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THERMAL DESIGN OF THE SPACE SHUTTLE EXTERNAL TANK

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ABSTRACT

The Shuttle External Tank thermal design presented many challenges to NASA and the aerospace industry in meeting the stringent requirements established by the structures, main propulsion systems, and Orbiter elements. The selected thermal protection design had to meet these requirements, as well as take into account ease of application, suitability for mass production considering low weight, cost, and high reliability. This development led to a spray-on-foam (SOFI) which covers the entire tank. This paper discusses the need and design for a SOFI material having a dual role of cryogenic insulation and ablator, and the development of the SOFI over SLA concept for high heating areas. Further issues of minimum surface ice/frost, no debris, and the development of the TPS spray process considering the required quality and process control are discussed.

INTRODUCTION

The External Tank (ET) is the largest of the Space Shuttle Vehicle elements and serves as the fuel tank for the liquid oxygen (LO_2) and liquid hydrogen (LH_2) propellants used by the Orbiter's three main engines. Since both of the propellants are cryogenic (LO_2 at -297°F and LH_2 at -423°F) the Thermal Protection System (TPS) for the ET must function as a cryogenic insulation to limit propellant boiloff and thus maintain the propellant quality. Furthermore, during ascent the ET experiences heating from aerodynamic and plume sources which requires the TPS to also function as an ablator providing protection against structural overheating. The challenge to the ET designers was: provide a low cost, lightweight, and reliable TPS, capable of providing the required protection while maintaining its integrity over the wide range of operating temperatures.

INITIAL ET TPS CONCEPT

Although the current ET design is much the same as the design which existed at the inception of the ET program in 1972, there has been a significant TPS design evolution driven by both changes in requirements and the design maturing. The ET with the initial TPS design as it existed in 1972 is shown in Figure 1. The ET consists of an ogive-shaped LO_2 tank with attached nose cap and a larger LH_2 tank supporting the orbiter and the SRBs. An intertank structure connects the two propellant tanks and reacts the SRB thrust load. Both the intertank and the nose cap form separate compartments which contain various electrical and propulsion components. Exterior to the tank are a multitude of protuberances on an otherwise smooth configuration — propellant lines, cable trays, and interface structural members.

The initial TPS design shown in Figure 1 consisted of BX-250 spray-on foam insulation (SOFI) on most of the LH_2 tank and an ablator, SLA-561s, on high heating areas of the LO_2 tank, the Intertank, and the LH_2 aft dome. Although these materials had been used on previous programs, both were being utilized in a novel way on the ET program.

BX-250 is a low density SOFI (2.0 pcf density) which was used on the Saturn S-II stage and also commercially as a cryogenic insulation. This material is a rigid closed cell polyurethane foam which has excellent insulation qualities at low temperatures. Therefore it is possible to limit the heat transfer through the insulation (and ultimately into the propellant) with a minimum of foam thickness. BX-250 was applied to the entire LH_2 tank (except the aft dome) to control boiloff of the super-cold LH_2 and to prevent the formation and runoff of liquid air. SOFI materials of this type are generally not thought of as being capable of withstanding the high temperatures expected from aerodynamic heating. For example, BX-250 begins to decompose at 255°F whereas the insulation surface temperatures could be expected to reach 1500°F or higher during ascent. Early testing of BX-250 however, showed that it responded favorably to moderate heating rates, and acted similar to an ablator, providing limited protection from ascent heating.

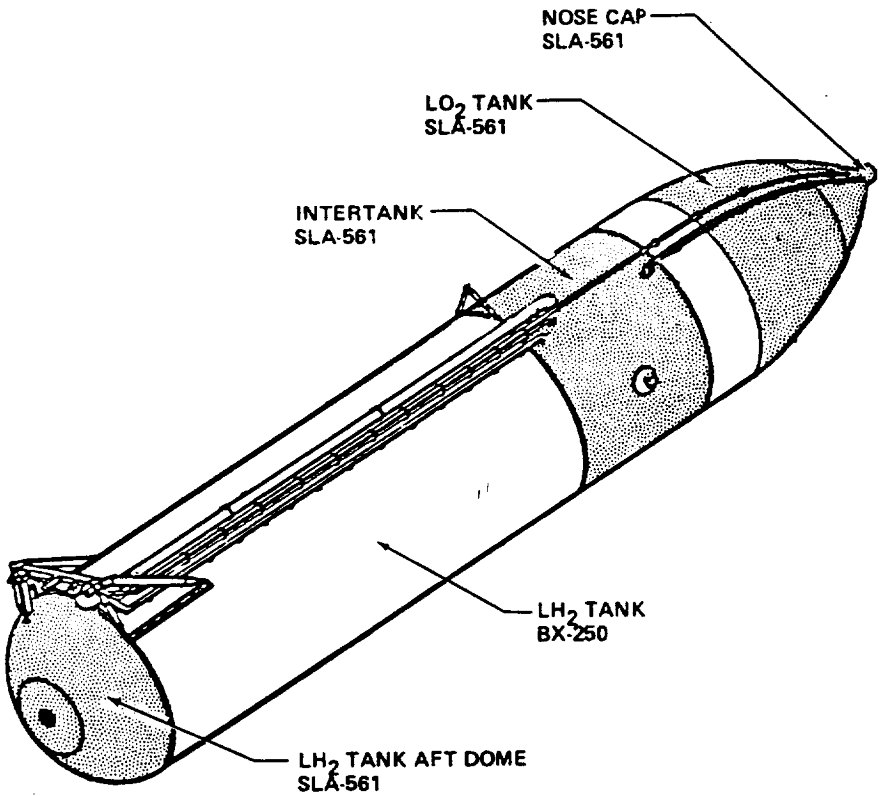

Figure 1. Initial ET TPS Configuration.

An "ablator" is a material which is designed specifically to withstand high temperatures through controlled sacrificial degradation (energy absorption) and is traditionally used in areas which require protection from high ascent or re-entry heating. Ablators can thus withstand significantly higher heating rates than foams, however, ablators are heavy and do not provide the same degree of cryogenic insulation properties. The ablator selected for the ET was sprayed SLA-561 (16 pcf density), a derivative of the light weight ablator developed on the Viking Project.

Prior to the ET program, whenever protection of cryogenic surfaces was involved, ablators were generally applied to separately attached heat shields or standoff panels. This was due in part to the implied problems of strain compatibilities between the substrate and the ablator material when subjected to cryogenic temperatures. The Martin Marietta Corp., recognizing the cost and weight advantage of applying an ablator directly to the cryogenic substrate, performed a feasibility program to develop this type system with SLA561. Based on the successful results of this study, the application of SLA561 directly to the LO_2 and LH_2 tanks was baselined initially and maintained to the current design.

Therefore the early design philosophy for the ET TPS was to utilize the capability of the SOFI wherever possible, supplementing with an ablator applied directly to the cryogenic tank wall. Although this was later impacted by an increase in the predicted ascent heating environments, this novel approach – using a foam insulation as both a cryogenic insulation and an ablator – continued throughout the program.

CHALLENGES IN THE TPS DEVELOPMENT

Since the ET is not reusable (discarded just prior to orbit) the total program cost of the ET is sensitive to recurring (per vehicle) costs, which of course is affected by TPS related labor and material costs. In addition, TPS contributes a small but significant part of the ET weight. The original thermal requirements imposed on the ET TPS design did not present significant challenges per se, since there were state-of-the-art TPS materials and techniques available; however, the weight and cost impacts would have been high. The challenge was to use the existing materials or their derivatives in new ways to develop a highly efficient, lightweight, and low cost system which would be consistent with a mass produced disposable vehicle such as the ET while maintaining the quality and reliability which is required of spacecraft design.

TPS MATERIALS

The first key to an efficient TPS was the use of a SOFI in a dual role of cryogenic insulator and ablator as discussed earlier. Shortly into the program, however, increases in predicted ascent heating quickly exceeded the capability of the original SOFI (BX-250) and the system became unable to accommodate the new environment without a major increase in the use of the SLA-561 ablator. A new urethane modified isocyanurate foam material (CPR-421) was selected to replace the BX-250 foam and concurrently reduce the quantity of SLA-561 required. This new material (selected from commercially available "high temperature" foams) possessed the required ablation characteristics and was compatible with the as-designed production concepts. This material was not flight qualified however and tighter control on processing parameters would be necessary to assure the desired material properties. It should be noted that this material was to be used in an environment unlike any previous commercial use and required considerably greater quality control than in previous applications. Material properties such as consistent density, bondline strength, tensile strength, and ablation characteristics were imperative for success on the ET whereas its low conductivity and durability were always important in commercial uses. Therefore, a complete development and verification program was initiated to characterize this "off the shelf" product. Early in this development program, the material was reformulated as CPR-488 to enhance its ablation characteristics and eliminate any toxic concerns.

A light-weight ablator capable of being applied directly to the cryogenic tank surface was the second key to an efficient TPS. Beginning with the baseline sprayed version, the ET SLA development extended to the installation options of mold-in-place, premold-and-bond, and hand packed in place, each tailored to suit design and production needs. However as the ablator thickness was later increased in response to growing ascent heat loads, problems began to appear with bondline integrity. This led to a substantial analytical and test effort to understand and characterize the stress relationships between the ablator and the substrate at cryogenic temperatures. Because of the inherent variability of material properties in composite mixtures and the low margin in the SLA561 bondline strength at flight conditions, the challenge was to control processing parameters and thus material properties to maintain a positive margin.

ICE/FROST AND DEBRIS REQUIREMENTS

The problem of ice forming on the ET cold surfaces during loading operations and then dislodging during ascent and damaging some other portion of the vehicle was recognized initially as a potential issue. However, no attempt was made in the design to preclude ice until later in the program when a "no ice/debris" requirement was imposed. At that time the LO_2 barrel was bare and was thus a large potential area for ice formation.

Icing on the general tank acreage was eliminated by adding an appropriate layer of SOFI, with the associated weight impact. The prevention of ice at the numerous protuberance areas however was a much more difficult design problem since these areas typically involved movable joints which are difficult to insulate. In addition, SOFI and ablator stress cracking, SOFI divots due to cryopumping, and other minor TPS discontinuities or anomalies which had previously been tolerated now could be a source of ice and/or TPS debris which had to be addressed. The challenge of preventing ice and TPS debris was compounded by the fact that most of the vehicle and ground systems had already been designed.

ET TPS DESIGN

Today's TPS design, although greatly evolved from the original concept, still adheres to the basic design philosophies of a SOFI and ablator system. Because of increased heating environments and the requirement to preclude ice formations on the tank, 1 in. or more of SOFI is now specified for all cryogenic surfaces. In those local areas where the capability of the SOFI is exceeded (generally near and around protuberances) an underlayer of ablator is retained. The latest ET TPS design showing the areas of SOFI and ablator coverage as well as other TPS elements is shown in Figure 2. The majority of the ET TPS is CPR488 SOFI, used on all exterior acreage except the aft dome. Ancillary to this, urethane foams (BX-250 or PDL-4034) are used for closeouts, ramps, and the domes within the intertank due to these foams' liberal application parameters. Another commercially available polyisocyanurate SOFI, modified for ET use (NCFI 22-65) was selected for the LH_2 aft dome where the high plume heating rates exceeded the capability of CPR488.

The ET TPS uses two ablator materials. SLA-561 is for primary use on the general acreage under the SOFI and most attachments, and MA25s is used on a few select protuberances which experience heating in excess of the capability of SLA-561.

The SOFI design thicknesses are basically defined by prelaunch requirements where maintaining good quality and stable propellants and minimizing ice are the primary considerations. A low heat transfer rate through the insulation is desired to maintain the insulation surface temperature above freezing (to preclude icing) and to limit propellant boiloff. Figure 3 shows propellant boiloff rate and a

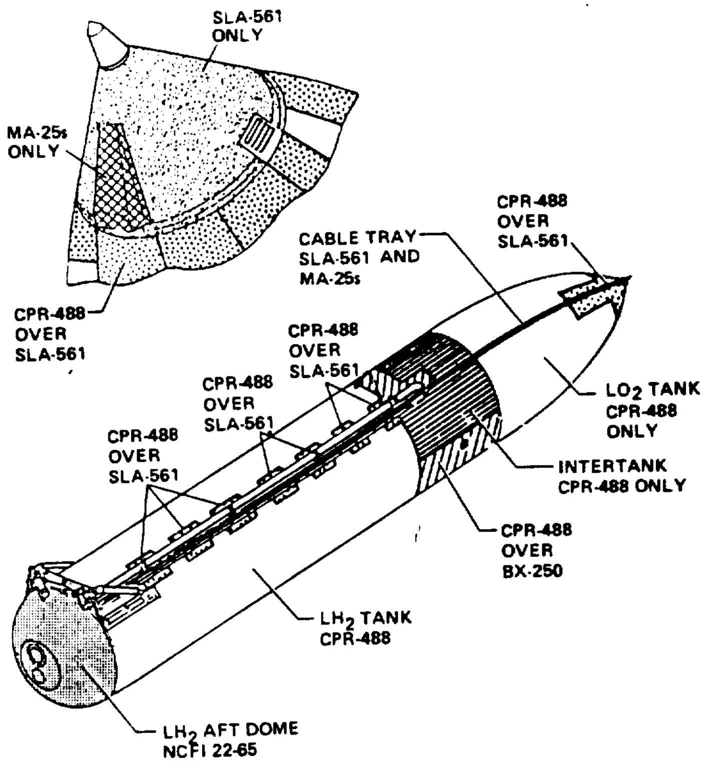

Figure 2. TPS Configuration.

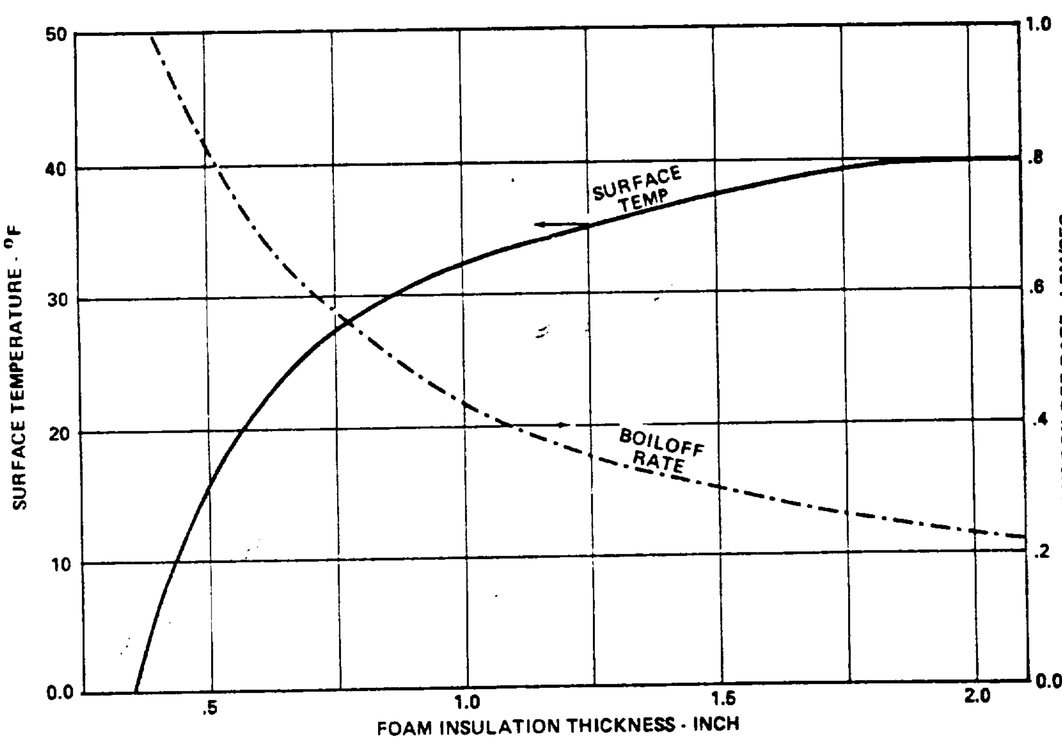

Figure 3. Typical Foam Insulation Performance.

typical surface temperature versus CPR-488 SOFI thickness for the LH_2 tank. The surface temperature is based on a set of combined ambient environments (i.e., wind, temperature, humidity, etc.) which represent the 95% cold conditions at KSC. As seen from this data, a 1-in. thickness of SOFI is very effective in limiting propellant boiloff and maintaining a surface temperature close to ambient (ambient is 55°F for this case). Increasing the thickness above 1 in. provides minimal improvement considering the corresponding weight increase. Therefore, a 1-in. thickness of SOFI was selected for the LH_2 and LO_2 tank acreages.

The ascent environments primarily define the ablator thicknesses and the thickness of the SOFI on the LH_2 tank aft dome. Maintaining the primary structure and subsystem components within the design temperature limits and minimizing unusable propellants due to stratification are the primary considerations. The thermal environments considered include: aerodynamic heating, radiant and convective heating from the engine exhaust plumes, SRB separation motor plume impingements, radiative and conductive transfer with the other elements of the Shuttle vehicle, and internal heating due to the autogenous tank pressurization gas. The TPS requirements were determined analytically based on these environments and material properties derived from test data.

Another function of the TPS occurs during ET entry when structural temperatures and tank pressures contribute to the ET fragmentation and the subsequent debris size and impact footprint. Except for some sections of the cable tray where MA25s ablator was added to protect RSS charges, the TPS defined by ascent requirements is adequate to meet entry requirements.

Other elements related to the thermal design of the ET include thermal isolators on LH_2 tank attachments, cryopumped propulsion lines, electrical heaters primarily for ice/frost protection, and several heated purge systems for compartment conditioning and also for ice/frost protection. The icing problem at the several brackets and attachments which penetrate the general acreage SOFI was eliminated by various combinations of insulation, isolators, heaters and purges.

TPS DEVELOPMENT AND VERIFICATION

The development and design verification effort for the ET TPS involved many activities ranging from material ablation testing to development of manufacturing techniques. All of these efforts can not be addressed adequately in a paper of this scope, therefore the following is an overview of the more important and unique aspects of this effort.

PROCESS DEVELOPMENT

As discussed earlier, the original SOFI, BX-250 was replaced by a new material CPR-421 which was subsequently reformulated as CPR-488. Results from the CPR-488 development program revealed that the critical materials properties required by the flight mechanical and thermal loads were predominantly influenced by applications processing parameters. Systematic material evaluations were undertaken to develop the acceptable ranges of critical processing conditions. It was determined that material temperatures at application had to be controlled to near 135°F and tank substrate temperatures to 140°F. The spray envelope of ambient temperature and relative humidity was also found to be more restricted than the conventional BX-250 foam as shown in Figure 4. In addition, spray overlap time and gun-to-tank separation distance was also found to be critical and needed proper controls.

With these very restrictive processing controls in place, critical material performance parameters, including structural strain compatibility, insulation efficiency, and ablation rates, were all consistent and repeatable.

The SLA-561 ablator material exists in three application modes; bonded premolded panels, a sprayable mix for large or complex surfaces, and a trowelable version for small closeouts and repairs. The primary focus of the development program was to develop a process for the sprayable ablator material which yielded repeatable physical properties of strength and density. Systematic material evaluations determined the need to seal the surfaces of the cork filler particles to minimize the absorption and subsequent retention of the spray solvent. The retention of this solvent was found to inhibit the cure of the silicone resin binder with resultant large variations in material properties. A cork coating process utilizing an epoxy resin was developed to control the solvent absorption and provide the required consistency in materials properties.

MANUFACTURING AND QUALITY CONTROL

The key to meeting high volume production, schedule, and budget constraints was to facilitate for critical TPS manufacturing operations using automated techniques in large environmentally-controlled TPS application cells. The two-component SOFI is sprayed using totally automated apparatus, where a

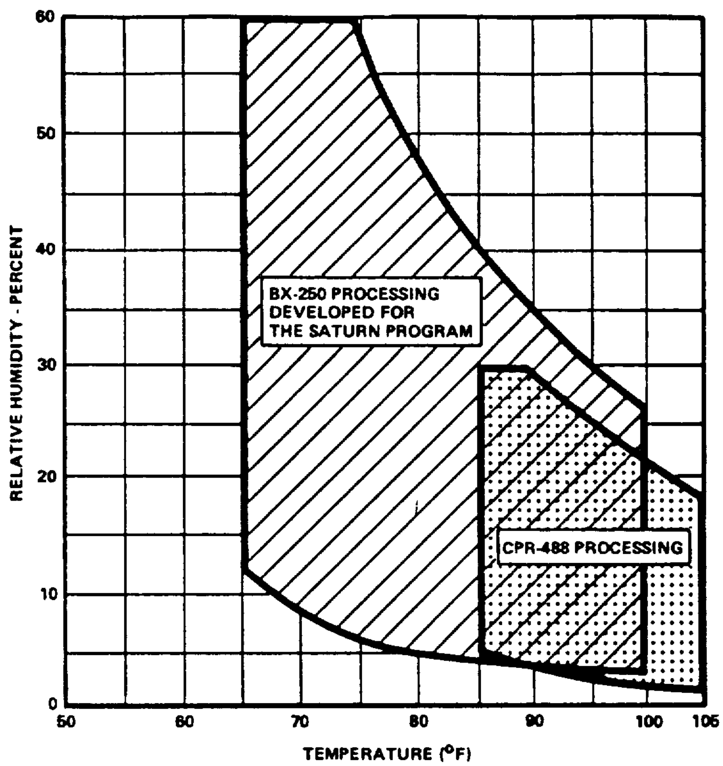

Figure 4. Temperature Versus Relative Humidity Requirements.

computer system is employed to control and monitor spray applications to the LH_2 tank and the LO_2/IT subassembly. All parameters during production runs are controlled and recorded throughout the spray operation. Each tank is coated as a single unit on a large rotating turntable. Spraying begins at the bottom and the gun carriage progresses upward at a predetermined rate to produce a uniform foam thickness of shingled layers. This technique with tight control on the processing parameters enables the system to produce a finished product on one pass with no machining of the "net sprayed" surface required afterward.

In acreage applications, SLA ablator is premolded to required thickness, cured, and then bonded to the tank using vacuum bagging techniques since this pliant material conforms to the large curvature. In complex configuration/large applications, such as the nose cap and the crossbeam, the SLA, thinned to a slurry with solvent, is hand sprayed, heat cured, and trimmed. Small complex components such as cable tray brackets have mole-in-place SLA applied.

Quality control for both SOFI and ablator application is effected by inspections, testings, and process control starting with material receiving and continuing through application to the tank. In the case of automated processes, all parameters are recorded and the process is automatically aborted if parameters are out of range. For material formulation and applications, witness coupons are simultaneously produced and later tested for certain key properties. Selected post-manufacturing tests including TPS plug pulls and an adhesive plough test are also performed at selected tank locations.

The most critical defect which must be controlled and is also difficult to detect, is a weak TPS bondline. This type of defect typically manifests itself only under loaded cryogenic conditions at which time the TPS may debond. Therefore, the first several tanks produced have had a cryogenic "TPS proof test" performed at KSC, which is basically a propellant loading test designed to load the TPS above flight loads. The successful results from these tests have verified the quality and process controls currently in place, and the "TPS proof test" are to be discontinued.

TPS STRUCTURAL CONSIDERATIONS

The ablator material, which is an elastomer at room temperature undergoes a change of state at approximately -180°F and becomes brittle due to the silicone resin binder passing through the brittle transition temperature. In this state the strain capability of the ablator is limited and excess loads, particularly in thick layers undergoing flexure, may cause the material to crack and ultimately debond. During the ET design phase, the predicted heating environments increased significantly driving the program to ablator thickness in excess of 0.75 in. under which the induced loads were near the strain capability of the material with little margin remaining. To regain an acceptable TPS structural margin, a re-ordering of the launch procedures was implemented to take advantage of the change in the ablator's properties with temperature.

Normally the ET is loaded with little or no ullage pressure in the tanks. The tanks are then subsequently pressurized for launch and ascent to meet structural and propellant delivery requirements. The two primary loads applied to the ablator are due to the cryogenic chilldown of the tank (thermal strain) and to the internal tank pressure (substrate strain). It was shown analytically (and by test) that if the strain due to tank pressurization is applied prior to loading when the ablator is near room temperature, rather than after loading when the ablator is at cryogenic temperatures, that the resultant ablator structural margin is increased. This is because the final stress in the ablator is dependent on the temperature varying material properties that exist when the loads are applied. Therefore, the current propellant loading procedure for the ET LH_2 tank is to pressurize prior to loading. This provides an increase in TPS margin at the expense of a longer propellant loading timeline. Reductions in ablator coverage since the first vehicle have lessened the need for this procedure, and continuing efforts to reduce ablator application will hopefully eliminate the need altogether.

ET TPS ABLATION TESTING

Design and testing of ablative heat shields for spacecraft had traditionally been performed by testing materials in plasma arc facilities and correlating the erosion and pyrolysis reactions with kinetic chemical reactions in complex computer programs. This technique was necessary because most applications involved subjecting materials to the wide range of environments associated with re-entry or ballistic applications which did not always match the test (plasma arc) environment. Although this approach was used in the initial ET design phase it was subsequently replaced with a simpler semi-empirical design method.

The most significant difference between the ET program and other programs was that the primary ET TPS material was an extremely low density polyurethane foam rather than the normal higher density ablators. The foam recession appeared to be more of a char erosion process than actual ablation. In addition facilities and techniques were used which allowed TPS testing in environments which closely approximated those expected during flight.

The von Karman Gas Dynamic Facility's tunnel "C" at the Arnold Engineering and Development Center (AEDC) was selected to characterize the ET TPS as an alternative to plasma arc testing. This facility was chosen because it had the capability of matching flight recovery temperature and had a flow field which could produce turbulent flow heating and shear conditions over the range of flight environments. It was also capable of testing large samples (17 x 24 in. versus 6 x 6 in. for plasma arc) and had multiple test specimen capability.

Ablation test data obtained from this facility was correlated with test conditions and an empirical relationship established between material recession rate and heating rate. This relationship was then used with the predicted flight heating profile to perform the flight erosion analysis. This simplified technique (rather than the previously mentioned complex computer programs) was possible due to the close approximation of the test to flight environment offered by this facility.

When the design environments were increased, however, the AEDC facility could not produce turbulent heating at levels as high as required, particularly for ablator testing. The AMES 3.5 ft hypersonic wind tunnel with the Mach 7 nozzle was selected for TPS testing since it could achieve the higher environments and offered most of the same desirable aspects as the AEDC facility. Tests in the Ames facility provided the ablation data at the higher heating rates, and also duplicated and provided confidence in the AEDC simulation.

TPS testing in these facilities also included tests with simulated interference heating. This was accomplished by generating shock waves which were allowed to impinge on the TPS sample. Another approach used was to include a protuberance directly on the TPS sample. The heating rates obtained in these tests, however, were often too high to test SOFI materials due to the inherent tunnel conditions (Mach numbers of 7 and 10 as compared to 4 in flight). Lower heating rates could only be obtained with flat plate (shockless) testing although in many cases they represented an interference heating area on the ET. This presented a design concern, since the flat plate type testing was not representative of these special design cases, and because of this a substantial design margin was used for TPS sizing.

In early 1981, AEDC modified the Tunnel "C" facility with the addition of a Mach 4 nozzle. No facility previously existed at Mach 4 with corresponding total temperature and heating rate capabilities.

Test hardware was developed and calibrated to obtain the shock impingement environments in a flow field duplicating the flight environments at the lower heating rates needed for SOFI testing. Results of tests on the SOFI materials have provided the data and confidence to further utilize the ablative capabilities of CPR-488 with a subsequent substantial reduction of ablator coverage required on the ET.

Testing techniques are currently being developed to further reduce the ablator coverage on the ET. The technique involves a specimen design which will form an extension of the Mach 4 nozzle and take advantage of the relatively thick boundary layer in the nozzle (as compared to testing on a wedge shaped holder in the stream centerline). Specimens simulating the ice/frost ramps on the hydrogen tank at the cable tray/barry mount locations will be tested to demonstrate that the protection provided by the ramps is sufficient to virtually eliminate ablator coverage on the hydrogen tank at the attachment region of these brackets.

TPS VERIFICATION TESTING

The verification test program for the ET TPS included the wind tunnel tests discussed above together with several cryogenic, radiant heating, and combined environment tests. These tests were designed to verify the TPS integrity under the various predicted flight induced environments. There was no one test (other than flight) which simulated all of the pertinent flight parameters. Confidence in the TPS system was achieved by the successful results of these tests taken together.

Minitank tests were used to evaluate TPS cryo-strain compatibility, primer and TPS adhesion, and TPS cracking and susceptibility to cryopumping. The minitanks were 3-ft diameter aluminum tanks with TPS applied and were tested under repeated cryogenic fill, drain and pressurization cycles. These tests did not simulate any ascent pressure, heating or acoustics loads.

A larger 10-ft cryogenic tank was also tested similar to the minitanks to assess any large scale application issues. The 10-ft tanks, like the minitanks, were tested with LH_2 under repeated cryogenic cycles. The 10-ft tank also included a radiant heat test to assess TPS recession and propellant quality on a large scale tank. Note that the 10-ft tank was the largest scale application of flight type TPS prior to STS-1.

Radiant heating tests were conducted to verify the TPS recession characteristics under the aft dome environment where heating was due primarily to the exhaust plume radiation and recirculation, rather than aeroheating as simulated in the windtunnel tests. These tests were conducted in two facilities; one simulated radiant heat and acoustics, and the other radiant heat and ascent pressure decay.

The key element in the TPS qualification was the "combined enviornment" tests conducted on 4 x 4 ft TPS panels configured to represent the substrate and TPS in specific critical areas on the ET. These panels were subjected simultaneously to biaxial substrate loads, cryogenic backface temperature, ascent heat load (radiant), and either acoustics or ascent pressure. The panels were tested in a thermal/vacuum chamber, and/or in a thermal/acoustic facility depending on the specific test objective. Both facilities employed a large load cell structure which could be programmed to induce biaxial load profiles in either tension or compression. These were used to simulate various degrees of predicted flight substrate loads to demonstrate the TPS structural margin. The panels were cooled with liquid helium to simulate hydrogen tank substrate temperatures and the flight heat loads were simulated by an infrared lamp bank. These unique tests allowed the TPS to be subjected to nearly all the flight conditions (except aeroheating) and on a scale large enough to verify production methods.

FLIGHT RESULTS

The first six ETs were instrumented to provide flight verification of the thermal analysis and TPS design. The prime interest in evaluation of TPS performance are the heating rate and structural temperature data. Many of the instruments were intended to measure the conditions in the boundary layer on the ET TPS surface. The challenge was to mount this instrument such that it would not create a disturbance which would alter the parameters being measured, and such that the instruments would not be exposed to cryogenic temperature which would render them useless. Since the general TPS thickness on the ET is 1 in., there was very little room within the ET moldline to mount the transducers and still maintain isolation from the cryogenic tank wall. Furthermore the TPS surface may be receeding due to heating during flight which would uncover the instruments. To overcome these problems, the transducers were mounted in hardened instrumentation islands as shown in Figure 5. The islands were "hardened" to the aeroheating environment by surrounding the island core with a non-receeding ramp which would maintain a constant geometry during ascent. The transducers were mounted within the core which was isolated and insulated from the tank wall. This resulted in transducer temperatures that were acceptable and a stable surface surrounding the measurements.

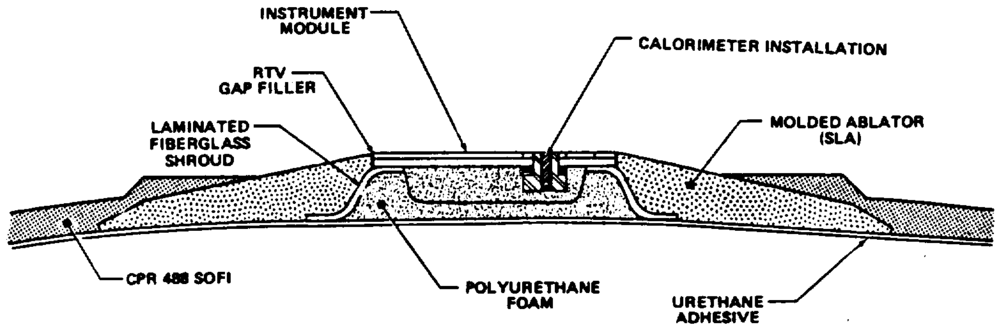

Figure 5. Hardened Instrumentation Island.

Flight instrumentation included aerodynamic heating measurements at 30 locations on the tank sidewall and at 8 locations in the base region. In addition, temperatures at 59 locations on the tank structure were also measured during flight. A comparison of the predicted temperature and measured temperatures from STS-5 for the LO_2 tank and LH_2 tanks are shown in Figures 6 and 7, respectively. The good correlation was part of the total TPS verification. In addition to the flight instrumentation data, cameras mounted in the orbiter were used to photograph the ET during ascent and after ET/orbiter separation. These photographs confirm the location of the higher heating regions and the adequacy of the ET TPS.

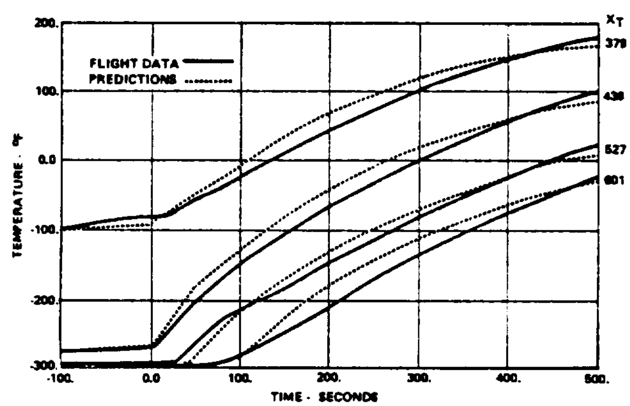

Figure 6. LO_2 Tank Structural Temperatures.

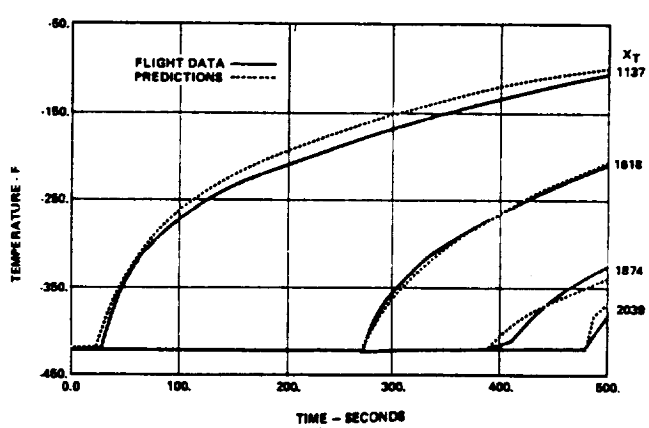

Figure 7. LH_2 Tank Structural Temperatures.

SUMMARY AND CONTINUING CHALLENGES

The major remaining challenges of the ET TPS design are to further reduce cost and weight and to enhance launch operations. In 1982, the first lightweight ET was delivered with a reduction of 6000 lb (including TPS) from the vehicle weight of the first series of tanks. Additional improvements are near term, such as the ablator reduction effort discussed earlier. Other efforts include the reduction of design heating rates and alternate designs which would not require ablators. This is basically a "fine tuning" of the design based on the experience and confidence gained from both production and flight results.

Components which still receive a covering of SLA will have automated ablator application using one of the newly developed methods of GIM, VIM or SAM (gas injected, vented injected, screen and applied molding), each of which eliminates final machining. Components requiring foam coverages or ramps made of foam will use the industrially familiar RIM (reaction injection molding). These are processes which will save material, reduce touch labor, and yield consistent sub-products.

Alternate and back ups to the primary TPS materials and their precursor ingredients are being identified and qualified in the event of supplied problems. Also, since the SLA-561 is essentially being underworked in ET environments, efforts are underway to tailor formulations into cheaper/lighter ablators (and/or cheaper/heavier foams) to replace the SLA-561 completely.

Efforts related to the enhancement of launch operations include the elimination or simplification of TPS closeout required at the launch site, simplification of the active thermal control systems (purges and heaters), and elimination of the residual Ice/Frost problem, particularly at the Vandenberg launch site.

Based on the present TPS design, weather conditions at VAFB result in a much higher probability of ice formation and thus launch delays than at KSC (23% versus 5%). Increasing the SOFI thickness would decrease the icing probability, however, the weight impact would not be acceptable at VAFB where Shuttle performance is already marginal. Many concepts were investigated to decrease the probability of forming ice. The concept finally selected, based on an analytical effort performed by CHAM of North America, Inc., consisted of two hot air jets directed vertically between the ET and Orbiter. The two jets entrain and mix with the ambient air and result in a warm high velocity flow over the tank surface which maintains the surface above freezing. The concept is currently undergoing scale model testing and the results look very promising.

The challenge of developing an efficient TPS system for the ET has been accomplished. Additional effort is continuing to upgrade the design based on flight and production experience. Today's ET TPS designers and engineers are addressing the technologies which will assure the future delivery of an already good product.

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