating of all panel surfaces are a problem primarily for a brazed honeycomb sandwich. The high surface area to volume ratio of the thin gage uncoated honeycomb core makes this panel type extremely vulnerable to oxidation after any external panel damage. Regions in fastener areas may also present hidden surfaces and, in general, the use of coated refractory metal fasteners is undesirable and should be minimized. All coated surfaces should be visible prior to installation because, in our opinion, completely acceptable nondestructive evaluation procedures for coated panels with interior coated surfaces are unavailable at this time. The double beaded panel construction suffers on this point. In fact, NDE for recertification of any panel configuration in place on a structure is in its infancy.

The avoidance of braze/coating contact is an important factor in coating protectiveness for the braze and coating systems currently under consideration. This problem is most significant for the honeycomb sandwich but brazing may also be required for the double beaded panel if other fabrication methods are unsatisfactory for full-size shields. The final item here is avoidance of contact of the coated heat shield or its support with incompatible materials which could cause possible chemical or eutectic reactions. Material compatibility is a general problem for any heat shield design.

Maximize edge and corner radii

Minimize faying surfaces prior to coating

Minimize sliding surfaces supporting airloads

Minimize coating strains

Allow all surfaces to be coated

Coated surfaces visible prior to installation

Minimize use of coated fasteners

Avoid braze/coating contact

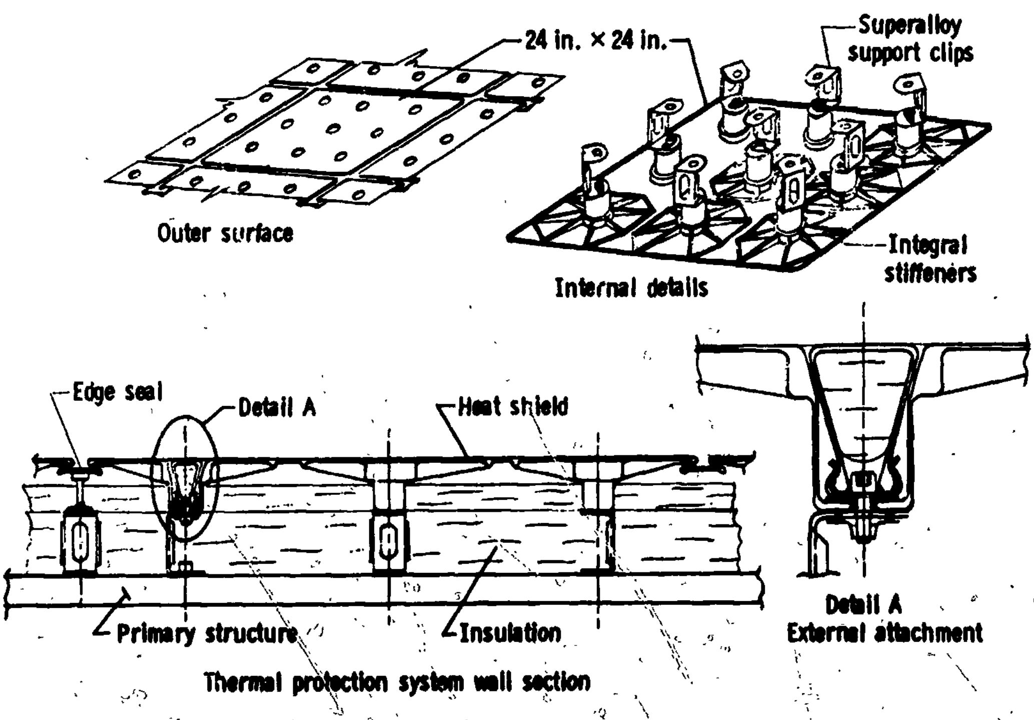
Avoid contact with incompatible materials

COATED REFRACTORY METAL HEAT SHIELD CONCEPT

Consideration of these configuration restraints noted in the preceding slide for existing designs has led to new ideas for coated refractory metal heat shields. An example of one such concept is shown. It must be pointed out that as of this date the 2-ft. square heat shield shown here is a concept only; it has not been fully analyzed and has not been fabricated.

The aerodynamic surface of the heat shield is shown at the upper left and the internal details are shown on the upper right. These details are not to scale; they are highlighted so that the concept may be more easily understood. A wall section of a complete TPS using this heat shield is shown at the lower left and a detail of the external attachment is shown at the lower right. The aerodynamic outer surface of the shields is relatively smooth. Thermal bowing is minimized by the use of shallow depth integral stiffeners which are radially oriented beams supporting the pressure difference loading on the shield. Overall bowing of the panel is prevented by the use of multiple supports. Differential expansion between shield and primary structure is accommodated by the flexible superalloy support clips which are oriented to support drag shear in any direction while imposing little restraint to thermal expansion. The shields are attached to the structure from the outside by superalloy screws. The access hole to the screw is plugged with a coated refractory metal cup filled with insulation. The connections to the superalloy clips are made at the 1500° F isotherm. This concept utilizes overlapping segmented edge seals.

Each design constraint (previous slide) is considered in this concept. Generous edge and corner radii are used in all refractory metal members. No faying surfaces exist before coating application and no load-carrying coated sliding surfaces are used. Coating strains have been minimized. All surfaces are coated and all coated surfaces are visible prior to installation. No coated fasteners are required and no brazing is used. The only contact with possible incompatible materials is in the external attachment plug which could be readily replaced and at the 1500° F connection to the superalloy supports. Further design, thermostructural analysis, fabrication, and realistic testing of this coated refractory metal heat shield concept is necessary to evaluate its performance.



COATED REFRACTORY METAL TECHNOLOGY RESPONSIBILITIES

A list of technology areas and responsible centers for the coated refractory metal technology programs is shown. Coating development and scaleup programs are under Lewis Research Center. Design, fabrication and evaluation studies on full-size thermal protection system panels are the prime responsibility of the Langley Research Center with the Ames Research Center taking a significant role in the evaluation phase. Development of nondestructive evaluation and defect-repair processes are the responsibility of the Marshall Space Flight Center.

It should be noted that, although these centers have prime responsibility for the areas listed, other NASA centers will contribute their efforts where specific expertise exists. To develop this technology, both contractual and in-house studies have begun as listed in the following slides.

TECHNOLOGY AREA	RESPONSIBLE CENTER
Coatings Development and Process Scaleup	LeRC
Design, Fabrication, and Evaluation of Full-Size, Integrated TPS Panels	LaRC (ARC)
Development of NDT and Defect-Repair Process	MSFC

COATED REFRACTORY METAL TPS TECHNOLOGY PROGRAMS

Contractual Programs

This slide lists the NASA contractual programs in coated refractory metals for FY 70 and FY 71. All studies listed here have either recently been initiated or will begin in the very near future and no comprehensive technical results will be available for several months yet. It is expected that these efforts will need to be supplemented as design criteria become more specific and as additional problem areas are identified by the existing studies.

Most of these studies involve columbium alloys because of the availability of at least one coating/substrate system which appears ready for scaleup to full-size components. This is the purpose of the Langley Research Center coated columbium study listed first. A refurbishment cost study also undertaken by Langley will include coated refractory metal shields. Under Lewis Research Center support, materials scaleup and the optimization of existing columbium systems is under way and contracts to develop an improved tantalum alloy coating will begin soon. As noted previously, tantalum heat shield technology awaits the development of such a coating. Marshall Space Flight Center has four contractual programs, all for coated columbium alloys. These include evaluations of commercially available coatings and coating emittance to establish design properties and studies of field repair methods and defect tolerance and crack propagation behavior.

TITLE	RESPONSIBLE CENTER
Coated Columbium-Alloy Heat Shields for Space Shuttle Application	LaRC
Refurbishment Cost Study of a Space Shuttle Vehicle	LaRC
Scaleup and Optimization of Fused Slurry Silicide Coatings for Columbium Alloys	LeRC
Development of Improved Fused Slurry Silicide Coatings for Tantalum Alloys	LeRC
Evaluation of Oxidation Resistant Coatings for Columbium Alloys	MSFC
High Temperature Emissivity Measurements on Coated Columbium Alloys	MSFC
Field Repair Methods for Coated Columbium	MSFC
Defect Tolerance and Crack Propagation Behavior of Coated Columbium Alloys	MSFC

COATED REFRACTORY METAL TECHNOLOGY PROGRAMS

NASA in-house coated refractory metal studies which are either under way or will begin soon are listed. At the Langley Research Center studies are under way to provide a good definition of the local thermal protection system environment for coated refractory metal shields. Langley will evaluate the performance of coated columbium sheet in the best simulation of this environment which we can provide.

Lewis Research Center is exploring both improvements to existing fused slurry coatings for columbium and advanced coatings for columbium alloys. At the Marshall Space Flight Center there is an in-house effort to obtain and codify design properties for coated columbium alloys and there is in-house work to develop columbium alloy fabrication procedures. The final item listed is an assessment by the Manned Spacecraft Center of the properties of refractory metals with commercially available coatings and of nondestructive evaluation procedures for these material systems.

TITLE	RESPONSIBLE CENTER
Definition of Local TPS Environment	LaRC
Performance of Coated Columbium Alloy Materials in a Simulated Shuttle Environment	LaRC, ARC
Exploratory Studies of Fused Slurry Coatings for Columbium Alloys	LeRC
Exploratory Investigation of Advanced Coatings for Columbium Alloys	LeRC
Design Properties for Coated Columbium Alloys	MSFC
Columbium Alloy TPS Fabrication Studies	MSFC
Refractory Alloy Studies - Materials and NDE Assessment	MSC

CONCLUDING REMARKS

A review of the NASA technology programs in coated refractory metal heat shields for shuttle application has been presented. It is anticipated that these programs and others to come in the future will provide the necessary materials system developments, heat shield designs, and the experimental verification required to demonstrate the applicability of coated refractory metal thermal protection systems for space shuttle.

N70-42990

NONMETALLIC REUSABLE THERMAL PROTECTION SYSTEMS

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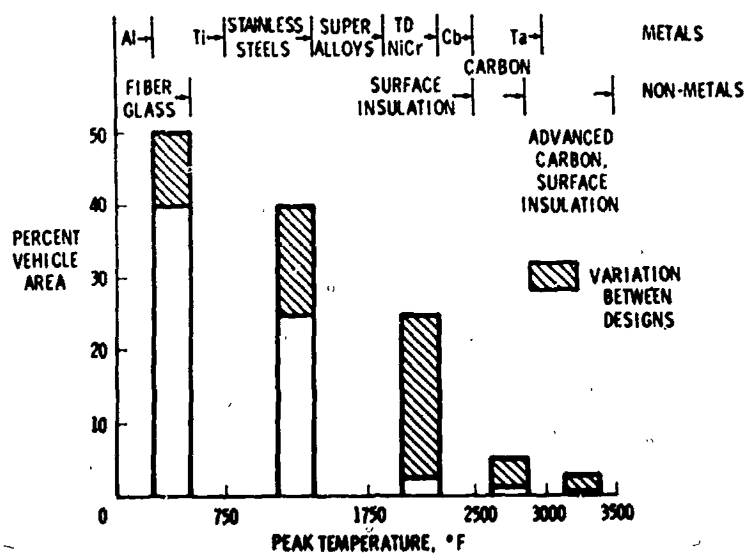
INTRODUCTION

The most highly developed and most thoroughly investigated reusable thermal protection materials for use on the shuttle are metals which have been alloyed and specially treated to exhibit oxidation resistance and high strength at elevated temperatures. However, there exists nonmetallic materials which exhibit inherent chemical stability and/or high strength properties under entry conditions, and thus are potentially better choices for reusable heat shield application. The development of two such materials, refractory oxide and carbon composites, are being pursued to assure that the potential advantages of these materials are adequately investigated.

APPLICATION OF MATERIALS TO THE SHUTTLE ORBITER

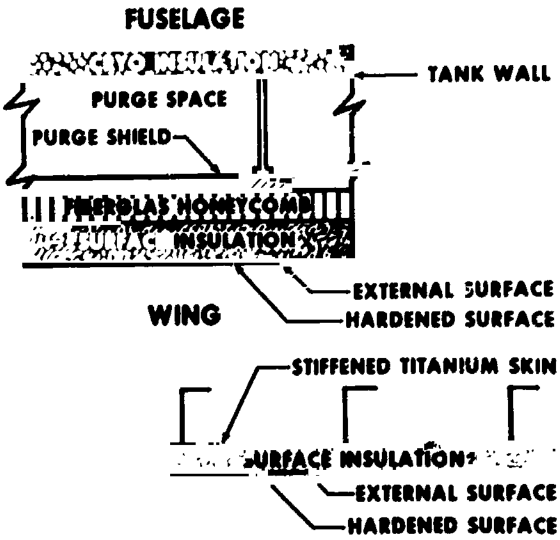
THE FIRST CHART SHOWS THE AREAS OF THE ORBITER WHICH WILL EXPERIENCE PEAK ENTRY TEMPERATURES IN FOUR RANGES. THE SHADED AREAS INDICATE THE VARIATION IN THESE AREAS WHICH MIGHT OCCUR DEPENDING ON THE EXACT VEHICLE CONFIGURATION, ENTRY ATTITUDE, AND TRAJECTORY.

IT WILL BE NOTED THAT A LARGE AREA CAN BE PROTECTED BY STRUCTURAL METALS FOR WHICH THE STATE-OF-THE-ART IS WELL ESTABLISHED. HOWEVER, PROTECTION OF THE REMAINING HALF OF THE AREA REQUIRES THE USE OF MORE ADVANCED MATERIALS, SUCH AS THOSE INDICATED AT THE TOP OF THE FIGURE. NOTE ALSO THAT THE WEIGHT, COST, AND DIFFICULTY OF APPLICATION OF THE METALS GENERALLY INCREASES TOWARD HIGHER TEMPERATURES. THE SURFACE INSULATION CONCEPT, HOWEVER, IS COMPETITIVE IN WEIGHT AND SIMPLICITY WITH THE SUPERALLOYS, AND YET ITS TEMPERATURE RANGE EXTENDS TO THE HIGHEST TEMPERATURE FOR WHICH DEVELOPED METAL SYSTEMS ARE AVAILABLE. THE CARBON SYSTEM EXTENDS IN CAPABILITY TO EVEN HIGHER TEMPERATURES, AGAIN WITHOUT DIFFERENT WEIGHT OR COST PENALTY.



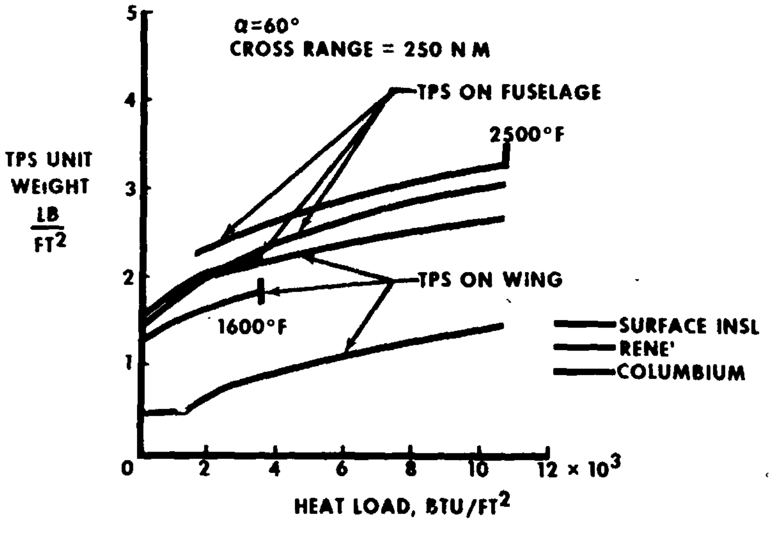
NONMETALLIC TPS CONCEPTS

THE APPLICATION OF REFRACTORY OXIDES BEING DEVELOPED IS THAT OF SURFACE INSULATION SYSTEMS. IN THIS CONCEPT, THE REFRACTORY OXIDE IS USED IN THE FORM OF A FIBER INSULATION MAT WHICH HAS BEEN RIGIDIZED WITH A CERAMIC BINDER AND COVERED WITH A DENSE, TOUGH SURFACE LAYER. THIS MATERIAL IS THEN BONDED TO SUBSTRATE PANNELS WHICH MAKE UP THE AEROSHELL OF THE VEHICLE, OR BONDED DIRECTLY TO THE LOAD CARRYING STRUCTURE. THUS, SOME OF THE SIMPLICITIES OF THE FAMILIAR ABLATOR SYSTEMS ARE REALIZED.



WEIGHT COMPARISON OF NONMETALLIC AND METALLIC SYSTEMS

TYPICAL METALLIC AND SURFACE INSULATION SYSTEMS HAVE BEEN SIZED FOR A SHORT RANGE SHUTTLE ORBITER. THE UNIT WEIGHTS OBTAINED ARE SHOWN AS A FUNCTION OF HEAT LOAD, WHICH VARIES WITH LOCATION ON THE VEHICLE. WEIGHTS ARE SHOWN BOTH FOR APPLICATIONS REQUIRING THE USE OF A SUBSTRATE PANNEL (FUSELAGE) AND THOSE FOR WHICH A LOAD BEARING STRUCTURE CAN BE USED AS THE SUBSTRATE (WING). NOTE THAT THE WEIGHTS OF ALL THREE SYSTEMS STUDIED (SUPERALLOY, COLUMBIUM, AND SURFACE INSULATION) ARE COMPARABLE FOR THE CASE WHERE A SUBSTRATE MUST BE CHARGED TO THE TPS, HOWEVER, WHERE A LOAD CARRYING STRUCTURAL SURFACE IS AVAILABLE, A 50% WEIGHT ADVANTAGE IS INDICATED FOR THE SURFACE INSULATION.



SURFACE INSULATION SYSTEM

THERE ARE SEVERAL COMBINATIONS OF FIBERS AND BINDERS WHICH CAN BE USED TO FORM A SURFACE INSULATION MATERIAL. IN GENERAL, ALL THESE MATERIALS SHARE THE ADVANTAGES OF SIMPLICITY, STABILITY, AND WEIGHT-EFFECTIVENESS. HOWEVER, THEY ALSO SHARE PROBLEMS. ALTHOUGH LESS BRITTLE THAN THE USUAL CERAMICS, MECHANICAL PROPERTIES CONSTRAIN THE SYSTEM DESIGN. THE OTHER PRIMARY PROBLEM IS THAT OF A COATING WHICH IS SUFFICIENTLY TOUGH AND WATER RESISTANT TO SURVIVE THE TOTAL SHUTTLE MISSION ENVIRONMENT.

FIBER	BINDER
SiO_2	COLLOIDAL SiO_2
$Al_2O_3 \cdot SiO_2$	PYROLIZED SILICONE RESINS
	PHOSPHATES
ZrO_2	ZIRCONIUM SALTS

ADVANTAGES	PROBLEMS	POSSIBLE SOLUTION
THERMAL EFFICIENCY	DEVITRIFICATION	TRACE INHIBITORS
REUSE	WATER ABSORPTION	PORE GEOMETRY AND NON-WETTING FIBERS; COATINGS
OVERTEMP CAPABILITY	EROSION RESIST. COATING	SURFACE DENSIFICATION
DESIGN SIMPLICITY		
COMPATIBLE WITH WEIGHT-EFFICIENT STRUCTURES	LOW MECHANICAL STRENGTHS	REINFORCEMENTS
CONVENTIONAL ATTACHMENT AT LOW TEMP	LOW STRAIN LIMIT	DESIGN
INSPECTION		

CARBON-CARBON COMPOSITES

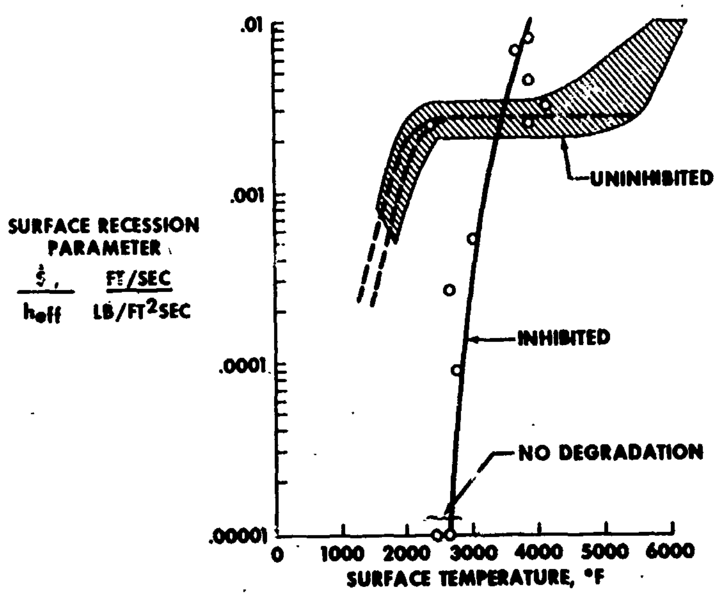
THERE ARE SEVERAL DIFFERENT TYPES OF CARBON FIBER WHICH CAN BE USED TO MAKE THE CARBON-BONDED CARBON COMPOSITE MATERIAL. HOWEVER, THOSE BEING PURSUED FOR SHUTTLE APPLICATION ARE USED IN A CLOTH FORM, IMPREGNATED WITH A PHENOLIC RESIN. THIS PRE-PREG IS LAID UP IN MOLDS MUCH LIKE FIBERGLAS, AND CURED. SYSTEMS DESIGNED USING THIS MATERIAL TAKE THE SAME BASIC FORM AS THOSE DESIGNED USING METALS. AFTER THE LAY-UP IS CURED, THE ARTICLE IS SUBJECTED TO FURTHER HEATING AT TEMPERATURES ABOVE 2000°F, AND THE PHENOLIC RESIN AS PYROLIZED INTO CARBON. INHIBITORS USED WITH THIS SYSTEM ARE USUALLY METALS WHICH FORM CARBIDES ON HEATING WITH THE CARBON MATRIX, AND OVER-LAYS OF A REFRACTORY OXIDES. THE ADVANTAGES OF THIS SYSTEM OVER METALLICS LIE IN THE INHERENT HIGH TEMPERATURE STRENGTH OF CARBON, AND THE HIGH CHEMICAL STABILITY OF THE CARBIDES AND OXIDE USED TO INHIBIT OXIDATION. IT SHOULD BE NOTED THAT BORIDES CAN ALSO BE USED ON A SECONDARY INHIBITOR WITH CARBIDES. THE RESULTING SYSTEMS SHARE THE HIGH TEMPERATURE ATTACHMENT PROBLEMS WITH METALS, AND ARE NOT DUCTILE, THUS FACING DESIGN AND FABRICATION PROBLEMS.

MATERIAL	INHIBITING ELEMENT	PHYSICAL DISTRIBUTION
REINFORCEMENT	BORON	IN-DEPTH DIFFUSION
CARBON, GRAPHITE	SILICON	DISPERSED WITH BINDER
CLOTH OR FIBER	ZIRCONIUM	SURFACE OVERLAY
BINDER	OTHER CARBIDE AND OXIDE FORMERS	COMBINATION OF ABOVE
PYROLYZED RESIN		

ADVANTAGES	PROBLEMS	POSSIBLE APPROACH
HIGH TEMP STABILITY	INHIBITOR ABOVE 3000° F	MATERIAL DEVELOPMENT
LIGHT WEIGHT	STRUCTURAL COMPATIBILITY	DESIGN DEVELOPMENT
HIGH STRENGTH AND MODULUS, DIRECTIONAL	THERMAL MISMATCH BETWEEN INHIBITOR AND CARBON	MATERIAL AND DESIGN DEVELOPMENT
COMPLEX SHAPE FABRICABILITY	COST	SYSTEM STUDY
INSPECTABILITY	INTER-LAMINAR STRENGTH	MATERIAL AND DESIGN DEVELOPMENT
FAILSAFE CHARACTERISTICS		

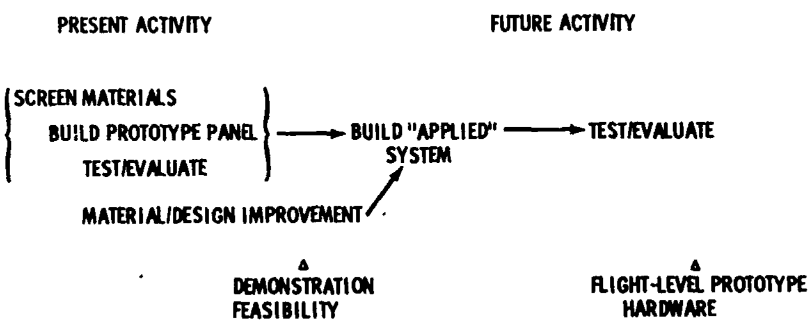
THERMAL DEGRADATION OF CARBON-CARBON

CARBON IN ITS PURE STATE IS SUBJECT TO OXIDATION AT TEMPERATURES OVER 1000°F. HOWEVER, THIS OXIDATION IS GENERALLY AT A LOWER RATE UP TO 4000°F, AND CARBON ACTS AS A RELATIVELY EFFICIENT ABLATOR UNDER ENTRY CONDITIONS. THE CLASSICAL OXIDATION OF CARBON IS INDICATED IN TERMS OF A SURFACE RECESSION PARAMETER AS A FUNCTION OF TEMPERATURE BY THE SHADED BAND ON THE CHART. THE STATE-OF-THE-ART FOR THE SIMPLEST CARBIDE-INHIBITED CARBON SYSTEMS IS INDICATED BY THE NEARLY VERTICAL LINE AT 2700°F. THIS PERFORMANCE HAS BEEN REPEATEDLY DEMONSTRATED IN ARC HEATED FACILITY TESTS AT REALISTIC CONDITIONS. HOWEVER, WORK IS PRESENTLY UNDERWAY ON THE OVERLAY AND BORIDE SYSTEMS WHICH SHOW PROMISE UP TO 3500°F OR 4000°F.



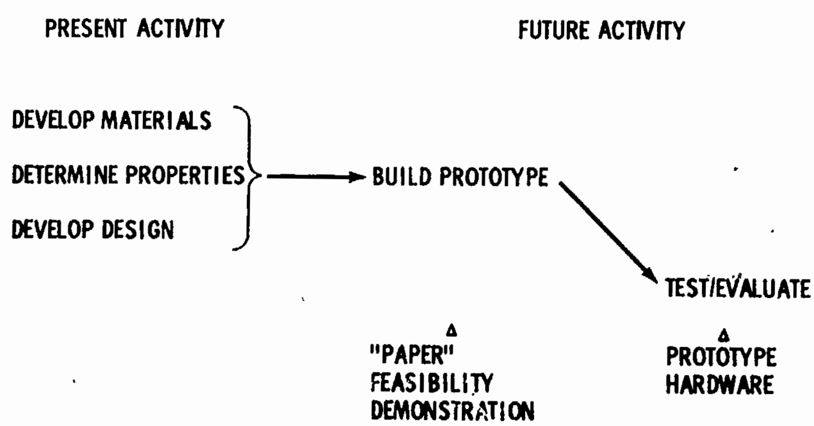
SURFACE INSULATION DEVELOPMENT APPROACH

THE DEVELOPMENT OF SURFACE INSULATION WAS INITIATED BY SCREENING SEVERAL MATERIAL/COATING SYSTEM CANDIDATES: THIS ACTIVITY WILL CONTINUE INTO THE THIRD AND FOURTH QUARTERS OF 1970. AT THIS TIME, THE SCREENING TESTS HAVE BEEN SUFFICIENTLY ENCOURAGING TO INITIATE TWO MAJOR CONTRACTED EFFORTS: ONE, A TWO-CONTRACT DEVELOPMENT PROGRAM WHICH IS AIMED AT DEMONSTRATING, THROUGH TESTS OF COMPLETE THERMAL PROTECTION SYSTEMS, THE PRACTICAL FEASIBILITY OF BUILDING REUSABLE THERMAL PROTECTION SYSTEM USING MATERIALS WHICH CAN BE MADE NOW. THE SECOND MAJOR EFFORT IS AIMED AT IMPROVING THE BASIC MATERIAL AND COATING AND ITS APPLICATION TECHNIQUES, AND IN APPROACHING THE HIGHER TEMPERATURE REGIMES ABOVE 3000°F. AFTER THESE TWO EFFORTS ARE COMPLETED, FURTHER STUDY AND DESIGN EFFORTS SHOULD RESULT IN FLIGHT PROTOTYPE PANELS FOR TYPE QUALIFICATION.



CARBON-CARBON DEVELOPMENT APPROACH

THE DEVELOPMENT APPROACH TO THE CARBON SYSTEM ORIENTED INITIALLY TOWARD MATERIALS DEVELOPMENT, EXTENSIVE PROPERTY DETERMINATION, AND DESIGN STUDIES. THIS IS NECESSARY PARTLY BECAUSE TEMPERATURE STABILITY ABOVE 3000°F IS REQUIRED FOR MANY POTENTIAL APPLICATIONS, AND BECAUSE DESIGN AND FABRICATION ARE MORE COMPLEX. THIS EFFORT WILL BE THROUGH TWO PARALLEL CONTRACTS, AND WILL RESULT IN A "PAPER" DEMONSTRATION OF FEASIBILITY BY EARLY 1971. THESE CONTRACTS WILL BE FOLLOWED BY A SINGLE CONTRACT WHICH WILL RESULT IN FLIGHT-TYPE HARDWARE PROTOTYPES FOR TESTING.



CONTRACT PROGRAM SUMMARY

	RESPONSIBLE CENTER
SURFACE INSULATION SYSTEMS	
SCREENING - 6 - 15K CONTRACTS: LMSC, MDAC, UCC, GE, AVCO, WHITTAKER	MSC
SYSTEM DEVELOPMENT - 2 CONTRACTS FOLLOWED IN FY 71 BY ONE FOLLOW-ON UNSELECTED	MSC
MATERIALS DEVELOPMENT - ONE CONTRACT, UNSELECTED	LaRC
CARBON-CARBON SYSTEMS	
SCREENING - 6 - 15K CONTRACTS: MDAC, LTV, UCC, GE, HITCO, AVCO	MSC
DEVELOPMENT - 2 CONTRACTS FOLLOWED IN FY 71 BY ONE FOLLOW-ON UNSELECTED	MSC
NOTE: OPTION HELD OPEN FOR FOLLOWING OTHER PROMISING APPROACHES AT A LOWER LEVEL	MSC, LaRC

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N70-42991

LOW-COST ABLATIVE HEAT SHIELDS

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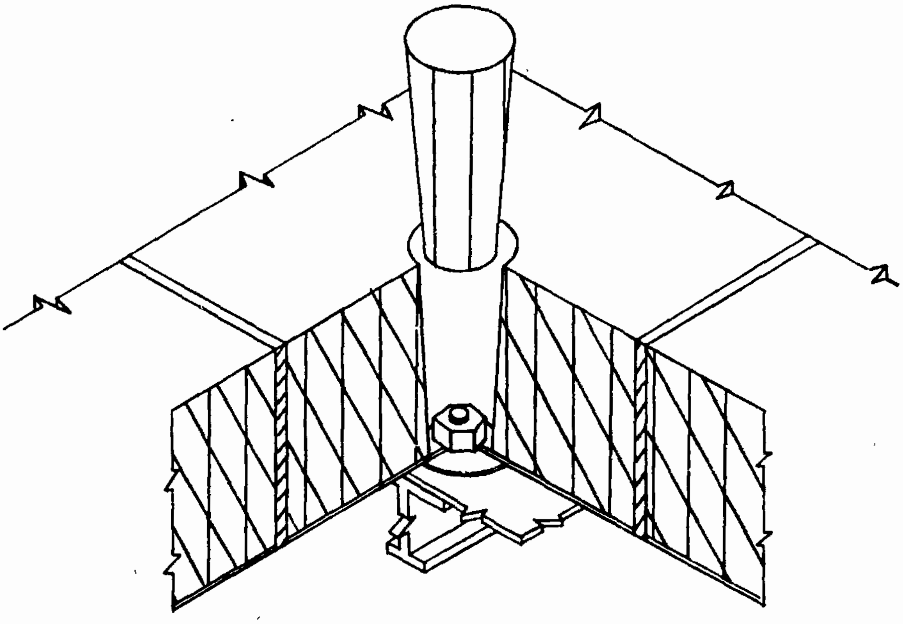
INTRODUCTION

The objectives and content of the ablative heat shield technology program are outlined on the first figure. While some attention is directed to thermal performance, the program emphasizes cost and replacement aspects of the problem. Manufacturing, quality assurance, and panel replacement costs and procedures are being studied for particular high heating rate areas such as leading edges as well as for broader vehicle areas where other technology may not develop as rapidly as expected.

OBJECTIVE:	**TO DEVELOP RELIABLE, LIGHTWEIGHT, LOW-COST ABLATORS AS PRIMARY OR BACK-UP SPACE SHUTTLE TPS**
MSC	**Design, Fabricate, and Test Ablative Leading Edge**
LRC	**Select Ablative Heat Shield Panel Configuration**
	Identify and Characterize Nominal Space Shuttle Ablator
	Develop and Verify Analysis of Space Shuttle Ablator
	Reduce Ablative Heat Shield Costs
	Manufacturing
	Quality Assurance
	Panel Replacement

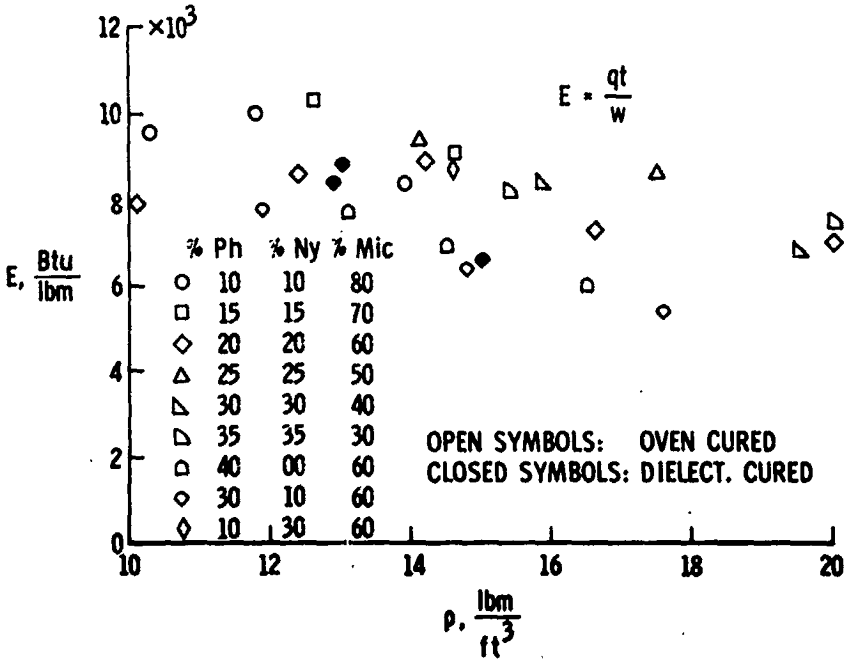
REPLACEABLE HEAT SHIELD PANEL CONFIGURATION

The heat shield panel configuration under consideration, which is shown in the figure, has high reliability and is easily replaced. The panels are constructed by first bonding a honeycomb matrix to a back-up sheet, then filling the honeycomb with an ablation material and curing. Positive attachment to the structure and easy panel replacement are obtained by mechanical fastening through holes in the ablator. These holes are then filled with a plug of ablation material. So far as thermal performance is concerned, this approach is similar to that used on Gemini and Apollo. However, the ablation material itself will probably be elastomeric based composition having a density of about 15 lbs/sq. ft. It should be noted that a continuous external structural skin is presupposed. If such a skin is not provided as part of the structural design, then it must be included as part of the heat shield system.



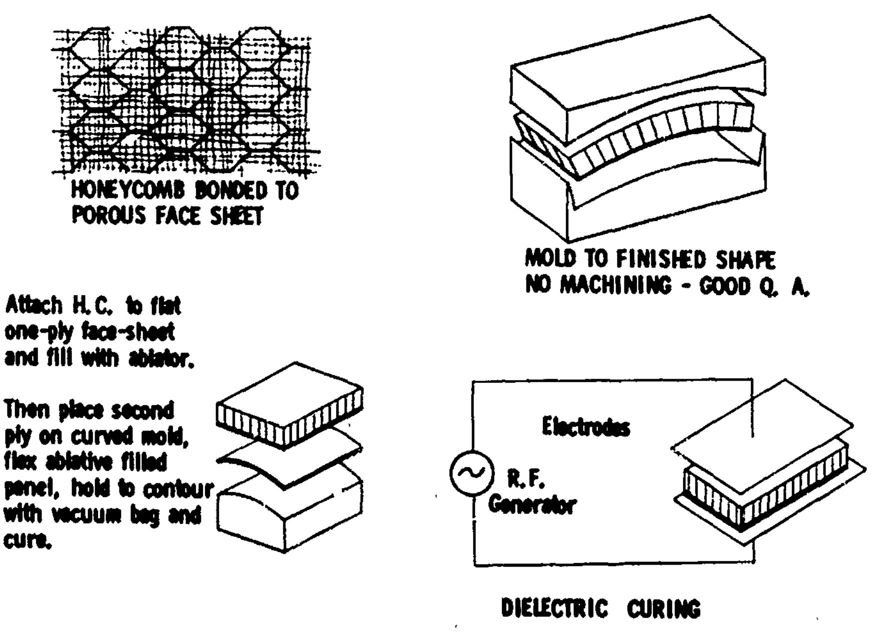
EFFECTIVE DENSITY AND COMPOSITION ON EFFECTIVENESS

The effects of density and composition on the ablative effectiveness of phenolic nylon ablator are shown on the figure. In general, the effectiveness varies by about ±10 percent from the average value at a given density and also varies slowly with the density itself. This relative insensitivity of performance to composition and density combined with the high inherent reliability of the panel configuration under consideration provides the basis for our belief that ablative heat shield costs can be substantially reduced.



LOW-COST MANUFACTURING APPROACHES

Five contractors are studying ablator manufacturing costs. Each contractor will build several 2'x4' panels and based on this actual fabrication experience, he will then estimate the cost of heat shields in varying quantities. The various contractors are trying several different approaches to reduce fabrication costs. Several of these ideas are shown on the figure. One contractor is using porous face-sheet bonded to the honeycomb to aid in filling the cells with ablation materials. Another is investigating the cost of molding to finished shape with hard tooling to eliminate machining and provide good quality assurance. An interesting approach to the manufacture of curved panels is to attach the honeycomb to a flat face shee and fill with ablator. Then place this assembly on a second ply, place the panel on a curved mold, and hold to contour with a vacuum bag during curing. Another approach which is being studied in house is the use of dielectric heating to reduce curing time.



IDENTIFY CRITICAL DEFECTS IN ABLATIVE HEAT SHIELDS

In view of the inherent reliability of the panel configuration as well as the relative insensitivity of performance to density and composition, quality assurance is perhaps the most promising area for cost reduction. Our program to identify which defects are actually critical is outlined in the figure. Defects will be identified and characterized and their effect on performanc on the entry environment and other environments will be examined. Based on these experimental results, methods for certifying that ablative heat shield panels are free of critical defects will be developed. Then the impact of allowable defects on heat shield fabrication costs will be determined.

OBJECTIVE: TO DEFINE THE INFLUENCE OF DEFECTS ON ABLATIVE HEAT SHIELD PERFORMANCE AND COST

APPROACH:

TASK I.- Characterize and Identify Defects

- Identify potential defects
- Establish performance requirements for space shuttle
- Establish inspection techniques

TASK II.- Develop Plans for Following Tasks

TASK III.- Determine Defects Critical in Entry Environment

- Determine effect of material and processing defects on performance
- Determine which defects are critical in entry

TASK IV.- Determine Defects Critical in Non-entry Environments

TASK V.- Develop Methods for Certifying Ablative Heat Shield Panels

TASK VI.- Determine Effects of Allowable Defects on Heat Shield Fabrication Costs

SUMMARY OF CONTRACT EFFORT

The contract effort on a low-cost ablative heat shield for FY 70 and 71 is outlined in the figure. In FY 71 work on the first three items will be primarily a follow-on to work initiated in FY 70. Two new starts are visualized in FY 71. Once a nominal space shuttle ablative material has been defined, then it will be exercised over the range of environments encountered by the space shuttle, wherever facilities are available. The other major new activity is the development of an approach suitable for limited high heating rate areas such as the leading edge.

	FY 70	FY 71
DEVELOP LOW COST ABLATIVE HEAT SHIELD FABRICATION TECHNIQUES (LRC)	X	X
IDENTIFY THERMAL PROTECTION SYSTEM REFURBISHMENT COSTS AND PROCEDURES (LRC)	X	X
IDENTIFY CRITICAL DEFECTS IN ABLATIVE HEAT SHIELDS (LRC)	X	X
CONDUCT ABLATIVE TESTS IN SPACE SHUTTLE ENVIRONMENTS (LRC)		X
VALIDATE ABLATION ANALYSIS (LRC)	X	
DESIGN, FABRICATE, AND TEST ABLATIVE LEADING EDGE (MSC)		X

N70-42992

TEMPERATURE CONTROL FOR THE SPACE SHUTTLE

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ABSTRACT

The influence of the thermal control system on the design requirements and operational capability of the Space Shuttle Orbiter are discussed in terms of the thermal environments experienced during the pad-hold, launch, on-orbit, entry, post-landing, and ferry phases of a shuttle mission. Potential temperature excursions are compared to temperature allowables of various equipment and the effect of some candidate temperature control concepts on vehicle weights/design requirements are illustrated. Candidate thermal control techniques are discussed and an evaluation/design approach for selecting the optimum thermal control system - an integrated thermal management system - is described.

Introduction

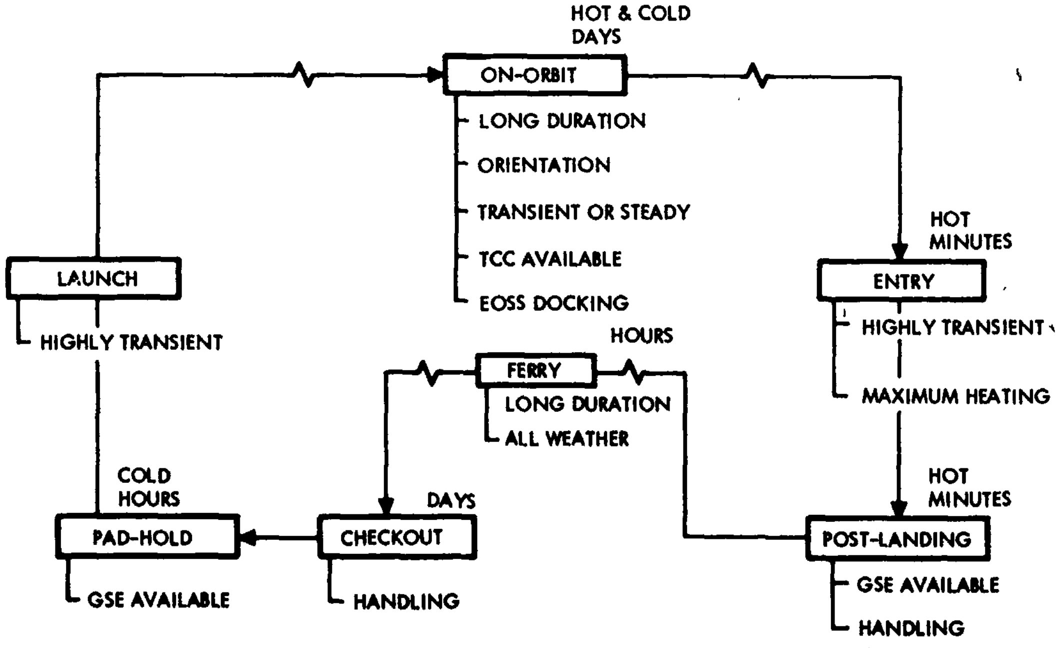
The Space Shuttle will experience extreme temperature environments - both hot and cold. Equipment located throughout the spacecraft as well as the structure must be maintained within allowable temperature limits. There are many techniques available, passive and active, by which this temperature control can be provided. For this spacecraft, a variety of techniques or temperature control elements are required to provide over-all acceptable thermal environments for equipment at the component and subsystem level. The combination of temperature control techniques and elements constitutes a temperature (thermal) control system (TCS) which is important by virtue of its influence on the weight and operational capability of the spacecraft.

Mission Environment

A wide range of environments are experienced by the Space Shuttle in the various mission phases. All of these phases are important to the Thermal Control System (TCS) design. For example, the minimum temperatures near the main propulsion tanks occur during fueled-up pad hold and launch phases. The high temperature environment occurs during entry and after landing.

Although any one (or more) of the mission phases may cause the maximum/minimum temperatures for a particular subsystem component, the on-orbit phase represents a particularly complex design condition. Both hot and cold conditions can be reached during the phase; however, neither a highly transient environment (amenable to passive control) nor GSE are available during this phase. The long duration of the on-orbit phase will involve increased consumables (for active heating or cooling) or operational constraints (vehicle orientation). The space station interface or docking constraints may also impact the TCS design. Although all phases are important to TCS design, the on-orbit phase may very well have the strongest impact on Space Shuttle design and operation.

The various mission phases affect TCS design both directly (through environmental extremes) and indirectly. An important example of this indirect impact is the effect of orbital conditions on TPS requirements during entry. The TCS design affects orbital temperatures, and hence, the temperatures at the beginning of entry. These temperatures have a major influence on the weight of insulation required for entry, which in turn, affects the orbiter payload capability.

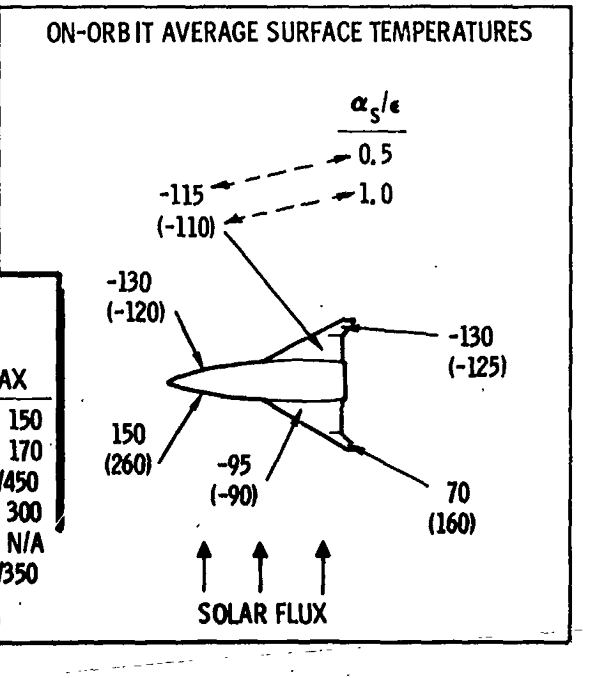


On-Orbit Temperatures and Temperature Allowables

During orbital operations, temperatures at any given location on a Space Shuttle can be either hot or cold. In order to effectively limit these extremes to acceptable values, the TCS design must include certain basic design considerations. These considerations include vehicle orientation to the sun, orbit inclination, and surface optical properties.

High temperatures can be reduced by a thermal control coating (TCC) with low solar absorptivity and high infrared emissivity. The TCC can thus reduce insulation and active cooling requirements. However, a TCC is not effective in raising the minimum temperatures. These minimum temperatures can be controlled by vehicle orientation. Orientation can also improve (lower) the maximum temperatures. Vehicle orientation control may represent an operational constraint in some cases. In those cases, mission requirements and operational flexibility must be weighted against alternate TCS techniques, such as active heating or cooling.

Limitations on potential orbital temperature extremes are required by equipment and subsystems which will not function at those extremes. Only a small fraction of the Space Shuttle volume will be environmentally controlled (ECS). For those components located outside ECS areas, temperature control must be provided through passive and/or active techniques.



EQUIPMENT ALLOWABLES - TYPICAL

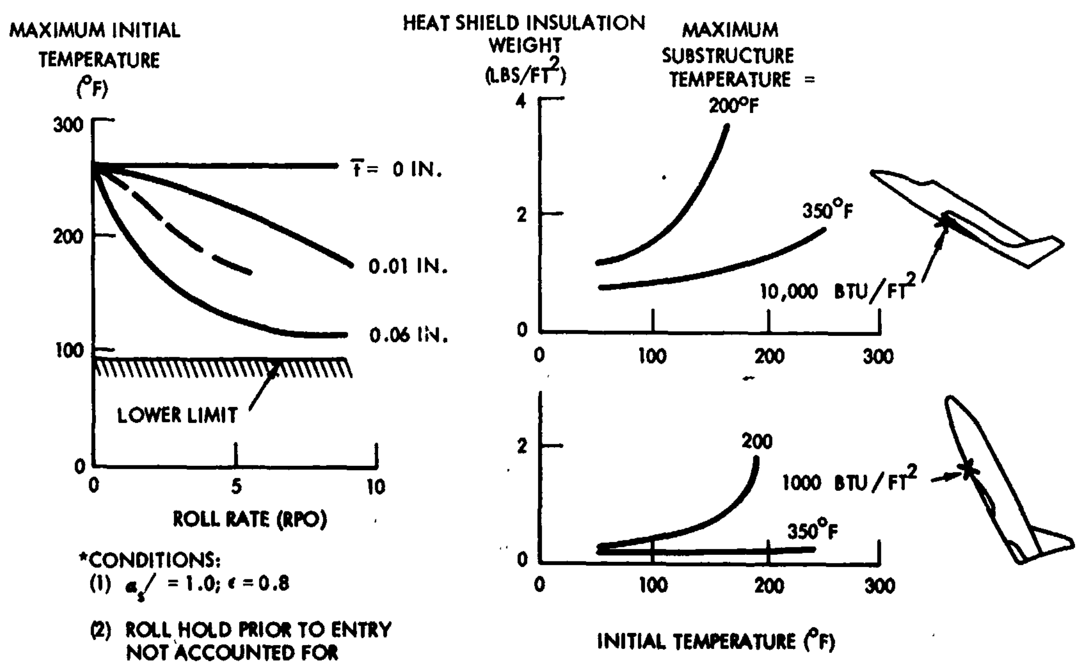
	LIMITS (°F) MIN	MAX
SOLENOID VALVES	0	150
SIGNAL CONDITIONERS	-65	170
HYDRAULIC FLUID	-65/-20	250/450
TIRES	-65	300
WASTE WATER TANK	32	N/A
LH_2 CRYOGEN INSULATION	-423	200/350

Effect of TCS on Heat Shield Design

Maximum temperature conditions affect active cooling and/or insulation requirements of components located throughout the vehicle. Perhaps the most significant effect of maximum orbital temperature conditions is reflected in the heat shield design. The heat shield insulation weight required for entry increases substantially as the initial (pre-entry) temperature increases. Further, the applicability of certain materials is dependent on pre-entry temperatures. For example, if the cryogenic tank foam insulation is limited to a maximum temperature of 200°F, heat shield insulation weights become prohibitively high as the initial temperature increases from very moderate levels.

The maximum orbital temperature, and hence the reentry system weight, can be controlled by thermal control coatings (TCC) or vehicle orientation. The use of TCC's implies some refurbishment. It is doubtful that the current TCC candidates could survive the entry environment without having their optical properties affected. This potential refurbishment requirement would conflict with one of the shuttle design goals - minimum maintenance and refurbishment.

Rolling the vehicle can be quite effective in reducing temperature extremes. This reduces the consumables needed for active cooling or heating. However, consumables required for the attitude control system may increase. Also, roll hold for docking and pre-entry attitude would limit the potential reduction in temperature extremes provided by the roll mode technique.



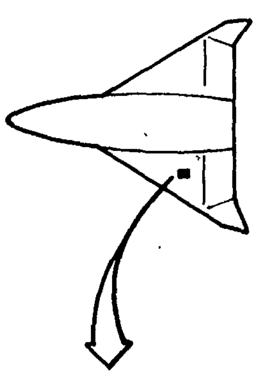
Component Requirements

Equipment and components requiring thermal control will be located throughout the space shuttle, including both the fuselage and wings. Because much of the equipment will be remotely located, the relatively small ECS areas will not be available to protect this equipment. Many of the components will require heat addition and/or rejection to maintain temperatures and operational reliability at acceptable levels. Although passive thermal control techniques may protect some of the equipment, it is probable that active methods will also be required. For long duration missions, the use of active cooling and heating systems will be reflected in increased consumables to provide power.

Because of the large number of components needing thermal control, the consumables required for active thermal systems may represent major design considerations. As an example, consider one component, the elevon actuator. In a cold environment, 8 elevon actuators will require heater power totaling 100's of watts. When the numerous other components requiring power for thermal control are considered, the total TCS power requirements may amount to several kilowatts.

Component thermal protection may present significant design challenges in several areas. One area involves equipment located in the wings. Placement of long lines (whether electrical, hydraulic, etc.) through the wings may require clever thermal design to avoid a TCS which is heavy and inefficient.

Protection of components from the different mission environments encountered by the space shuttle is also a major problem area. The conflicting requirements for these environments are typified by the preference for a high α/ϵ surface to minimize heater power requirements as opposed to the low α/ϵ surface favored by heat shield design. These parameters, and others, are influenced by and affect the TCS definition. These parameters must be thoroughly evaluated in concept trade studies.



HYDRAULIC ELEVON ACTUATORS
-20°F MINIMUM
300/450 MAXIMUM
8 ACTUATOR UNITS TOTAL

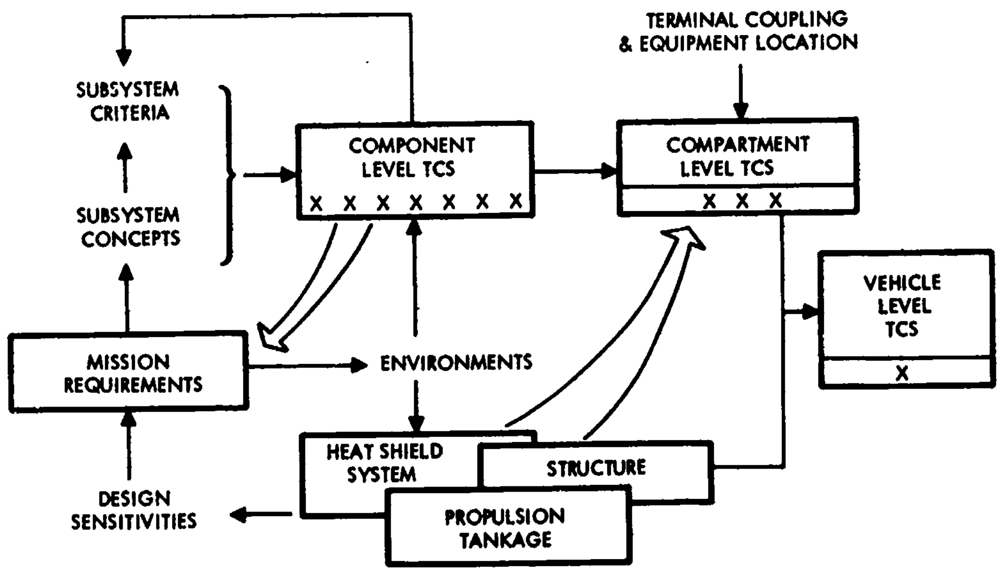
ESTIMATED POWER REQUIREMENTS (WATTS)

LOCAL INSULATION	α_s/ϵ 0.5	α_s/ϵ 1.0
NONE	500	400
0.5 IN. CRYOFOAM	400	300
0.5 IN. DYNAFLEX	350	375
0.5 IN. TG15000	300	200

TCS Design Approach

An integrated TCS design approach is vital to the development of an effective space shuttle TCS. The TCS design is characterized by multiple interactions and frequently conflicting requirements between the various systems, subsystems and mission characteristics. The TCS design that will best satisfy the design objectives of the space shuttle must be developed with cognizance of the impact of mission and subsystem design criteria. Available thermal control equipment (e.g., louvers, heat pipes, thermal coatings, thermal switches, electrical heaters, cold plates, etc.) must be considered and evaluated as to applicability, reliability, relative efficiency, maintenance, and cost. Thermal coupling and decoupling of the subsystem components is a part of the TCS concept evaluations. In essence, these design and concept evaluations which include consideration of all mission environments constitute the elements of a trade study which must be conducted to identify TCS design problems, applicable solutions, and design sensitivities. These trade study results will provide the basis for establishing design criteria and defining a TCS concept.

This design and concept evaluation involves analyses employing three-dimensional nodal network models of the spacecraft. These models are necessarily configuration and arrangement dependent; unfortunately these spacecraft characteristics may change as the design is developed. However, significant conclusions regarding concept applications, design sensitivities, and operational trade-offs should be valid for reasonable variations in configuration. The value of early definition of these trade data is evidenced by the impact of the TCS on design, operational capability, and spacecraft weight.



Thermal Control System Definition - Requirements

To provide an effective and efficient thermal control system for the space shuttle, an integrated thermal management system is required. Thermal management requires a major TCS study to provide a basis for decision making. This TCS study should include extensive analyses on all scales, ranging from detailed component evaluation to 3-D networks. These networks may include major areas (e.g., a wing) or the entire vehicle. Design sensitivity and trade-off study are also a necessary part of the TCS study.

As a part of the TCS study, an evaluation of design criteria and mission requirements is also required. These "groundrules" can have a major impact on the TCS; therefore, they should be continually reviewed with regard to their impact on the TCS and vehicle design and operational requirements.

There are several thermal control techniques and tools which have been used on previous spacecraft, and may prove useful on the space shuttle. However, some characteristics of shuttle operation and requirements are unique (e.g., 100 mission life, minimum refurbishment, and the wide range of environments). Therefore, it is expected that these techniques (such as louvers, heat pipes, thermal control coatings, and thermal switches) will require additional development work to be compatible with shuttle requirements.

As a prerequisite to the TCS study, decisions and definitions regarding property and degradation data are required. Although material property data is generally available for "new" materials, additional data are required to determine the effect of a 100-mission vehicle life on these properties. These data will affect both the choice of materials and the design limits usde for those materials. Definition is required for parameters such as internal pressures during the on-orbit and entry phases (this pressure affects the insulation conductivity, and hence the weight of the required insulation). Also needed are definitions of what constitutes acceptable performance degradation.

Conclusions

Temperature control of the many Space Shuttle subsystems can be effected by presently available techniques. The applications and combinations of these techniques will, however, have a substantial influence on the design, mission capability, and operational requirements of the Space Shuttle. It is necessary that these influences be as non-restrictive as is technically possible within the framework of the Space Shuttle requirements and technology capability - this can be accomplished by thermal control management.

N70-42993

THERMAL PROTECTION SYSTEM REFURBISHMENT COST STUDY

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STUDY OBJECTIVES AND PHASES

The purpose of the Thermal Protection System Refurbishment Cost Study is "to identify the costs associated with inspection, repair, and replacement of components, and to develop efficient techniques for performing these operations." This is accomplished in a two-phase program: Phase I includes definition of problem areas and study/test plans to aid in their solution, and Phase II implements the study/test plans.

Prime objectives for the Phase I effort are described in the chart. While refurbishment cost data are of great significance and interest to the space-shuttle program, in this study there is equal emphasis on identifying the degree of cost uncertainty of major elements of TPS maintenance operations. With priority uncertainty areas determined, study and/or test plans are to be formulated to reduce these maintenance cost uncertainties. Heat shields, and associated hardware, for use in these tests are to be designed to be compatible with a full-scale mockup at Langley Research Center.

A major portion of the study described in this paper was initiated as a company-sponsored activity in January 1970. The present program consists of a Langley Research Center-sponsored effort under contract No. NAS 1-10094 initiated on 16 June 1970. Most of the data, observations, and results presented herein reflect the earlier contractor-supported activity. The methodology, study approach, and schedule shown in the paper are descriptive of the on-going Langley-sponsored effort.

PHASES

I PERFORM TPS REFURBISHMENT COST ANALYSIS, DESIGN TEST-PANELS AND DEFINE ALTERNATE TEST-PROGRAM STUDY PLANS

II FABRICATE TEST-PANELS IN CONJUNCTION WITH MOCKUP PROGRAM PERFORM TEST/STUDY TASKS

PHASE I OBJECTIVES

- CHARACTERIZE TYPICAL TPS CANDIDATE SUBSYSTEMS, COMPONENTS, ATTACHMENTS AND ADJACENT PRIMARY STRUCTURAL VARIATIONS
- DEFINE REFURBISHMENT OPERATIONAL COSTS AND COST UNCERTAINTY AREAS, AND DETERMINE PRIORITY FOR FURTHER STUDY/TEST WORK USING FULL-SCALE PANELS
- DESIGN HEAT-SHIELD PANELS COMPATIBLE WITH FULL-SCALE LRC DEVELOPMENT-TEST MOCKUP, FOR FURTHER STUDY/TEST WORK

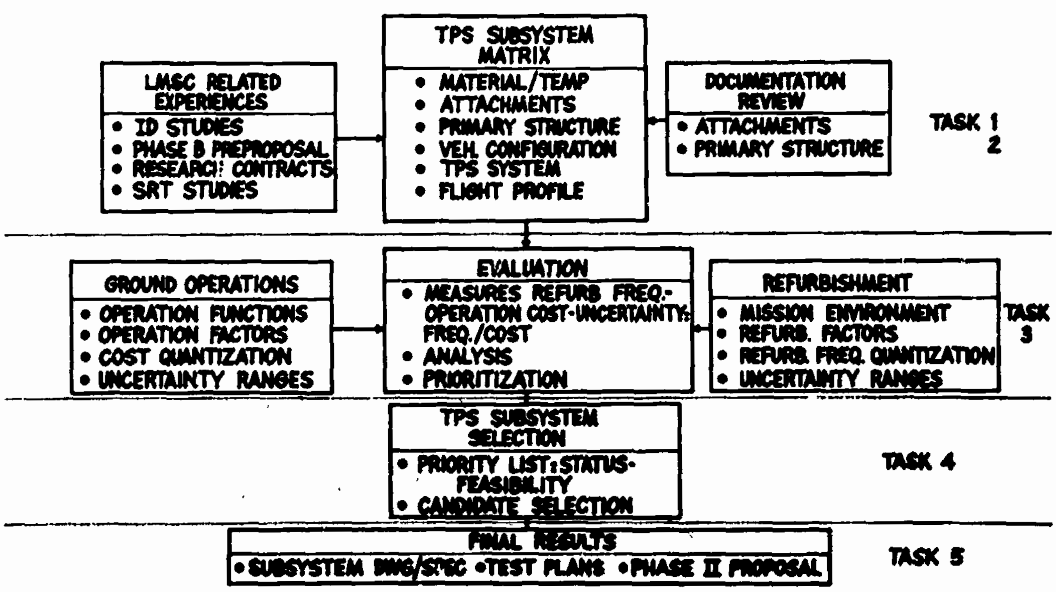
STUDY METHODOLOGY

The Phase I study is divided into 5 tasks. Tasks 1 and 2 identify primary structural components and heat shield attachment techniques, as determined from on-going studies, applicable literature, and space-shuttle preliminary design documentation. In these tasks, various TPS design candidates are considered in relation to the possible primary structure and attachment configurations. Configuration definition is accomplished utilizing design efforts on other on-going space-shuttle studies to provide a basis for refurbishment cost analyses.

Task 3 includes the detailed operational cost estimating of the candidate matrix. Refurbishment frequencies are estimated for each TPS subsystem area, together with estimated uncertainty ranges for these. The various operational functions are analyzed, and cost and cost-uncertainty factors are determined.

Task 4 utilizes the cost magnitude and uncertainty ranges to identify most promising TPS candidates and the development problem areas. The cost magnitudes, combined with the uncertainty ranges shown, allow for quantitative alignment of priorities for further study and tests.

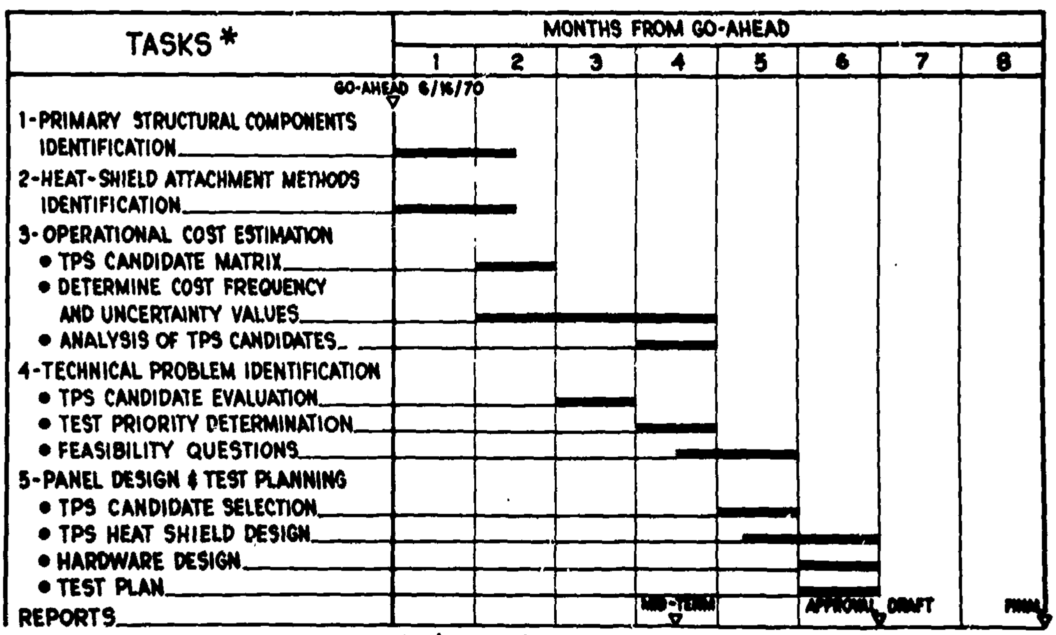
In Task 5, TPS candidates are selected for study and test in conjunction with a full-scale mockup at Langley. Study and test plans are formulated and test article designs are completed. A proposal for the Phase II program is then assembled.



PHASE I STUDY SCHEDULE

The Phase I program is completed in eight months, with the approval draft of the final report submitted at the end of the sixth month. Study go-ahead occurred on 16 June; each of the tasks and its related prime activities is scheduled as shown on the chart.

A midterm report is to be submitted in the fourth month of the Phase I program.



*NASA/LaRC CONTRACT NO. NAS1-10094

TPS CONFIGURATION CONSIDERATIONS

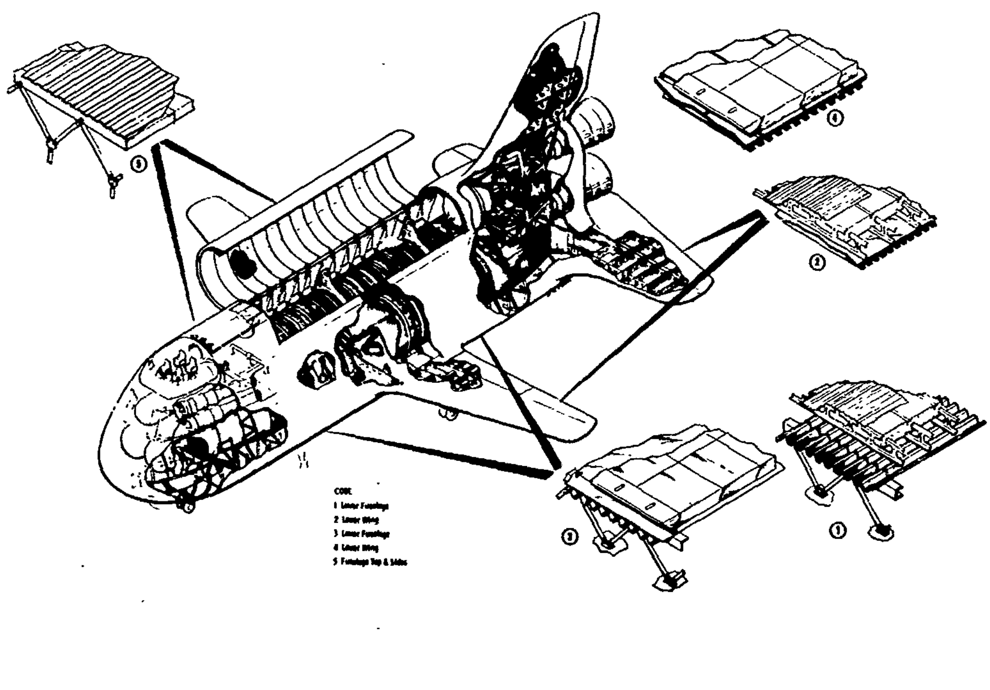
The matrix of configuration variation influence is described in the chart. Vehicle system mission and configuration factors exert an overall effect on the TPS, introducing changes in shape, gross areas of TPS subsystem types, and TPS weight. Variations in vehicle detail design introduce different primary structure arrangements to which the heat shields are attached.

The heat shield configuration alternatives include variations in the basic techniques for coping with the severe environments encountered. At least four primary types are considered; these include two reradiative-type systems, one ablative, and one actively cooled. Subsystem location influences both the design of heat shields and their attachment details. With most of the design variations, several alternative metallic and/or nonmetallic material candidates are being considered.

VEHICLE SYSTEM VARIATIONS		HEAT SHIELD CONFIGURATION VARIATIONS	
CONFIGURATIONS:	DELTA-BODY STRAIGHT-WING DELTA-WING	TYPE HEAT SHIELD:	METALLIC RADIATIVE NONMETALLIC RADIATIVE ABLATIVE ACTIVELY COOLED
CROSS RANGE REQUIREMENT:	200 NM 1500 NM	HT. SHIELD LOCATION:	NOSE CAP LEADING EDGE UNDERSIDE TOPSIDE BASE SHIELD
ALTERNATE PRIMARY STRUCTURES:	RINGS/STANDOFFS/ STRINGERS; TANKS; SMOOTH OR CORRUGATED SKINS; FLAT/SINGLE CURVATURE/ DOUBLE CURVATURE CONTOURS	SUBSYSTEM ALTERNATIVES:	DIFFERENT METALS DIFFERENT NONMETALS

TYPICAL HEAT SHIELD SUBSYSTEMS

The chart illustrates alternate heat shield types in representative locations on a typical orbital vehicle configuration. Several different primary structure arrangements are shown. The TPS is then characterized further by the different thermal-exposure regimes, and the associated variations in insulation, etc., to provide the required level of protection to the primary structure.



TYPICAL TPS AREA/WEIGHT DISTRIBUTIONS

Summary weights and areas are shown for two candidate TPSs for a specific-mission orbiter configuration. The primary TPS subsystems are identified for each of these, with some of them appearing for both candidates.

The two candidates shown are carried in this paper as examples of the study methodology being employed. Also, they provide a basis for preliminary results and observations with regard to refurbishment costs of metallic versus nonmetallic TPS candidates.

	TPS TYPE (MAT'L/TEMP)	DELTA BODY-METALLIC			DELTA BODY-NONMETALLIC		
		AREA (FT^2)	WEIGHT (LB)	PSF	AREA (FT^2)	WEIGHT (LB)	PSF
020	TANTALUM (2500°-3000°)	70	989	14.10	70	989	14.20
030	COLUMBIUM (2000°-2500°)	4576	18454	4.04	-	-	-
060	HAYNES 188 (1600°-2000°)	2132	9592	4.50	-	-	-
070	RENE' 41 (1000°-1600°)	1845	5973	3.23	-	-	-
041	LI-1500 (2000°-2500°)	-	-	-	5431	18387	3.40
042	LI-1500 (1600°-2000°)	-	-	-	1277	4257	3.35
043	LI-1500 (1000°-1600°)	-	-	-	1845	6027	3.27
080	TITANIUM (UNDER 1000°)	6078	4642	0.76	6078	4642	0.76
044	LI-1500 BASE SHIELD	1610	2816	1.75	1610	2816	1.75
101	DYNA-FLEX FLAP SHIELD	1100	632	0.57	1100	632	0.57
	TOTAL	17411	43098	2.48	17411	37750	2.17

TPS TRADE STUDY COST ELEMENTS

The primary cost matrix utilized is as defined in the Space Shuttle Phase B statement of work. This requires that costs be gathered in three contract/phase packages, one consisting of nonrecurring DDT&E, and the remaining two of recurring production and operations costs. In gathering cost data in a bottom-up costing approach, it is then necessary to identify applicable functional activities and obtain estimates from responsible organizations for their assigned tasks. Nine such functional groups are utilized in the TPS costing activities.

Costs must also be categorized in subsystem groupings. There is overlapping utilization of subsystems from one TPS candidate to another, and relating cost-level to the alternate subsystem candidate is essential in making TPS selections and study/test effort priority assignments.

Each functional area then provides further detail costing breakdown as required in its activity area for assembling useful cost versus uncertainty-area data.

PRIMARY COST MATRIX ~ PER SPACE SHUTTLE PHASE B SOW

- TIME PHASES (3)
  - •NONRECURRING COSTS (DDT & E)
  - •RECURRING COSTS - PRODUCTION
  - •RECURRING COSTS - OPERATIONS
- FUNCTIONAL ACTIVITIES (9)
  - •6 ENGINEERING ACTIVITIES : (DESIGN/ANALYSIS/TESTS/LIAISON)
    - •DESIGN
    - •LOADS & CRITERIA
    - •MATERIALS
    - •STRUCTURAL
    - •THERMAL
    - •MASS PROPERTIES
  - •MANUFACTURING
  - •QUALITY ASSURANCE
  - •OPERATIONS
- SUBSYSTEMS (APPROXIMATELY 6 TO 12)
  - •NOSE CAP
  - •UPPER SURFACES (2 OR 3 TYPES)
  - •LEADING EDGES (2 OR 3 TYPES)
  - •BASE SHIELD
  - •UNDERSURFACES (4 TO 6 TYPES)

DETAIL COST ELEMENTS ~ AS REQUIRED IN EACH FUNCTIONAL AREA

MAINTENANCE COST DETERMINATION

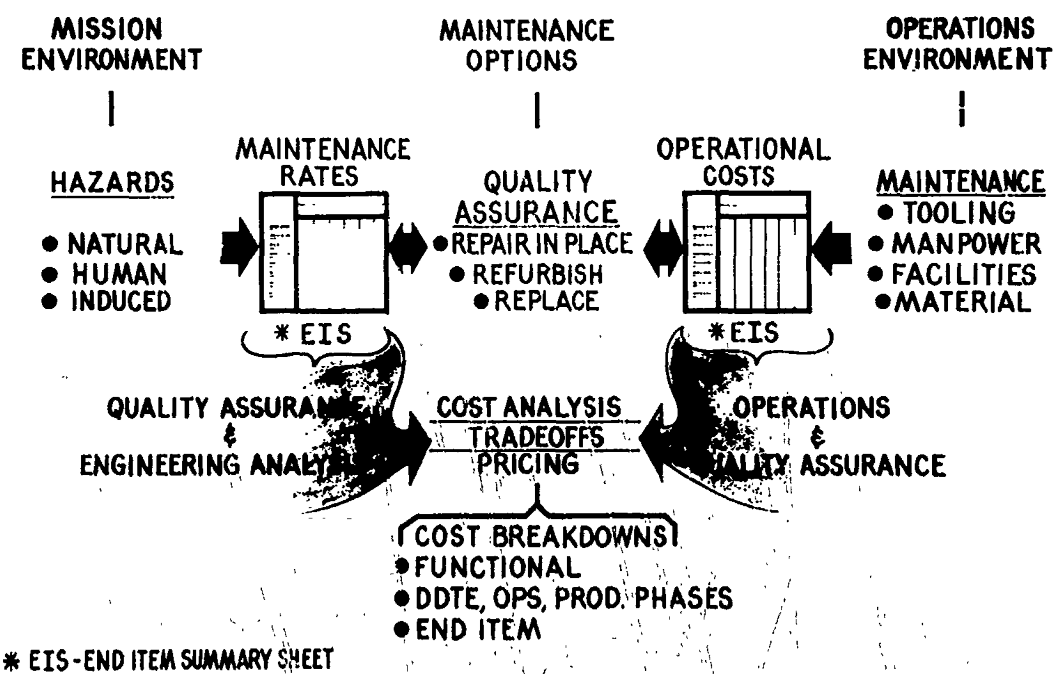
Operational tradeoff and pricing exercises require the delineation of maintenance rates, maintenance options, and operational functions.

Engineering and quality assurance perform mission environmental studies to identify and establish the maintenance rates for natural, human, and induced hazards. Maintenance rates are the integrated effect of all hazards experienced by the TPS system during a mission profile. For a given TPS system, the rates are recorded on End Item Summary (EIS) sheets.

Quality assurance makes use of the maintenance rate information to establish inspection requirements and maintenance options. Here, inspection concepts are projected and the ground rules for material, spares, and labor expenditures by operations are defined.

Operations will perform various maintenance functions in accordance with the findings of inspection, which will be to "repair in place," "refurbish," or "replace." The cost of this effort will be contingent on the maintenance rate. Results from maintenance and inspection are recorded on the Operations EIS sheet.

NASA cost ground rules are used to present cost data for tradeoff and pricing activities. This allows for rational allocation of cost to organizational function, acquisition phases, and hardware-related TPS subsystems.



BASIS FOR COSTING

In order to conduct a costing exercise it is essential that a baseline system be known to the study team. The collection of system requirements used for Phase B costing is embodied in a single document referred to as "Cost Estimating Requirements for Space Shuttle". Elements of this document that are directly applicable to the TPS total program are compiled in a "TPS Cost Estimating Requirements" document, which meets the more specialized needs of TPS. Some of the more significant TPS total program items are listed in the table and represent an agreed-to baseline.

For TPS maintenance operations, a similar collection of cost-estimating requirements for the assessment of refurbishment costs is shown.

In general, costs are collected according to NASA cost-collection specifications, while the costs themselves are constrained by various criteria or predetermined definitions. Within this framework, cost tradeoffs are conducted for system design optimization purposes.

TPS TOTAL PROGRAM	TPS MAINTENANCE OPERATIONS
• 11 VEHICLES: 2 GTV; 3 FTV; 6 PV FTV NO. 2 & NO. 3 REFURBISHED TO PV • 10 YEAR OPERATIONAL LIFE • 75 FLTS/YR; 2 LAUNCH SITES • GTV COSTS ASSUMED 2 X PV COSTS FTV COSTS ASSUMED 1.5 X PV COSTS • TOOLING: 1 SET DEVELOPMENT 2 SETS PRODUCTION • PROGRAM MGT & SYSTEMS INTEGRATION DEVELOMENT/OPERATIONS COSTS NOT PRORATA-SHARED BY TPS FOR TRADE-STUDY COSTING	• MAINTENANCE & INSPECTION COSTS BASED ON TOTAL-PROGRAM PRESUMPTIONS • NO LAUNCH / MISSION-OPS COSTS PRORATED • TPS PRODUCTION & DDTE COSTS EXCLUDED • SPARES COSTS EXCLUDED: BY NASA DEFINITION INCLUDED IN PRODUCTION COSTS • FACILITIES COSTS EXCLUDED: ASSUMED TO BE IN GENERAL LAUNCH / MISSION-OPS COSTS

METALLIC TPS MAINTENANCE FREQUENCY

The combined effect of all mission hazards encountered by a TPS system while flying a selected mission profile will determine the nature and extent of operational refurbishment. Inspection, maintenance, and logistic TPS activities (and costs) are essentially a direct function of the operations that must be undertaken as a result of the hazards experienced.

The matrix of TPS Maintenance Frequencies provides values that indicate the degree to which a selected TPS subsystem will respond to a given hazard. Integrating the spectrum of hazards over the mission profile provides a maintenance rate. Both frequencies and rates are interpreted as "the number of flights the TPS subsystem will experience before some maintenance action is required."

Both frequency and uncertainty are iteratively developed measures derived from existing documentation and best engineering judgments. They allow estimates of the breakdown of the projected maintenance levels; and as the program progresses and uncertainties diminish, they approach the actual true level.

The Metallic TPS Maintenance Frequency Matrix Chart shows that the expected number of flights per panel that will occur before some maintenance action will be required is 37, and this value can range from 30 to 87 flights, based on the information presently available. In other work being performed, the target for development of a reusable heat shield is "minor" refurbishment with 100 reuses. This target is presumed to reflect flight-environment-induced refurbs; hence, based on present preliminary data, the target is judged to be compatible with the optimistic estimate of maintenance frequency reflected in the uncertainty estimates.

A parallel data summary for nonmetallic TPS is shown in a separate chart.

TPS SUBSYSTEM CODE/TYPE	TEMP. EXPOSURE	COMBINED TEMP/LOAD	COMBINED TEMP/PRES	COMBINED TEMP/PRE/LD	HANDLING	ENVIRONMENT	RMS TOTAL	MAINT. RATE (FLTS/PANEL)	
	$f_T \pm U$	$f_{TL} \pm U$	$f_{TP} \pm U$	$f_{TPL} \pm U$	$f_H \pm U$	$f_E \pm U$	$F \pm U$	$\frac{1}{F}$	MAX/MIN
020/TANTALUM (NOSE)	22±12	104±33	56±20	151±33	53±33	109±33	93±33	11	17/6
030/COLUMBIUM	13±2	38±20	48±11	24±20	20±3	29±8	31±20	32	90/20
060/HAYNES	20±3	32±7	36±11	14±5	18±3	18±3	24±11	41	75/28
070/RENE 41	36±12	28±6	30±9	17±6	18±2	29±6	27±12	37	66/25
080/TITANIUM	17±5	33±7	38±8	14±5	20±6	25±9	26±9	39	[illegible]
044/LI-1500 (BASE SHLD)	16±3	37±10	30±10	10±4	31±10	32±11	28±11	36	[illegible]/26
TPS AVERAGE								37	87/30

NOTES: f = INDIVIDUAL-CONDITION PANEL-MAINTENANCE FREQUENCY PER 1000 FLIGHTS
F = SUBSYSTEM COMPOSITE FREQUENCY, BASED ON ROOT-MEAN-SUM² (RMS) AVERAGE
U = ESTIMATED FREQUENCY UNCERTAINTY RANGE
CONFIGURATION = DELTA-BODY, 1500 NM CROSS RANGE, METALLIC TPS

NONMETALLIC TPS MAINTENANCE FREQUENCY

The combined effect of all mission hazards encountered by a TPS system while flying a selected mission profile will determine the nature and extent of operational refurbishment. Inspection, maintenance, and logistic TPS activities (and costs) are essentially a direct function of the operations that must be undertaken as a result of the hazards experienced.

The matrix of TPS Maintenance Frequencies provides values that indicate the degree to which a selected TPS Subsystem will respond to a given hazard. Integrating the spectrum of hazards over the mission profile provides a maintenance rate. Both frequencies and rates are interpreted as "the number of flights the TPS subsystem will experience before some maintenance action is required."

Both frequency and uncertainty are iteratively developed measures derived from existing documentation and best engineering judgments. They allow estimates of the breakdown of the projected maintenance levels; and as the program progresses and uncertainties diminish, they approach the actual levels.

The nonmetallic TPS Maintenance Frequency Matrix Chart shows that the expected number of flights per panel that will occur before some maintenance action will be required is 39, and this value can range from 27 to 81 flights, based on the information presently available. In other work being performed, the target for development of a reusable TPS heat shield is "minor" refurbishment with 100 reuses. This target is presumed to reflect flight-environment-induced refurbs; hence, based on present preliminary data, the target is judged to be compatible with the optimistic estimate of maintenance frequency reflected in these uncertainty estimates.

A parallel data summary for Metallic TPS is shown in a separate chart.

TPS SUBSYSTEM CODE/TYPE	TEMP. EXPOSURE $f_T \pm U$	COMBINED TEMP/LOAD $f_{TL} \pm U$	COMBINED TEMP/PRES $f_{TP} \pm U$	COMBINED TEMP/PRES/LD $f_{TPL} \pm U$	HAND-LING $f_H \pm U$	ENVIR-ONMENT $f_E \pm U$	RMS TOTAL $F \pm U$	MAINT RATE FLTS/PANEL $\frac{1}{F}\left\{\frac{MAX}{MIN}\right.$
020/TANTALUM (NOSE)	22±12	104±33	56±20	151±33	53±33	109±33	93±33	$11\left\{\frac{17}{8}\right.$
041/LI-1500 (2000°-2500°F)	18±3	79±20	35±10	12±4	36±10	39±11	44±20	$23\left\{\frac{41}{16}\right.$
042/LI-1500 (1600°-2000°F)	17±3	68±20	30±10	10±4	31±10	33±11	36±20	$27\left\{\frac{61}{18}\right.$
043/LI-1500 (1000°-1600°F)	15±6	57±33	25±15	8±4	26±15	28±15	31±20	$33\left\{\frac{94}{20}\right.$
080/TITANIUM	17±5	33±7	38±8	14±5	20±6	25±9	26±9	$39\left\{\frac{59}{29}\right.$
044/LI-1500 (BASE SHELD)	17±3	37±10	25±10	10±4	31±10	32±11	28±11	$36\left\{\frac{59}{26}\right.$
030 COLUMBIUM (FLAP)	14±2	38±20	48±11	24±20	20±3	29±8	31±20	$32\left\{\frac{90}{20}\right.$
070 RENÉ 41 (FLAP)	36±12	28±6	30±9	17±6	18±2	29±6	27+12	$37\left\{\frac{66}{25}\right.$
TPS AVERAGE								$39\left\{\frac{81}{27}\right.$

NOTE: f = INDIVIDUAL-CONDITION PANEL-MAINTENANCE FREQUENCY PER 1000 FLIGHTS
F = SUBSYSTEM COMPOSITE FREQUENCY, BASED ON ROOT-MEAN-SUM² (RMS) AVERAGE
U = ESTIMATED FREQUENCY UNCERTAINTY RANGE
CONFIGURATION = DELTA-BODY, 1500 NM CROSS RANGE, LI-1500 TPS

FUNCTIONAL COSTS VERSUS PHASES FOR METALLIC TPS

The total system cost for a metallic TPS system is 378 million dollars. Using NASA costing ground rules the system cost is divided into Nonrecurring DDT&E (85.6 million), Recurring Production (45.1 million), and Recurring Operations (247.3 million).

The contribution by each of nine functional groups can be summarized under 39.2 million for manufacturing, 227.9 million for operations, 77 million for engineering; and 33.8 million for quality assurance. Thus, the total system can be observed as to its major cost drivers in terms of functionally responsible program organizations, for each of the three specified contract-phase activities.

A parallel data summary for nonmetallic TPS is shown in a separate chart.

	NONRECURRING	RECURRING		
FUNCTIONS	DDTE	PRODUCTION	OPERATIONS	TOTAL
MANUFACTURING	17.3	21.9	—	39.2
OPERATIONS	—	—	227.9	227.9
ENGINEERING - STRESS	4.7	2.2	—	6.9
- WEIGHTS	3.4	0.7	—	4.1
- LOADS	2.0	0.6	—	2.6
- THERMO	4.1	1.1	—	5.2
- DESIGN	3.6	1.8	—	5.4
- MATERIALS	42.7	10.1	—	52.8
ENGINEERING - TOTAL	60.5	16.5	—	77.0
QA - MANUFACTURING	5.6	6.7	—	12.3
- OPERATIONS	2.2	—	19.4	21.6
QA TOTAL	7.8	6.7	19.4	33.9
TOTAL	$85.6	$45.1	$ 247.3	$378.0

NOTES: • COSTS IN MILLIONS OF DOLLARS
• CONFIGURATION IS DELTA-BODY, 1500 NM CROSS RANGE, METALLIC TPS

SUBSYSTEM COSTS VERSUS PHASES FOR METALLIC TPS

For tradeoff purposes, a hardware orientation of system cost is necessary. Major TPS subsystem cost drivers are then identifiable and the activity distribution (measured in costs) between contract phases is evident.

Design alternatives intended to improve operational costs will produce satisfactory changes in production and development costs. Thus, over a series of design iterations, both system cost and performance are intimately related and their cost effectiveness measureable. In this manner a series of system/design points is established for use with selected MOEs (Measures of Effectiveness) in optimization studies. The TPS/RCS methodology assures that, while concentrating attention on the maintenance operations costs, their relationship and effect on TPS total-program costs are not ignored.

A parallel data summary for nonmetallic TPS is shown in a separate chart.

SUBSYSTEM	NONRECURRING	RECURRING		TOTAL
	DDTE	PRODUCTION	OPERATIONS	
020 TANTALUM	7.4	2.5	8.4	18.3
030 COLUMBIUM (S)	11.7	5.4	58.1	75.2
030 COLUMBIUM (C)	15.5	10.1	45.8	71.4
060 HAYNES (LEAD EDGE)	5.7	2.3	12.0	20.0
060 HAYNES	6.0	2.6	11.8	20.4
070 RENÉ 41	10.3	4.1	20.4	34.8
080 TITANIUM	16.1	13.3	66.6	96.0
044 LI-1500	9.0	3.8	24.2	37.0
101 DYNA-FLEX	3.9	1.0	0	4.9
TOTAL	$95.6	$45.1	$247.3	$378.0

NOTES: COSTS IN MILLIONS OF DOLLARS
CONFIGURATION IS DELTA-BODY, 1500 NM CROSS RANGE, METALLIC TPS

FUNCTIONAL COSTS VERSUS PHASES FOR NONMETALLIC TPS

The total system cost for a nonmetallic TPS system is 301.6 million dollars. Using NASA costing ground rules, the system cost is divided into Nonrecurring DDT&E (55.2 million), Recurring Production (28.0 million), and Recurring Operations (218.5 million).

The contribution by each of nine functional groups can be summarized under 29.6 million for manufacturing, 199.8 million for operations, 46.9 million for engineering, and 25.3 million for quality assurance. Thus, the total system can be observed as to its major cost drivers in terms of functionally responsible program organizations, for each of the three specified contract-phase activities.

A parallel data summary for metallic TPS is shown in a separate chart.

FUNCTIONS	NONRECURRING	RECURRING		TOTAL
	DDTE	PRODUCTION	OPERATIONS	
MANUFACTURING	14.0	15.6	–	29.6
OPERATIONS	–	–	199.8	199.8
ENGINEERING – STRESS	4.6	2.2	–	(6.8)
– WEIGHTS	3.4	0.7	–	(4.1)
– LOADS	2.0	0.6	–	(2.6)
– THERMO	4.1	1.1	–	(5.2)
– DESIGN	1.1	0.5	–	(1.6)
– MAT'LS	21.7	4.9	–	(26.6)
ENGINEERING TOTAL	36.9	10.1	–	46.9
QA – MANUFACTURING	2.1	2.3	–	(4.4)
OPERATIONS	2.2	–	18.7	(20.9)
QA TOTAL	4.3	2.3	18.7	25.3
TOTAL	\$ 55.2	\$ 28.0	\$218.5	\$ 301.6

NOTES: • COSTS IN MILLIONS OF DOLLARS
• CONFIGURATION IS DELTA-BODY, 1500 NM CROSS RANGE, LI-1500 TPS

SUBSYSTEM COSTS VERSUS PHASES FOR NONMETALLIC TPS

For tradeoff purposes, a hardware orientation of system cost is necessary. Major TPS subsystem cost drivers are then identifiable and the activity distribution (measured in costs) between contract phases is evident.

Design alternatives intended to improve operational costs will produce satisfactory changes in production and development costs. Thus, over a series of design iterations, both system cost and performance are intimately related and their cost-effectiveness measureable. In this manner a series of system/design points is established for use with selected MOEs (Measures of Effectiveness) in optimization studies. The TPS/RCS methodology ensures that, while concentrating attention on the maintenance operations costs, their relationship and effect on TPS total-program costs are not ignored.

A parallel data summary for metallic TPS is shown in a separate chart.

SUBSYSTEM	NONRECURRING	RECURRING		TOTAL
	DDTE	PRODUCTION	OPERATIONS	
020 TANTALUM	7.2	2.4	8.4	17.9
041 LI-1500	19.6	8.6	87.0	115.1
042 LI-1500	4.6	2.0	16.6	23.2
043 LI-1500	6.5	2.9	19.2	28.6
080 TITANIUM	13.4	10.1	63.1	86.6
C44 LI-1500	3.9	2.0	24.2	30.2
TOTAL	55.2	28.0	218.5	301.6

NOTES: COSTS IN MILLIONS OF DOLLARS
CONFIGURATION IS DELTA-BODY, 1500 NM CROSS RANGE, LI-1500 TPS

METALLIC TPS COST UNCERTAINTY SUMMARY

Nominal costs to perform the DDT&E, Production, and Operation phases reflect the depth of informational detail available to all functional groups. The costs shown are based on a mix of subjective judgment, "similar to" knowledge, and definitive information. The extent to which definition is lacking will appear in the magnitude of the uncertainty factor. For conditions existing at the time these data were produced, the metallic TPS system can cost 4.16 times nominal or 1.573 million dollars. Technological uncertainty can result in a 1/3.20 reduction in the nominal cost (118 million dollars for a metallic TPS system).

The importance of this information is twofold: (1) it provides perspective which allows the establishment of priorities for further development activities that will effectively lead to uncertainty reduction and definitive costing, and (2) the data can be directly related to a function, activity, or end item, permitting critical appraisal of design and system tradeoffs and maintenance of program objectives.

During a Phase B effort when definition is meager, this approach gives order to normally volatile and disjointed activities and ultimately produces a consolidated position with known uncertainty.

A parallel data summary for nonmetallic TPS is shown in a separate chart.

COST UNCERTAINTY FACTORS & COST RANGE	PROGRAM PHASES			TOTAL
	DDT&E	PRODUCTION	OPERATIONS	
HIGH UNCERTAINTY FACTOR	3.48	2.26	4.73	4.16
	1	1	1	1
LOW UNCERTAINTY FACTOR	2.82	1.82	3.93	3.20
HIGHEST TPS COST	$ 299 M	$ 103 M	$ 1,170 M	$1,573 M
NOMINAL TPS COST	85	45	247	378
LOWEST TPS COST	30	25	63	118

NOTES:
- UNCERTAINTY FACTORS ARE SUMMATION VALUES REFLECTING ALL COST-ELEMENT UNCERTAINTY ESTIMATES
- THE HIGH & LOW FACTORS ARE MULTIPLIERS TO ESTIMATED NOMINAL COSTS, TO OBTAIN ESTIMATED HIGH & LOW COST LIMITS AT TIME OF ESTIMATE
- THESE DATA REFLECT A TYPICAL METALLIC TPS COST ESTIMATE FOR A DELTA-BODY ORBITER, 1500 NM CROSS RANGE

NONMETALLIC TPS COST UNCERTAINTY SUMMARY

Nominal costs to perform the DDT&E, Production, and Operation phases reflect the depth of informational detail available to all functional groups. The costs shown are based on a mix of subjective judgment, "similar to" knowledge, and definitive information. The extent to which definition is lacking will appear in the magnitude of the uncertainty factor. For conditions existing at the time these data were produced, the nonmetallic TPS system can cost 4.51 times nominal or 1,361 million dollars. Technological uncertainty can result in a 1/3.60 reduction in the nominal cost (84 million dollars for a nonmetallic TPS system).

The importance of this information is twofold: (1) it provides perspective which allows the establishment of priorities for further development activities that will effectively lead to uncertainty reduction and definitive costing, and (2) the data can be directly related to a function, activity, or end item, permitting critical appraisal of design and system tradeoffs and maintenance of program objectives.

During a Phase B effort when definition is meager, this approach gives order to normally volatile and disjointed activities and ultimately produces a consolidated position with known uncertainty.

A parallel data summary for metallic TPS is shown in a separate chart.

COST-UNCERTAINTY FACTORS & COST RANGE	PROGRAM PHASES			TOTAL
	DDT & E	PRODUCTION	OPERATIONS	
HIGH UNCERTAINTY FACTOR	2.77	2.09	5.26	4.51
LOW UNCERTAINTY FACTOR	$\frac{1}{3.97}$	$\frac{1}{1.69}$	$\frac{1}{4.09}$	$\frac{1}{3.60}$
HIGHEST TPS COST	$153 M	$ 58 M	$1,150 M	$1,361 M
NOMINAL TPS COST	55	28	219	302
LOWEST TPS COST	14	17	53	84

NOTES:
- UNCERTAINTY FACTORS ARE SUMMATION VALUES REFLECTING ALL COST-ELEMENT UNCERTAINTY ESTIMATES
- THE HIGH & LOW FACTORS ARE MULTIPLIERS TO ESTIMATED NOMINAL COSTS TO OBTAIN ESTIMATED HIGH & LOW COST LIMITS AT TIME OF ESTIMATE
- THESE DATA REFLECT A TYPICAL NONMETALLIC TPS COST ESTIMATE FOR A DELTA-BODY ORBITER, 1500 NM CROSS RANGE

METALLIC TPS OPERATIONAL COSTS

A matrix of operational expenditures and uncertainty in shown in the chart. The values shown are provided by experienced operations and quality assurance personnel. Each hourly estimate represents the expected expenditure of resources and arises from "Time Line" analysis on a selected panel.

The total operations rate is a measure of the contribution made by each operational task. Operations rate, maintenance rate, vehicle configuration, and maintenance model combine to give a total recurring cost. In this manner, the interrelationship between system, operations, subsystem, and task can be managed and evaluated. As an example, panel installation and the maintenance tasks are the most costly operational items. This might indicate that a repair-in-place design approach would be an attractive vehicle option in the event that resulting development or production costs were not prohibitive.

All computational steps are not included in the charts. Also, the composite dollar-uncertainty values are not shown here, but are available in the backup data.

A parallel data summary for nonmetallic TPS is shown in a separate chart.

TPS SUBSYSTEM CODE/TYPE	PANEL INST.		PANEL REMOVAL		INSPECTION		PACKAGING AND HANDLING		STORAGE		MAINTENANCE		TOTAL OPERATIONS RATE		TOTAL RECURRING COST
	H	U	H	U	H	U	H	U	H	U	H	U	F_r	U	($M)
020 TANTALUM	150	2.5	50	25	24	6	16	4	8	4	300	10	548	6.3	8.4
030 COLUMBIUM	24	5	8	5	5	5	3	4	3	4	40	5	83	4.9	103.9
060 HAYNES	18	3	6	3	1	4	1	2	2	2	32	5	60	4.0	23.8
070 RENÉ	18	1.5	6	1.5	1	1.5	1	1	1.5	1	36	5	63	3.5	20.4
080 TITANIUM	18	1.5	6	1.5	1	1.5	1	1	1	1	24	5	51	3.1	66.6
044 LI-1500	40	7	8	7	15	5	1	5	3	4	23	5	90	6.0	24.2
TOTALS	268	3.3	84	3.1	47	5.3	23	3.1	10	3.4	455	8.5	995	5.9	247.3

NOTES:
- H = MANHOURS PER OPERATION
- U = UNCERTAINTY FACTOR, MULTIPLIED FOR HIGH VALUE, AND DIVIDED FOR LOW VALUE
- LABOR RATE, ETC; COST CALCULATIONS OMITTED FOR BREVITY

NONMETALLIC TPS OPERATIONAL COSTS

A matrix of operational expeuditures and uncertainty in shown in the chart. The values shown are provided by experienced operations and quality ssurance personnel. Each hourly estimate represents the expected expenditure of resources and arises from "Time Line" analysis on a selected panel.

The total operations rate is a measure of the contribution made by each operational task. Operations rate, maintenance rate, vehicle configuration, and maintenance model combine to give a total recurring cost. In this manner, the interrelationship between system, operations, subsystem, and task can be managed and evaluated. As an example, panel installation and the maintenance tasks are the most costly operational items. This might indicate that a repair-in-place design approach would be an attractive vehicle option in the event that resulting development or production costs were not prohibitive.

All computational steps are not included in the chart. Also, the composite dollar-uncertainty values are not shown here, but are available in the backup data.

A parallel data summary for metallic TPS is shown in a separate chart.

TPS SUBSYSTEM CODE/TYPE	PANEL INSTAL.		PANEL REMOVAL		INSPECT.		PACKAGING & HANDLING		STORAGE		MAINT-ENANCE		MAINT-ENANCE RATE		TOTAL RECURRING COST
	H	U	H	U	H	U	H	U	H	U	H	U	Fr	U	($M)
020 TANTALUM	150	1.4	50	2.5	24	6	16	4	8	4	300	10	548	6.3	8.4
041 LI-1500	34	1.6	12	1.6	2	5	1.8	5	3	4	2	2.5	55	2.0	87.0
042 LI-1500	34	1.6	10	1.6	2	5	1.8	5	3	4	2	2.5	53	2.0	16.6
043 LI-1500	31	1.6	10	1.6	2	5	1.8	5	3	4	2	2.5	50	2.0	19.2
080 TITANIUM	18	1.5	6	1.5	1	1.5	1.0	1	1	1	24	5.0	51	3.1	63.1
044 LI-1500	40	7	8	7	15	5	1.9	5	3	4	23	5.0	90	6.0	24.2
TOTAL	307	2.2	96	2.5	46	5.4	23.4	4.1	21	3.9	353	9.2	847	5.6	218.5

NOTES: H = MANHOURS PER OPERATION
U = UNCERTAINTY FACTOR, MULTIPLIED FOR HIGH VALUE, AND DIVIDED FOR LOW VALUE
LABOR RATE, ETC., COST CALCULATIONS OMITTED FOR BREVITY

PRELIMINARY RESULTS AND OBSERVATIONS

The results and observations derived from the metallic and nonmetallic TPS systems cost analyses are summarized in the chart.

- TPS REFURBISHMENT COSTS IMPACT ALL 3 PROGRAM SEGMENTS: DDT&E, PRODUCTION, AND OPERATIONS
- FOR OPERATIONS SEGMENT, TERM "MAINTENANCE COSTS" IS BETTER THAN "REFURB COSTS"; INCLUDES REPAIR, REFURBISHMENT, AND/OR REPLACEMENT OF TPS PANELS/ELEMENTS
- DEFINITION OF SUBSYSTEM AND FUCTIONAL ACTIVITY COST MAGNITUDE AND COST UNCERTAINTY RANGE PROVIDES POWERFUL TOOL FOR:
    - SUBSYSTEM SELECTION IN TRADE-STUDY ACTIVITIES
    - DEFINING AND PRIORITIZING DESIGN/ANALYSIS/TEST DEVELOPMENT ACTIVITIES
    - DOLLAR QUANTIFICATION OF TECHNOLOGICAL RISK
- METALLIC TPS MAINTENANCE COSTS ESTIMATED 13% HIGHER THAN FOR NONMETALLIC TPS
- MAINTENANCE COSTS ESTIMATED 64% AND 72%, RESPECTIVELY, OF METALLIC AND NONMETALLIC TPS TOTAL-PROGRAM COSTS
- TPS COST UNCERTAINTY HIGHEST IN OPERATIONS PROGRAM SEGMENT; NOMINAL VALUE IS APPROXIMATELY $250M WITH EXTREME LIMITS RANGING FROM LESS THAN $100M TO MORE THAN $1,000M

N70-42994

TEST FACILITIES FOR SPACE SHUTTLE THERMAL PROTECTION SYSTEM

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INTRODUCTION

Because of the reuse requirement and the weight restraint imposed by the necessity for an economically efficient shuttle, all competitive concepts for the thermal protection system must be accurately assessed and their relative overall merit established. This is necessary to insure that the correct choice of thermal protection system or systems is incorporated into the shuttle design.

The intent of this presentation is to review briefly some of the problems associated with experimental evaluation of thermal protection systems and with the associated testing capability requirement.

On the first figure, four examples of concepts for the space shuttle thermal protection are shown.

Metallic. — The metallic heat shield is supported on flexible stand offs from an insulated structure. The heat shields are pictured as having overlapping joints with omega type seals.

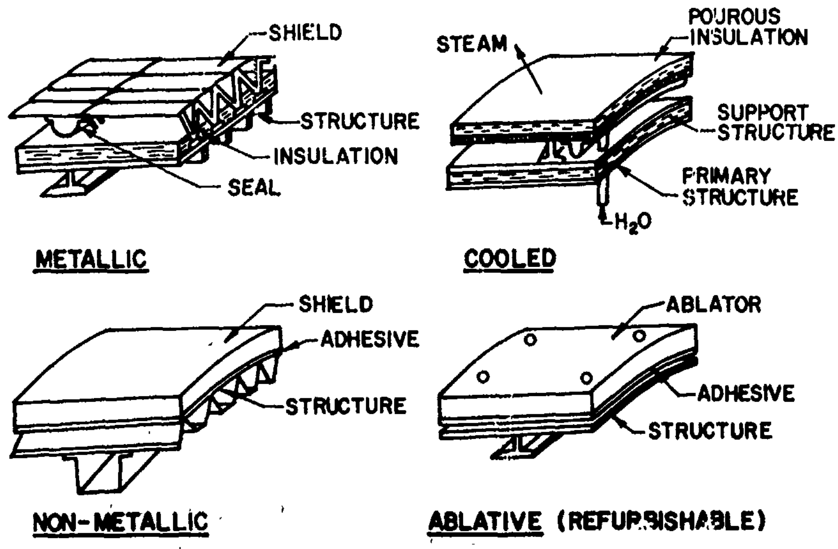
Non Metallic. — The system is composed of an external insulation such as rigidized ceramic foam bonded to a metal substrate. The external insulation withstands both the temperature and aerodynamic forces.

Cooled. — The cooled system shown used water supplied through a manifold to a distribution panel from which it enters a porous insulation such as felt metal. The incoming heat is absorbed and blocked by the phase change and the steam transpires to the surface of the insulation.

Ablator. — The ablator is considered as a back up to the other thermal protection systems . The ablator is bonded to a metal surface which is bolted to the load carrying structure. The access holes for bolting are closed with plugs of ablative material which can be removed for refurbishment.

The major point to be made from figure 1 is that a number of different approaches to thermal protection are available which have potential application. There are many variations in design for each type shown. Each type and design has its own specific problems and areas of concern and each has possible failure modes different from the others. As a consequence, the evaluation program set up to establish the relative merits of the different systems must be carefully planned such as to avoid the inadvertent introduction of bias through inadequate or improper testing techniques. Misleading test data could result in the incorrect choice of the systems. Inasmuch as there is not a great backlog of experience with reusable thermal protection systems; experimental screening is believed necessary to access the relative merits of different designs.

TYPICAL THERMAL PROTECTION CONCEPTS



In the screen activity, it appears that the interaction of the stream and structure should not be a primary consideration and that major emphasis should be placed on the thermal protection efficiency and gross life expectancy. This approach will permit relatively small specimens to be used. In the overall evaluation of the final contenders, however, the total interaction between the environment and T.P.S. appears necessary.

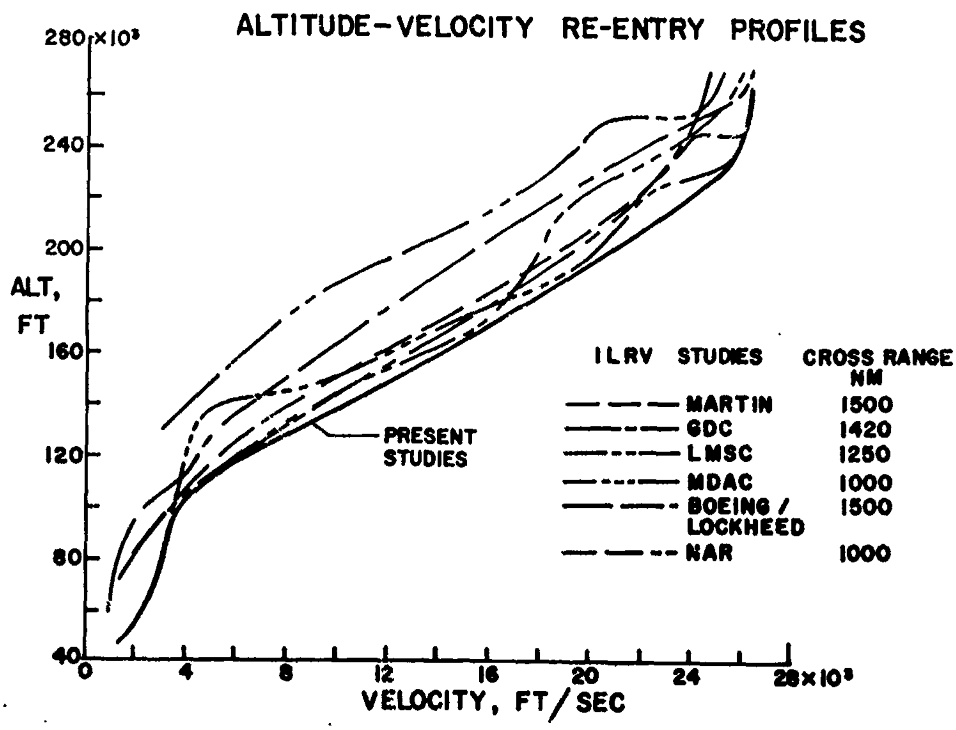
For screening tests some of the more important simulation requirements are indicated for some of the more common failure modes. The dots indicate which ones must be equal to specified values for evaluation of a specific failure mode. You will note certain failure modes for the TPS are also failure modes for materials but must be included in the screening process. For example: to evaluate the effects of oxidation on the life of a thermal protection system require that the wall temperature and pressure must be flight design values since these parameters have a pronounced effect on the chemistry. Further, since mass transfer gradient must correspond to a specific value to get the desired simulation of oxygen diffusion. The value of the ratio depends upon how you achieve the wall temperature. Of course accurate oxidation damage accumulation requires that you simulate the flight heating cycle or the wall temperature history.

The two important points to be made from this figure are (1) it appears that as many as five parameters must be matched simultaneously to specified values to insure a correct evaluation of a failure mode: and (2) it is observed that the evaluator has some latitude on how he achieves the simulation; Note that stream enthalpy or convective heat flux do not appear as important simulation parameters. Hence, radiant heating can be used to achieve wall temperatures and may possibly be coupled with a stream to get combined heating and loading for these screening tests.

SIMULATION REQUIREMENTS
TPS CONCEPT SCREENING
SPECIMEN SIZE ~ 6"×6"×2"

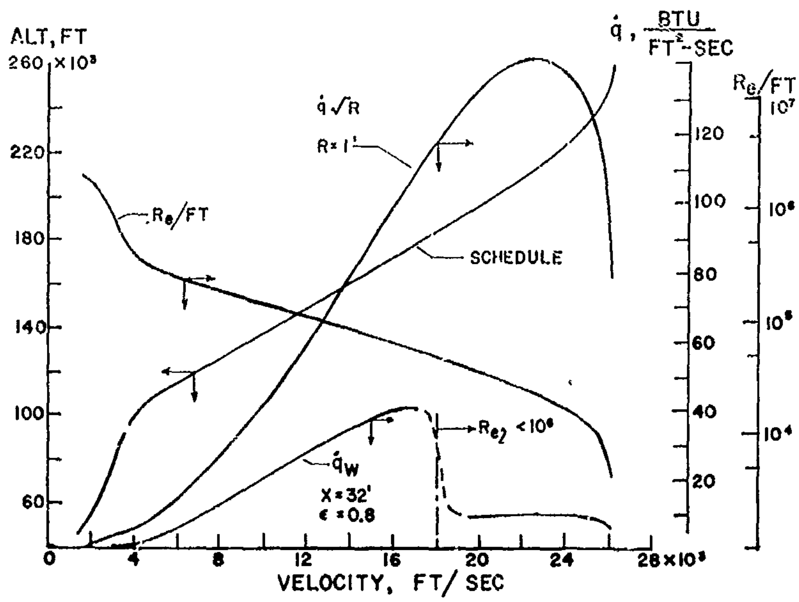
TYPE	FAILURE MODE	$\dot{q}_{HW}$	$\frac{d\dot{q}_{HW}}{dt}$	T_W	$\frac{dT_W}{dt}$	$\frac{dT}{d\eta}$	P_W	$\frac{dP_W}{dx}$	τ_W	h_∞	$\frac{K_{O_2}}{\Delta h}$
METALLIC	OXIDATION			●	●		●				●
	CREEP			●	●	●	●	●			
	CYCLIC FATG			●	●	●					
NON METALLIC	OXIDATION			●	●		●				●
	MECH. REMOVAL			●	●			●	●		
ABLATIVE	SWELLING	●	●				●				
	OXIDATION		●	●			●				●
	MECH. REMOVAL			●				●	●		

The environmental parameters that we have to match for simulation are defined by the entry trajectory of the vehicle. To establish a trajectory to be used for study purposes all of the Phase A ILRV study trajectories for shuttles with moderate cross range capability were compared on figure 3. You will note that with the exception of two, the trajectories appear relatively independent of configuration and band together very well. The lower bound of the trajectories indicated by the heavy solid line was selected as the present study trajectory. Because of its lower altitude it defines a more difficult set of conditions for simulation than a higher trajectory would. The trajectory selection, however, does not alter the general test environment picture that evolves.



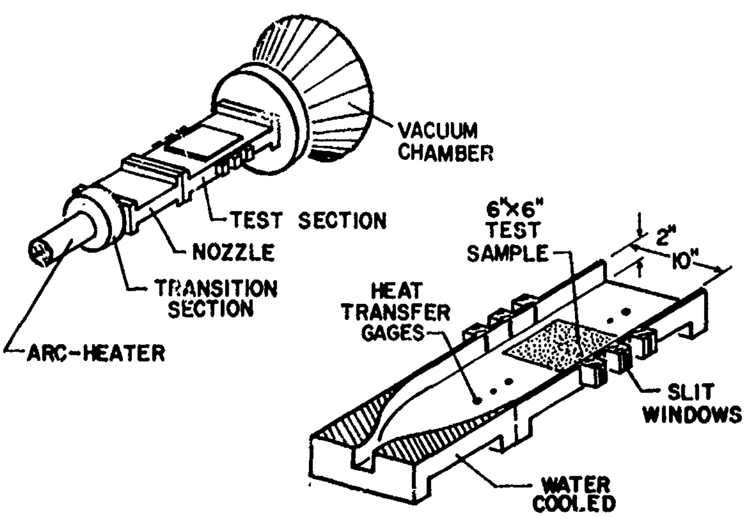
For the selected entry schedule stagnation heating rate, wall heating rate and stream unit Reynolds number were computed and are presented on figure (4). If one looks at stagnation heating and assumes that the vehicle heating follows it the indication is that the critical heating will occur at a very high altitude and soon after reentry begins. However, we find that Reynolds number increases as you proceed down the entry schedule and that local Reynolds number required for transition moves forward on the vehicles such that the peak heating for certain parts of the thermal protection system may be well into the earth's atmosphere where there is substantial aerodynamic loads. Hence, in experimental evaluation we not only have to be able to test for the peak stagnation heating environment but also be able to impose turbulent heating to the TPS at fairly high dynamic pressures.

HEATING AND REYNOLDS NUMBER VARIATION FOR RE-ENTRY



One method currently being considered for achieving this combination of turbulent heating and aerodynamic loading, is indicated on the next figure. Arc heated air jets are being modified such that they produce two dimensional flow. For the typical screening activity we've assumed a 6" x 6" specimen which is felt may be about as small as you can go in specimen size without getting edge effects on the thermal gradient through the system. To accommodate this size specimen you will need some room around it to avoid heating gradients near the nozzle wall. In the example presented, we have chosen a 10 inch wide duct and, to minimize flow requirements, the width of the duct was limited to 2 inches. Thus the exit area of the example apparatus is 20 square inches.

TYPICAL RECTANGULAR FLOW APPARATUS



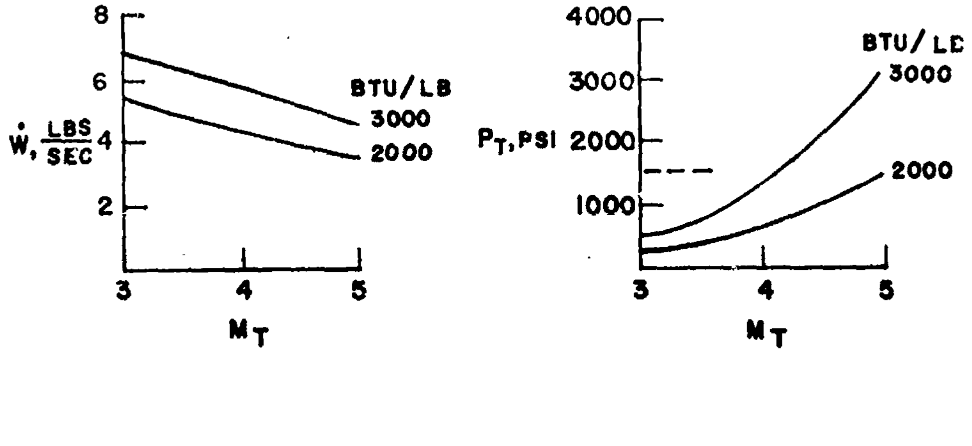
On the next figure (figure 6) an indication of the arc heater capability required to achieve the desired simulation is indicated. The specified conditions are (1) that the nozzle exit area is 20 square inches as developed on the previous figure; (2) that the unit Reynolds number be 10^6 to assure reasonable confidence that turbulent heating can be achieved, and (3) that a test cycle of the order of 2000 sec. is attainable since this is about the duration of the heating pulse for the moderate cross range shuttle.

The curve on the lower left indicates the weight flow of air for total enthalpys of 3000 and 2000 Btu/lb. as a function of test Mach number. Note that the product of weight flow and enthalpy give an indication of the power in the test stream. For example, at 4 lb/sec. and 2000 Btu/lb. the total power in the stream is 8000 Btu/sec. or the equivalent of approximately 8 MW of electrical power. Inasmuch as most arc heaters operate at efficiencies of 50 percent or less, the electrical power source required to achieve the simulation is 16 MW or more. The curve on the right indicates the total operating pressure in the arc chamber for the same set of parameters. The dashed line at 1500 psi is approximately the state of the art for arc heated jets. Generally, for long testing intervals the operating pressure is below 1000 psi. It appears, therefore, that if this approach to simulation is taken, the test Mach number will be low and consequently the mass flow and the power requirement will be high. These source requirements severely limit the facilities that may be able to contribute to the testing requirements of certain parts of the shuttle thermal protection screening. It should be noted that if wedges or deflected surfaces are used to get the local flow conditions needed, the total flow rate and consequently power requirements are somewhat larger than for this approach. Let's take a brief look now at the problem of overall thermal protection system evaluation.

ARC TUNNEL OPERATING REQMTS

$R_e/l = 10^6, FT^{-1}$ $A_x = 20$ IN.^{2}

$\Delta t \sim 2000$ SEC



The set of parameters shown on figure 7 are some of the more important from the heating and loading point of view. (Accoustic and vibration parameters are not included). Correct Reynolds number is required to assure that the flow characteristics including boundary layer, heat transfer, and detailed flows – such as through joints and cracks are correctly simulated. Stream energy level as the driving potential for heating should be correct, especially for the evaluation of the effects of flows through joints and cracks as well as to assure correct temperature levels. Heating history is very important since in many instances the largest thermal strains have been developed as a result of temperature lags within the structure. Hence, heating history will have significance with regard to thermal fatigue. Aeroelastic parameter must be matched to assure a correct loading. Our experience has indicated that the details of fabrication are significant to the performance of structures which undergo thermal cycles. Consequently, the hardware tested should be that to be incorporated in the flight vehicle insofar as the design detail, materials, and fabrication is concerned. Finally, there is a clear indication that edge mach number will have a pronounced effect on heating resulting from surface roughness and joints.

If you add these simulation requirements you realize that, ideally, you want to test full scale TPS panels in the real and critical environment. The indications are that this may not be possible prior to an actual flight of the vehicle.

Accordingly, it is believed that an effort should be made to determine what simulation requirements can be relaxed and by how much and still obtain a correct overall evaluation of the general thermal protection system.

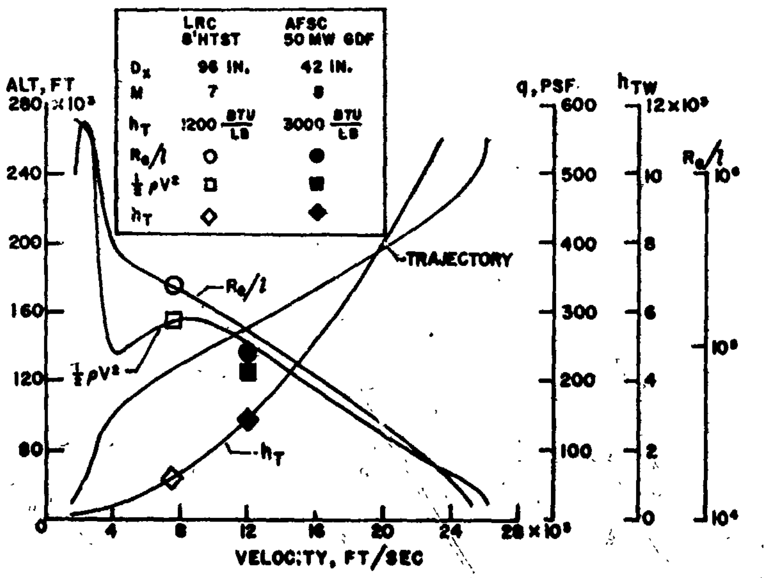
SIMULATION NEEDED FOR OVERALL TPS EVALUATION

- REYNOLDS NUMBER, $\frac{\rho_\infty V_\infty l}{\mu_\infty}$
- STREAM ENERGY LEVEL, h_{T_∞}
- CONVECTIVE HEATING HISTORY, $\frac{d\dot{q}_{HW}}{dt}$
- AEROELASTIC SIMULATION PARAMETER, $\frac{\rho_\infty V_\infty^2}{E}$
- DETAILS OF TPS DESIGN MATERIALS AND FABRICATION
- SCALE 1.1
- EDGE MACH NUMBER, M_e

Irregardless of the ultimate environmental simulation deemed acceptable, it is believed that the hardware to be tested will be essentially full size panels. Such panels are typically of the order of 2' x 2' square or larger. After supporting these panels in a fixture the total transverse dimension may be 2.5 to 3 ft. minimum. Currently, there are only two facilities in the country having the potential of testing such large panels in a realistic hypersonic stream. They are the NASA – LRC – 8' HTST and the AFSC-FDL-50MW GDF. Figure 8 presents the variation of stream dynamic pressure, unit Reynolds number, and total enthalpy along the flight reentry trajectory. On the figure the performance of LRC 8' HTST and the Air Force 50 MW Gas Dynamics Facility is indicated. Inasmuch as the 8' HTST was designed specifically for structures evaluations it matches the flight environment very well at M=7.0. Note that for this particular entry schedule this is very near maximum dynamic pressure. The ability of the facility to duplicate flight environment is offset by the fact that it is combustion heated (contaminants and oxygen deficient) and only has a testing interval of about 200 seconds. The performance point for the 50 MW facility are for its 3.5 ft. diameter nozzle which is felt to be about as small a nozzle as can be used for testing panels of this size. Because of operating pressure limitation, all of its parameters do not exactly match flight values. For this comparison the stream enthalpy has been matched while the dynamic pressure and Reynolds number fall below the flight values. This facility uses arc heated air and can test for an interval of 1800 seconds. Both of the facilities may encounter difficulty in achieving turbulent heating unless some means of tripping the boundary layer is devised.

OVERALL TPS EVALUATION

SPECIMEN SIZE ~ 24" x 24"



SUMMARY

The problem of experimentally establishing the relative merits of thermal protection concepts proposed for the space shuttle is made complex by the different simulation requirements for the different possible failure modes.

Test facilities capable of simultaneous simulation of different parameters in combination as required for TPS screening do not currently exist. Plans are currently underway in certain NASA Centers to hopefully alleviate this problem. However, the performance of these new or modified facilities is not certain.

Overall evaluation of thermal protection systems apparently requires testing of large specimens under correct combinations of aerodynamic heating and loading. Facilities currently available for such evaluation and verification tests are believed to be inadequate.

It is believed that considerable additional effort should be spent defining acceptable simulation for overall TPS evaluation and verification so that facility requirements can be more adequately established.

N70-42995

MATERIALS TECHNOLOGY - INTRODUCTION AND OVERVIEW

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This session will be concerned with the advanced materials technology that is required for the space shuttle. First, however, it might be pertinent to consider why the space shuttle presents unique materials problems. The environment in which the shuttle will operate has already been covered in this conference but it would be worthwhile to review very briefly the salient features that apply to the selection of the structural materials. In the table on top of Fig. 1, the space shuttle is compared to both aircraft and to current Apollo type reentry vehicles. The shuttle will operate at much higher temperature than current aircraft, will have a payload that is a considerably smaller fraction of gross weight, and will use hydrogen fuel. The differences one encounters when comparing the space shuttle to present reentry vehicles are all concerned with the ability of the vehicle to land and be reused. The shuttle should be capable of landing subsonically, of being ready for another flight a fairly short time after landing, and of accumulating a total of one hundred flights with a minimum of repair or refurbishment.

When these requirements are translated into materials terminology, it becomes apparent that this is the first aircraft type device in which parts of the structure will be creep limited. It is also evident that the oxidation resistance required of the shuttle will be many times that of previous aircraft. The constructional materials must, of course, not be embrittled by the hydrogen which will be used as a fuel and many of the bearings, seals and hydraulic fluids will have to operate intermittently, with great reliability, and at temperatures in excess of those met in current practice. In addition there is of course the overall requirement that all constructional materials, both in the structure of the space shuttle and in its power plants, be as light as possible.

The several papers in this session will deal with various aspects of these problems. Before coming to the papers, a brief discussion of the current Materials Technology Program, supplemented with an indication of program trends for the coming year, is in order. This is done in Fig. 2. The first major topic shown there is High Temperature Metals. This includes the technology of columbium, tantalum, and dispersion strengthened alloys and of the coatings which protect them for use above 1800°F. This topic was discussed -- for continuity purposes -- during the thermal protection systems session yesterday. You may recall that the program is a joint one covering activities at the Lewis, Manned Spacecraft, and Marshall Centers. In the fiscal year just ending, two million one hundred thousand dollars of research was conducted under this major topic. It is anticipated that this program will remain at about its current level during the coming year. The next major topic, Structural Composite Materials, is concerned with materials technolc as opposed to the application studies covered yesterday. These materials are expected to contribute materially to achieving the light weight structure that the shuttle must have. The program is being conducted by the Lewis and Marshall Centers and in the past year was just barely initiated with a one hundred thousand dollar effort. This will be substantially increased in the coming year. The third major topic, Environmental Compatibility, is concerned with the ability of the materials to avoid hydrogen embrittlement, corrosion, stress corrosion, and flammability problems. The program was carried out at the Marshall Center in the past year at a level of three hundred thousand dollars. In the forthcoming year it is anticipated that the continuing program at Marshall and a program to be initiated at the Manned Spacecraft Center will, in combination, be somewhat greater than the present effort. Bearings, Lubricants, and Seals studies were carried out by the Lewis and Marshall Centers at a level of about four hundred thousand dollars in the past year.

In the forthcoming year this program will be kept at approximately the same level. With so many new materials being used in the shuttle, the accumulation of statistically significant design data poses a rather considerable problem. To alleviate this problem about one half million dollars was spent in the past year and a modest increase is anticipated in the coming year. Finally, as the fracture mechanics and other discussions indicated yesterday, any defects that develop during a flight must be found by inspection and quickly repaired so as to allow prompt recertification for the next flight. The program at the Marshall Space Flight Center to ensure that this can be done has been running at the rate of about five hundred thousand dollars a year and is expected to increase.

A more detailed breakdown of these major topics is given in Figures 3 and 4. In Fig. 3, for example, the high temperature program is broken down into the specific studies involving dispersion strengthened alloys, columbium and tantalum. And in the final figure, for example, the specific topics to be covered in environmental compatibility are detailed.

The program that is covered by these figures represents an effort to attack the most critical problems expected with the materials for the space shuttle. A very conscious effort has been made to concentrate initial efforts on those areas where the problems appear to us to be most severe and which require the longest lead times for their solution. As experience and confidence in these areas are gained it can be anticipated that the program will be modified to encompass a broader spectrum of material problem areas.

The papers to follow will attempt to assess the technology in several of the outlined areas. Many will indicate gaps that we seek your assistance in filling.

UNIQUE OPERATIONAL CHARACTERISTICS	
COMPARED TO AIRCRAFT	COMPARED TO REENTRY VEHICLES
• HIGHER TEMPERATURES (TO 3000° F)	• 100-MISSION REUSE
• SMALLER PAYLOAD WEIGHT FRACTION	• SUBSONIC MANEUVERABILITY
• HYDROGEN FUEL	• FAST TURN AROUND

PLACE NEW DEMANDS ON MATERIALS

- CREEP RESISTANT STRUCTURE
- DYNAMIC OXIDATION RESISTANCE
- COMPATIBILITY WITH HYDROGEN
- HOTTER SEALS, BEARINGS, HYDRAULIC FLUIDS

FIGURE 1

SPACE SHUTTLE MATERIALS TECHNOLOGY PROGRAM

MAJOR TOPIC	CENTER	FY 70 FUNDS (thousands)	FY 71 (trend)
HIGH TEMPERATURE METALS	LeRC,MSC,MSFC	2100	LEVEL
STRUCTURAL COMPOSITE MATERIALS	LeRC,MSFC	100	INCREASE
MATLS. ENVIRONMENTAL COMPATIBILITY	MSC,MSFC	300	INCREASE
BEARINGS,LUBRICANTS AND SEALS	LeRC,MSFC	400	LEVEL
DESIGN PROPERTIES OF MATERIALS	LeRC, MSFC	500	INCREASE
ADVANCED INSPECTION & REPAIR TECHNIQUES FOR MATERIALS	MSFC	500	INCREASE

LeRC—LEWIS RESEARCH CENTER
MSC—MANNED SPACECRAFT CENTER
MSFC—MARSHALL SPACEFLIGHT CENTER

FIGURE 2

SPACE SHUTTLE MATERIALS TECHNOLOGY

ELEMENTS OF PROPOSED NASA FY 71 R&D PROGRAMS

STRUCTURAL COMPOSITE MATERIALS

EVALUATE BORON-ALUMINUM COMPOSITE SHEET & SHAPES
DEVELOP HIGH MODULUS ORGANIC FIBER COMPOSITES
ADVANCE TECHNOLOGY OF RESIN-FIBER COMPOSITES - IMPROVE RESINS, PROVIDE IMPROVED ANALYTICAL DESIGN METHODS, STUDY THERMAL FATIGUE, RESIDUAL STRESS, & FLAW PROPAGATION

HIGH TEMPERATURE METALS

EVALUATE PROPERTIES OF Ni-Cr-ThO_2, INCLUDING OXIDATION RESISTANCE
IMPROVE & SCALE-UP METHODS OF PRODUCING Ni-Cr-ThO_2 SHEET
DEVELOP METHODS FOR FORMING & JOINING Ni-Cr-ThO_2
CONSIDER ALTERNATE SUPPLIERS OF Ni-Cr-ThO_2 SHEET & MODIFIED COMPOSITIONS
DEVELOP IMPROVED COATINGS FOR COLUMBIUM & TANTALUM ALLOYS & EVALUATE IN 2000^0-2700^0 F ENVIRONMENTS
DEVELOP MANUFACTURING TECHNOLOGY FOR FORMING COATED Cb & Ta HEAT SHIELDS

245

FIGURE 3

MATERIALS ENVIRONMENTAL COMPATIBILITY

EVALUATE COMPATIBILITY WITH HYDROGEN, SIMULATING PRESSURE, LOAD, & TEMPERATURE OF FLIGHT HARDWARE
DETERMINE RESISTANCE OF SHUTTLE ALLOYS TO CORROSION & STRESS CORROSION
STUDY FLAMMABILITY & OXYGEN COMPATIBILITY OF SHEET MATERIALS, POTTING COMPOUNDS, & ELECTRICAL WIRE

BEARINGS, LUBRICANTS, & SEALS

DEVELOP FLUID & DRY LUBRICANTS SUITED TO HIGH TEMPERATURE OPERATIONAL REQUIREMENTS
DEVELOP SPECIAL DESIGNS FOR SLIDING & ROLLING ELEMENTS
DEVELOP FACE TYPE SEALS FOR LIQUID HYDROGEN PUMPS

DESIGN PROPERTIES OF MATERIALS

DETERMINE STRENGTH VS TEMPERATURE & CREEP DATA ON SUPERALLOYS, Ni-Cr-ThO_2, COATED Cb & Ta SHEET
OBTAIN VALID EMISSIVITY DATA UNDER OPERATIONAL CONDITIONS
APPLY FRACTURE MECHANICS TO PRESSURE VESSEL DESIGN

FIGURE 4

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MATERIALS RE-USE, EVALUATION, INSPECTION AND REPAIR

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Objectives

The primary objectives of the space shuttle vehicle are to provide reliable and low cost transportation from earth to orbit and return in a variety of missions. This requires a re-useable and a light-weight vehicle. Re-useability requires that we must minimize the amount of turn-around time and effort required for inspection, refurbishment, wearout, replacement maintenance and repair between flights. Light-weight certainly requires that we use high strength/low density materials in the thinnest possible gages and that we minimize temperatures so that the strength of the materials will not fall off rapidly. Also, we must have a re-radiation thermal protection system to get rid of the energy rather than a system which consumes material to absorb energy. These requirements lead us to consider the re-useable metal instead of ablative systems in the heat shield, rocket nozzles, attitude control and auxiliary propulsion systems throughout the vehicle (figs. 1 and 2).

Background

The experience that we have with all-metallic re-radiative re-entry vehicles is impressive. The six Asset vehicles in addition to the several boost glide re-entry vehicles used coated columbium and tantalum alloy re-radiative panels. The side heat shield panels of both Mercury and Gemini, those highly successful manned spacecraft, employed super-alloy re-radiative heat shield panels. Our total experience and confidence with this type of thermal protection is sufficient to proceed with larger vehicles (fig. 3)(Ref. I).

Requirements

The space shuttle requires a combination of aircraft technology, booster technology, and orbital spacecraft technology. For instance, an airplane such as the DC-8 or 707 is built to sustain on the order of 100,000 takeoffs and landings; an Apollo vehicle, only one; and the space shuttle, somewhat more than 100. We must build a spacecraft to go through boost and re-entry with aircraft type of re-useability. Looking at the primary structure, the temperature variation in an airplane is approximately 200°F; in a spacecraft it is over 600°F where cryogenic tankage is involved. In the shuttle it will probably be over 700°F, because we need to allow the aluminum tanks to attain as high a temperature as possible. We can look at the structure in terms of pressure differential, safety factors, mass fractions or anything else and we see that we are required to build a vehicle which must accomplish the mission of a spacecraft with the re-useability of an aircraft (fig. 4).

We must consider the many time-and-temperature-dependent properties which change with the life of the vehicle. From the point of view of mechanical properties these are, of course, the strength and ductility, in addition to the creep deformation and remaining fatigue life. Low cycle fatigue is important where we look to the number of component operations such as pressurization cycles or loads on landing gears. High cycle fatigue is important in terms of time of powered flight at frequencies of engine resonance and turbulence in fuel lines. Finally, acoustic fatigue is important mainly during boost and re-entry. We must also consider fracture mechanics to look at the material in terms of its toughness and resistance to crack propagation rather than its strength.

All of these mechanical properties are more or less dependent for their lifetime on the metallurgical or solid state of the material. We are talking here about the microstructure in which the thermal history is extremely important. Not only initial properties but the change in the properties depends upon whether the material is in a solution annealed or "as-cast" condition or whether it has been precipitation aged. Also, the all important creep deformation depends on whether the material has reached a temperature at which recrystallization and changes in the grain structure have occurred. Finally, we must look at the surface of the material for oxidation of coatings, diffusion of coatings, into the substrate or of oxide into the coatings, and volatilization of alloying elements into the vacuum of space, which may change emissivity, probably the one most important property we must consider in a re-radiative temperature limited system. Of course, erosion and surface topology of the coatings is very important also (fig. 5). At this point we must add that the re-useability of the fibrous chrome-aluminum silicate, high-temperature insulation is vital to the protection of the primary structure. Changes in fiber length or compaction due to vibration or to moisture pick-up on cooling must be minimized.

Mechanical Properties

Figure 6 is an actual creep curve on panels which the McDonnell-Douglas Corporation tested for the Air Force. The panels deformed further with each 60-minute cycle at 2400°F under load. The total deflection allowable for this type of panel was determined to be 40 mils and this particular panel reached that limit in 21 cycles. The study indicated that panels of this sort reach their creep limit well before they reach their life limit by any other mode of degradation. This curve also shows the difference in creep rate of an undamaged surface-coated columbium panel and one in which there has been damage to the coating over a spotweld. The major concern is with contamination of the columbium substrate with oxygen and the ensuing embrittlement as evidenced by the steeper creep rate. Fortunately, the referenced study shows that coating damage does not appear to have catastrophic effect on the performance of the silicide coated columbium panels at reduced pressures in a static test environment. (Ref II).

Figure 7 is an example of recently obtained fatigue data from the University of Minnesota on Rene' 41 at 1600°F. This is a nickel alloy, very popular because of its strength but very sensitive to thermal-mechanical history. The "A" values of the different curves are for various ratios of cyclic to mean stresses in the specimen indicating how cyclic stresses imposed on top of steady state loads degrade the total stress capability of the material. (Ref. III).

Figure 8 is a typical fracture toughness curve from Tiffany at Boeing showing normalized stress intensity versus life. K_{1i} in the numerator is the initial crack stress intensity due to a crack which is determined to be in the material by either non-destructive inspection or an initial proof test. K_{1c} in the denominator is the critical stress at which fracture occurs in the material due to rapid and unstable propagation of a crack. The A over Q values are equivalent crack dimensions. Both the K_{1c} value and the shape of the curve can be determined for any given material, a few of which are listed in the upper right hand corner. The important point here is that if one determined that 0.4 of the fracture stress intensity has been imposed upon the material, then something like 200 cycles of life are remaining. If "A" cycles are applied in the life of the vehicle, then the stress intensity will increase to 0.6 of the critical fracture stress intensity for this material. We expect to extend this type of analysis of pressure vessels to other areas of the cyclicly loaded structure. Ref. IV.

Metallurgical Structure

For figure 9 in the area of metallurgical or micro-structural stability we have turned to the engine manufacturer, Pratt and Whitney, Division of United Aircraft Corporation, who has much experience with superalloys at high temperature. In this case we see the difference in the microstructure of IN 100, a complex nickel alloy, in which the annealed structure shows large ir-regular grains and fairly clean grain boundaries. The microstructure on the right, which has been cycled to 2000°F and then aged at about 1600°F shows a much finer and much more regular crystal structure with a large amount of carbide precipitation in the grain boundaries. The point is that the mechanical properties of this material have been greatly changed with the change in microstructure due to the thermal history. Later Mr. Cataldo will discuss a similar situation involving grain growth in HS 188, a cobalt alloy which many people have suggested for use on the space shuttle.

Surface Structure

Next in importance, we must emphasize the re-useability of silicide coatings on columbium alloys. This involves oxidation of the coating, diffusion of the oxide into the coating and of the coating into the substrate, volatization, erosion, spalling, and cracking of the coating. Stoichiometric changes of the oxide cause volumetric changes, hence cracking, in addition to differential thermal expansion between the silicide coating and the columbium substrate (see fig. 10). Figure 11 is a typical diagram, shown by Perkins of Lockheed, of the many different kinds of cracks and defects one should look for in these coatings.

Figure 12 is an indication of the effects of both temperature and thickness indicating that the heavier coatings will provide longer life because of the greater depth to which the oxides penetrate before they reach the columbium substrate. Fortunately, many studies have now shown that many cycles of safe operation can be obtained even after the oxides reach the columbium substrate (Ref. V).

Emissivity

Figure 13 is a diagram of the change of emissivity with the number of cycles to 2000°F. The bottom curve is a typical Sylvania R512E silicide coating on columbium showing a rather large 0.19 change in emissivity. McDonnell Douglas Corporation has shown in the middle curve how they can modify this coating and obtain a much smaller change in emissivity.

The top curve is from some data provided by General Electric Company from Snap 27 flight reactor experience on the Vac-Hyd 109 silicide coating on columbium 129Y, which shows both a higher value and no change of emissivity over a long period of time. Much more data on these coatings is needed, however, to substantiate these curves.

Non-Destructive Inspection

I will not dwell on non-destructive inspection because Mr. Steinbring, General Electric Company, will give us much better detail on this subject in the next paper. However, we know that there are several types of measurements that can be made both initially and between flights very efficiently. For instance, looking at the coating of silicide or columbium, we can see oxidation because of the yellow-green coloration of the oxided area. We can measure the change in thickness with rapid scanning eddy current and thermoelectric probes. Finally, cracks can be determined using dye penetrant and radiographic techniques. There are several ways we can also look at the important variables in the metallic substrates. The grain size changes which were mentioned earlier can be determined by both ultrasonic and thermoelectric techniques which, again, can be applied in a rapid scan mode. Radiographic techniques can be applied to find major cracks, and beta gage scanning devices can be used to determine changes in total thickness. The moisture content of the high-temperature insulation can be determined by dielectric techniques (fig. 14).

Repair:

We are looking at several field repair techniques for quick fixes of the silicide coating on the panel. These involve slurry spray coating, electrophoretic deposition, plasma arc deposition, hot patches, of both low melting silicate or exothermic paste on a film which will exclude air while the material is being heated. Also, coatings such as the platinum paint or ceramic cement may be useful for patching oxided areas. We have not yet determined which of these might be the most effective for use directly on the vehicle. Most of them will require some sort of high temperature tool to place over the patched area. In terms of refurbishment, there are many causes which may damage a coating. Some of these are listed on figure 15, such as the weather, ground debris, someone dropping a tool, brittle cracks which occur during thermal changes, ablation products from some other part of the vehicle coating the surface, meteoroid impact, and the deformations which are put into the material, and the failure of fasteners which may shake loose. We are attempting to determine the tolerance of the various kinds of panels to these forms of damage and the best methods which san be used to repair them.

Summary

In summary, the primary thermal-protection-shield life-limitations as we now see them, are first creep and the total deformation in the panel: second, thermal stability of the microstructure or changes in the grain size, and finally flaws that may occur in the coating. We have development and evaluation programs active in measuring properties and in determining the flaw effects on the reuseability of the material, on the non-destructive inspection techniques which can be applied, and finally in the repair techniques which can be used to speed the vehicle onto its next mission. All of this is aimed at making a space shuttle reuseable and, therefore, both reliable and low cost in that one vehicle can do many jobs.

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OBJECTIVES

RELIABLE, LOW-COST TRANSPORTATION

FROM EARTH TO ORBIT AND RETURN

REQUIRES A REUSEABLE, LIGHT-WEIGHT VEHICLE

REUSEABILITY REQUIRES MINIMIZING

INSPECTION, REFURBISHMENT

WEAR-OUT, AND REPLACEMENT

MAINTENANCE AND REPAIR

LIGHT-WEIGHT REQUIRES

HIGH STRENGTH/LOW DENSITY

THIN GAGES

MINIMUM TEMPERATURES

RE-RADIATION OF ENERGY

Figure 1

BACKGROUND WITH METALLIC RE-RADIATIVE RE-ENTRY VEHICLES

ASSET

BGRV

MERCURY

GEMINI

Figure 2

COMBINATION OF AIRCRAFT, BOOSTER AND ORBITAL SPACECRAFT REQUIREMENTS

	AIRPLANE	SPACECRAFT	SHUTTLE
FLIGHTS	10^5	1	10^2
PRIMARY STRUCTURAL			
a. TEMPERATURES, °F			
MAXIMUM	+140.	+200.	+300.
MINIMUM	-60.	-423.	-423.
b. PRESSURES, PSIA	+10.	30.	?
C. SAFETY FACTORS	1.5-2	1.1 - 1.4	?

Figure 3

TIME AND TEMPERATURE DEPENDENT PROPERTIES

MECHANICAL

- STRENGTH AND DUCTILITY
- CREEP DEFORMATION
- FATIGUE
  - LOW CYCLE - COMPONENT OPERATIONS
  - HIGH CYCLE - TIME OF POWERED FLIGHT
  - ACOUSTIC - BOOST AND RE-ENTRY
- FRACTURE TOUGHNESS

METALLURGICAL SOLID STATE

- THERMAL CONDITION
- PRECIPITATION
- RECRYSTALLIZATION
- GRAIN GROWTH

SURFACE CONDITION

- OXIDATION
- DIFFUSION
- VOLATILIZATION
- EMISSIVITY
- EROSION

Figure 4

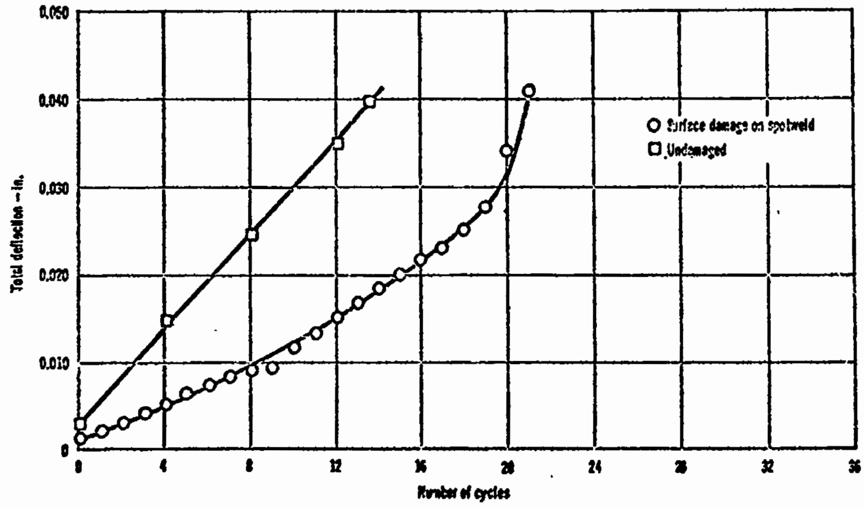


Figure 5 – Total Deflections of Undamaged and Surface Damaged Spotwelded Flat Corrugation Stiffened Panels Tested at External Pressure/High Stress

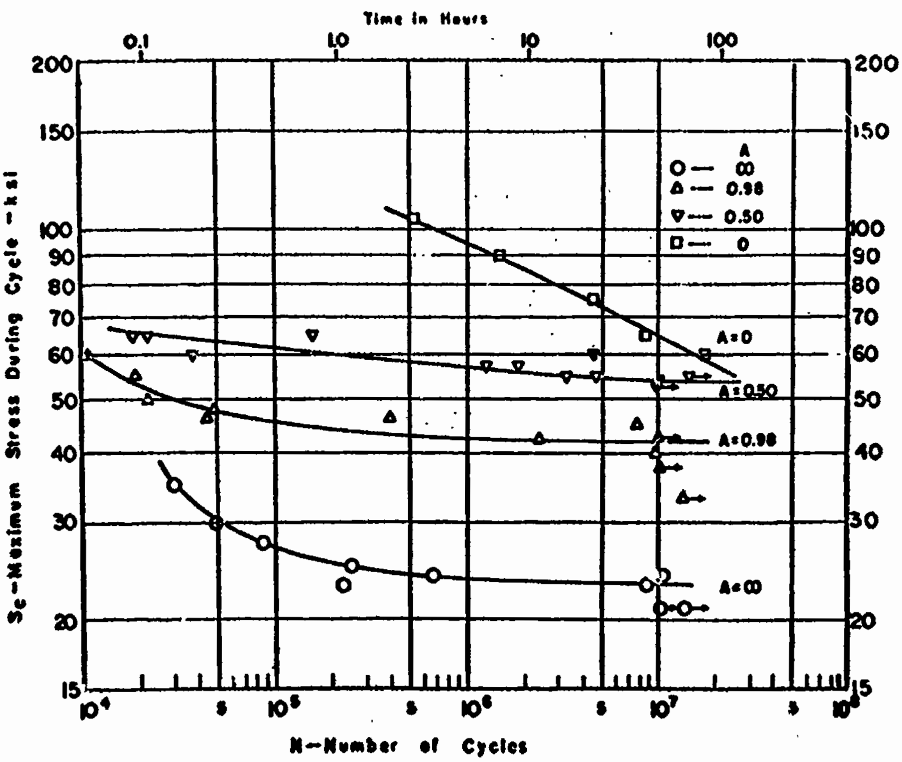


Figure 6 - S-N Diagrams for Notched Specimens (K_t = 3.0 of René 41 at 1600°F.

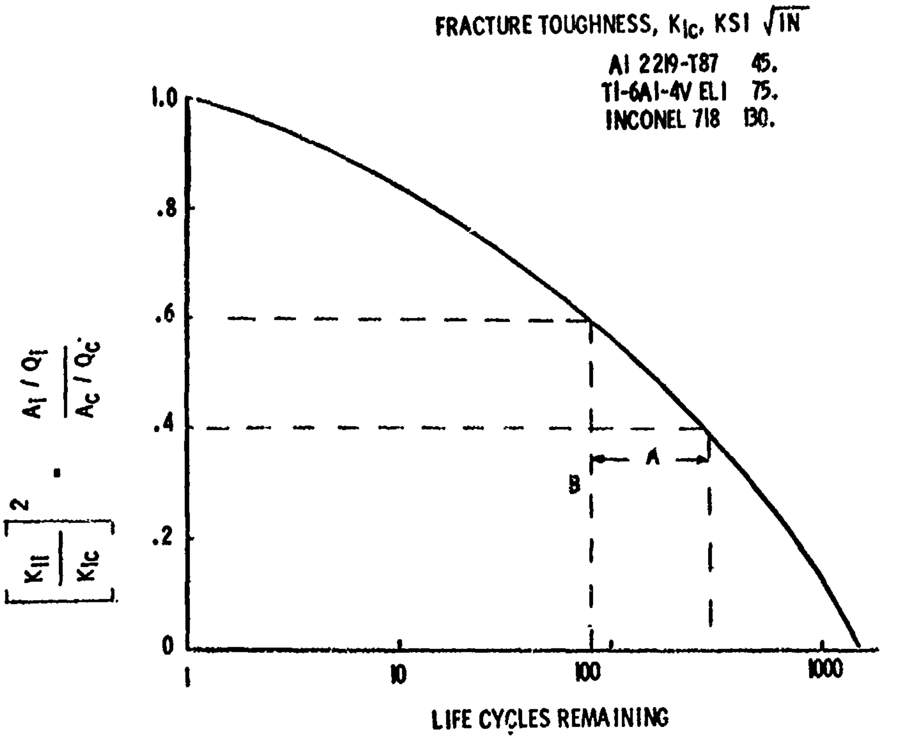


Figure 7

MICROSTRUCTURAL CHANGES IN 20% COLD WORKED HS-188 ALLOY

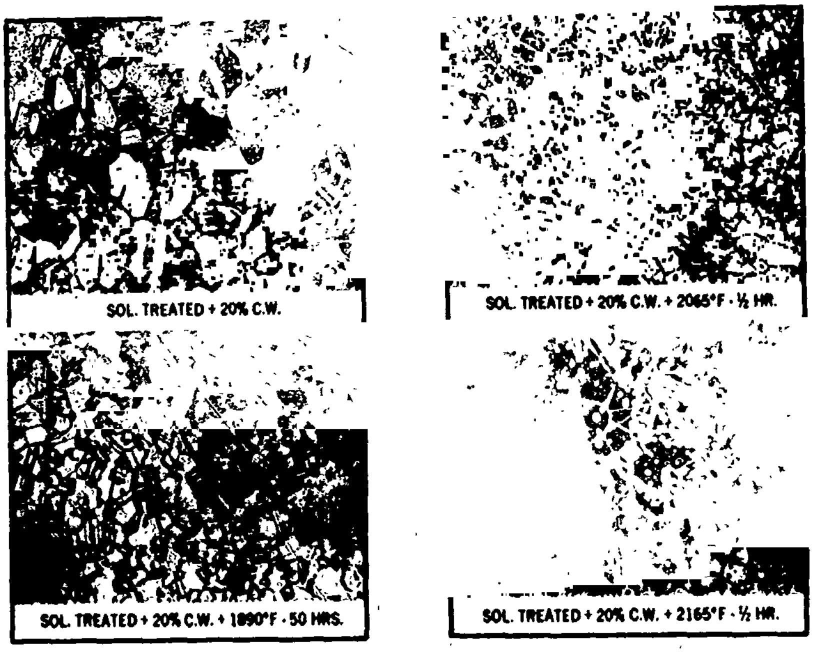


Figure 8

SILICIDE COATINGS ON COLUMBIUM ALLOYS

OXIDATION

DIFFUSION

VOLATILIZATION

EROSION/SPALLING

STIOCHIOMETRIC/VOLUMETRIC CHANGES

DIFFERENTIAL THERMAL EXPANSION

Figure 9

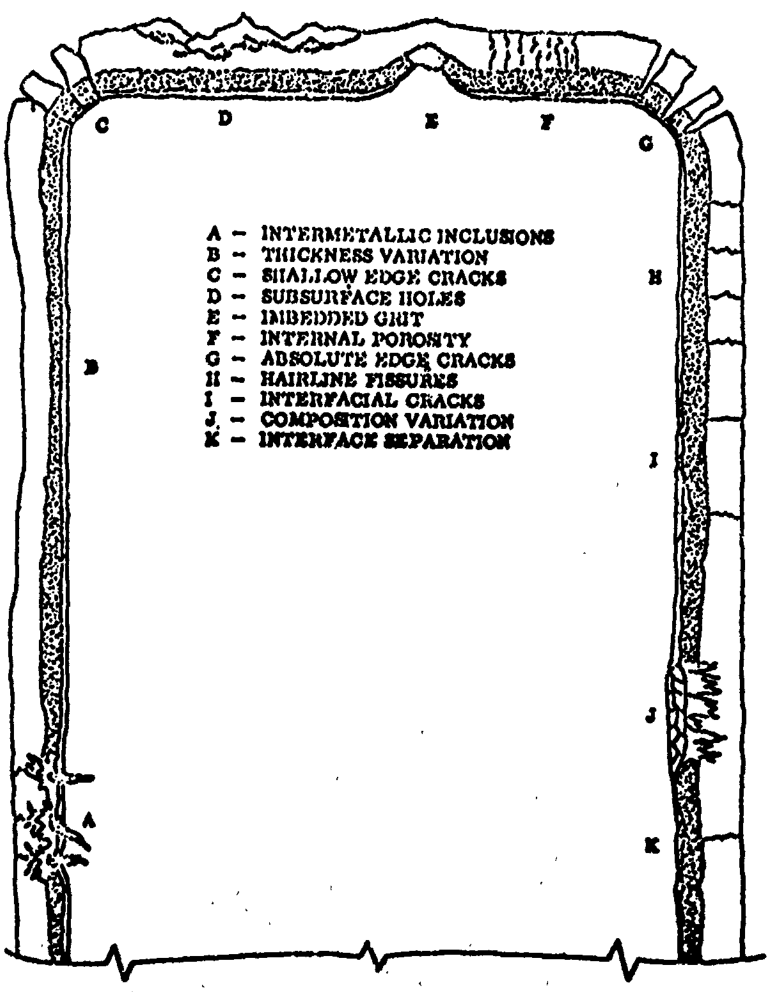
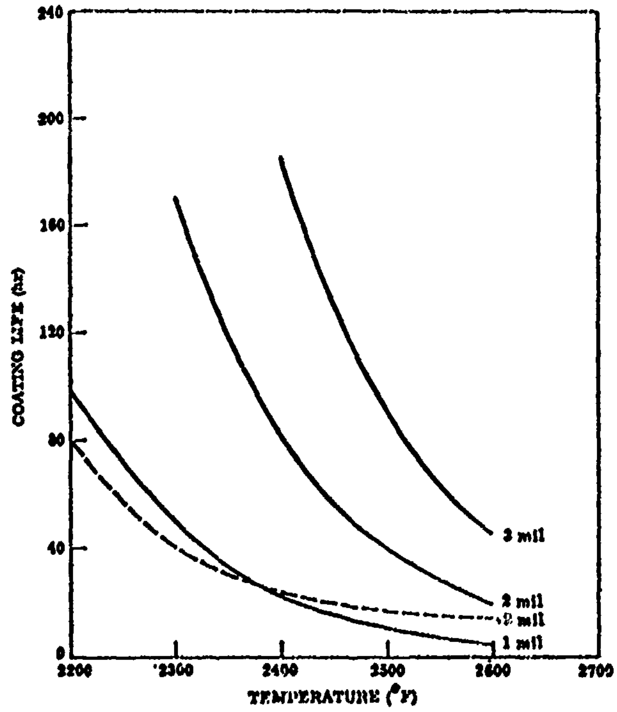
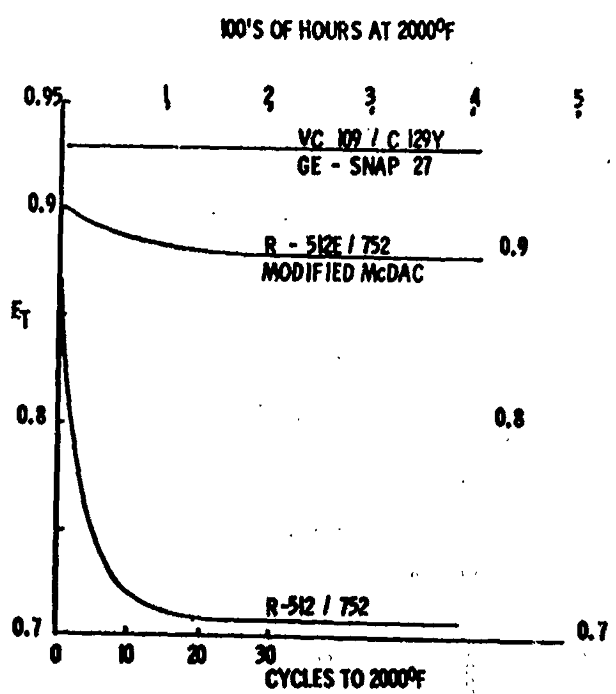


Figure 10



SILICIDE COATING LIFE

Figure 11



TOTAL EMITTANCE OF Si / Cb

Figure 12

NON-DESTRUCTIVE INSPECTION

INITIALLY AND BETWEEN FLIGHTS

COATING

OXIDATION	- VISUAL YELLOW-GREEN COLOR
THICKNESS	- EDDY CURRENT THERMOELECTRIC
CRACKS	- DYE PENETRANT RADIOGRAPHY

SUBSTRATE

GRAIN SIZE	- ULTRASONIC - THERMOELECTRIC
CRACKS	- RADIOGRAPHY
THICKNESS	- BETA GAGE

Figure 13

REFURBISHMENT

DAMAGE CAUSES

HAIL, SLEET, RAIN
GROUND DEBRIS
TOOLING DROP
BRITTLE CRACKS
ABLATION PRODUCTS
METEOROIDS
DEFORMATIONS, BOWING, FLEXING·
FASTENER FAILURE

DAMAGE TOLERANCE

REPAIR TECHNIQUES

SLURRY SPRAY COATING
ELECTROPHORETIC DEPOSITION
PLASMA-ARC
HOT PATCH - LOW MELTING SILICATE
- EXOTHERMIC PASTE
PLATINUM PAINT
CERAMIC CEMENT

Figure 14

N70-42997

THE IMPORTANCE OF NONDESTRUCTIVE TESTING FOR THE COATED METALLIC THERMAL PROTECTION SYSTEM

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THE RELIABILITY REQUIREMENTS

The key to success for the space shuttle program is the reliable performance of the vehicle for a projected one hundred flights. A catastrophic failure at any time prior to the full 100 missions, resulting in the loss of lives and the vehicle, would mark the end of the entire space shuttle program and possibly of future manned space missions. Each flight of the shuttle vehicle must have the same high confidence as the first flight and yet the requirements of economics and fast turn-around time serve as formidable constraints.

In addition, it can be anticipated that in the course of 100 missions some launches and re-entries will not be textbook perfect. For example, it may be necessary to perform evasive maneuvers or to pass through hail, rain, dust etc. or to undergo a hard landing. Each of these situations cause thermal and/or mechanical overstressing of the vehicle materials and unpredictable damage or abnormal degradation may occur.

How then can we assure the high reliability that is required? Essentially, this depends on three primary elements:

- Proper Design
  The vehicle must be designed with emphasis on fail-safe aspects so that local failure of one or several components, such as panels on the TPS, will not cause complete failure of the system.

- Materials Characterization
  There must be a thorough knowledge and understanding of the materials behavior under the thermal and mechanical stresses of multiple re-entries.

- Nondestructive Testing (NDT)
  A complete set of sensitive, rapid, and effective NDT methods must be developed and applied for evaluating the status and monitoring the properties of the vehicle materials as a function of exposure to the re-entry environment. These methods must be able to detect the onset of incipient failure and to measure abnormal degradation.

THE SYSTEMATIC APPROACH TO NDT IS ESSENTIAL

NDT will have a direct impact on every stage of manufacture and operation (Figure 1) as dictated by the requirements for re-usability for 100 missions. The starting point for NDT development is the thorough knowledge of the potential failure mechanisms and the material variables which influence behavior. Based on this knowledge, NDT techniques, which are sensitive to the important variables and defects, can be developed and time and energy will not be spent developing techniques for detecting conditions which have no relevancy.

1. In-Process NDT Evaluation

It is important to note that variability in the material properties is "locked-in" during the manufacturing process stages. As a result, it is essential that those variables which lead to premature failure are detected and minimized. This can be done by the effective application of NDT techniques for process monitoring and control, as well as for final inspection.

2. Systems Checkout and Prelaunch Evaluation

NDT techniques will be used for evaluating the space shuttle vehicle before the first launch. This will be primarily to assure the initial integrity of the vehicle and to obtain base-line data for future reference. The assessment of degradation from subsequent re-entry exposures will be dependent on comparisons with prior NDT data.

The NDT techniques applied at this stage must be rapid and the data obtained should be compatible with computer storage and readout systems.

Damage due to handling and assembly will be detected during this evaluation and the decision for repair or replacement of components will be based, largely, on the NDT information available. In addition, the repair site or replacement component will require NDT inspection and verification.

3. NDT Assessment of Re-usability

As mentioned above, NDT data must be obtained rapidly at this stage because of the turn-around time constraints. As a result, it will probably not be possible to examine all areas of the vehicle and reliance must be placed on identifying critical areas for special attention. These areas may be identified as a result of the design analysis where flight testing or windtunnel tests indicated the areas of highest temperature or stress. In addition, these areas may be identified through the use of implanted or attached sensors which may indicate that an over-stress or over-temperature condition has occurred in local areas. Examples of these sensors may be such things as "thermochromic implants" which change color irreversibly when a particular temperature has been exceeded. A quick visual inspection of the TPS would then permit these areas to be quickly identified and NDT evaluation techniques for measuring the amount of degradation could then be applied to these local areas to determine if sufficient residual life is still left or if the component must be replaced or refurbished. Another example of "condition" sensors could be the use of functional magnetic fasteners (such as bolts) which are composed of an alloy which has a specific "curie point". When the temperature at the structure has been exceeded the magnetic bolt would lose its magnetism. Externally mounted probes could then quickly scan the vehicle and detect those areas where the demagnetized bolts are located thereby delineating areas which have been overheated and thereby weakened. A further example would utilize materials which irreversibly change their electrical resistance when strained beyond a specific value. When these sensors are attached to the internal supporting structures they will indicate that an overstress condition has taken place and further NDT verification of the flight-worthiness is required. These are only a few examples of the types of unique and novel approaches that can be taken to reduce the amount of NDT scanning that will be necessary to assure the reliability of the vehicle for another re-entry.

However, in order to be sure that the TPS materials are behaving in a characteristic fashion it will be necessary to obtain NDT data from specific selected areas after each flight or series of flights. In this case, the data would be compared with previously acquired NDT data, probably by using a computer system, and assessments of reusability would be made by observing the amount of degradation which has occurred. The computer would have pre-programmed information as a result of the extensive environmental testing during the materials development and characterization stages and comparisons of the NDT data accumulated after each re-entry would allow deviations from normal behavior to be detected. The combination of implanted and attached sensors and the evaluation of the vehicle at specific selected locations will permit very rapid assessments to be made. A framework unit with appropriate NDT transducers could then be an integral part of the maintenance and inspection facility and the shuttle vehicle would then be placed under the frame when the inspection is to be made.

WHAT IS THE STATE-OF-THE-ART OF NDT FOR COATED METALLICS?

During the past several years, a program under the direction of the Air Force Materials Laboratory (Ref. 1-4) has been concerned with the development of nondestructive testing techniques for evaluating high temperature coatings on refractory alloys and superalloys. This program is an excellent example of the considerable value that can be obtained when NDT development occurs as a parallel effort with the materials development. Primarily, this program sought to answer the following questions (and did successfully):

- Why do the coatings fail?
- When will the coatings fail?
- Where will the coatings fail?
- Which material variables are associated with failure?
- How can these variables be eliminated during the processing step?

As a further result of this program, coating producers and contractors are currently using NDT techniques for process control, acceptance testing, and for evaluating specimen degradation during environmental testing.

For the sake of brevity, three effective NDT techniques will be described in this paper. These techniques were applied to specimens and structures having pack cementation and slurry coatings on columbium, tantalum and molybdenum alloys.

A. The Thermoelectric Probing Technique

The problem of detecting coating variability on the edges of specimens is a particularly troublesome one. Yet, it is necessary to evaluate edges since the greatest number of failures during static and cycling oxidation occur at edge sites.

A technique was developed in which a heated metallic electrode was brought into contact with the coating surface thereby generating a thermoelectric voltage. The amplitude of this voltage is dependent on the composition and the diffusion thickness of the coating. When it was applied to specimens with the slurry applied coating considerable variations in the thermoelectric response could be seen on the flat surfaces and on the edges. When some of these specimens were sectioned and evaluated metallographically the variations in thermoelectric voltage output could be traced to variations in the coating thicknesses. It was possible to calibrate the test in terms of coating thickness and to make predictions with an accuracy of about ± .0002". Further, it was found that slurry coated specimens, which were oxidation tested at about 2600 degrees F. under static and cyclic conditions, consistently failed at the site where the lowest thickness was predicted. As a result, this variable was identified as the primary one which leads to failure and, therefore, must be controlled. One coating vendor monitored this condition in numerous specimens and made adjustments in the coating procedure which resulted in consistent coating uniformity at edges and on the surfaces with a greatly improved predictability for performance, as well as longer life.

In addition this technique was applied to internal surfaces of complex specimens such as hollow, gas cooled turbine vanes made of columbium alloy and coated with the Sylvania R 512 E coating. By bending the probe tip into hook shape and grinding the tip to a point, it was possible to accurately measure and assure the coating thickness in very restricted areas of the internal leading and trailing edges. This demonstrated that the limitations of part geometry have no noticeable effect on the accuracy of this technique. Examples of the simple instrumentation required to perform this test and a schematic showing the arrangement of parts are given in Figures 2 & 3.

B. The Backscatter X-Ray or Electron Emission Test

Another test which was developed under this program and showed considerable usefulness was based on an unorthodox radiographic technique (Ref. 5.). In this technique a radiograph can be made of the coating alone without the interfering effects from the thicker substrate or a coating on the opposite surface of the specimen. This is achieved by placing a piece of radiographic film in direct contact with the upper surface of the specimen or structure as shown in Fig. 4 a. A vacuum is then drawn on the special cassette, which has a rubber bladder as the top cover, and the film is thereby pressed firmly into contact with the specimen surface (Fig. 4 b). The cassette is then placed under a standard X-ray tube and is exposed under conditions which are determined empirically. As the X-rays proceed through the film they will cause some uniform darkening or fogging of the film and as they impinge on the coating surface and the substrate they generate photo-electrons and secondary X-rays which are then scattered in a backward direction onto the film.

When this technique was applied to coatings, which were prepared by a pack-cementation process using pure powders in the pack, the back-scatter radiograph showed numerous areas of mottled appearance. Subsequent, oxidation testing of specimens at 2600 F. and 3000 F. in a static furnace and plasma arc showed that failure always occurred at one of the mottled areas. This failure had the appearance of a pinhole through the coating when viewed under a 20X microscope after failure had been seen visually. Metallographic sections through some of these mottled sites showed coating structures which were radically different than the surrounding normal coating areas (Fig. 5). Electron beam probe analysis of this structure showed that it was low in chromium and also completely lacked the uniform laves phase of columbium dichromide at the coating/substrate interface. As a result, it was hypothesized that the lower chromium content of these islands in the coating, resulted in reduced oxidation resistance in these local areas. Further, the lack of the laves phase, which serves as a diffusion barrier to the outward migration of columbium, continued to decrease the oxidation resistance at these local areas and thereby triggered the failure of the coating at these sites. Figure 6 shows the typical appearance of a Cr-Ti-Si coated specimen with the numerous islands of faulty composition. A review of the coating process and an analysis of the pack materials permitted the source of this problem to be identified. The chromium particles used in the pack showed varying amounts of oxide films on their surfaces. Since this coating procedure is performed in two steps, where the chromium and titanium are diffused into the columbium substrate in the initial step, it was seen that the oxide layer on the pack particles could mask the substrate causing a vacancy to be formed at this site. Upon subsequent siliciding in the second step of the process, the vacancy would permit the direct diffusion of the silicon and the columbium substrate. As a result, the local area would have, essentially, a $CbSi_2$ composition rather than the more oxidation resistant $CrSi_2$ composition. Again the detection of this variable by NDT techniques permitted the process to be analyzed and the failure-influencing variable to be minimized or eliminated by removal of the defective pack materials and replacement with fresh non-oxidized materials. As a result, subsequent coatings of this composition prepared by pack-cementation showed considerably longer lives in the oxidation test.

C. The Eddy Current Technique

Eddy current test techniques are commonly used for measuring coating thicknesses on metallic substrates, as well as for detecting such defects as cracks and porosity. In this test a high frequency electrical field is introduced into the test specimens by means of a coil transducer and produces eddy currents which are affected by changes in thickness, composition and structure (Figure 7). Since this technique can be used for rapid scanning, it was employed for evaluating the coating thicknesses on the flat portions of panels and specimens. However, this technique is severely affected by changing surface contours and displays edge distortion effects near or at the edges of specimens. As a result, the thermoelectric test was used for evaluating the as-coated edges and complex shapes while the eddy current technique was used for looking at flat panels primarily. In addition, the thermoelectric technique depends on the surface of the coating being an electrical conductor and cannot be used when the surface has been oxidized. This limitation does not hold for the eddy current test.

Therefore, after the specimens had been placed in an oxidation test furnace for varying lengths of time at 2600 F.,they were evaluated by the eddy current technique. Obvious changes in the eddy current response could be seen as the specimens accumulated high temperature exposure time. Specimens were sectioned and evaluated and it was found that the eddy current technique was monitoring the changes in diffusion depth and oxide buidup.(Figure 8).

Other coatings than the one illustrated in the graph in Fig.8 showed the same type of behavior, although the slope of the parabolic curve changed in relation to the speed of coating growth. In one coating system (Cr-Ti-Si), the coating grew parabolically until a certain point was reached, and then abruptly changed to a linear growth curve. This change in the form of the curve has been identified as the "breakaway" and signals impending failure of the coating system due to wearout and the consequent loss of oxidation protection. As a result, this phenomenon may be used in the space shuttle for signalling incipient failure of the coating.

In addition, the eddy current technique has been used by Sylvania in evaluating their slurry applied coating while it is in the "green" stage. From their experience with this technique, they have been able to strip the coating or to apply additional slurry before the final fusing step. As a result, they have been able to produce more consistent and reliable coatings for the refractory alloys.

SUMMARY

When nondestructive testing techniques are developed in a systematic way in conjunction with the materials development, they can produce numerous benefits. For example:

1. NDT can be an effective characterization tool for assuring that the manufacturing process is reproducible.
2. These techniques can be used for identifying and locating variables and defects in coatings that lead to failure.
3. Processes can be monitored and controlled to minimize or eliminate failure-influencing variables, thereby assuring maximum life and reliability.
4. Wearout or degradation of the coatings, which signal incipient failure can be detected, thus permitting specimens or structures to be removed before serious damage has occurred.

WHAT IS STILL NEEDED ?

The program outlined above is only a beginning. There are numerous problems still to be solved, for which NDT can play a significant role. Among the things still required are those listed below:

- N.D.T. Instrumentation-Computer Analysis Systems for making reuse decisions in real-time.

- Automation of present N.D.T. techniques to permit rapid, accurate and sensitive evaluation.

- Development of novel techniques or sensors which can be used to rapidly detect over-temperature, over-stress or other abnormal conditions.

- NDT techniques for detecting substrate degradation (embrittlement, thinning,debonding etc.)

- Criteria for determining reusability and NDT for assuring reliability during reuse.

- Early detection of loss of oxidation resistance (incipient failure).

- Detection of failure of coatings on internal surfaces which are inaccessible.

- Development of new N.D.T. techniques which are specific to new coatings as they are developed.

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THE ROLE OF N.D.T. IN THE SPACE SHUTTLE COATED METALLIC THERMAL PROTECTION SYSTEM

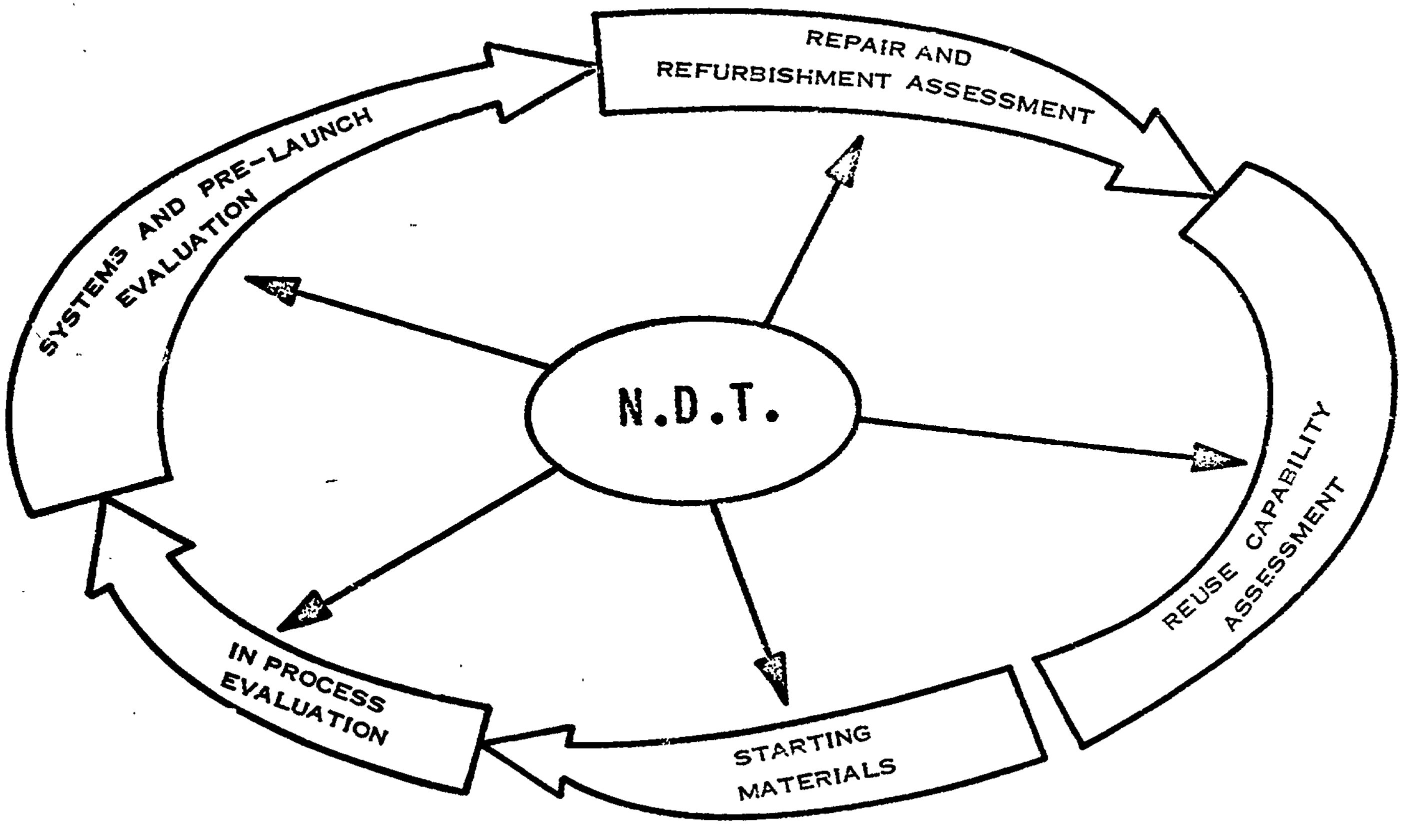

FIGURE 1

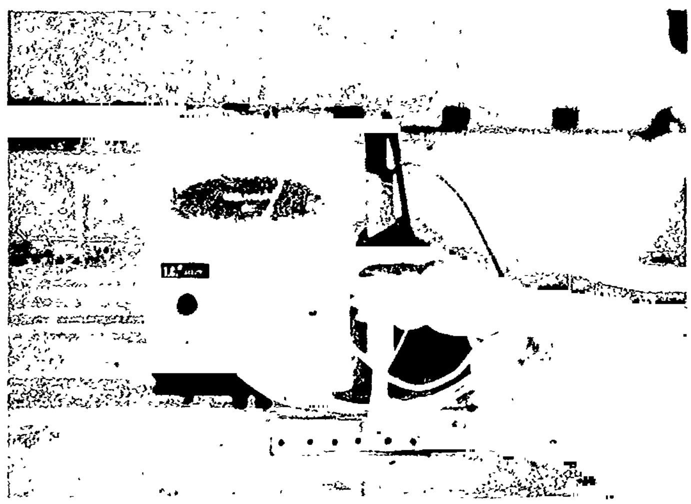

FIGURE 2 - The Thermoelectric Instrument and Probe

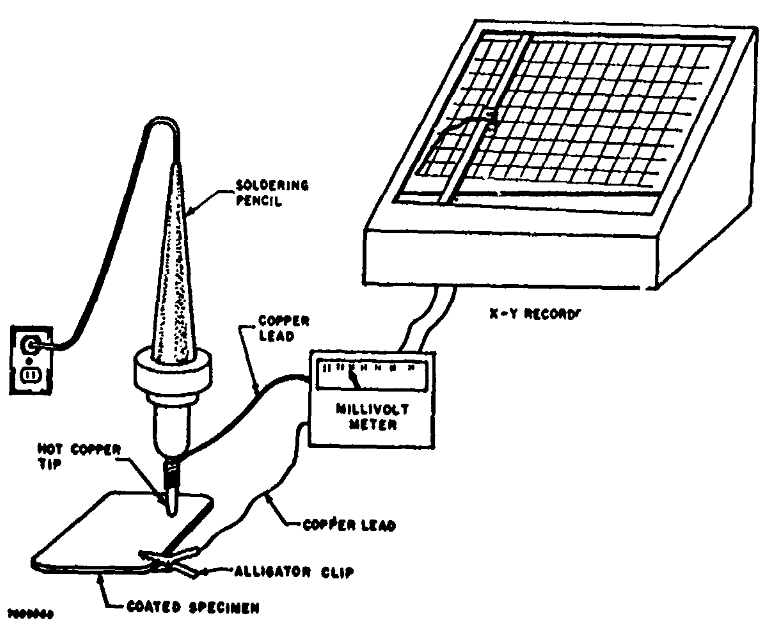

FIGURE 3 SCHEMATIC OF THERMOELECTRIC PROBING METHOD

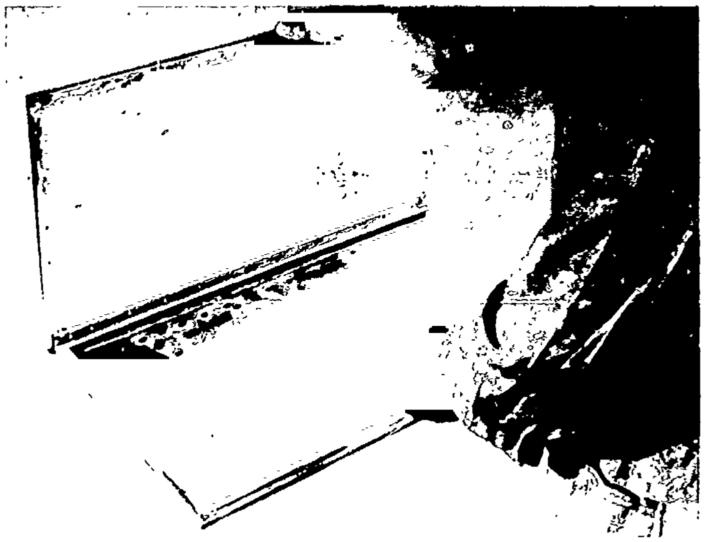

FIGURE 4A - Placement of Film on Coated Columbium Panel
Note that Bare Film is Placed On Top of Panel

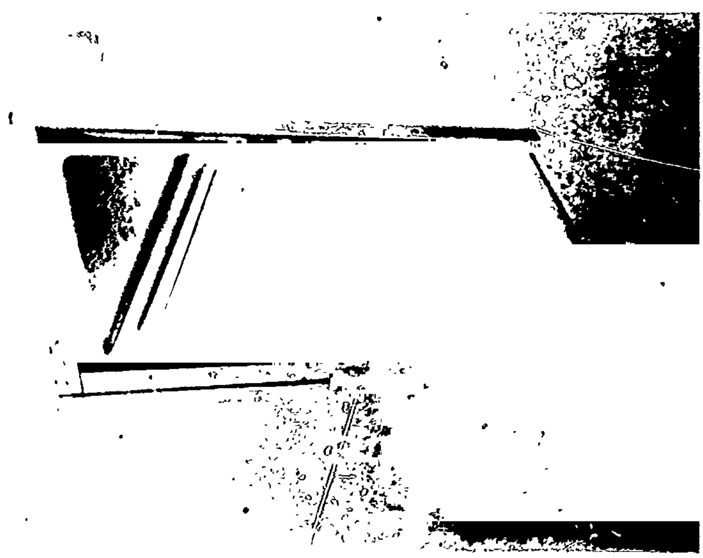

FIGURE 4B - Vacuum Cassette With Enclosed Panel and X-ray Film. Cassette is Ready for Exposure to the X-ray Source.

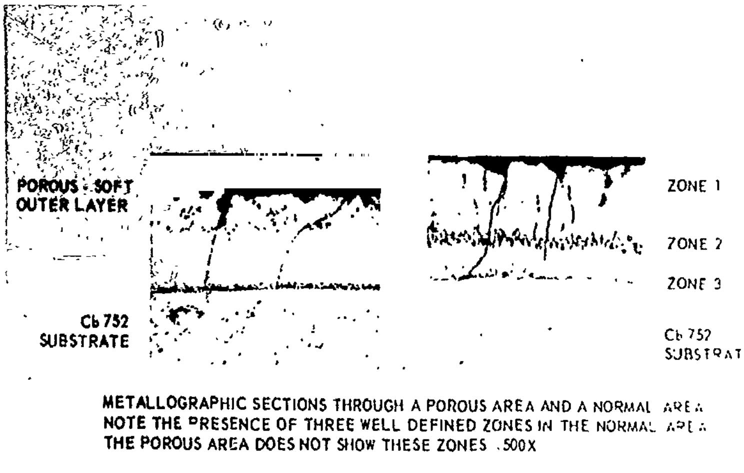

FIGURE 5 - Sections Through Normal and Abnormal Areas Detected By the Backscatter X-ray Technique

FIGURE 6 - X-ray Print Showing Numerous Islands of Chemical and Structural Variation. Dark Spots on Print Show Abnormal Areas that Are Always Associated With Failure in The Pack Cementation Cr-Ti-Si Coating.

FIGURE 7 - Eddy Current Instrument for Measuring Coating Thickness

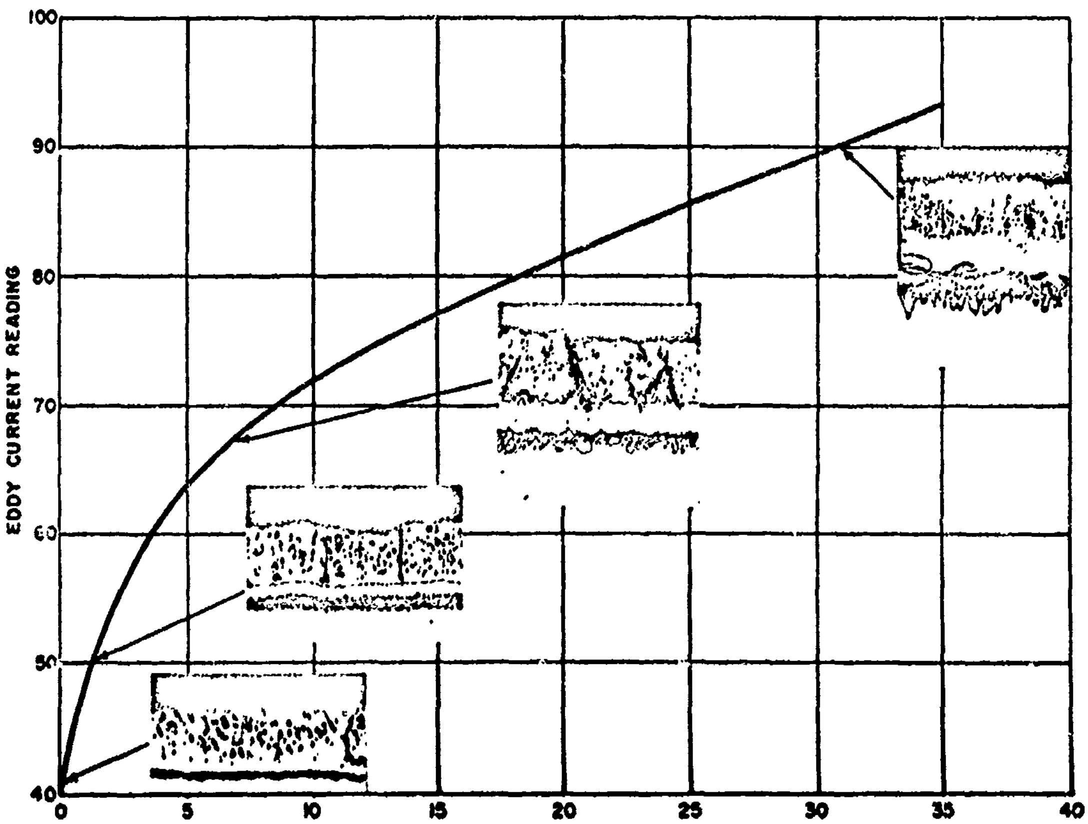

FIGURE 8 EDDY CURRENT AND METALLOGRAPHIC RESULTS VERSUS TIME IN FURNACE AT 2600°F

N70-42998

MATERIALS PROPERTIES AND ALLOWABLES FOR THE SPACE SHUTTLE

L. K. Crockett
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Downey, California

and W. E. Witzell
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MAJOR MATERIAL AREAS

THE ORGANIZATION OF THE DISCUSSION FOLLOWS THIS OUTLINE WITH IDENTIFICATION OF SPECIFIC MATERIALS AND THEIR ASSOCIATED REQUIREMENTS MADE IN EACH OF THE MAJOR AREAS LISTED.

METALLIC TPS & HOT STRUCTURE

NONMETALLIC TPS - REUSEABLE

PRIMARY STRUCTURE & TANK MATERIALS

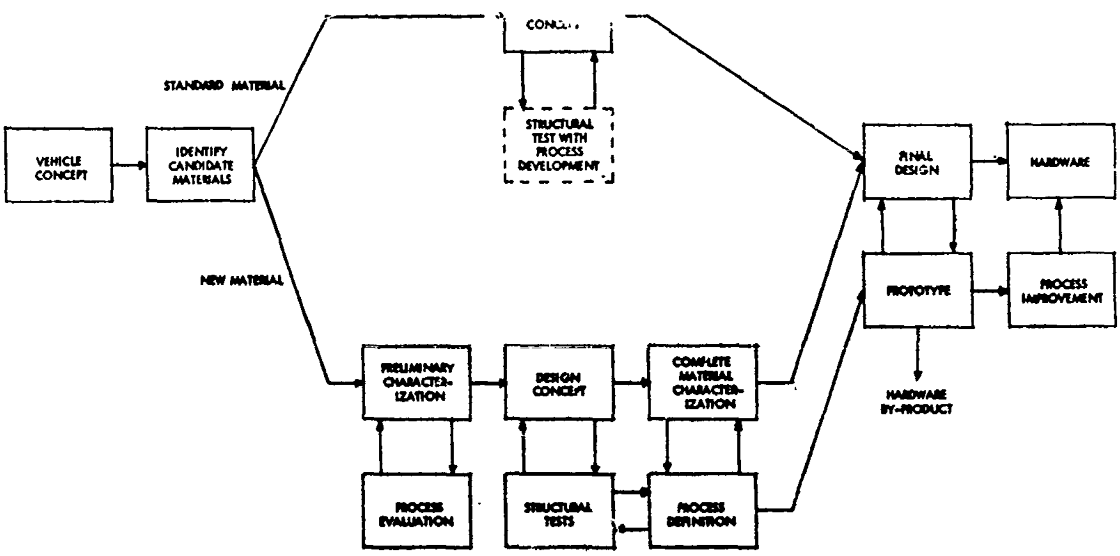
HIGH TEMPERATURE & CRYOGENIC INSULATION

CREW COMPARTMENT MATERIALS

NON-STRUCTURAL MATERIALS

MATERIAL EVOLUTION

THE EVOLUTION OF MATERIAL TECHNOLOGY FOLLOWS THIS FLOW CHART FOR STANDARD MATERIALS SUCH AS ALUMINUM ALLOYS AND THE NEW MATERIALS SUCH AS TD Ni-Cr. THERE IS A GREY AREA WHERE ONLY CERTAIN SPECIFIC CHARACTERISTICS WILL REQUIRE EXTENSIVE DEVELOPMENT.



METALLIC TPS & HOT STRUCTURE

THE SPECIFIC METALLIC MATERIALS THAT ARE PRIMARY CANDIDATES FOR BOTH TPS AND HOT STRUCTURE CONCEPT ARE LISTED IN ORDER OF THEIR MAXIMUM OPERATING TEMPERATURE. THE CHOICE BETWEEN INCO 625 AND HAYNES 188 WILL BE MADE DURING PHASE B.

THE FABRICATION PROCESSES THAT ARE LISTED INCLUDE ONLY THOSE THAT ARE LIKELY TO AFFECT THE PROPERTIES OF THE CANDIDATE MATERIALS.

APPLICATIONS

MATERIAL	FABRICATION PROCESSING
Cb	COATING, WELDING, DIF BONDING, FORMING
TDNiCr INCO 625 HS 188	BRAZING, WELDING, DIF BONDING, FORMING
RENE 41 (1650 AGE)	BRAZING, WELDING, DIF BONDING, FORMING
INCO 718	BRAZING, WELDING, DIF BONDING, FORMING
Ti 6AL-4V (ELI)	— WELDING, DIF BONDING, FORMING

METALLIC TPS & HOT STRUCTURE

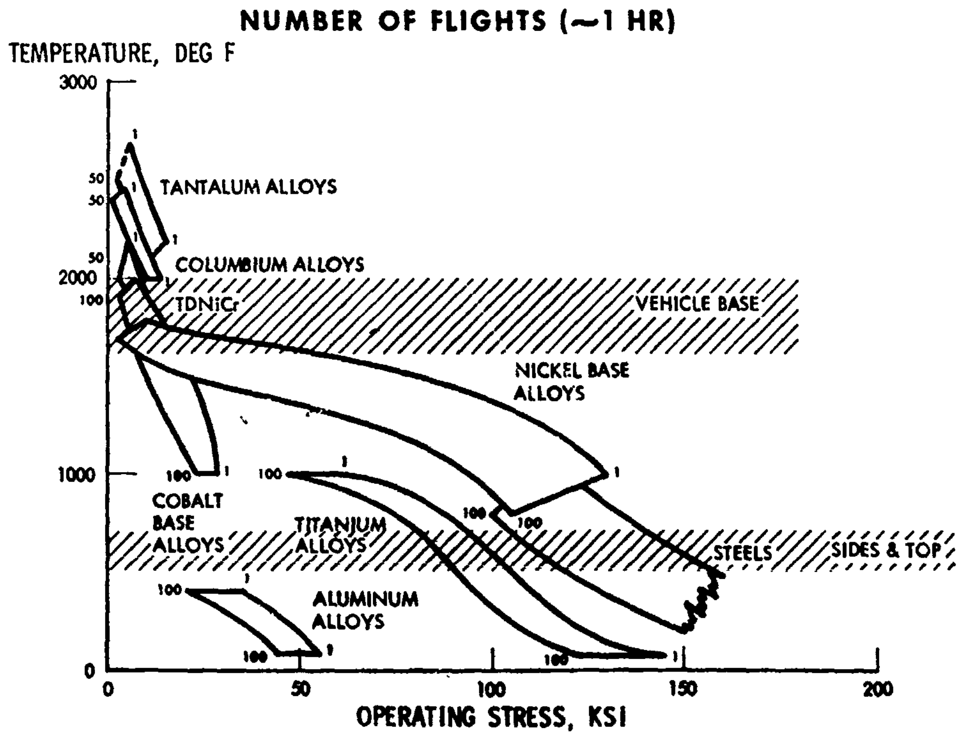
THE REQUIREMENT FOR DATA VARIES SIGNIFICANTLY WITH THE PHASE OF THE OVERALL PROGRAM. AN ATTEMPT IS MADE TO REFLECT THIS IN THE DEFINITION OF THE REQUIREMENTS. PHASE B DATA WILL FORM A PORTION OF THAT REQUIRED FOR THE COMPLETE CHARACTERIZATION REQUIRED DURING PHASE C. HOWEVER, THE PRIMARY EMPHASIS DURING PHASE B WILL BE TO PROVIDE CONFIDENCE IN THE SELECTION OF CANDIDATE MATERIALS AND PROVIDE DATA TO ALLOW A BELIEVABLE MASS FRACTION.

PROPERTIES VS FABRICATION HISTORY, THERMAL HISTORY, (LIFE CYCLE)

PROPERTY	PHASE "B" - PRELIM ALLOWABLE	PHASE "C" - DESIGN ALLOWABLE
MECHANICAL	COATED Cb, INCO 625, RENE 41	ALL
PHYSICAL - INCL OPTICAL	COATED Cb, INCO 625,	ALL
FATIGUE - INCL SONIC	COATED Cb, INCO 625, RENE 41	ALL
CREEP	COATED Cb, INCO 625,	ALL
FRACTURE PARAMETERS		

METALS REUSABILITY

TYPICAL LIFE CYCLE EFFECT ON MECHANICAL PROPERTIES ARE SHOWN TO ILLUSTRATE THIS REQUIREMENT.



NONMETALLIC TPS - REUSEABLE

THE REUSEABLE NONMETALLIC TPS MATERIALS ARE DEVELOPMENTAL AT THIS TIME. HOWEVER, THEY HOLD CONSIDERABLE PROMISE FOR VIABLE APPLICATION. THE CARBON-CARBON COMPOSITE OR REINFORCED PYROLIZED PLASTICS HAVE EXTREME TEMPERATURE CAPABILITY LIMITED PRIMARILY BY OXIDATION PROTECTION.

THE CONCEPT OF EXTERNAL INSULATION HAS MANY DESIRABLE ATTRIBUTES AND MUST BE GIVEN A HIGH PRIORITY FOR DEVELOPMENT.

APPLICATION

MATERIALS	FABRICATION PROCESSING
EXTERNAL INSULATION (COATED)	COMPONENT FABRICATION & ATTACHMENT
CARBON-CARBON COMPOSITES (COATED)	COMPONENT FABRICATION & ATTACHMENT

NONMETALLIC TPS - REUSEABLE

THE PRIMARY EMPHASIS OF TESTING THESE MATERIALS MUST BE TO DEMONSTRATE FEASIBILITY ESPECIALLY THEIR COMPATIBILITY WITH ALL ENVIRONMENT TO BE ENCOUNTERED THROUGH A TOTAL LIFE CYCLE.

PROPERTIES VERSUS LIFE CYCLE

PROPERTY	PHASE "B" - REQUIREMENTS	PHASE "C" REQUIREMENTS
MECHANICAL	PRELIMINARY CHARACTERIZATION	DESIGN ALLOWABLES
PHYSICAL (INCLUDING OPTICAL)	PRELIMINARY CHARACTERIZATION	DESIGN ALLOWABLES
ENVIRONMENTAL	PRELIMINARY CHARACTERIZATION	DESIGN ALLOWABLES
VIBRATION		
MOISTURE		

PRIMARY STRUCTURE & TANK MATERIALS

MIL-HANDBOOK 5 CONTAINS THE BASIC ALLOWABLE REQUIRED FOR MOST OF THESE METALS. HOWEVER, THE INFLUENCE OF LIFE CYCLE AND SOME SPECIFIC PROPERTIES WILL REQUIRE ASSESSMENT.

CONSIDERABLE EMPHASIS MUST BE GIVEN TO THE DEVELOPMENT OF THE MORE EFFICIENT COMPOSITES DURING PHASE B. THIS IS REQUIRED TO MINIMIZE THE RISK ASSOCIATED WITH THE EARLY SELECTION OF THESE MATERIALS.

FRACTURE PARAMETER AND FRACTURE CONTROL PHILOSOPHY REPRESENT MAJOR AREAS OF CONCERN. HOWEVER, THE PRIORITY FOR ACQUISITION OF DATA DOES NOT SEEM TO BE AS HIGH AS IT SHOULD BE.

PROPERTIES VERSUS LIFE CYCLE

PROPERTIES	PHASE "B" PRELIMINARY ALLOWABLE	PHASE "C" DESIGN ALLOWABLE
MECHANICAL	ESSENTIALLY ADEQUATE - SUPPLEMENT AS REQUIRED	ALL
PHYSICAL	ESSENTIALLY ADEQUATE - SUPPLEMENT AS REQUIRED	ALL
FATIGUE	ESSENTIALLY ADEQUATE - SUPPLEMENT AS REQUIRED	ALL
FRACTURE PARAMETER		
K_c	ESSENTIALLY ADEQUATE - SUPPLEMENT AS REQUIRED	ALL
K_{Ic}	ESSENTIALLY ADEQUATE - SUPPLEMENT AS REQUIRED	ALL
K_{Iscc}	ESSENTIALLY ADEQUATE - SUPPLEMENT AS REQUIRED	ALL
dA / dN	ESSENTIALLY ADEQUATE - SUPPLEMENT AS REQUIRED	ALL
FRACTURE CONTROL PHILOSOPHY	DEFINE & EVALUATE	IMPLEMENT

PRIMARY STRUCTURE & TANK MATERIALS

THE DEFINITION OF HOT STRUCTURE AND PRIMARY STRUCTURE INCLUDES A RATHER ARBITRARY TEMPERATURE LIMIT. THEREFORE, Ti-6Al-4V AND INCO 718 APPEAR ON BOTH LISTS.

APPLICATIONS

MATERIAL	FABRICATION PROCESSING
INCO 718	WELDING DIF BONDING FORMING
Ti 6Al-4V	WELDING DIF BONDING, FORMING
2219-T87 7075-T6 & T73	WELDING, FORMING
METAL MATRIX COMPOSITES	FASTENING, BONDING
NONMETAL MATRIX COMPOSITES	FASTENING, BONDING

HIGH TEMPERATURE & CRYOGENIC INSULATION

THE PRIMARY CANDIDATE INSULATION MATERIALS ARE LISTED ESSENTIALLY IN ORDER OF OPERATING TEMPERATURE.

	APPLICATION
MATERIAL	FABRICATION PROCESS
ZIRCAR (ZrO)	INSTALLATION - PACKAGING
DYNAFLEX (SiO_2, Al_2O_3 Cr_2O_3)	INSTALLATION - PACKAGING
TG 15000 (GLASS)	
MIN K (PROPRIETARY)	COMPONENT,
MARINITE (ASBESTOS)	COMPONENT
POLYURETHANE	SPRAY FOAM, COATING, PRIMERS
MULTILAYER H. P. I.	SYSTEM INSTALLATION

HIGH TEMPERATURE & CRYOGENIC INSULATION

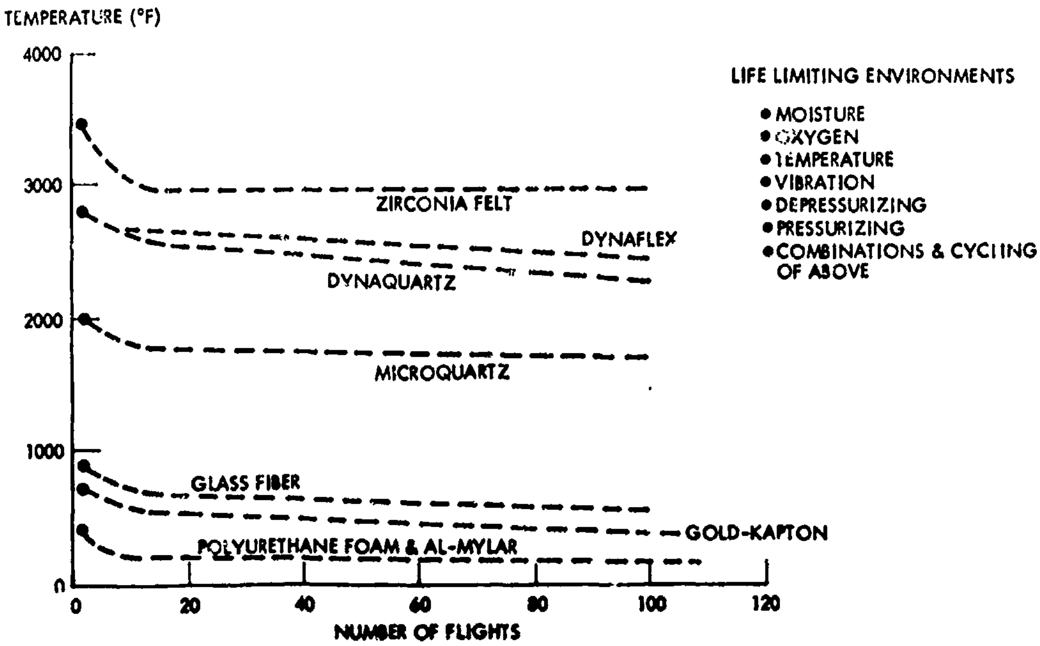
VERY LITTLE DATA RELATED TO THE EFFECTS OF THE LIFE CYCLE ENVIRONMENTS ON ALL OF THE INSULATION MATERIALS IS AVAILABLE. THE CHOICE OF HIGH TEMPERATURE MATERIALS IS SOMEWHAT LIMITED AND REQUIRES EARLY CHARACTERIZATION TO SUPPORT CANDIDATE SELECTION.

PROPERTIES VERSUS LIFE CYCLE (THERMAL, MOISTURE, VACUUM, VIBRATION, ETC.)

PROPERTIES	PHASE B REQUIREMENTS	PHASE C REQUIREMENTS
PHYSICAL	CONFIRMATION OF SELECTION CRITERIA - i.e., OPERATION LIMITS	COMPLETE CHARACTERISTICS
MECHANICAL	CONFIRMATION OF SELECTION CRITERIA - i.e., OPERATION LIMITS	COMPLETE CHARACTERISTICS
COMPATIBILITY	CONFIRMATION OF SELECTION CRITERIA - i.e., OPERATION LIMITS	COMPLETE CHARACTERISTICS

PROJECTED INSULATION REUSE CAPABILITY

VERY LIT[illegible] MISSION LIFE CYCLE DATA IS AVAILABLE. THE MAXIMUM PERMISSABLE OPERATING TEMPERATURE WILL PROBABLY DECREASE WITH INCREASED EXPOSURE. THIS ILLUSTRATION EXTRAPOLATES THIS CHANGE BASED ON JUDGEMENT RATHER THAN DATA.



CREW COMPARTMENT MATERIALS

THE DATA ACCUMULATED TO SUPPORT THE APOLLO PROGRAM IS ADEQUATE FOR PHASE "B". HOWEVER, THE MORE BENIGN ATMOSPHERE OF THE SHUTTLE MAY ALLOW THE USE OF SPECIFIC MATERIALS WHERE THE COSTS OF TESTING ARE JUSTIFIED. THEREFORE, PHASE "C" WILL REQUIRE SOME TESTING. AN EARLY EFFORT WILL BE TO ESTABLISH THE FORMAL CONTROL PROCEDURES.

MATERIALS	PHASE "B" REQUIREMENTS	PHASE "C" REQUIREMENTS
VARIOUS ORGANICS	FLAMMABILITY, TOXICITY	VERIFY DATA
CLOTH	OUTGASSING DATA	FOR LESS SEVERE
ADHESIVES	FROM APOLLO	O_2-N_2 ENVIRONMENT
COOLANTS		
SEALS & GASKETS		
LUBRICANTS		

NONSTRUCTURAL MATERIALS

THE EMPHASIS DURING PHASE "B" SHOULD BE TO IDENTIFY THE REQUIREMENT WITHIN THE VARIOUS SYSTEMS. THE DEMONSTRATION OF 100 MISSION LIFE CYCLE CAPABILITY WILL REPRESENT A MAJOR TEST PROGRAM DURING PHASE "C" FOR MANY OF THESE MATERIALS.

MATERIALS	PHASE B REQUIREMENTS	PHASE C REQUIREMENTS
LUBRICANTS	IDENTIFY REQUIREMENTS	CHARACTERIZE MATERIALS
SEALS & SEALANTS	IDENTIFY REQUIREMENTS	CHARACTERIZE MATERIALS
WINDOWS	IDENTIFY REQUIREMENTS	CHARACTERIZE MATERIALS
ELECTRONICS	IDENTIFY REQUIREMENTS	CHARACTERIZE MATERIALS

CONCLUSIONS

PHASE "C" WILL REQUIRE CONSIDERABLE TESTING TO COMPLETELY CHARACTERIZE EACH MATERIAL TO THE LEVEL OF STATISTICALLY SIGNIFICANT DESIGN ALLOWABLES AND THE EFFECT OF TOTAL LIFE CYCLE.

PHASE "B" TESTING SHOULD PROVIDE CONFIDENCE IN THE PROPER SELECTION OF MATERIAL AND FABRICATION PROCESSES WITH SUFFICIENT CHARACTERIZATION TO GIVE BELIEVABLE MASS FRACTION AND TO PROVIDE A BASIS FOR A REALISTIC APPRAISAL OF PHASE "C" EFFORTS.

BEARING, LUBRICANTS, AND SEALS FOR THE SPACE SHUTTLE

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INTRODUCTION

The unique mission requirements of the space shuttle will expose some of the bearings, lubricants, and seals to extreme conditions of temperature and pressure. In this presentation we will consider three major problem areas: (1) lubrication and design of airframe control surface bearings, (2) thermally-stable fluids for hydraulics and for jet engine lubrication, and (3) shaft seals for liquid hydrogen pumps.

DISCUSSION

An estimate of some of the conditions to which the control surface bearings on the orbiter will be exposed during a shuttle mission is given in figure 1. There will undoubtedly be a prelaunch checkout requirement for the aerodynamic control bearings under ground atmospheric conditions and at light bearing loads. During ascent, the control surfaces and their associated bearings are expected to reach about 1400-1500° F from aerodynamic heating. The control surface bearings may not be required to actually function during ascent; control may be handled by rocket engine gimbals and reaction control jets. The bearings on the orbiter will then be exposed to a long soak in vacuum during orbit at temperatures possibly as low as -100° F. Upon re-entry, the bearings will heat up to about 1600° F.

The control surface bearings will become functional at some time during re-entry. They will then be required to provide oscillatory motion of the control surfaces under high loads during the balance of the re-entry and later during jet-powered subsonic flight.

Figure 2: This is an estimate of the airframe skin temperature in the vicinity of the control surface bearings during re-entry. For one concept of the orbiter vehicle, the high temperatures (1400-1600° F) persist for about one-half hour. The bearings will not heat as rapidly as the airframe skin but they will retain their heat longer and may have to operate at temperatures above 1000° F into the jet-powered portion of the flight. In some less optimistic analyses, temperatures as high as 1800° or 1900° F are anticipated in the areas of the control surface bearings.

The temperature extremes to which the airframe bearings will be exposed eliminates the possibility of using oil or grease lubricants except perhaps during ground checkout. Therefore, solid lubricants are being considered. In our previous research on solid lubricants, it was demonstrated that coatings and composites utilizing CaF_2 as the lubricant are promising high temperature solid lubricants. For example, in simple slider experiments ceramic-bonded CaF_2 lubricated Rene 41 to 1900° F. Ball bearings have been successfully lubricated with CaF_2-BaF_2 coatings at temperatures up to 1500° F. Figure 3 is a CaF_2-lubricated superalloy bearing running at 1500° F and 5000 rpm. This bearing is shown to indicate the high-temperature capability of the lubricant and the bearing materials. It is in unidirectional rotation and is carefully aligned with the shaft. In comparison, airframe bearings undergo oscillating motion and must be capable of accepting severe shaft misalignment while the bearing is operating. (Airframe deflections and thermal gradients can easily cause dynamic misalignments of $\pm 5°$.)

Boeing in the DYNASOAR re-entry glider program and a following program have evaluated CaF_2 coatings for control surface hinge bearings. Boeing found that ceramic-bonded CaF_2 and fused CaF_2-BaF_2 coatings were effective at 1000 and 1500° F, but very poor at room temperature. Martin in its PRIME lifting body program found that oil could be successfully used as a co-lubricant for the low temperature ground checkout condition. Both investigators found that a special gold plate developed at Boeing and designated X.88 was an effective lubricant for plain spherical bearings up to 1000° F.

Areas of investigation which should be pursued include: (1) bearing design, (2) improvement of low temperature properties, and (3) formulation of coatings and fillers for composites that not only lubricate, but also provide oxidation protection to the substrate or porous metal.

Three types of self-aligning airframe control surface hinge bearings are shown in figure 4. The ball bearing aligns on the outer raceway which is ground as a segment of a circle with its center at the bearing center. The concave roller bearing can carry both a radial and a thrust load and alignment occurs at the roller-inner raceway contact. The inner raceway is a spherical segment. In both of the rolling contact designs, the solid lubricant will either be applied as a bonded coating on the cage surfaces indicated or the cages will be made of a self-lubricating composite material. The remaining bearing is a sliding contact, plain spherical type. Oscillating motion occurs on the thrust washer and on the bore of the bearing. The spherical segment is for self-alignment only. The lubricant coating is applied to the shaft, the flat thrust surface, and the spherical surface of the bearing.

In simple slider experiments, improvement in low temperature lubricating properties have been achieved by adding finely dispersed metallic silver to the fluoride coating compositions. The effectiveness of this addition on the lubrication of airframe bearings will be studied.

Figure 5: Self-lubricating sintered porous metals infiltrated with fluoride lubricants are effective up to 1500° F but the life is limited by oxidation

of the sintered metal in the composite. We have begun work on modifying CaF_2 infiltrants to provide oxidation protection to the metal. The slide shows the weight increase with time in air at 1500° F caused by oxidation of the sintered metal in the composite. In 50 hours, fluoride-filled Hastelloy X has oxidized more than unfilled Hastelloy X. However, the use of a CoO-based ceramic instead of the fluoride reduces oxidation drastically. Studies are now underway to modify the fluoride infiltrants by the addition of oxide ceramics.

Thermally-Stable Fluids for Hydraulics and Jet Engine Lubricants

Some heat transfer studies by aircraft companies indicate that heat soak during and just after re-entry may increase hydraulic and lubricating fluid temperatures to as high as 800° F. These fluids should, therefore, have good thermal stability, good oxidation resistance and low vapor pressures in order to minimize sealing and thermal protection problems for the hydraulic and lubricating systems. Figure 6 gives some pertinent properties of some candidate fluids.

The polyphenyl ethers and the so-called C-ethers appear to have the necessary high temperature properties. Polyphenyl ethers have the disadvantage of high pour point (+40° F) and certain deficiencies as boundary or thin film lubricants. The C-ethers, although they have a slightly lower thermal decomposition temperature, do have the advantage of a low pour point (-20° F), and they have good susceptibility to additives for improving wettability on metals and for improving boundary lubricating ability. We plan to conduct both in-house and contract programs with C-ethers and polyphenyl ethers. The contract program will be a program (1) to formulate additives for the C-ether and polyphenyl ethers to improve their lubricating characteristics and other properties, (2) to determine suitability of the lubricants for ball bearing lubrication, and (3) to evaluate the fluids in a hydraulic system. The in-house program will be to investigate the fundamental lubricating characteristics of the various candidate fluids at the temperatures of interest.

Seals for Propellant Pumps

The last item I would like to discuss relates to the other temperature extreme. It is concerned with rotating seals for liquid hydrogen pumps for the rocket engines. Current thinking is that to get the high flow rates needed in a compact package, the pumping pressure may be anywhere up to 10 times the pressures found in current rocket propellant pumps. Therefore, seals are needed that will seal a high Δp and must be reusable for many flights.

Figure 7 compares the leakage rate for three types of shaft seals: labyrinth, sliding contact, and self-acting lift pad type of narrow gap seal designed here at Lewis. This seal provided lower leakage rates than either the labyrinth or the sliding contact seal. Also, because there is no solid contact, little or no seal wear occurs. Low seal wear rates are especially important to obtain the long seal life needed in a reusable vehicle. A schematic of the self-acting lift pad seal is shown on figure 8. This type of seal is characterized by a high gas film stiffness. This means that the lift force drops off rapidly if the seal tends to open and the closing force increases rapidly. If the seal tends to close, the opposite effect occurs. A small operating film thickness is thereby maintained and the result is that the high gas film stiffness forces the seal nosepiece to dynamically track any small runout motion of the seal face.

SUMMARY

Three major areas of research which have promising potential for space shuttle applications have been discussed. They are:

1. High temperature airframe bearings lubricated with solid lubricant film and composites. Modifications to provide good low temperature lubrication and to reduce metal oxidation rate are needed.

2. Formulation and evaluation of high temperature modified polyphenyl ethers for lubricant and hydraulic fluids, and

3. Self-acting lift pad shaft seals for hydrogen pumps. They give low leakage and little or no seal face wear.

ESTIMATED MISSION REQUIREMENTS FOR ORBITOR'S CONTROL SURFAC. BEARINGS

PHASE OF MISSION	CONDITIONS	BEARING FUNCTIONS
PRELAUNCH	80^0 F 1 ATM AIR LIGHT BEARING LOADS	OSCILLATION GROUND CHECK OF CONTROL SURFACE OPERATION
ASCENT	80^0 TO 1600^0 F	STATIC LOAD SUPPORT
ORBIT	-100^0 F VAC MINIMAL BEARING LOADS	OSCILLATION CHECK OF CONTROL SURFACES BEFORE REENTRY
REENTRY	-100^0 TO 1600^0 F VAC TO HIGH DYNAM PRESSURES PEAK BEARING LOADS	OSCILLATION ATTITUDE CONTROL, MANEUVERS, & ACCOMMODATE HINGE PIN MISALINEMENTS
JET-POWERED FLIGHT	HEAT SOAK FROM REENTRY ATMOSPHERIC PRESSURE NORMAL FLIGHT LOADS	OSCILLATION ATTITUDE CONTROL, MANEUVERS

FIGURE 1 CS-54833

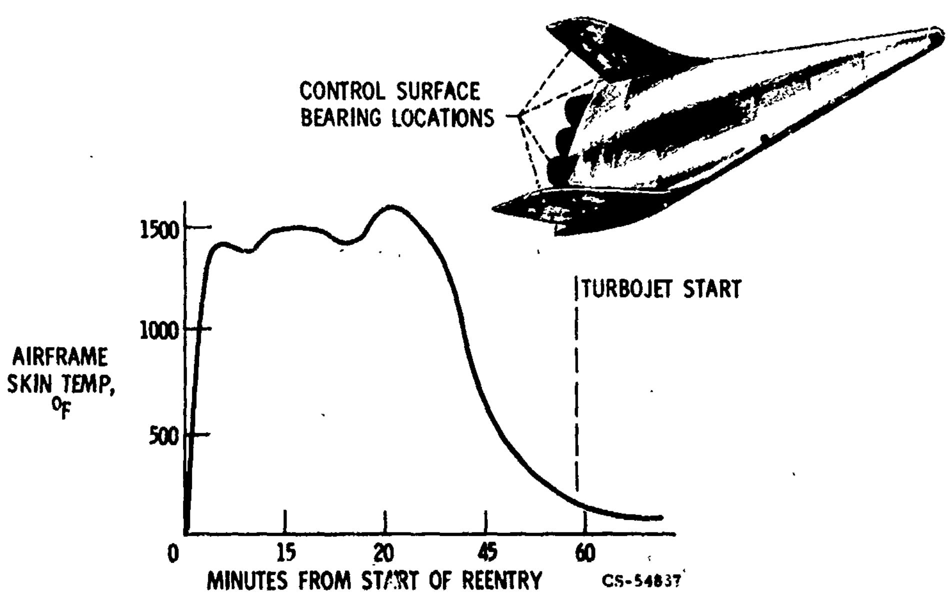

FIGURE 2

HIGH TEMPERATURE BEARING TEST

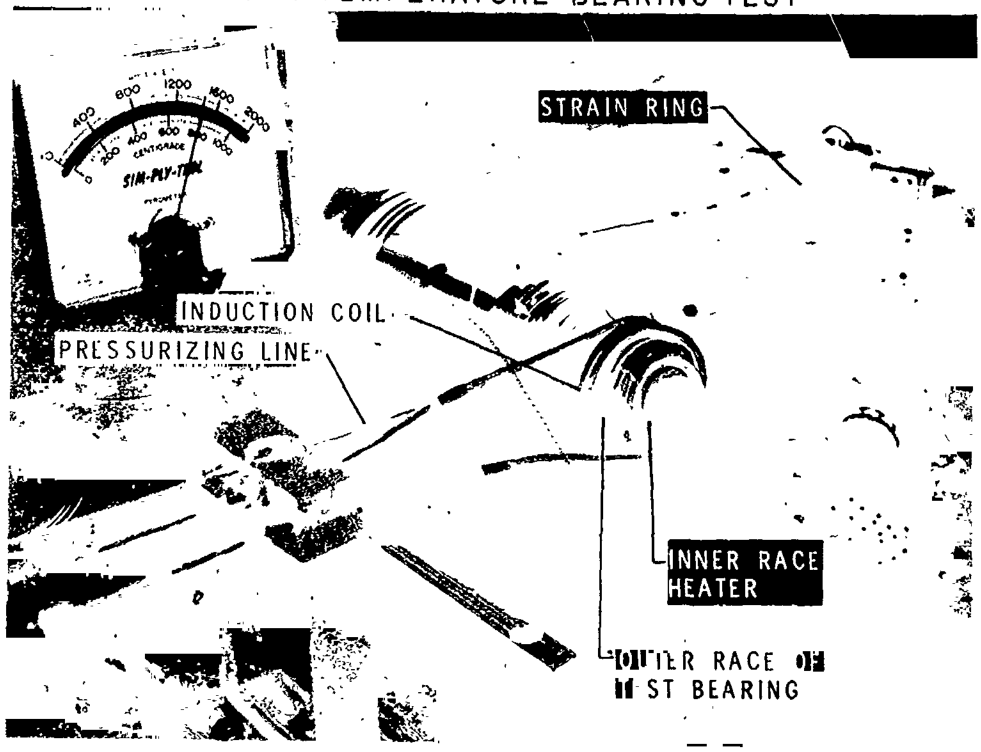

FIGURE 3

TYPES OF SELF-ALIGNING AIRFRAME BEARING

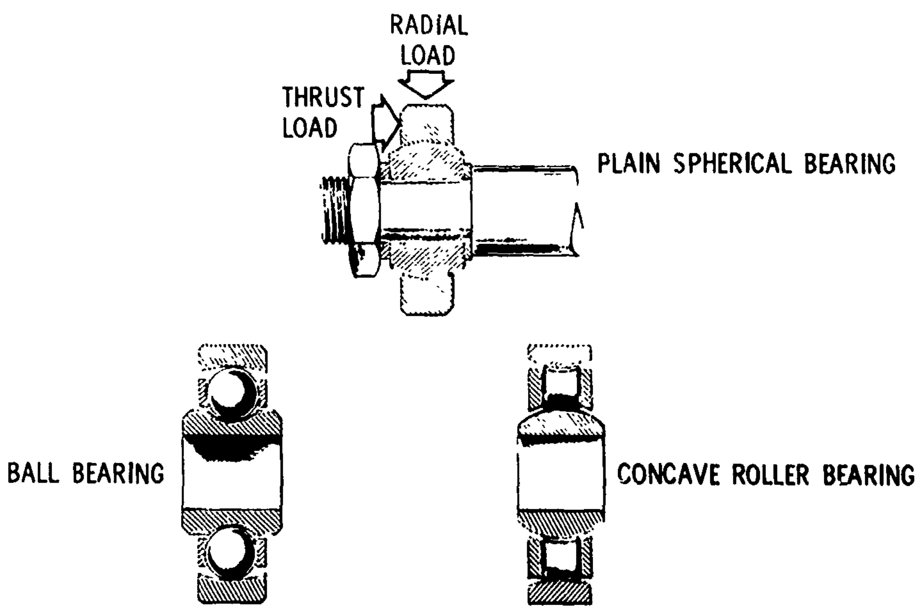

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FIGURE 4

EFFECT OF LUBRICATING FILLERS ON OXIDATION OF SINTERED HASTELLOY X IN AIR AT 1500° F

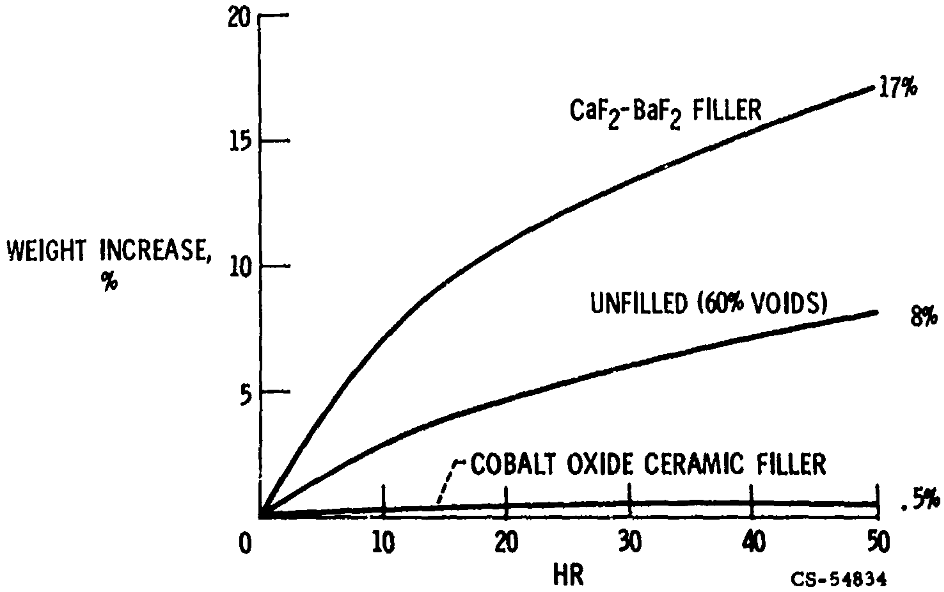

FIGURE 5

COMPARISON OF SOME HIGH-TEMPERATURE FLUIDS

FLUID \ PROPERTY	OXIDATIVE STABILITY LIMIT, °F	THERMAL STABILITY LIMIT, °F	SPONTANEOUS IGNITION TEMP, °F	POUR POINT, °F	COMPRESSIBILITY
SUPER-REFINED MINERAL OIL	350	600	700	-30 TO -70	MEDIUM
ESTERS TYPE II	450	600	800	-30 TO -70	LOW
SILICONES	500	600	800	-60 TO -100	HIGH
C-ETHERS	550-600	700	950	[illegible]	VERY LOW
POLYPHENYL ETHERS	650	850	1100	[illegible]	VERY LOW

FIGURE 6

CS-54835

SEAL LEAKAGE

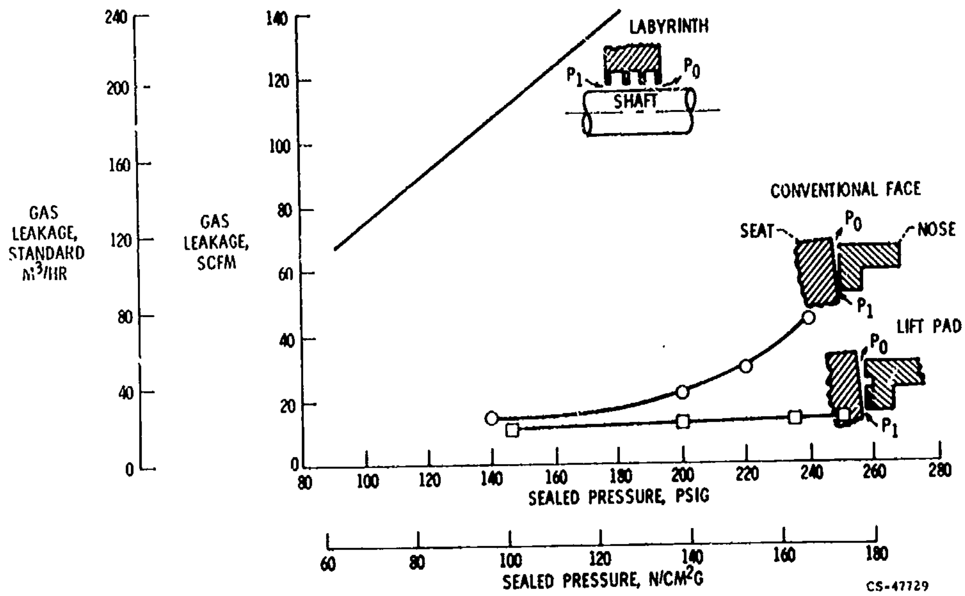

FIGURE 7

FACE SEAL WITH SELF-ACTING LIFT PADS

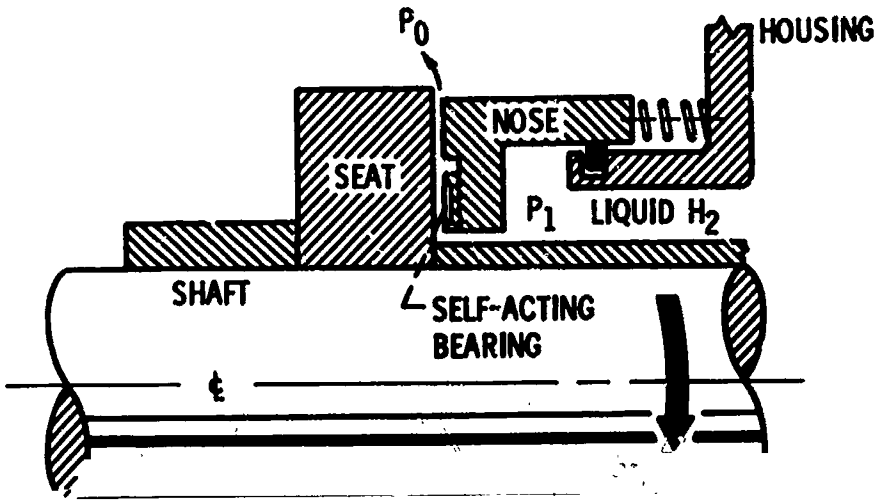

FIGURE 8

N70-43000

CRYOGENIC INSULATION

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STATUS

Before considering Shuttle requirements, it seems appropriate to differentiate between two types of cryogenic insulation. When containment of the cryogen for only a few hours is required, and when conduction and convection mechanisms prevail, it is permissible to simply interpose a substance of low thermal conductivity in the heat transfer path. In the context of this paper, these are short-time insulations, and the insulation material can be applied to either the interior or exterior of liquid hydrogen tankage. However, since all known practical short-time insulations contain potentially hazardous organic materials, their use must be restricted to the exterior of liquid oxygen (Lox) tankage.

For the prolonged orbital storage of cryogens on the order of several days, e required tankage design will necessarily minimize all conductive contributions, and the multilayer reflective insulations (Figure 1) become the logical choice. We have only limited single flight experience with these insulations. To protect this type of insulation from the degrading effects of humidity and to provide reasonable thermal performance while the tankage is on the ground, multilayer insulation is usually purged with helium. The insulation evacuates during ascent to attain the low pressure radiation heat transfer environment where it is most effective.

Figure 2 shows that the critical long-time storage requirement for this vehicle is the Orbiter tankage, which must store propellants for de-orbit and landing maneuver. Some type of multilayer high performance insulation will be needed for these long-term Orbiter requirements, but it would appear that the Booster and Orbiter short-time performance requirements can be met by appropriately modified existing insulations. Figure 3 shows several possibilities that exist in selecting and applying an insulation for the Booster and Orbiter short-time requirements. We have assumed that insulation of all Lox tankage will be required; however, this may not be the case with Booster tankage. The list is restricted to insulations that have either flown or have been the subject of heavy developmental work. If required, we would not exclude any insulation from application to the exterior of oxidizer tankage, but reactivity considerations dictate that it be protected from direct contact with liquid oxygen spillage or vented gaseous oxygen. A requirement for elevated temperature resistance may exist for both short-time and multilayer insulation for reasons that will be clarified later.

With this background, it is instructive to consider specific examples of insulations now in service or under development with an eye toward their potential Shuttle application. We will first emphasize concepts rather than materials.

Two examples of internal liquid hydrogen insulation are shown in Figure 4a. This foam insulation has flown on all S-IV and S-IVB flights. It is biaxially reinforced with glass filaments and preformed lap-edged tiles are bonded adhesively to the tank interior. The inner surface, which actually contacts the cryogen, is a wet layup of fiberglass - polyurethane resin. The other internal insulation is an example of the gas barrier type where the liquid cryogen is physically and thermally cushioned on a layer of its own vapor which is anchored by capillary forces within the cells. This gas barrier approach is still a developmental concept.

Two external insulations have flown successfully on the Saturn S-II stage. The first version (Figure 4b) consisted of a phenolic honeycomb partially filled with foam down to within approximately 1/8 - 1/4 inch of the tank wall. Saw cuts were made through the honeycomb cell walls next to the tank to permit maintenance of a helium purge over the tank's outer surface. The outside surface of the insulation is closed by a nylon-phenolic laminate with a Tedlar outer film. The finished insulation was adhesively bonded to the tank exterior. The second and current S-II insulation has significant advantages over the first from the standpoint of weight, thermal performance, manufacturing ease, and installation. It is a machine-deposited layer of polyurethane foam applied directly to the tank exterior and coated with modified polyurethane resin under a white vinyl coat. In areas where localized aerodynamic heating could create serious erosion, a layer of cork is adhesively bonded to the foam exterior surface.

Another external insulation which has been carried to an advanced state of development, is the so-called dual seal insulation, shown in Figure 5. It is a double layered honeycomb sandwich with a perforated outer honeycomb layer which accomodates a purge. The inner Mylar honeycomb layer approaches a multiplicity of individual hermetically sealed volumes that cryopump to a low-conducting gas pressure regime when liquid hydrogen is tanked.

Again, for Orbiter long-time applications, any short-time insulation could in principle be employed beneath a suitable multilayer insulation, so there are several possibilities.

At this point, we can weigh some of the advantages and disadvantages of these and various other insulation approaches against Shuttle requirements. The first choice that may arise is the problem of internal versus external short-time insulation for the Booster and Orbiter ascent hydrogen tankage. On the basis of our experience with the insulation discussed earlier, Figures 6 and 7 summarize the advantages and disadvantages of these approaches. If the argument is restricted to insulations we have actually flown, the external spray foam insulation is the lightest, but this would not necessarily be a general advantage of external insulation. The internal gas barrier type can potentially approach this weight advantage. Average tank wall temperatures in service would be lower in the case of external insulations and this would theoretically permit the use of higher allowables for a lighter tank design. It perhaps bears repetition that any insulation required for oxidizer tankage will have to be applied externally for safety reasons. Finally, internal insulations of S-IVB type are subject to a progressive and adverse change in thermal performance as the cryogen permeates the foam.

Turning to internal insulations, as a class they would be less vulnerable to environmental or accidental damage and any damage that is incurred would likely be more localized and without the potentially serious consequences of external insulation damage. It is hard to say which approach might be easiest to inspect and refurbish. If an external insulation is more likely to require separate reentry protection, then an internal insulation might be easiest to inspect and repair. A summary of the performance of these systems is contained in Figure 8.

With the Orbiter long-term insulation, the multilayer reflective insulation is the only conceptual possibility, but there are recognized problems in using it on any tankage, and there are some additional problems peculiar to the Shuttle.

Deficiencies

Unless insulations can be found that will withstand the full extremes of this mission, from cryogenic tanking to reentry conditions, all external cryogenic insulation used on the Shuttle will probably underlie some type of high temperature insulation. How much of this high temperature protection to provide, and how to integrate it structurally and thermally with the cryogenic insulation are questions which must be answered. This requires consideration of the temperature limits of the materials commonly used to fabricate insulation and tankage.

A maximum use temperature of 300°F seems a logical base point for an aluminum tankage alloy. Therefore, as a first attempt at optimization, a cryogenically adequate insulation capable of withstanding this temperature would require a minimum thickness and weight of associated high temperature insulation.

Unfortunately, the ability of most materials used in present day cryogenic insulations to survive repeated cyclic exposures to this temperature is unproven if not doubtful, particularly when these exposures are combined with other aspects of the Shuttle environment.

The maximum temperature to which the S-II/S-IVB type urethane foams can be exposed on a static or sustained cyclic basis will be in the range of 200 - 225°F. If we take the S-II external foam insulation approach, the need for other plastic foams with improved thermal stability is indicated. A number of possibilities are tabulated in Figure 9. To date only one isocyanurate foam has been examined, and it did not significantly exceed the temperature capability of the existing urethane foams. If a foam with improved high temperature capability is found, we may then be compelled to look at some other adhesives for both thermal and bond strength reasons. The adhesive presently used in the S-IVB internal insulation has temperature limitations of its own and may not bond satisfactorily to a foam that differs chemically and physically from the existing urethane, If an external foam insulation is considered, it seems unlikely that any nonurethane foam system can be adapted to exterior spray application within the Shuttle time frame. Urethane foams have been with us for some time, but the spray application of urethane to S-II scale fuel tankage required the solution of a number of problems.

At this time, the non-foam containing short-time insulations look less foreboding from this high temperature standpoint. High temperature honeycomb materials are now available which merit and are receiving consideration for dual-seal type external insulations and the gas barrier internal variety. The gas barrier approach is being considered for a liquid methane-fueled version of the supersonic transport, where temperatures approaching 700°F may be transiently encountered and a related program is underway within the same contractor organization that is oriented specifically toward the Shuttle requirements.

In the case of the long-time Orbiter insulation, this same high temperature problem will be compounded by other environmental considerations. The two key elements of the multilayer insulation are the reflector and spacer layers. Various metallized plastic films (usually a fraction of a mil in thickness) are under consideration as reflector elements and the intervening spacer layers will probably be some open netting or fabric. Installing insulation of this type on tankage of the scale we are encountering is a substantial problem, but once this insulation is on the tank, we have the same questions regarding its durability under repetitively cyclic temperature extremes, pressures, shock, and vibration. It seems unlikely that a controlled humidity can be guaranteed for this insulation on the Shuttle and humidity - temperature excursions that can lead to interlayer moisture condensation and prolonged retention are very destructive to the aluminizing treatments normally applied to polyester films. Other film substrates may be needed on the Shuttle for thermal reasons and it will also be necessary to investigate other metallizing treatments, (Figure 10). Goldized polyimide film, is obtainable and a special germanium vaporcoating has been applied to aluminized polyimide film to minimize this corrosion problem. Finally, just because these multilayer insulation elements have no load carrying capability does not guarantee that they won't be stressed. There is a substantial disparity between their thermal expansion behavior and that of the candidate tankage materials. In addition, vibration, evacuation, and repressurization all induce stresses on the insulation. The relatively poor reproducibility of multilayer insulation is a disadvantage that must be bought with their outstanding insulation performance. Determining the extent of this variation and the magnitude of any service induced performance deterioration will be necessary and this may well require the parallel development of a new test methodology. The flat calorimeter configurations best adapted to cyclic environmental exposures and subsequent thermal testing of insulation assemblies are the poorest in our present state of knowledge for yielding reproducible and realistic thermal performance data.

EVALUATION OF CURRENT MATERIALS

The suitability of existing foams has been studied by tensile measurements, high temperature aging and thermogravimetric (TGA) testing. The latter technique involves measuring the weight loss of the sample in an environment where the temperature is maintained constant or alternatively, is increased at a programmed rate. Isothermally aged glass filament reinforced S-IVB foam insulation samples exposed for six days to temperatures of 210, 250, and 300°F showed weight losses of 0.7%, 1.1%, and 6.6% respectively. Between 250° and 300°F, the foam became discolored, expanded slightly, and showed evidence of separation from the reinforcing glass fiber network.

The flatwise tensile test results, shown in Figure 11, verify the rapid decline in foam strength that would be expected over the temperature range in question. Although the failures visually appeared to occur in the foam, the nature of the adhesive may have an effect on the actual strength.

Similar tests were conducted by applying heat to the metal side of large insulated panels which simulated the current S-IVB insulated tank. The metal surface was brought to the maximum test temperature in twenty to thirty minutes and held at the test temperature for an additional thirty minutes. Single tests at maximum temperatures between 200 and 250°F caused no observable failures but repetitive testing is incomplete. Dissection of samples heated to 300°F revealed that cracks had developed in the foam adjacent to the bondline and this effect was noticeably more pronounced in samples that had been heated to 350°F. Overall, these results support our earlier conviction that this system will be limited to a maximum use temperature in the 200 - 225°F range, and this limitation now seems to be inherent in the urethane foams rather than the adhesives.

Figure 12 demonstrates that we have several adhesives with reasonably good strengths up to 300°F, again, cyclic exposure data is incomplete. In common with a number of people, we have been studying high temperature strength-enhancing additives for adhesive systems. Additives are known which markedl' nprove the high temperature and age resistance of our standard u. .thane cryogenic adhesive and the beneficial effect of using this additive is also shown in Figure 12.

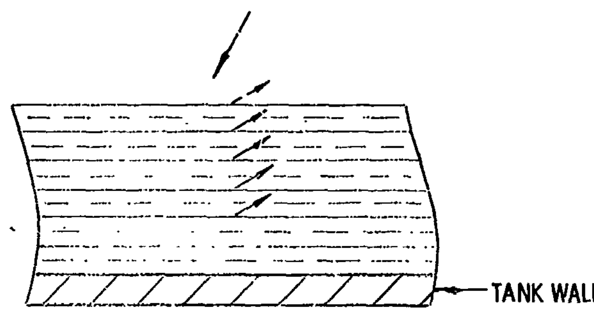
So, the most high temperature vulnerable element in urethane foam insulation is the foam itself. Several other foams, including the polyisocyanurate materials mentioned earlier, are under study and silicone foams also merit immediate consideration. These and other candidate foam systems are being screened by TGA, high temperature aging and mechanical tests. For externally applied foams, a suitable corrosion resistant primer may be needed for application to the tank exterior surface before bonding. Consequently, the parallel evaluation of adhesives, primers, and candidate exterior foams is indicated.

Evaluation of potential multilayer insulation materials must encompass at least the activities summarized in Figure 13. We have had representative state-of-the-art multilayer insulation test panels exposed to weathering tests at Kennedy Space Center for over a year. These unpurged assemblies are stored in an open shed and although they are protected from direct precipitation, we have noted adverse changes in the optical properties of some of the aluminized reflector materials. There is also evidence that some spacer materials contain agents that accelerate this corrosive attack. This study is being broadened to include materials of greater interest to the Shuttle. Some of the metallized films that may prove less subject to humidity-induced optical changes may not be as durable physically, so methods of evaluating metal adhesion and wear resistance will be investigated. Representative Spacer materials that show greater promise than the currently used nylon and dacron include netting of beta glass, Nomex, polybenzimidazole and Kynol, a flame resistant phenolic fiber.

We are also enlisting aid through several contracts aimed at developing or improving insulation suitable for Shuttle service (Figure 14). The first program is basically an effort to upgrade the type of internal foam insulation now flying on the Saturn S-IVB Stage. An internal gas layer approach is being pursued in another program by Martin-Marietta Corporation, and McDonnell Douglas Astronautics Company is involved in a third program oriented toward high performance insulation to meet the long-time Shuttle storage requirement.

FIGURE 1

MULTILAYER REFLECTIVE INSULATION



———— REFLECTIVE FOIL OR METALLIZED PLASTIC

– – – – INERT SPACER MATERIAL

INCIDENT RADIATION IS REFLECTED BY MULTIPLE FOIL OR METALLIZED PLASTIC LAYERS

REFLECTOR LAYERS SEPARATED BY INERT LOW CONDUCTIVITY SPACER SHEETS

FIGURE 2

SHUTTLE CRYOGENIC INSULATION REQUIREMENTS

STAGE	MANEUVER	REQUIRED CRYOGEN STORAGE DURATION	COMMENTS
BOOSTER	ASCENT	SEVERAL MINUTES	SHORT TERM LOW PERFORMANCE INSULATION SATISFACTORY
	LANDING	1 HOUR	
ORBITER	ASCENT	SEVERAL MINUTES	"
	APS & RELATED REQUIREMENTS	UP TO SEVEN DAYS	BOILOFF FROM DE-ORBIT/PROPELLANT TANKS
	DE-ORBIT	SEVEN DAYS	MULTILAYER HIGH-PERFORMANCE INSULATION NEEDED
	LANDING	SEVEN DAYS	"

FIGURE 3

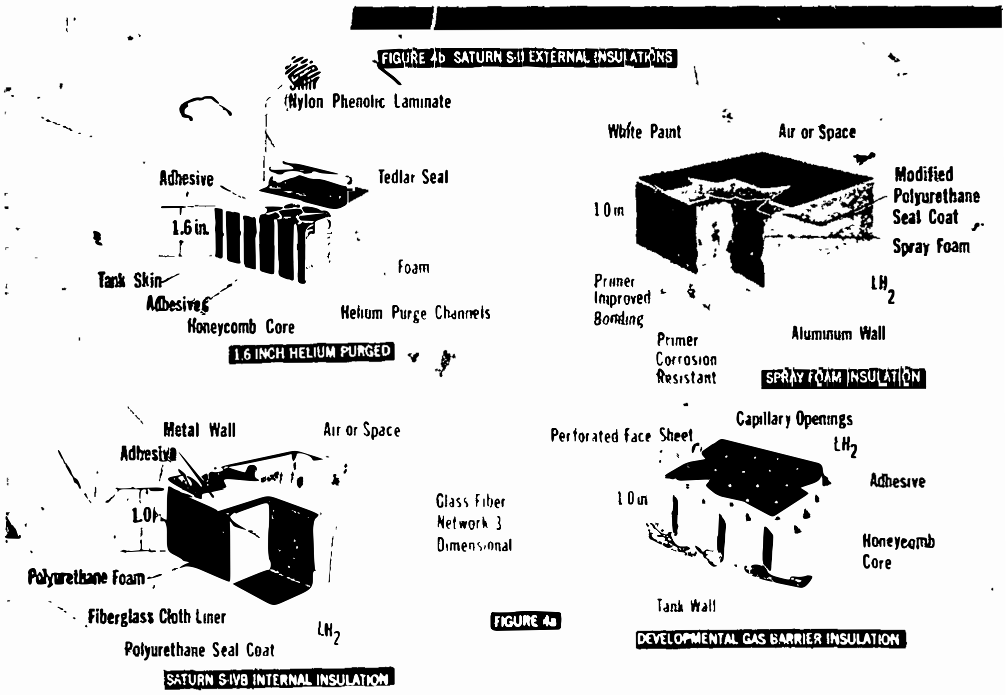
BOOSTER AND ORBITER

SHORT TIME INSULATION APPROACHES

LH$_2$ TANKAGE	LOX TANKAGE (IF NEEDED)
INTERNAL	EXTERNAL
1. REINFORCED FOAM (S-IVB)	1. FOAM FILLED HONEYCOMB (S-II)
2. GAS BARRIER	2. SPRAY FOAM (S-II)
	3. DUAL SEAL

GENERAL REQUIREMENTS

- ADEQUATE INSULATION PERFORMANCE
- NO INTERNAL INSULATION FOR LOX TANKAGE
- RESISTANCE TO CYCLIC SHUTTLE FLIGHT ENVIRONMENTS AND WEATHERING
- EASE OF DEFECT DETECTION, ISOLATION AND REPAIR
- INSULATION SHOULD HAVE ADEQUATE RESISTANCE TO ELEVATED TEMPERATURES
- CORROSION PROTECTION MUST BE PROVIDED WHEN NEEDED



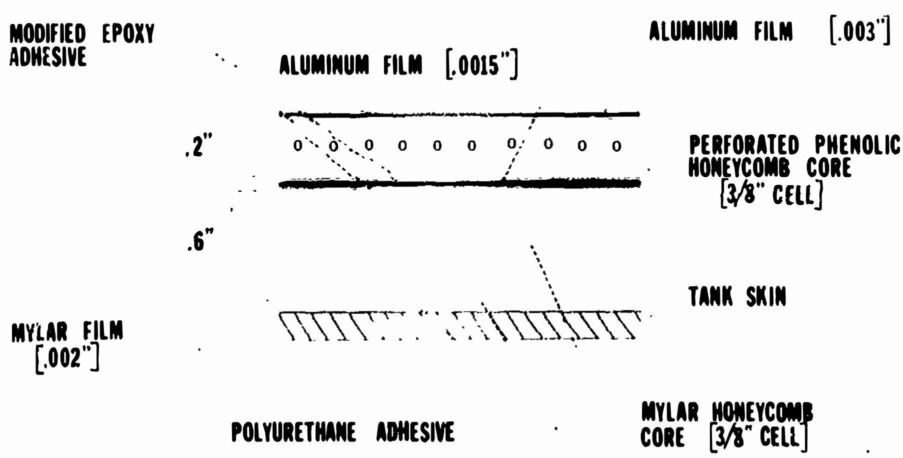


FIGURE 5 - DOUBLE SEAL INSULATION

FIGURE 6

EXTERNAL INSULATION

- MORE UNIFORM AND BETTER PERFORMANCE, LOWER INSTALLED COST
- PERMITS LIGHTER TANK CONSTRUCTION
- SUITABLE FOR OXIDIZER TANKAGE
- MORE VULNERABLE TO WEATHERING AND DAMAGE, CORROSION INHIBITION MAY BE REQUIRED

FIGURE 7

INTERNAL INSULATION

- LESS VULNERABLE TO SERIOUS DAMAGE, FAILURES TEND TO BE ISOLATED
- FEWER BONDING PROBLEMS
- LOWER CRYOGEN REQUIREMENTS DURING CHILLDOWN
- POORER THERMAL PERFORMANCE WHICH USUALLY WORSENS WITH TIME
- HEAT LOSSES THROUGH EXTERNAL TANK SUPPORTS AND ATTACHMENTS ARE MINIMIZED
- UNSUITED FOR LOX TANKAGE
- HEAVIER, THICKER TANK CONSTRUCTION REQUIRED

FIGURE 8

PERFORMANCE OF TYPICAL CANDIDATE SHORT-TIME INSULATIONS

INSULATION	K-BTU/Hr-Ft-°F	NORMAL THICKNESS	Wt - Lb/Ft^2
S-IVB INTERNAL FOAM	.035 - .045	1-INCH	0.61
INTERNAL GAS BARRIER	.06 - .08	1-INCH	0.28
S-II FOAM FILLED HONEYCOMB EXTERNAL INSULATION	.06	1.6-INCH	0.86
S-II EXTERNAL SPRAY FOAM	.01	1-INCH	0.27

FIGURE 9

CANDIDATE FOAMS FOR CRYOGENIC INSULATION

FOAM	ESTIMATED TEMPERATURE LIMIT
POLYURETHANE	200 - 250°F
POLYISOCYANURATE	300 - 350°F
SILICONE	500 - 550°F
POLYIMIDE	600 - 650°F
POLYBENZIMIDAZOLE	600 - 650°F

FIGURE 10
CANDIDATE MATERIALS FOR MULTILAYER INSULATION

SPACER MATERIALS	APPROXIMATE UPPER TEMPERATURE LIMIT °F	COMMENTS
BETA GLASS NETTING/FABRIC	ABOVE 600	HEAVY BUT PROMISING
DACRON NETTING	250	VERY PROMISING
NOMEX NETTING	450	PROMISING
PBI NETTING	650	PROMISING
TISSUGLAS	ABOVE 600	FRAGILE
SUPERFLOC TYPE	250	SUBJECT TO COMPRESSIVE LOADS
POLYURETHANE FOAMS	250	RELEASES DUST
THIN SLICED HONEYCOMB	250 - 650	MAY BE ABRASIVE
REFLECTOR MATERIALS		
ALUMINIZED MYLAR	250	SUBJECT TO MOISTURE DEGRADATION
ALUMINIZED KAPTON	650	SUBJECT TO MOISTURE DEGRADATION
ALUMINIZED KAPTON OVERCOATED WITH GERMANIUM	650	UNDER DEVELOPMENT
GOLDIZED MYLAR	250	PROMISING
GOLDIZED KAPTON	650	VERY PROMISING

FIGURE 11

HIGH TEMPERATURE PROPERTIES OF GLASS REINFORCED POLYURETHANE FOAM

SIX DAYS ISOTHERMAL AGING		FLATWISE TENSILE PROPERTIES	
TEMPERATURE °F	WEIGHT LOSS %	TEMPERATURE °F	TENSILE STRENGTH, PSI
210	0.7	-320	150 - 200
250	1.1	75	200 - 250
300	6.6	200	100
		250	80
		300	40
		350	10

FIGURE 12

PROPERTIES OF CANDIDATE ADHESIVES

LAP SHEAR STRENGTH - PSI

TYPE	-423°F	-320°F	-250°F	200°F	250°F	300°F	400°F	500°F	600°F
POLYURETHANE	6000	5500	5000	400	250	200	----	----	----
MODIFIED POLYURETHANE	8000	8000	7500	1200	1000	700	----	----	----
MODIFIED EPOXY	1600	1200	2300	500	350	280	----	----	----
EPOXY	1400	1600	1700	2300	1800	1300	700	500	----
MODIFIED EPOXY	1800	2200	----	----	----	2800	1800	800	----
EPOXY - PHENOLIC	1800	2800	----	3000	2700	2700	2200	1900	----
POLYIMIDE	----	----	----	2700	----	----	----	2300	1750

FIGURE 13

EVALUATION OF MULTILAYER INSULATION MATERIALS

A. SPACER MATERIALS

- GAS FLOW RESISTANCE
- AGING, HUMIDITY RESISTANCE
- RESPONSE TO CYCLIC COMPRESSION, VIBRATION, VACUUM AND THERMAL EXPOSURES
- PHYSICAL AND CHEMICAL EFFECTS ON METALLIZED REFLECTORS
- CONTAMINATION PROBLEMS

B. REFLECTOR MATERIALS

- AGING AND HUMIDITY RESISTANCE OF FILM AND METAL COATING
- RESPONSE TO CYCLIC THERMAL AND PHYSICAL ENVIRONMENTS
- METAL ADHESION AND WEAR RESISTANCE

FIGURE 14

CONTRACTED INSULATION PROGRAMS

CONTRACT NO.	TITLE	CONTRACTOR
NAS8-26006	DEVELOPMENT OF LIGHTWEIGHT MATERIAL COMPOSITES TO INSULATE CRYOGENIC TANKS FOR 30-DAY STORAGE IN OUTER SPACE	McDONNELL DOUGLAS/ WESTERN DIVISION
NAS8-25973	DEVELOPMENT OF ADVANCED MATERIAL COMPOSITES FOR USE AS INTERNAL INSULATION FOR LH_2 TANK (FOAM)	McDONNELL DOUGLAS/ WESTERN DIVISION
NAS8-25974	DEVELOPMENT OF ADVANCED MATERIAL COMPOSITES FOR USE AS INTERNAL INSULATION FOR LH_2 TANK (GAS LAYER)	MARTIN MARIETTA/ DENVER DIVISION

N70-43001

ENVIRONMENTAL EFFECTS ON SPACE SHUTTLE MATERIALS

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The study of environmental effects on space shuttle structural alloys encompasses a wide range of investigations, including general materials compatibility studies and flight operational aspects.

It is important, from a cost and schedule viewpoint, to select materials in the preliminary design phase that are fully compatible with the anticipated environment; yet, in practice, this point is often not given sufficient emphasis, and late program materials and design changes become necessary, incurring excessive costs. Too often, systems qualification tests identify materials compatibility problems that should have been recognized at the time of the initial selection of the material. By then, it is too late to make a substitution, in some cases, without an overall design change in the component. Thus, in the shuttle program, we intend to make a strong effort to consider the compatibility aspects in the preliminary design phase of the program, yet, the environment to be experienced by the shuttle will be extremely difficult to judge until further analytical and experimental programs are completed and operational specifications have been firmed up.

Some of the more significant problems already under investigation are listed in Table I.

Propellant Compatibility with Tankage Alloys

One important problem to consider is the compatibility of the proposed propellants with the tankage alloys. While we can benefit from considerable past experience with liquid oxygen and liquid hydrogen containment, as well as with storable propellants, we do not have experience with the reuse conditions to which the shuttle will be exposed, nor to the possible effects of repeated re-entries. Some materials that have proven suitable for various propellant containment are listed in Table II. The majority of the shuttle preliminary designs have proposed the use of 2219 aluminum for the primary tanks. There is considerable Saturn-Apollo experience with this alloy, although it has not been used for large LH_2 tanks. Based on the laboratory data available, however, this alloy will perform satisfactorily for LH_2 tankage as well as for LO_2 tankage.

For smaller propulsion systems, titanium has been used quite a bit for LH_2 tankage, but one must avoid the use of titanium where both GH_2 atmospheres and elevated temperatures are present. Titanium must not be used for LOX or GOX applications because of its high reactivity. This material has also been used satisfactorily for storable propellants, such as MMH and N_2O_4.

Overall, the shuttle requirements involve the evaluation of these and perhaps other proven materials under the reuse conditions. The major factors that must be evaluated are fatigue life, thermal cycling effects, and corrosion susceptibility. Some of this work is underway within the NASA Centers, supplemented by some contractual effort. Additionally, it is probable that the propellant tanks of the shuttle will be of more complex design than the usual cylindrical tanks of large space vehicles, thus some effort to study fabricability of the candidate alloys must be made.

The shuttle main propulsion engines and the attitude control engines may operate at somewhat higher pressures than current systems, and the problem associated with high pressure hydrogen and oxygen containment will have to be faced. The recent Apollo 13 incident emphasizes the problem that can result from high pressure oxygen. Very little data is available on threshold pressure and energy levels that would cause ignition of a material in liquid or gaseous oxygen. The limited data available suggests that materials which are suitable for use in gaseous O_2 at atmospheric pressure may react strongly at higher pressures. This applies to both metals and nonmetals. The energy level required for ignition in whatever form it may be, such as impact or electrical arcs, must be studied extensively where one expects to employ specific materials in such systems.

Thermal control systems have historically been a problem with respect to materials compatibility. Fluids such as water, methanol, ethylene glycol, and oils have been used for coolants, and it is difficult to build a complex system that contains good sealants, good thermal conductive materials, and efficient pumps and heat exchangers using materials fully compatible with such fluids. Very careful control of impurities, ph level, and temperatures have been necessary in the past to assure long life reliable operation of such systems.

Effects of High Pressure GH_2 on Metals

Considerable effort is being put into studies on the effects of high pressure hydrogen gas on alloys. This is especially important in the propulsion systems proposed for the shuttle, including the main propulsion engines, attitude control engines, and the air breathing engines which are to be converted from hydrocarbon burners to GH_2. The high temperature requirements in rocket engines generally dictate the use of nickel alloys which have been found to be quite sensitive to high pressure hydrogen embrittlement. Some general characterizations of materials have been evaluated already, but specific operating conditions of proposed engine materials have not been studied.

Figure 1 shows the effects of tests on several materials in 10,000 psi hydrogen (yield strength vs. reduction of notched strength). Most of such evaluations have been made at room temperature. A few alloys, such as Ti-6Al-4V and Inconel 718 have been evaluated at cryogenic temperatures, but essentially no work has been done at elevated temperatures. Yet, some very preliminary work on a few selected alloys indicates that high temperature effects may not be as severe as room temperature effects - depending upon the hydrogen adsorption factors. Programs now under way are directed toward determining more precisely the influence of factors listed in Table III. One such program is being conducted at Rocketdyne, Canoga Park, and a program including high temperature effects is to be awarded very soon. Data obtained to date indicates that:

1. Pressure - Mechanical property degradation increases with increasing GH_2 pressure, especially on notched or precracked specimens

2. Exposure time - The effects are not time dependent; strength loss occurs as soon as the alloy is exposed to GH_2.

3. H_2 purity - A very important parameter; approximately 1% O_2 can completely inhibit degradation.

4. Cyclic effects - Little effect is observed unless the endurance limit is exceeded.

5. Temperature - Insufficient work done above room temperature; limited work indicates high temperatures may be beneficial. Cryogenic temperatures (-200 to -300° F) eliminates degradation.

6. Stress - Degradation occurs between the onset of plastic strain and ultimate failure in GH_2 exposure.

7. Combination of the above environments have not been explored sufficiently, but appear to be additive.

Life Support and Environmental Control Systems

Most of the work done to date to evaluate materials for crew compartment applications and environmental control systems has been limited to the specific mission requirements for certain space vehicles. Most recent emphasis has been placed on space station requirements where long space tenure is expected; yet, very similar characterizations must be determined for shuttle operations. Some of the evaluations under way are listed in Table IV.

Considerable contributions to our knowledge about these factors have been made by the Apollo and Skylab programs. Many materials that are likely to be used in crew cabins are nonmetallic and there are very few such materials that are not flammable to some degree. Even metals can be a problem if used as thin foils. The flammability tendencies of materials are determined by several factors as indicated in Table V; therefore, one must characterize flammability with the ambient environment of the material. Where this environment is difficult to control, the best practice is to use the least flammable material available. Most of this type of data is being generated in NASA labs and comprehensive reviews of materials usages are conducted to avoid the use of susceptible materials.

Corrosion and stress corrosion resistance of materials has been a very important factor in the design of spacecraft in the past, and this will become an even more important factor with the shuttle vehicle, due to the many reuses and the long lifetime anticipated. Current corrosion protective coatings should be adequate to protect most of the candidate shuttle structural alloys, yet each system considered must be evaluated for the shuttle operating conditions.

One area of particular concern is that of the effects of elevated temperature cycling on stress corrosion resistance. Another concerns the effects of high temperatures or moderate temperature cycling on aluminum primer systems, where the vehicle is exposed to high humidity environments between each flight. Such conditions may result in a degradation of the primer, rendering it ineffective or even aggressive to the material. With respect to corrosion of superalloys, which may be used extensively in the TPS, we are not too concerned, but the major effect will be concentrated on the substructure which likely will be aluminum and titanium.

Other environmental factors, such as biological or sterilization characteristics, offgassing characteristics, contamination, toxicity and odor, are not necessarily problems peculiar to the shuttle; they are of much more importance to a long life space station. Yet, we must consider that the shuttle orbiter will dock for a period of time with a space station and certain compatibility standards must be developed. For example, if an internal crew cabin or cargo compartment atmosphere exchange with the space station is necessary, extreme precautions with biological control must be taken. The life support systems should be compatible. If external contaminants are emitted from the shuttle, in the form of thermal protection system coating particles, or insulation particles, these could cause serious problems with critical space station experiments, especially optical surfaces associated with the experiments or the operation of the space station. The TPS materials proposed to date will be particularly difficult to prevent such particles from being emitted, since the refractory metal or superalloy panels may overlap and faying surfaces are likely to release small particles. This possibility and its consequences must be considered in the TPS system design.

High Temperature Effects on Materials

The high temperature re-entry conditions represent the most critical environmental factor for the shuttle. Almost the entire external surface will consist of refractory metals or superalloys, and extensive evaluation programs must be conducted to determine the effects of such high temperature cycles on these materials. Some of the consequences anticipated as a result of these high temperature exposures are listed in Table VI.

Oxidation characteristics of many of the candidate materials are under investigation now, but the actual temperature and aerodynamic factors are not too well defined at this point. Additionally, test facilities for simulating high temperature high flow conditions are quite limited currently, particularly for large full sized panels.

Erosion tests simulating aerodynamic flow, atmospheric dust particles, and possibly rain on both bare and coated materials, will be needed to determine the resistance of the TPS materials and windshields to such conditions.

The extreme temperature differentials will result in a sizable problem to design the TPS for excessive thermal expansion and contraction. While this might be easily accomplished in the skin material, it may turn out to be a very difficult problem when one considers the coupling between the TPS and the major structure and/or the propellant tanks.

Initial tests on superalloys, including both nickel and cobalt alloys have shown that rapid grain growth can occur in these alloys after short exposures to temperatures in the 2100° F regime. An example of this is shown in Figure 2. The effects of such grain size changes or other microstructural changes on mechanical properties will have to be determined and the incremental changes during multiple re-entry flights must be considered.

Finally, the emissivity of high temperature materials has a considerable effect on the actual temperature attained by the material. A change of even 0.1 in emissivity can change the temperature from 45 to 75° F. Most of the emissivity data available currently represents ambient condition values, thus emissivity values for metals must be determined under high temperature conditions simulating re-entry conditions.

In summary, the shuttle system seems to present to us a set of environmental factors never before encountered for a space vehicle. The shuttle is a launch vehicle, a spacecraft, a re-entry vehicle, and an airplane all in one. Temperatures will range from -423° to over 3000° F in various areas of the structure. Atmospheres will range from high humidity ground exposure to space vacuum to high temperature re-entry. It is difficult to imagine a harsher environment. Yet, such conditions must be investigated in all of the promising materials available to us in order to allow the most practical and conservative design possible.

TABLE I

MATERIALS COMPATIBILITY

- PROPELLANT COMPATIBILITY WITH TANKAGE ALLOYS
- THERMAL CONTROL SYSTEMS
- EFFECTS OF HIGH PRESSURE GASES (GH_2)
- CREW COMPARTMENT ATMOSPHERE & LIFE SUPPORT SYSTEMS
- HIGH TEMPERATURE EFFECTS
- GENERAL & STRESS CORROSION

TABLE II

CANDIDATE MATERIALS FOR PROPELLANT TANKS

LIQUID HYDROGEN	• 2000 SERIES ALUMINUM ALLOYS • AUSTENITIC STAINLESS STEELS • TITANIUM 5Al-2. 5Sn • NICKEL ALLOYS
GASEOUS HYDROGEN	• 2000 SERIES ALUMINUM ALLOYS • AUSTENITIC STAINLESS STEELS
LIQUID & GASEOUS* OXYGEN	• 2000 SERIES ALUMINUM ALLOYS • AUSTENITIC STAINLESS STEELS • NICKEL ALLOYS*
N_2O_4 (STORABLE)	• TITANIUM 6Al-4V • NONMETAL BLADDERS
MMH (STORABLE)	• 2000 SERIES ALUMINUM ALLOYS • AUSTENITIC STAINLESS STEELS • NICKEL ALLOYS
HYDROCARBONS	• 2000 SERIES ALUMINUM ALLOYS • AUSTENITIC STAINLESS STEELS • NICKEL ALLOYS

TABLE III

HYDROGEN EFFECT PARAMETERS

- PRESSURE
- TIME
- HYDROGEN PURITY
- CYCLIC EXPOSURE
- TEMPERATURE
- STRESS & STRAIN
- SYNERGISTIC EFFECTS

TABLE IV

ENVIRONMENTAL STUDIES

- FLAMMABILITY
- CORROSION RESISTANCE
- BIOLOGICAL EFFECTS & STERILIZATION METHODS
- AESTHETIC PAINTS OR COATINGS
- OFFGASSING
- TOXICITY & ODOR
- REACTIVITY WITH FLUIDS & GASES

TABLE V

MATERIALS FLAMMABILITY FACTORS

- TYPE OF IGNITION
- ATMOSPHERE CHEMISTRY & PRESSURE
- MATERIAL CHEMISTRY
- MATERIAL THICKNESS
- SAMPLE ORIENTATION

TABLE VI

EFFECTS OF SHUTTLE RE-ENTRY EXPOSURE CYCLES

OXIDATION OF TPS MATERIALS

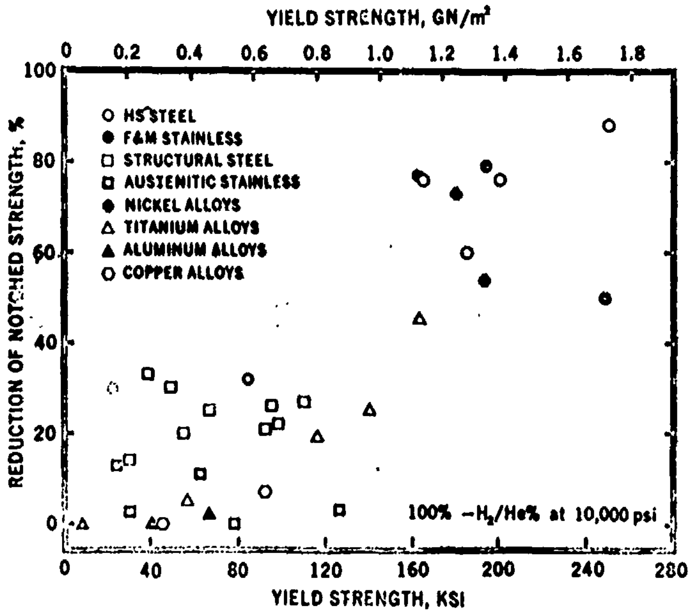
EROSION (AERODYNAMIC, DUST PARTICLES, RAIN, ETC.)

THERMAL EXPANSION AND CONTRACTION

GRAIN GROWTH IN STRUCTURAL ALLOYS

CHANGE IN EMISSIVITY VALUES

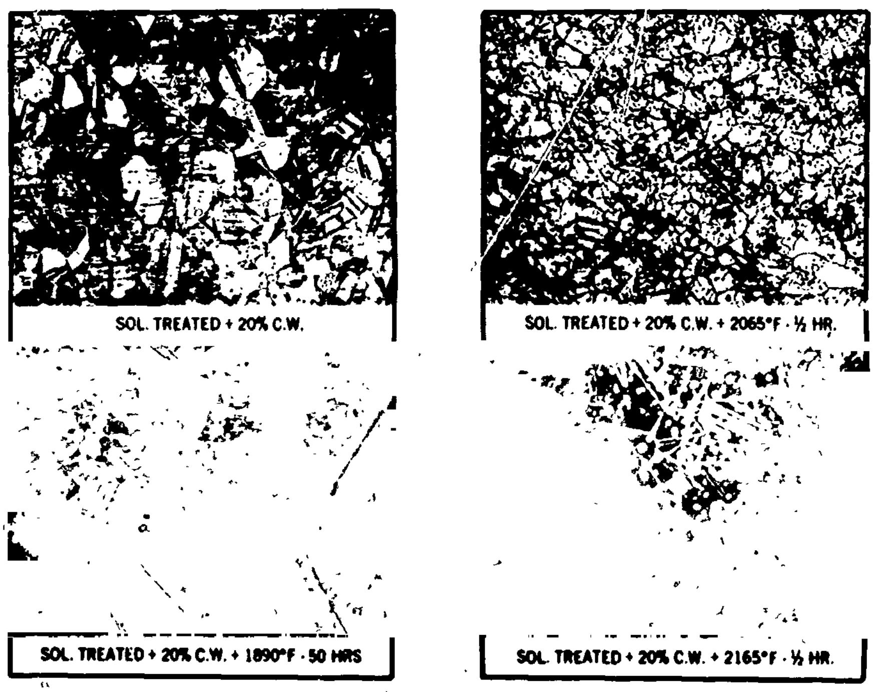
FIGURE 1



YIELD STRENGTH IN AIR VS REDUCTION OF NOTCHED STRENGTH

Figure 2

MICROSTRUCTURAL CHANGES IN 20% COLD WORKED HS-188 ALLOY



N70-43002

CARBON FIBRE COMPOSITES FOR RE-USABLE SPACE SHUTTLE STRUCTURES

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Since the discovery by RAE in the early 1960s of an economical process for converting a man-made polyacrylonitrile fibre into a highly oriented structure carbon fibre of outstanding specific strength and elastic modulus the Ministry of Technology and the aerospace industry have been supporting an intensive R&D programme aimed at exploiting the outstanding potential for weight saving in practical aerospace structures that this [illegible] provides. The applications of carbon fibr[illegible]stics, rather than reinforced metals, has been [illegible] principal concern, and while the primary objective has not been aimed specifically towards space vehicle structures the technology nevertheless has direct application in this field.

For the types of re-entry vehicles that are generally being considered for the space shuttle programme much of the load bearing structure will be thermally protected to provide an operating temperature such that aluminium alloys are acceptable (not greater than 200°C). At the same time the loading index would seem to be relatively low, as compared with that for more conventional high performance aircraft, and in consequence much of the structure will be designed on stability or even minimum skin gauge considerations. However, considerations of the letter kind are no novelty in the structures field, particularly for certain classes of light aircraft and for particular areas of structure for civil transports.

The efficiency of different materials for particular types of structure can broadly be determined in relation to certain merit indices and values of the merit indices for carbon fibre reinforced plastics for a wide range of loading indices show a marked superiority over those for conventional metallic materials. A typical comparison of carbon with conventional materials, for a skin-stringer compression panel, is shown in Fig. 1. Of course, this is far from demonstrating that the potential for weight saving, that superior merit indices implies, can be realised in a practical structure. It is towards the latter objective that the UK programme has been directed.

This programme is in two broad categories, namely;

(a) Evaluation of the properties of carbon fibre reinforced plastic composites and the accumulation of the data required for the proper design of carbon fibre reinforced composite structures.

(b) The design, fabrication and testing of a wide range of structural components in carbon composite material.

The somewhat mundane yet vital task under (a) must be undertaken whenever a novel aircraft structural material is introduced and the magnitude of the task should not be underrated. Within the time scale that is envisaged for the space shuttle system only thoroughly characterised structural materials of assured availability can seriously be considered, and it is only because of the extensive effort that has been deployed in the design data field over some years in the UK that carbon fibre reinforced plastics can be regarded as in this category.

The extent of the work in progress under (b) is best illustrated by a brief resume of a range of different types of structure that have been fabricated and tested, in some cases including flight testing.

The first of these (Fig. 2) is a floor beam developed by BAC in which the flanges of an aluminum alloy beam have been reinforced by unidirectional composite; beams up to 20 feet in length have been successfully fabricated. An application of this kind makes the maximum use of the potential of composite material. The composite element of the beam is only one-third of the weight of the metallic element that it replaces. Uses for the thrust grid structure of the space shuttle can be visualised.

Actuator rods for the Harrier that have been developed by HSA also seek to use the composite in a highly anisotropic form to achieve high weight saving (Fig. 3). Members of this kind could well find extensive applications in a space shuttle structure. A geodetic or braced space frame structure using fibre reinforced composite struts could well be more efficient than a stressed skin structure for carrying major structural loads, and may be an acceptable solution where a separate aerodynamically loaded outer shell is required to provide thermal protection from the inner structure.

Tabs for the HSA 748 and the BAC Strikemaster aircraft (Fig. 4) are examples of lightly loaded structures where thin composite skins are stabilised by a honeycomb core. This is a typical method of fabrication for lightly loaded structure that is likely to be used extensively in space structures. One UK application in which it has already been used is in a satellite structure (Fig. 5 and 6).

Examples of more heavily loaded structures that utilise a somewhat similar method of fabrication are a dive brake for the HSA Vulcan and a spoiler for the BAC/Breguet Jaguar (Fig. 7 and 8).

The HSA 748 torque tube for control surface operation is an example of a thin skin, stability designed filament wound structure. A transmission shaft for a WHL helicopter main rotor, formed by wrapping, is shown in Fig. 9. High rotational speed without whirling is required in this application.

Filament wound or wrapped structures are of particular interest for the space shuttle, both in applications of the above kind and for fuel tank or pressure vessel structures (Fig. 10).

Where pressure must be contained the permeability of thin sheet composite may necessitate a liner for the tank in the form of a flexible membrane or metal inner skin (Fig. 11), but the liner wrinkling problems common with glass fibre vessels should be largely absent from those in carbon fibre because the ultimate fibre strength for carbon is developed at a much lower level of strain.

Where metal and composite are used in combination the possibility exists of a judicious choice of the combination that enables the optimum strength to be developed by both components under the imposed load (Fig 12). Carbon fibre UK type 2 composite and titanium alloy seems to be a particularly advantageous combination, and the HSA Harrier, ferry-role, wing-tip is a particular structure where this combination has been put to test; here there is a titanium inner structure with a carbon composite skin. (Fig. 13).

The Harrier tip is a class 1 structure, i.e. its failure in flight could jeopardise the safety of the aircraft; this component is scheduled for flight tests in early 1971.

The Rolls-Royce work on compressor and fan blades is well known, and has particular application to rotating components of space shuttle systems, e.g. fuel pumps and the like.

The point has already been made that only properly categorized material of proven reliability and availability can be seriously considered for the space shuttle programme and nothing has so far been said of reliability and availability.

Throughout the period of the UK programme particular attention has been paid to process control variables for the carbon fibre itself and for the resin pre-impregnated composite, and material is now available to a controlled specification.

For most purposes the material is preferred in the form of a pre-impregnated tape with a resin formulation, fibre type and surface condition to suit the particular application, and it can be obtained in this form from most of the UK commercial suppliers.

Of particular interest for aerospace structural fabrication is the material available from Rolls-Royce Ltd. in the form of a continuous, 16 inches wide sheet of resin pre-impregnated, carbon composite that can be laminated to a 0.005 inches thick ply of high carbon fibre volume fraction. This material is ideal for fabricating thin gauge structures of large area with minimum joints and for the bulk of skinned structural applications; it has been intensively categorized in connection with Rolls-Royce engine applications.

So far as composite material supplies are concerned the current UK capacity for carbon fibre for aerospace structures applications is greater than 50 tons per annum with a ready capability for a further increase.

Numerous resin systems have been evaluated while others, including the high temperature polyimide resins, are being examined in the current Mintech programme.

There is no doubt that within the European context there is an effective contribution that can be made by the UK in the application of carbon composites to space shuttle structures.

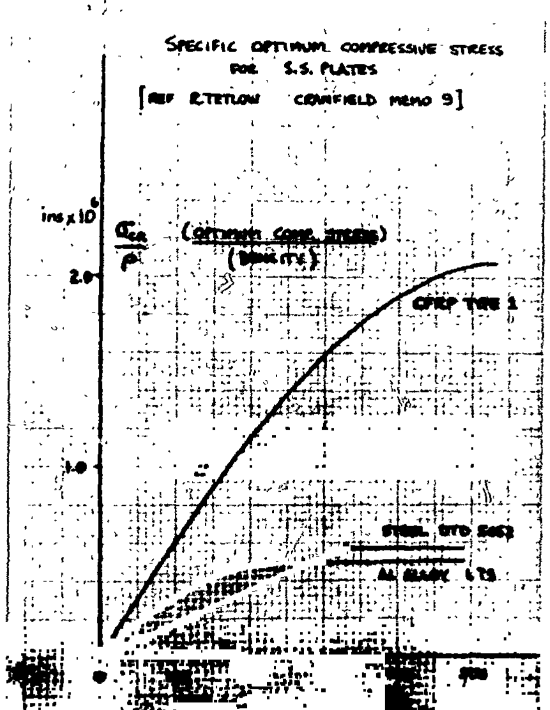

FIGURE 1

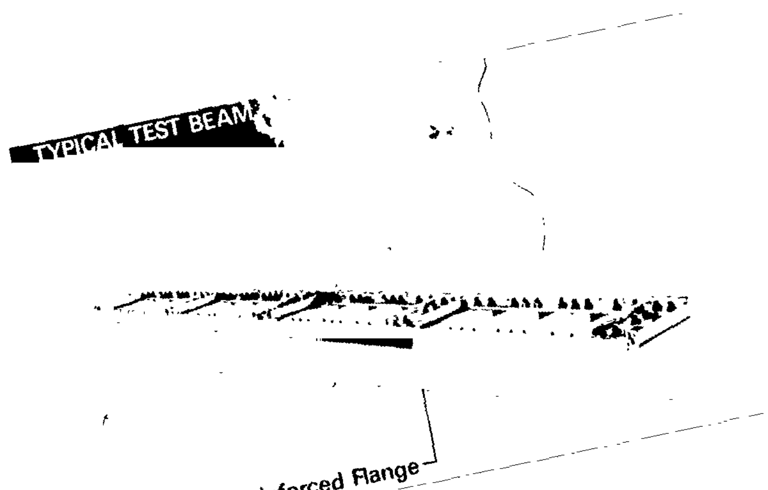

FIGURE 2

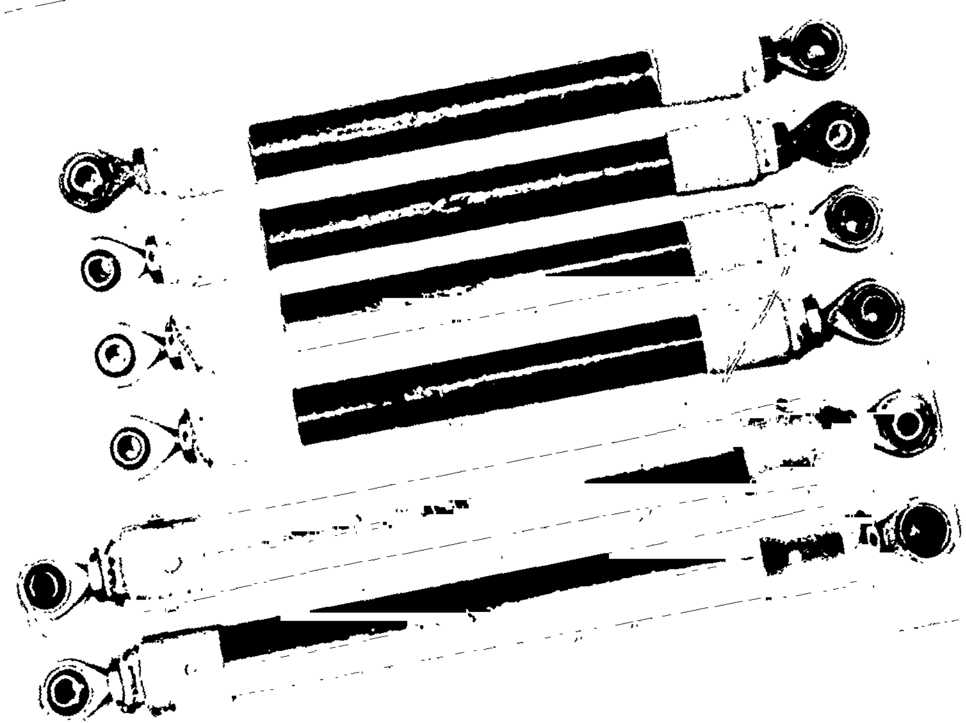

FIGURE 3

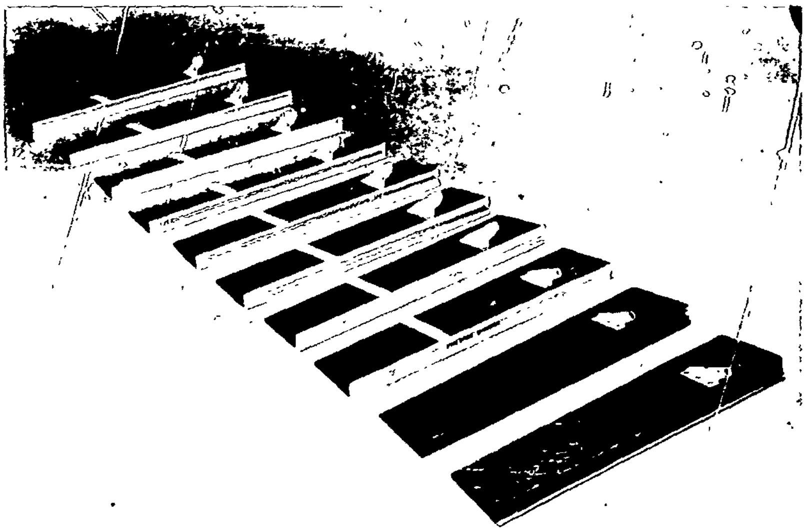

FIGURE 4

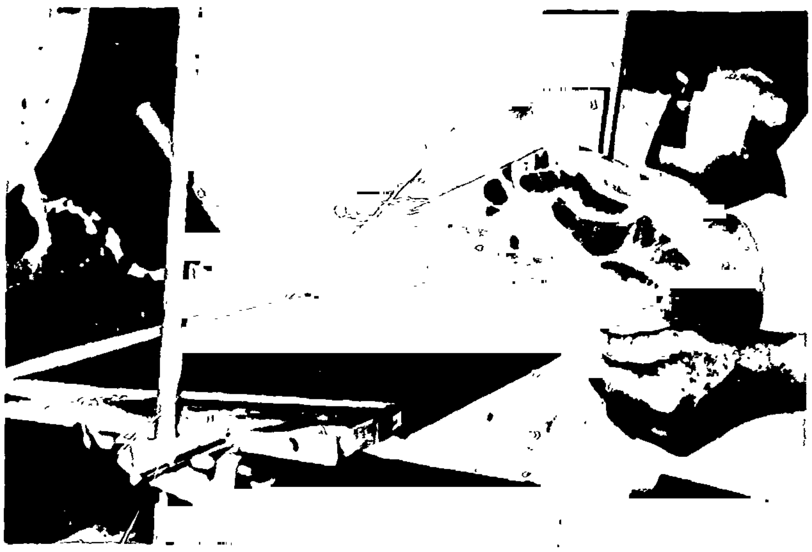

FIGURE 5

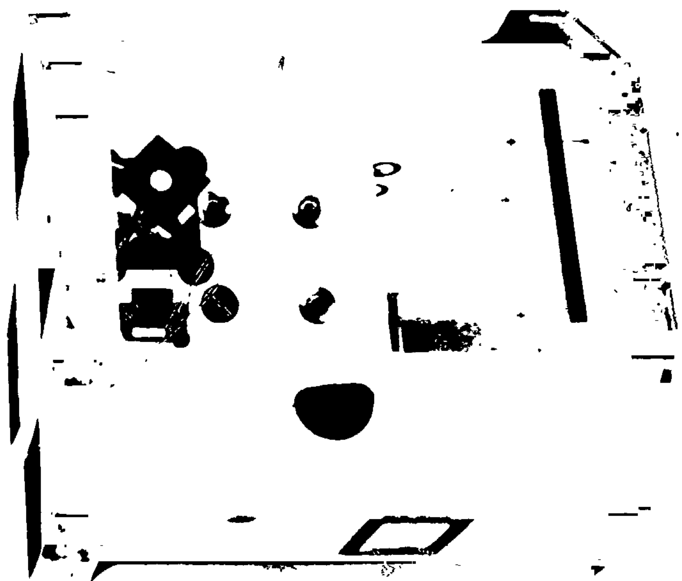
FIGURE 6

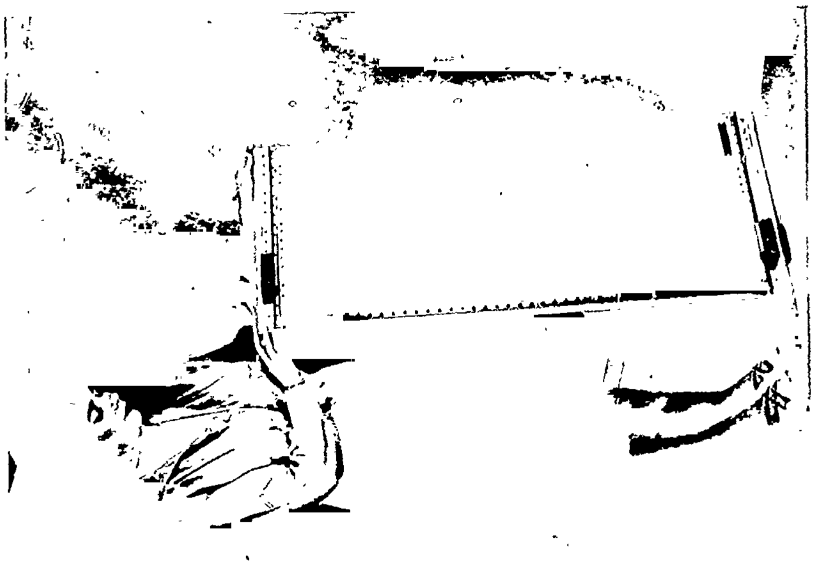
FIGURE 7

FIGURE 8

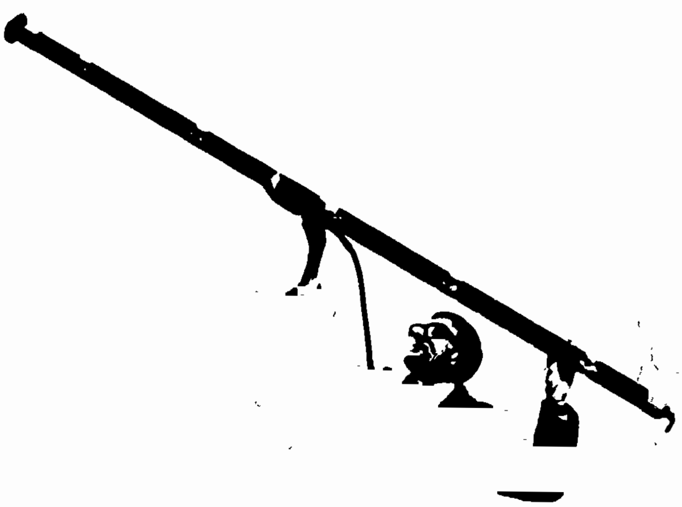

FIGURE 9

FIGURE 10

FIGURE 11

FIGURE 12

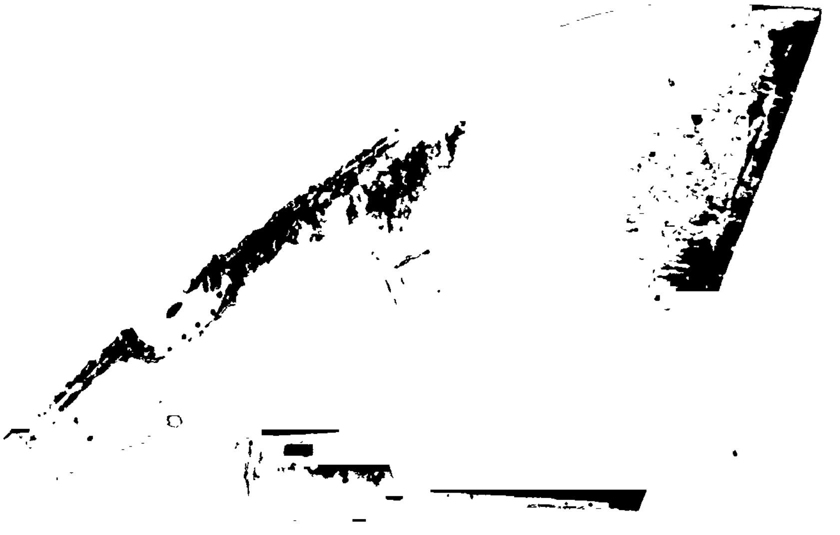

FIGURE 13

N70-43003

FIBER REINFORCED COMPOSITES FOR SPACE SYSTEMS

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Summary

M.A.N. has been engaged in the development of high-strength low weight fiber reinforced components for terrestrial and space application.

This paper deals with the experimental and qualification program of the last several years on the development of fiber reinforced cryogenic pressure vessels with special emphasis on CFR materials.

A weight comparison shows, that the structural weight for future space systems can be considerably reduced if the usual metal fuel and pressure tanks are replaced by fiber tanks.

Zusammenfassung

Die M.A.N. beschäftigt sich seit einigen Jahren mit der Entwicklung von hochfesten Leichtbaukomponenten aus Faserwerkstoffen für terrestrische und raumfahrttechnische Anwendungen.

Der Vortrag gibt einen Einblick in das seit 3 Jahren laufende Experimental- und Qualifikationsprogramm zur Entwicklung faserverstärkter Kunststofftanks unter besonderer Berücksichtigung der CFK-Bauweise.

Ein durchgeführter Gewichtsvergleich zeigt, dass für künftige Raumfahrtsysteme das Strukturgewicht erheblich reduziert werden kann, wenn die üblichen metallischen Treibstofftanks und Druckbehälter durch Fasertanks ersetzt werden.

Resumé

M.A.N. effectue depuis plusieurs années des recherches sur les composants légers à haute résistance en matériaux fibreux pour des utilisations terrestres et spatiales.

L'exposé donne un aperçu du programme expérimental et de spécification qui se déroule depuis 3 ans en vue de développer des réservoirs en matière plastique fibreuse renforcée, en mettant tout particulièrement l'accent sur la construction à l'aide de matières plastiques renforcées par fibres de carbone.

Une comparaison de poids qui a été effectuée montre que, pour des systèmes spatiaux futurs, le poids structural pourra être réduit, lorsque les réservoirs de carburants et réservoirs sous pression métalliques conventionnels seront remplacés par des réservoirs en matière plastique fibreuse.

1. Introduction

The performance of the space carrier systems depends basically on the specific impulse and the structural weight.

All structural and vessel concepts, which show promise of a minimum of weight should therefore be investigated. One method which shows promise is the use of fiber reinforced materials. M.A.N. has been working in this sector for many years.[1][2][3][4,]

The first aeronautical project at M.A.N. was the development of the low pressure casing for the jet engine M.A.N. RB 193 (Fig. 22), which at present is being flight-tested. The first space projects were the development of a space research rocket and also Apogee motor, both were successfully tested on the ground. Glass fiber composites were used in all these projects.

Some time ago the introduction of carbon fiber as a high-strength and stiff reinforcing material opened new and interesting possibilities. These CFR materials are widely used at M.A.N. in the nuclear technology for gas centrifuges. CFR should also prove superior to GFR for solid fuel rockets. Cracks in the composite structure are avoided by the high stiffness which results in minimum elongation making an additional liner unnecessary in most cases. A further advantage is the possible use of low ductility composite fuels as the stiffness of the structure prevents fuel cracking.

Several of the M.A.N. projects can be seen in fig. 1.

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2. Design Criteria

The basic criteria for structural materials is its specific strength S_L. It is the ratio of the ultimate strength to the specific weight. The Young's modulus is important for stability, therefore it is usual to plot the specific strength vs. specific modulus (fig. 23). The superiority of carbon fiber reinforcrd composites can be readily seen.

The change-over to tank design criteria is clear because the structural performance p.v/w is basically the same as only a number of factors have to be taken into consideration.

$$\frac{p \times v}{w} = S_L \sum_i^n C_i$$

as shown in fig. 24.

The results for a large tank design are shown in fig. 2. The comparison between a pressure gas and turbo-pump version is also shown. The next fig. 3 shows the results of the usual structural performance plot in the cryogenic temperature range.[2][3].

3. Structural Materials

The advantage of the fiber reinforced tank concept is determined by the mechanical properties of the composite material, for which, in turn, the properties of the individual components such as the fibers, the resin and last but not least the bonding between the two are of vital importance.

3.1. Epoxy-resins are preferably used for the matrix systems. These resins are modified as required in the individual application. For example for cryogenic tanks Epoxy resins with high elongations (greater than 30 %) are used to avoid brittleness at low ^{o}K. The ductility of such Epoxy systems based on long-chain aliphatic diepoxyds cured under appropriate conditions is still 2-3 % at cryogenic temperatures, and can be compared to the ductility of glass fibers. A large number of special Epoxy systems is available for other highly stressed fiber reinforced components.

3.2. As mentioned above, the bond between the resin and the fibers is of the highest importance for maintaining the properties of the fibers in the composite because the loads of broken fibers have to be carried by shear stresses on the fiber surface, which may also result from the shear mechanism on bending. Therefore the resin system must meet special requirements, which must be attained by optimal affinity to the fiber and appropriate reaction control during curing as well as suitable surface treatment of the fibers. These two points are of the greatest importance when making CFR composites.

The following scanning electron micropictures of fracture surfaces show clearly the bonding problems of CFR composites. The magnifications can be estimated by the known fiber diameters all of which lie in the range between 7 - 9 μm.

Fig. 25 shows an American fiber combined with a German resin system resulting in a large bonding gap. Fig. 26 shows a bonding test of American fiber with American resin system without any gap. A combination without a gap does not necessarily mean a good bonding, as fig. 27 shows (English fiber - German resin), as can be seen from the long fiber ends projection from the fracture surface.

The opposite can be seen in fig. 28 (German fiber - American resin), which shows very short fiber at the fracture surface. A further criterion for a good bonding is the condition of the resin fracture surface, as can be seen in fig. 29 (German fiber - German resin), which shows a clam shell type surface instead of a smooth surface. This is brought about by the internal stress in the resin which created in a good bonding process.

3.3. Glass and carbon fibers are used as reinforcement.

S- and R-glass fibers have strength values in the composite of approx. 280 kp/mm^2 at a specific weight of 2.5 gr/cm^3 and a Young's modulus of 700,000 kp/cm^2. Further limited increases of these values are to be expected. The main advantages of glass fibers in comparison to carbon fibers are the ease of processing and the low price.

The range of properties of carbon fiber is greater than that of the glass fiber. The strength figures lie within $180 \div 350$ kp/mm^2, the Young's modulus values are approx. $3.5 \div 2 \times 10^6$ kp/cm^2 at specific weights of $1.7 \div 2.0$ gr/cm^3.

The selection of the best type of fiber in each individual case depends on stress and stability requirements. The main advantage of the carbon fiber is its stiffness, which is 3 - 4 times greater than that of the glass fiber. Therefore, in most cases, the deciding factor is the strength.

The main object of composite technology is to achieve the greatest possible advantage of the fiber properties in the composite. Because of the scattering of the fiber properties, i.e. strength, Young's modulus and elongation, individual fibers break long before the fracture of the composite.

Fig. 31 shows the stress-strain diagram of composite.

Line A represents the nearly linear behaviour of the composite. Oppose to this, the statistical analysis of the individual fiber types based on the unbroken fibers in the sample shows a strong nonlinear behaviour of strength and stress-strain curves, B and C.

The intersection of curves B and C shows the expected fracture point F. Influenced by the bonding effect of the resin, we get, in practice, the curve $\overline{B}$ with the fracture point $\overline{F}$.

The attainable fiber performance lies between 80 and 95 % with a maximum achieved value at 98 %. These considerations lead us to far-reaching conclusions in classifying the quality of the fibers and possible improvements. (5)

3.4. The testing of the fibers and the composite is of great importance to the quality of the end product. More and more fiber strands are being tested in addition to the testing of individual fibers. Nevertheless, because of the good comparability with filament wound components, the NOL-ring method is still being extensively used. The doubts that the great stiffness of the carbon fibers and because of this the greater bending influence would adversely affect the NOL-ring testing have fortunately not been confirmed. Fig. 32 shows that carbon fiber have only a bending factor of less than 2 % as compared to 10 % and more with GFR. (6)

4. Tank design concept

We see in fig. 4 a tank concept for cryogenic fuel based on the application of GFR and CFR tank structure.

Particular attention should be paid to the type of liner taking into consideration the required permeation rate. A large number of various metal as well as plastic liners have been tested. A plastic applied in liquid form has proven successful, especially in GFR tanks. Thin aluminium foil was tested successfully in CFR tanks. By the use of an appropriate matrix system the liner can be eliminated with some fuels. For the storage of LH_2, the tanks must have a thermic insulation to minimise the evaporation rate.

On the basis of an experimental study (7) the tanks are being insulated with polyurethan foam and sealed with an aluminium-coated mylar foil to prevent diffusion and freezing of moisture in the insulation layer (cryo-pumping effect).

An added GFR open cross layer serves as an insulation reinforcement.

5. Manufacture

Whereas the development and manufacture of CFR pressure tanks is still a new technology which requires appreciable effort to find the optimal parameters, the filament winding of GFR pressure tanks is common practice.

We have wound tanks from 0.1 to 2.0 m in diameter.

Basic problems such as liner systems and composite concepts are being solved by testing small model tanks of 100 mm in diameter, series XI (fig. 5), e.g. galvanic applied liners. In addition, cylindrical bodies (fig. 6) and 400 mm dia. tanks, series XII and XIV, are being used in the experiments.

Details of insulated tank can be seen in fig. 7 and 9. The preformed insulation segments are glued into place and then covered with the above-mentioned sealing foil and then reinforced by an open layer.

The aluminium liners for CFR tanks seen in fig. 10 and 11 are deep-drawn, chemically etched down to 0.3 mm and then joined to the flanges by EBW.

A separate mandrel can be eliminated by this method as the liner stabilised by internal pressure can, if necessary, serve in its place.

6. Testing

In the main the tank testing consist of a series of pressure tests. H_2O, LN_2 and LH_2 are used in the tests.

An insulated GFR tank ready for testing in the test bed is shown in fig. 12. The tank is suspended on its upper flange where the supply lines are fitted.

A number of strain gauges are attached to the structure (fig. 8). The elongation of the whole tank is determined by means of an induction probe attached at the bottom of the tank.

The test program includes leak tests with LN_2 at low pressures over long periods, filling cycles and re-heating to room temperature, cyclic pressure tests and burst tests. Fig. 15 shows the filled tank with the induction probe at the bottom.

A non-insulated and therefore iced CFR tank filled with LN_2 is shown in fig. 17. After several load cycles up to operation pressure the pressure was raised to the bursting point. The fracture pieces of liner and structure can be seen in fig. 18.

The analysis of the obtained measurements agree in general with the original expectations (8).

7. Conclusions

The main aim of the experimental program described is the manufacture of fiber reinforced pressure tanks for the cryogenic, high energetic fuel components LH_2 and LOX.

These pressure tanks produced by the filament winding method could e.g. be later integrated in a modulus stage and subjected to a stage testing program, as shown in fig. 21.

Along with the fiber tanks, further-reaching necessary research is planned to determine the application possibilities of GFR and CFR in complex components.

In the light of the experience gathered at M.A.N with fiber reinforced units, it appears important to profit from the decided weight advantages offered by fiber reinforced fuel tanks and pressure vessels in future air space systems.

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[4] H. Kreft "Fasertanks für Trägerraketen, Vorteile des GFK-Filament-Winding für Treibstofftanks einer neuen europäischen Trägergeneration", Bericht 504 - Er - 001, M.A.N., März 1970

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[8] W.S. Lassmann "Beurteilung des Abdrückversuches am Tank XIII-CFK-2", Bericht 003 - Es - 002, M.A.N.

Fig. 1

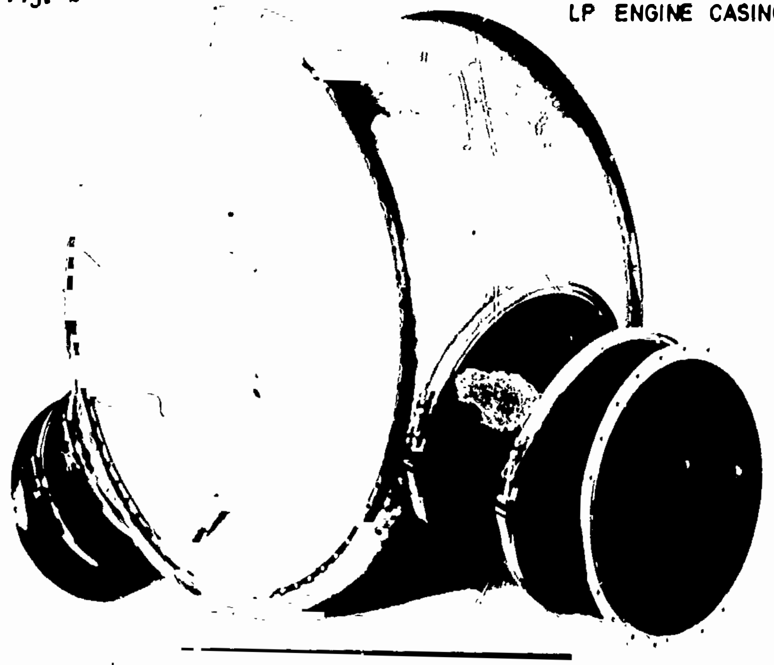
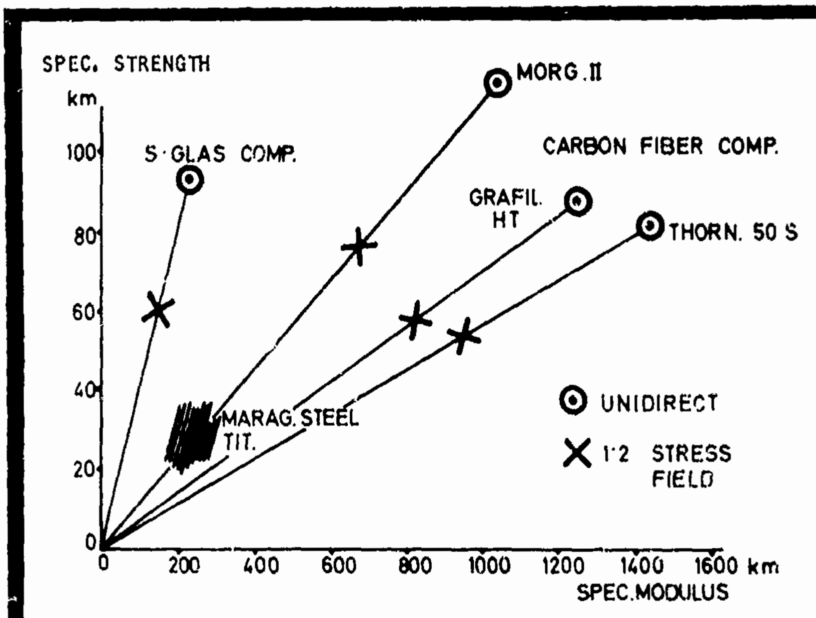


Fig. 2



MATERIAL PERFORMANCE — Fig. 3

$$\text{SPEC. STRENGTH} = \frac{\text{STRENGTH}}{\text{SPEC. WEIGHT}} = S_L \quad [\text{IN}]$$

$$\text{STRUCT. PERF.} = \frac{\text{PRESSURE} \cdot \text{VOL.}}{\text{WEIGHT}} \quad [\text{IN}]$$

$$\frac{P \cdot V}{W} = S_L \, (C_1 \cdot C_2 \cdot C_3 \cdot C_4 \cdot C_5 \cdot C_6 \cdot C_7 \cdot C_8)$$

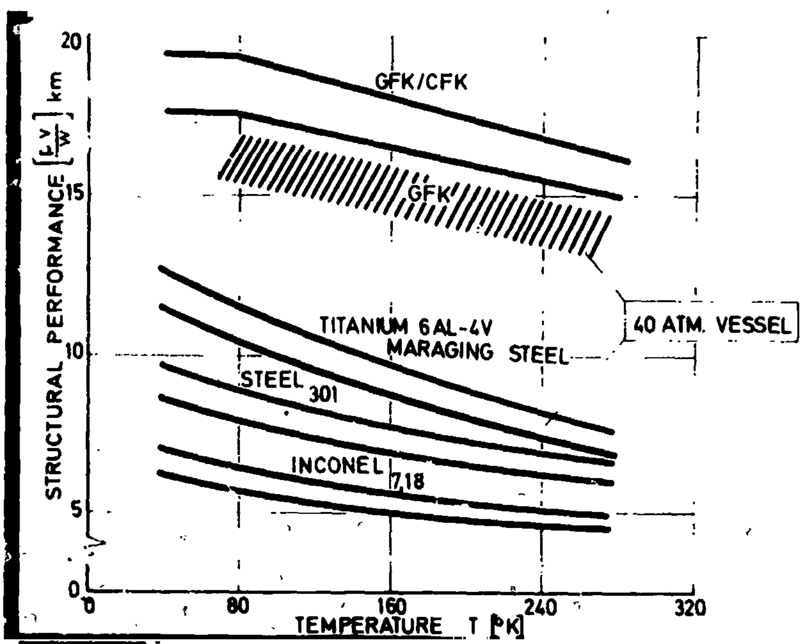
C_1	ANISOTROPIC - ISOTROPIC RATIO
C_2	RESIN FACTOR
C_3	LINER+POLPIECE WEIGHT RATIO
C_4	SHAPE FACTOR
C_5	TEMPERATUR INFL.
C_6	LOAD CYCLE INFL.
C_7	SHORT / LONG TERM CREEP
C_8	SAFETY / RELIABILITY MARGIN

STRUCTURAL TANK CRITERIA — Fig. 4

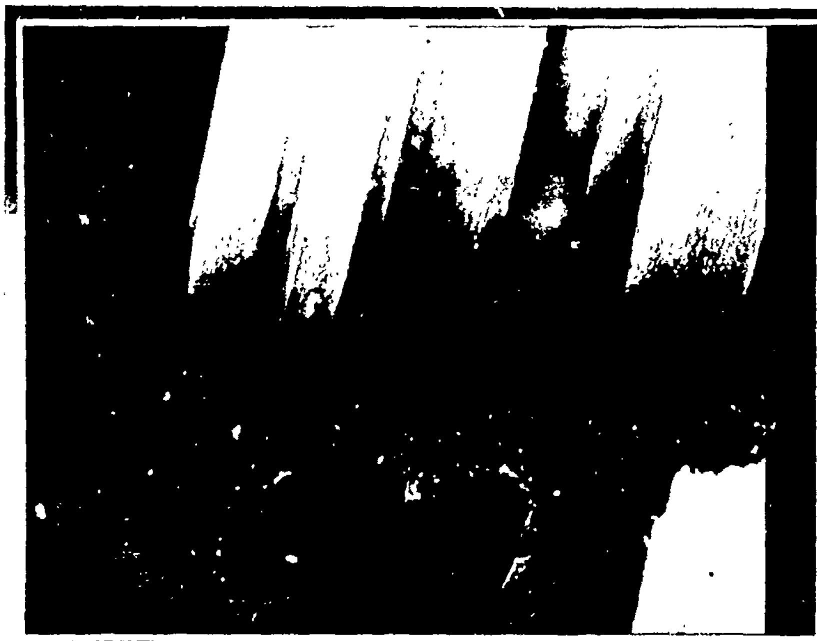
BASIC DIMENSIONS	
DIAMETER	: 2500MM
TOTAL LENGTH	: 8000MM
TANK DOME OPENING	: 400MM
VOLUME	: 37.5M^3
safety factor 1.4	
coeff. of variation 8%	
STRUCTURAL PERFORMANCE	
K $\frac{\text{PRESSURE} \cdot \text{VOLUME}}{\text{WEIGHT}}$	

RESULTS			
OPERATING PRESSURE	40 kp/cm^2		6 kp/cm^2
MATERIAL	GFK S-GLAS 904	STEEL MAR-AGING	STEEL VASCOJET 90
STRENGTH kp/mm^2	280/210	175	100
WEIGHT kp	1167	2300	800
K km	12.8	6.4	2.8

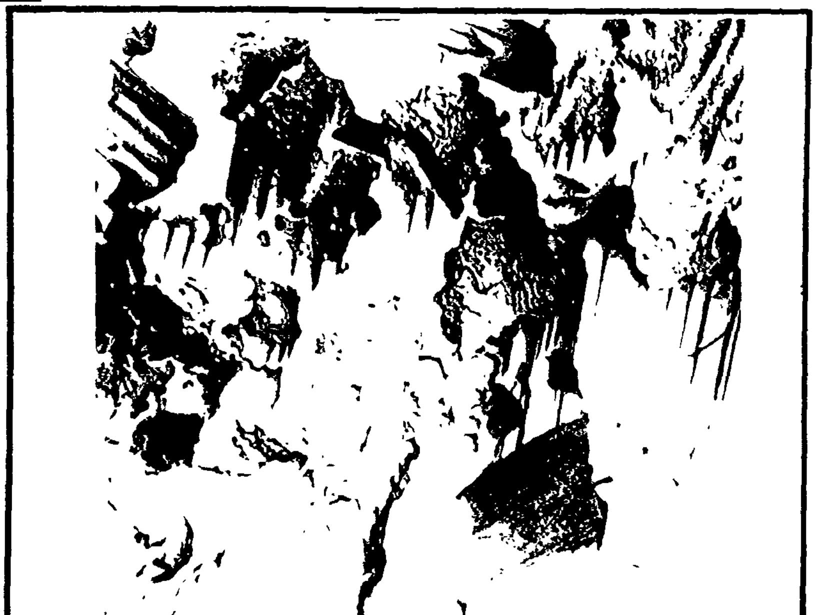
M·A·N STRUCTURAL PERFORMANCE OF TANK DESIGN **Fig. 5**



M·A·N STRUCTURAL PERFORMANCE OF PRESSURE TANKS **Fig. 6**



M·A·N THORNEL 50 - RESIN L 02 Fig. 7

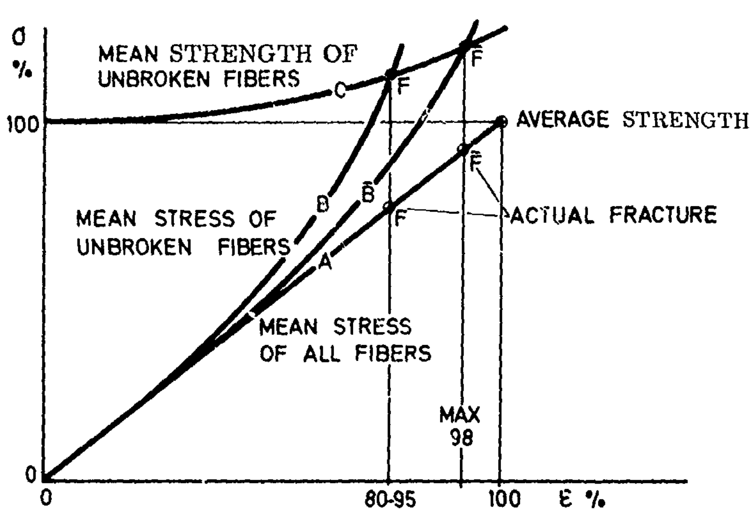


M·A·N THORNEL 50 / PVA-ERLA 4617 Fig. 8

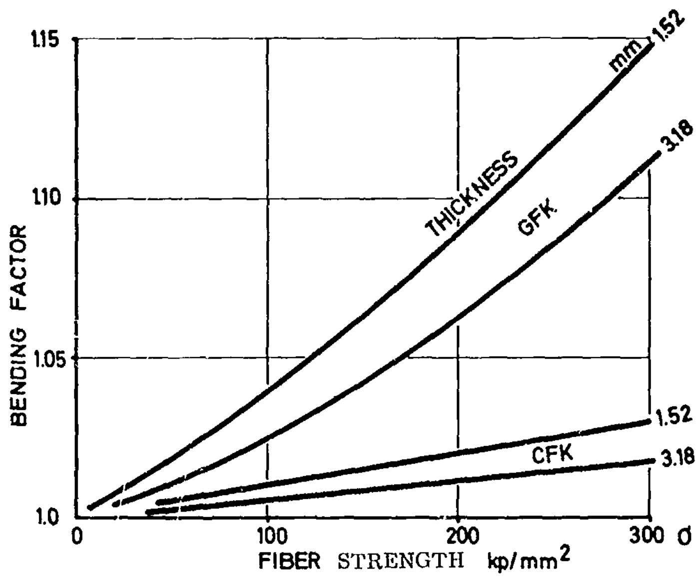
GRAPHIL HT - RESIN L 02 Fig. 9

SIGRAFIL - ERLA 4617 MOD. Fig. 10

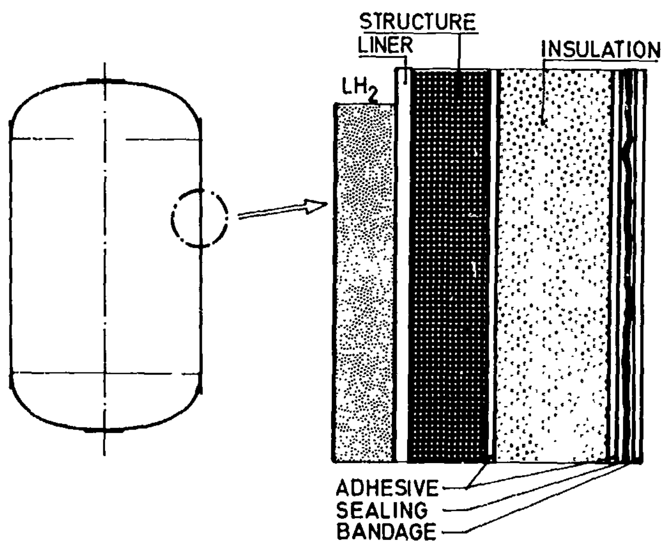
M·A·N SIGRAFIL - RESIN L 02 Fig. 11



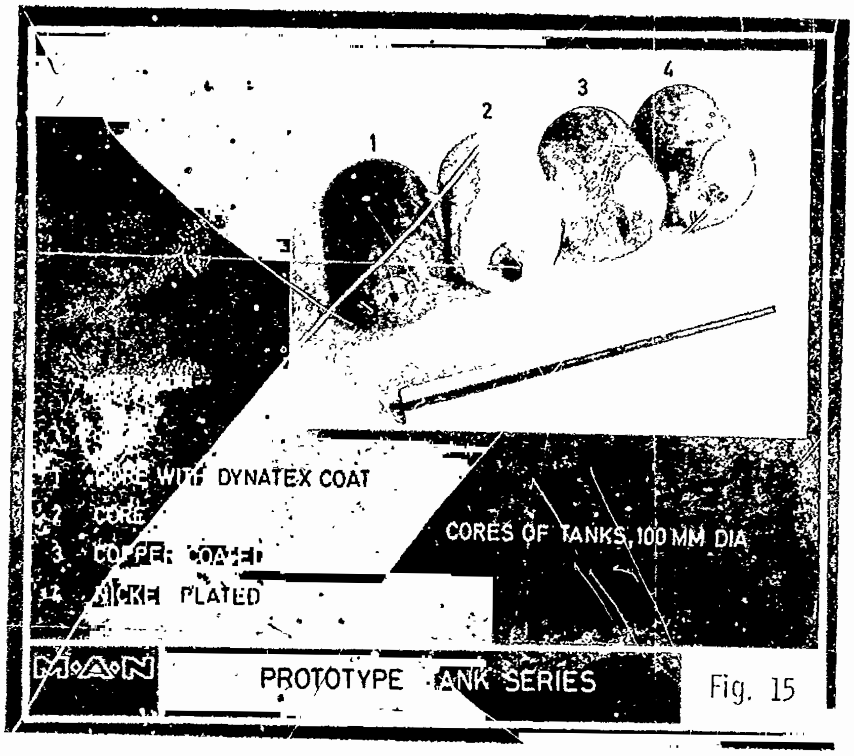
M·A·N STRESS AND STRAIN FOR COMPOSITES Fig. 12

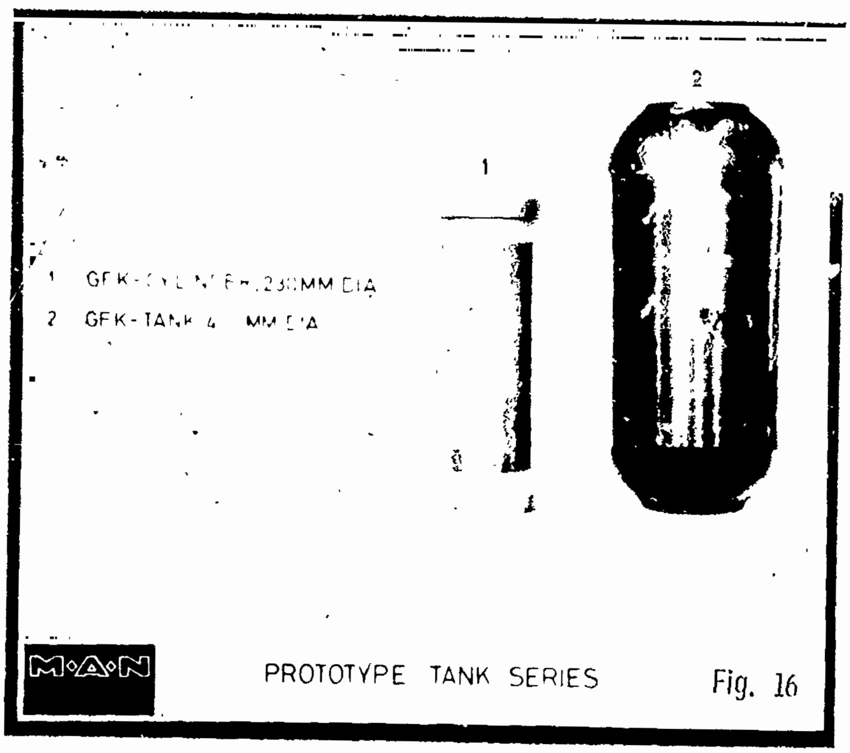


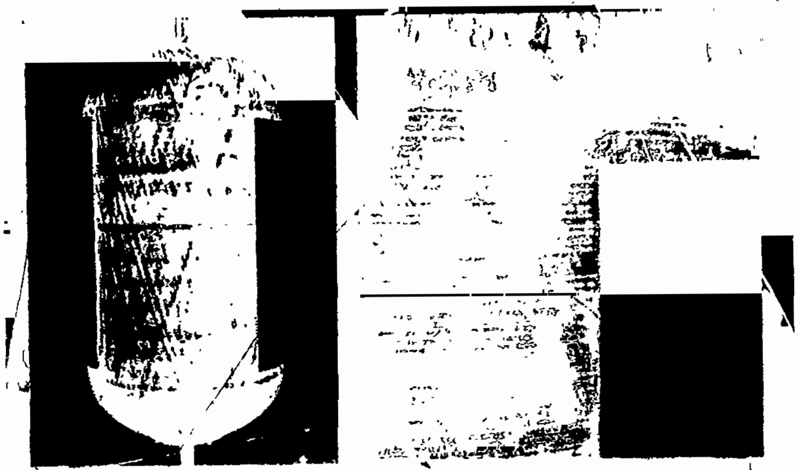
M·A·N STRESS INCREASE ON NOL-RING FRP TESTING Fig. 13



M·A·N STRUCTURE OF FILAMENT-WOUND TANKS Fig. 14


1 2 3 4
DYNATEX COAT
CORE
PLATED
CORES OF TANKS, 100 MM DIA
M·A·N
PROTOTYPE ANK SERIES
Fig. 15


1 2
M·A·N
PROTOTYPE TANK SERIES
Fig. 16

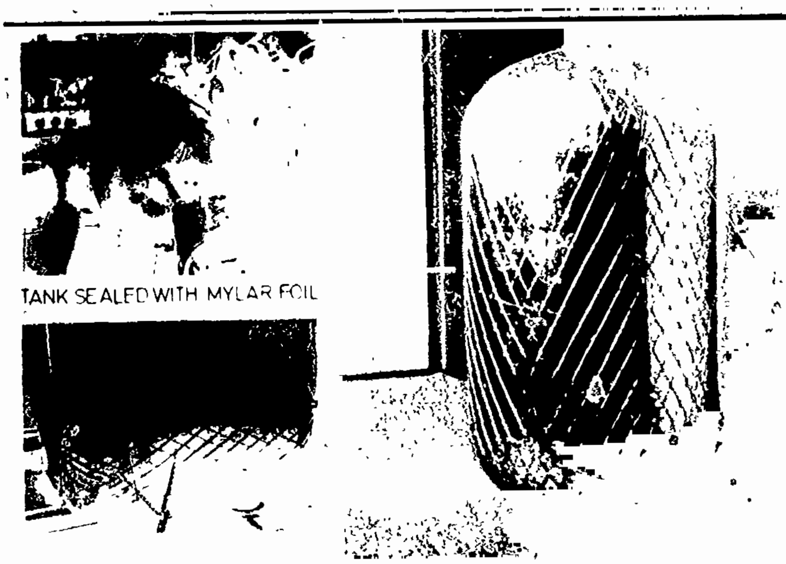


PARTLY INSULATED WITH POLYURETHANE FOAM

SECTIONAL VIEW OF INSULATING PANEL ON TANK

GFK-TANK OF SERIES XIII

Fig. 17



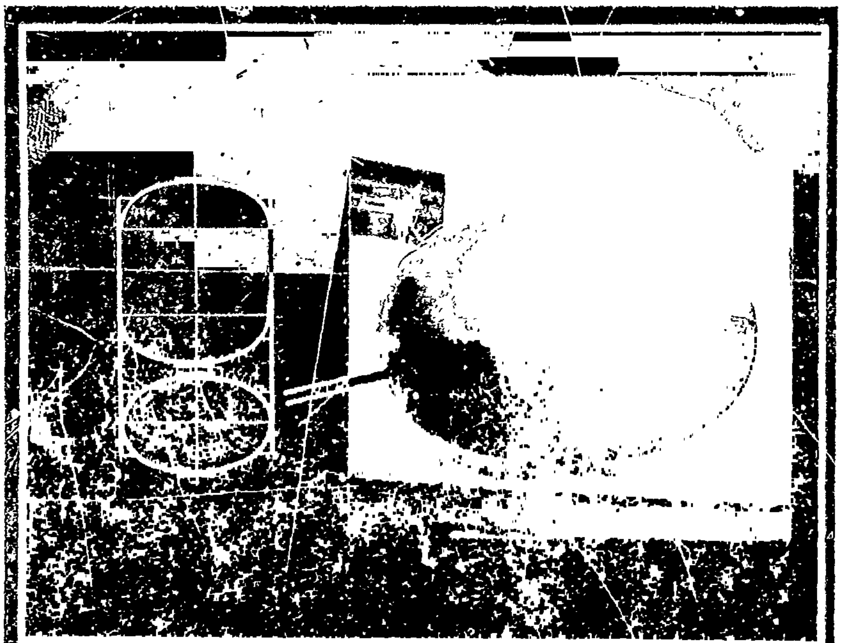
TANK SEALED WITH MYLAR FOIL

APPLICATION OF GFK BANDAGE

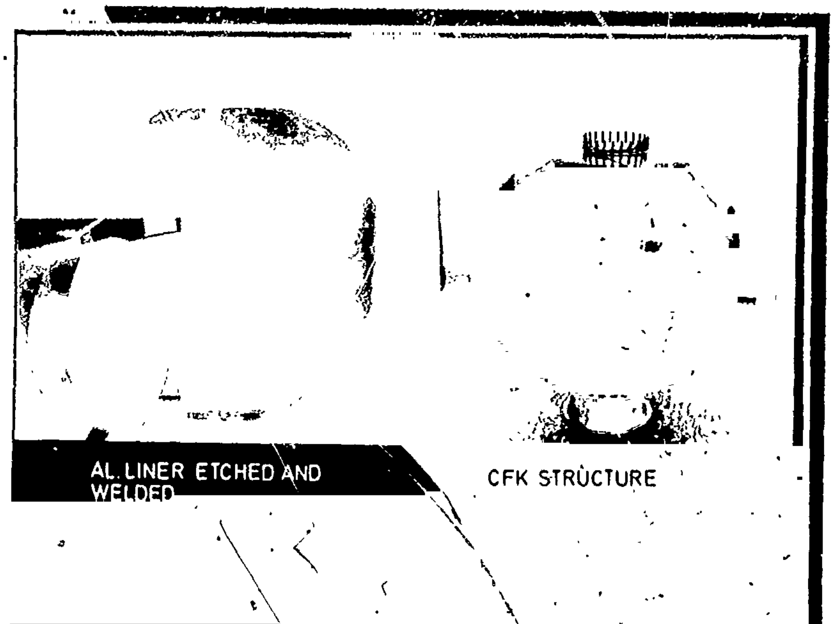
FINISHED GFK TANK

GF.K-TANK OF SERIES XIII

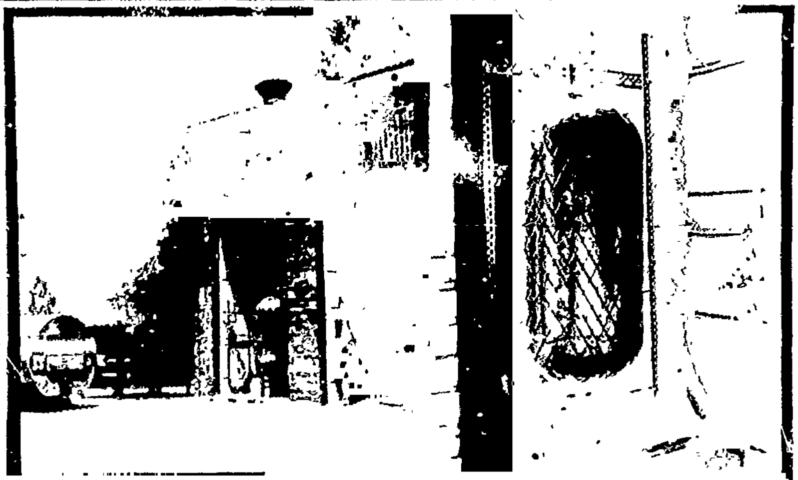
Fig. 18



AL. LINER OF TANK MODULE Fig. 19



CFK-TANK OF SERIES XIII Fig. 20



·VIEW TEST STAND   TANK IN TEST BED

M·A·N TESTING OF GFK-TANK OF SERIES XIII Fig. 21

STRESS MEASURING ON GFK STRUCTURE   FULLY INSULATED TANK

M·A·N GFK-TANK OF SERIES XIII Fig. 22

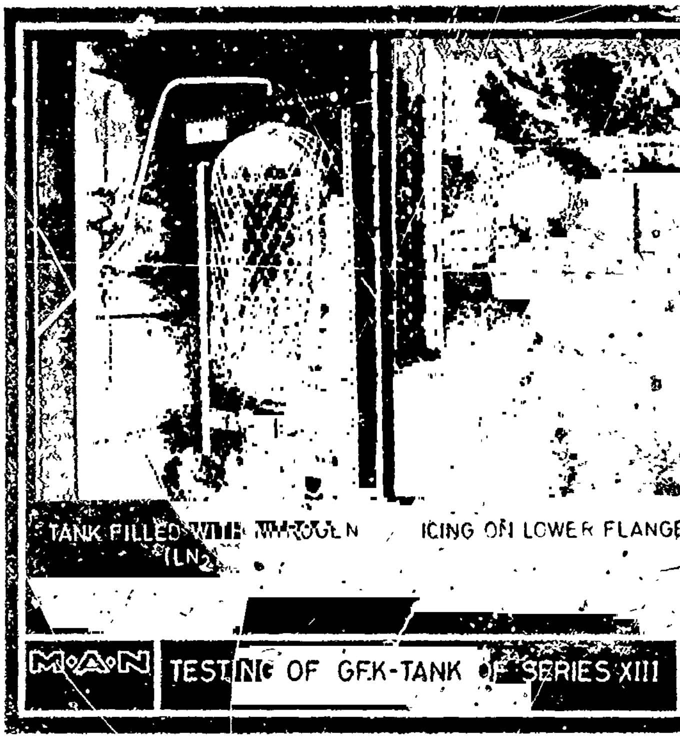

TANK FILLED
ICING ON LOWER FLANGE COVER
TESTING OF GFK-TANK OF SERIES XIII

Fig. 23

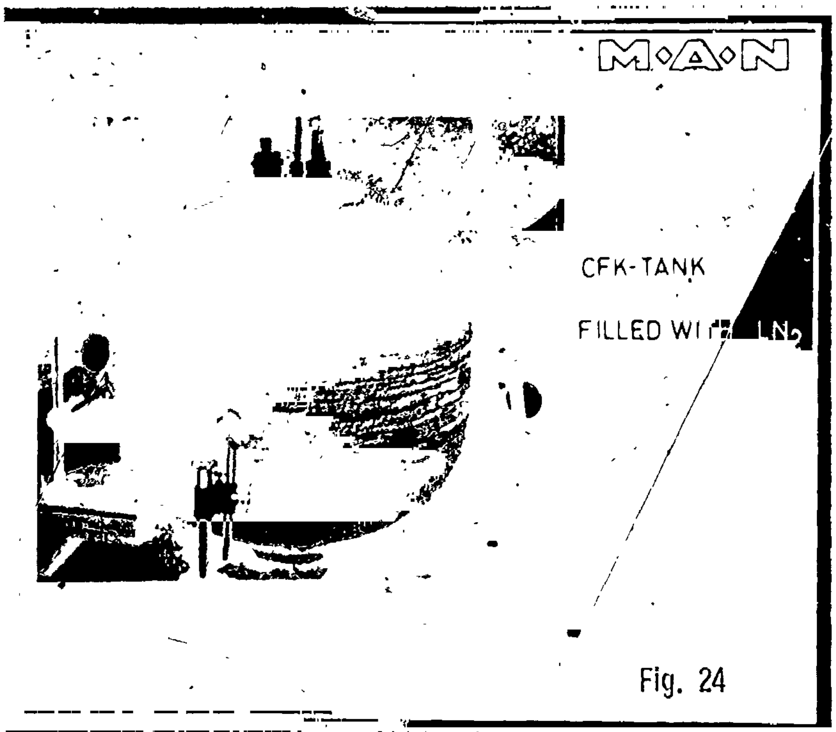
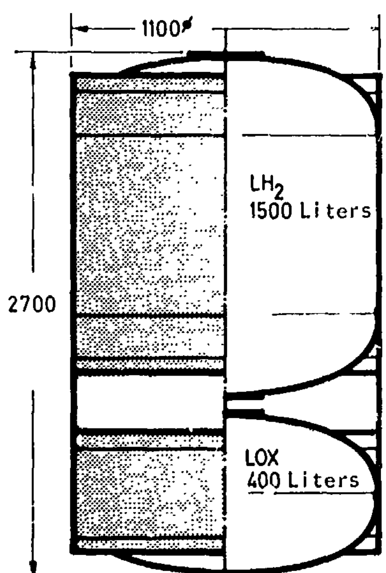

CFK-TANK

Fig. 24

FRAGMENTS OF CFK TANK AFTER DESTRUCTION TEST

Fig. 25



CFK-STRUCTURE, AL-LINER AND CONNECTING STRUCTURE

CFK TANK MODULE

Fig. 26