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SHUTTLE SYSTEM ASCENT AERODYNAMIC AND PLUME HEATING

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ABSTRACT

The Shuttle program provided a challenge to the aerothermodynamicist due to the complexity of the flowfield around the vehicle during ascent, since the configuration causes multiple shock interactions between the elements. An extensive wind tunnel test program was required to obtain data for the prediction of the ascent design heating environment which involves both plume and aerodynamic heating phenomena. This paper discusses the approach for the heating methodology based on ground test firings and the use of the wind tunnel data to formulate the math models.

INTRODUCTION

The Space Shuttle is America's most versatile space vehicle. It is the first to be designed for reusability. This concept of reusability not only provided the economic payoffs which led to the existence of the Space Shuttle, but also provided the designers with unique challenges requiring innovative solutions. The economic and reuse considerations, in conjunction with the ambitious launch schedule (up to 60 launches per year originally projected), dictated the parallel burn concept and required that each element's design enable ease of manufacture and assembly.

The Space Shuttle (fig. 1) is composed of elements which include an Orbiter, three main engines (SSME), two solid rocket boosters (SRB) and an external tank (ET). The complexity of this arrangement required an extensive wind tunnel test program. This wind tunnel test program provided data to predict the design heating environment for the launch vehicle. The design evolution of the launch vehicle configuration and the wind tunnel test program is discussed in reference 1.

The configuration of the Space Shuttle provided unique challenges to the aerothermodynamicists. Never before has there been a launch vehicle with so many elements and one held together with so many struts. The flowfield around and through this configuration, compounded by the presence of interacting shocks, is exceedingly complex. Adding to the complexity of the flowfield is the presence of distinctive protuberances on each element quite different from the aerodynamically shaped fairings used on previous launch vehicles. As examples, the SRBs have several thousand boltheads protruding above the skin line and several structural rings of T-shaped cross section which present a thin "lip" to the approaching flow leading to high localized heating. The ET has cable trays and various fuel and pressurization lines which are elevated above the surface of the tank. Also the spray-on foam insulation (SOFI) which covers the ET results in a rough and wavy external surface.

The ascent design environment involves two types of heating phenomena: plume heating and aerodynamic heating. Plume heating consists of radiative and convective heating from the solid rocket motor (SRM) and main engine plumes. Plume radiative environments were generated analytically with the use of several different computer models. The plume convective heating environments were developed with the use of a combined analysis/wind tunnel approach. The aerodynamic heating predictions were developed from math models formulated on wind tunnel data obtained for a range of angles of attack and yaw, Reynolds numbers and Mach numbers.

TRAJECTORY

Choosing a thermal design trajectory for ascent was an important challenge faced by the aerothermodynamicists. Total system integration analysis cycles are required to develop trajectories. Thus, trajectories were not frequently updated through the Shuttle design phase. Each cycle provided trajectories for a myriad of conditions such as launch from the Kennedy Space Center (KSC) and Vandenberg Air Force Base (VAFB), all potential flight missions, nominal and abort cases, and cases with dispersed atmospheric conditions, wind intensities, and wind directions. Thermal assessment showed the mission 3A trajectory launched from VAFB to be the most severe. Characteristics of this thermal design trajectory are: 1) launch into a polar orbit with a 32,000 pound payload; 2) a 95 per-

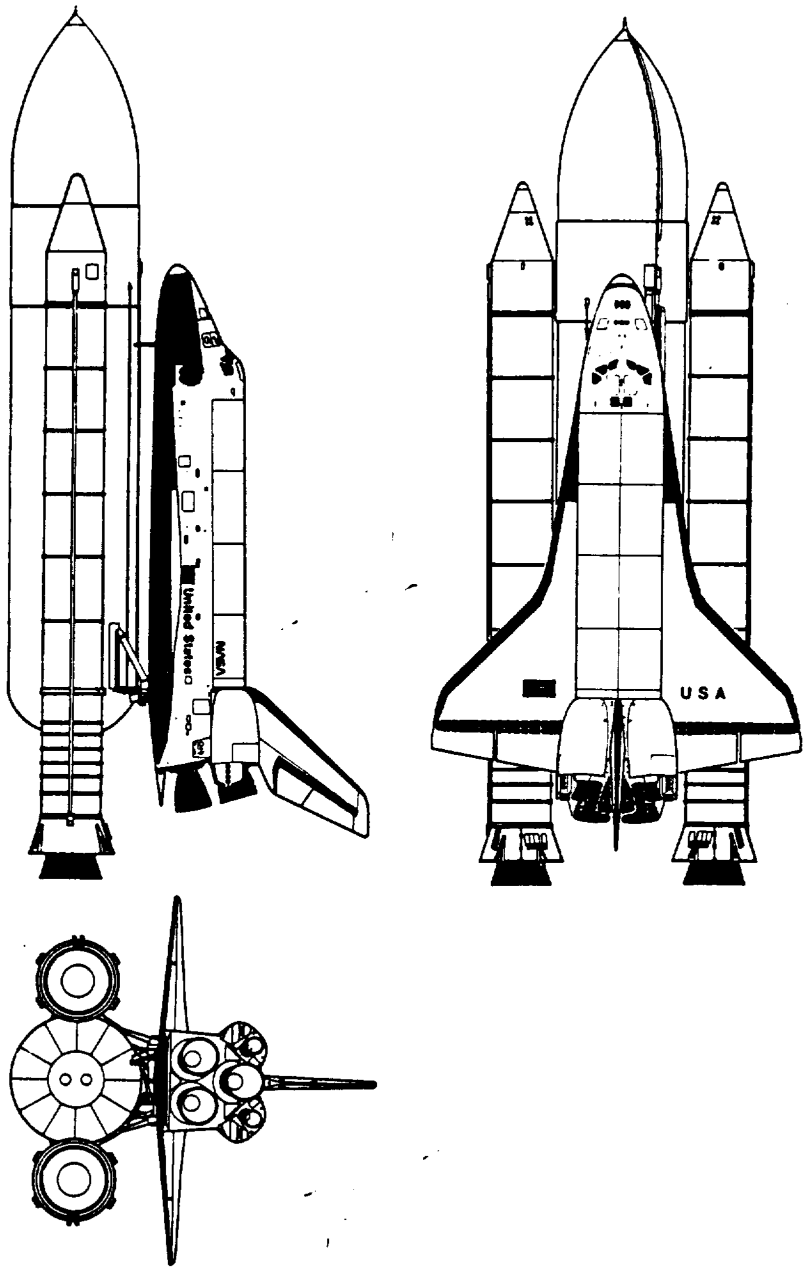

FIGURE 1.- SHUTTLE LAUNCH CONFIGURATION.

centile wind profile with a right quartering head wind direction; 3) the equivalence of 3σ dispersions on parameters that affect the trajectory (atmosphere, thrust, guidance and control parameters, etc.), and 4) one SSME out at 260 seconds into flight necessitating an abort-once-around (AOA) where the ET is separated at an altitude of 57 nautical miles and the Orbiter makes one revolution before landing at the launch site. Some properties of this thermal design trajectory are shown in figure 2. The heating indicator (q_{HI}) which is the heating to a one-foot sphere flying this trajectory, shows that the highest heating rates are associated with first stage flight. Figure 3 shows the envelope of possible vehicle attitudes during first stage flight. The excursions in sideslip angles around 115 seconds are caused by the potential differential thrust in the SRB's during tailoff.

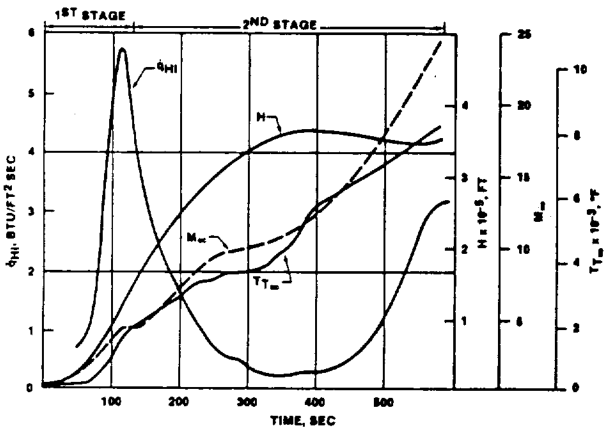

FIGURE 2.- ASCENT AERODYNAMIC HEATING DESIGN TRAJECTORY.

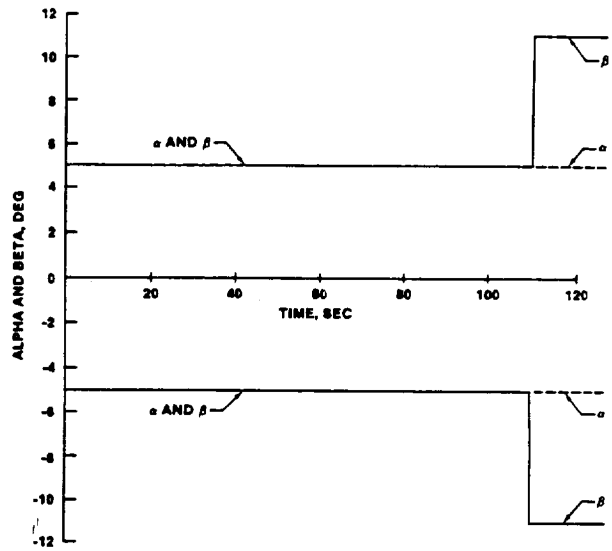

FIGURE 3.- ENVELOPE OF POSSIBLE ATTITUDES - FIRST STAGE FLIGHT.

During the second stage flight, many abort conditions are possible. Early in this portion of the flight, the Orbiter/ET can be turned around for a return to the launch site. This maneuver, called a return to launch site (RTLS) abort, is very risky and involves flying backwards into the SSME plumes. If an abort condition occurs later in the flight, an AOA or a Transatlantic (or Pacific) abort landing (TAL) would be elected depending on the number of SSME's which fail. Each of these aborts involves different flight conditions and were analyzed in the design cycle.

PLUME HEATING

An extensive effort to accurately predict the ascent base heating environment was undertaken early in the Shuttle program. A paper documenting the preflight Shuttle base heating methodology is given in reference 2. The ascent base heating environment is a combination of SSME and SRM plume radiation, freestream air convective cooling, and reversed plume flow convective heating. Each base region design point receives different levels of radiation and convective heating depending upon its location relative to the plumes, base gas absorption, structural blockage, general base configuration, and local surface temperature. The radiation environment varies with the plume shape and the incident radiation to any base location depends upon the emission/absorption and afterburning characteristics of each contributing plume and by the magnitude of attenuation of the base region gases. Convective cooling occurs during early first stage flight as cool freestream air is drawn through the base by the aspirating action of the plumes. At higher altitudes when the plumes become highly expanded and interact, hot gases from the SSME and SRM nozzle boundary layers are reversed into the base resulting in base convective heating to most base surfaces. The magnitude of this reversed flow convective heating during the last 30 to 40 sec. of first stage flight is significantly greater. It affects more surfaces than was anticipated before flight data were obtained. Photographs taken of the ET base during this time period show the reverse flow is so strong that the ET base appears to be burning. Actually, this observation is caused by aluminum oxide particles and ET aft dome ablation products glowing in the reversed flowfield.

Typical flight data measured at the center of the Orbiter heat shield illustrate the various environment components and their relative magnitudes throughout ascent in figure 4. Base heating is significant at this location from SSME ignition until main engine cutoff (MECO). Radiation during first stage flight is at a maximum near sea level, decreases as the altitudes increases, and is reduced by convective cooling during the first 70 seconds of flight. At 70 seconds, the plume boundaries intersect and begin to recirculate exhaust gases toward the base. There is then a rapid buildup in convective heating until SRM thrust tail-off begins which reduces the intensity of the reverse flow (and the heating). Approximately 7 seconds before SRB separation, a sharp spike in heating occurs associated with SRM shutdown. This "shutdown" heating spike is the result of motor liner material and other high radiators burning and flowing out the nozzle.

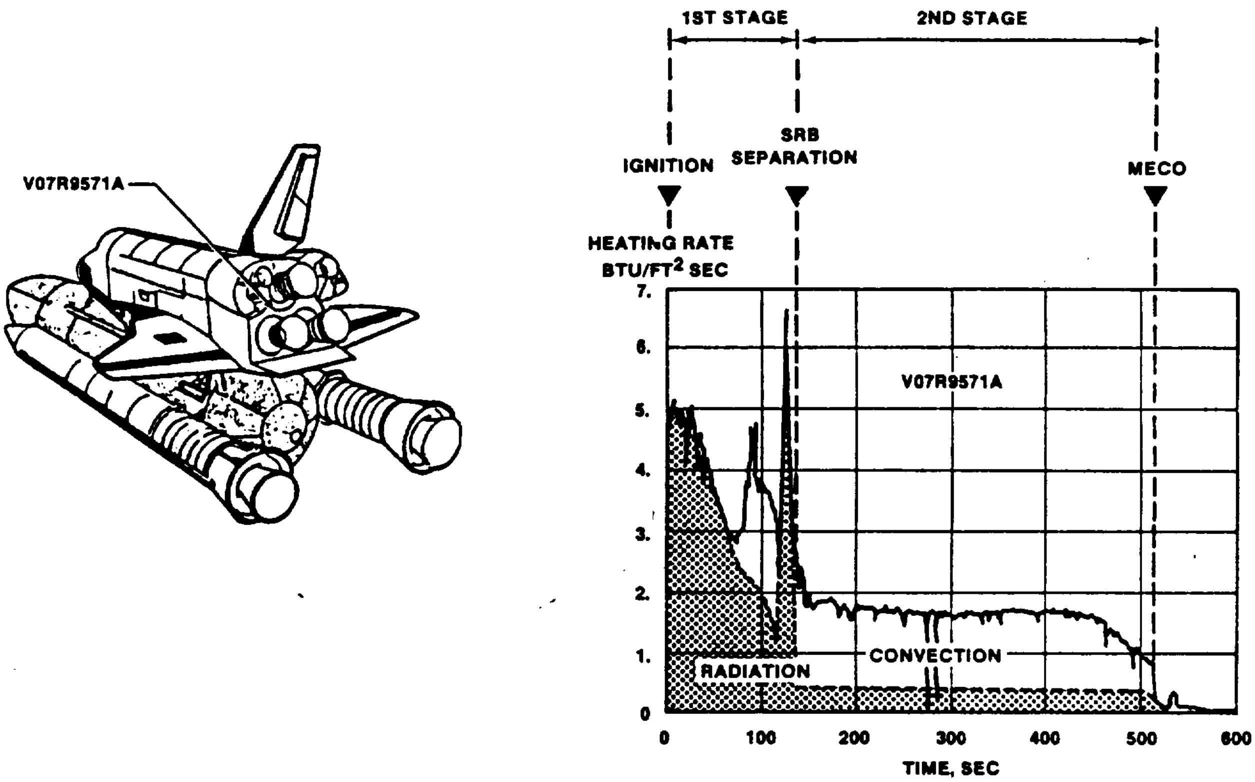

FIGURE 4.- TYPICAL ASCENT BASE HEATING ENVIRONMENT.

During second stage flight, the SSME plumes radiate at a nearly constant low level (note shaded area of figure 4). Convective heating is essentially constant as the flow into and out of the Orbiter base reaches a choked condition and becomes independent of altitude. This convective heating declines at the center of the Orbiter heat shield when the SSMEs throttle down at approximately 450 seconds. This decline is a result of flowfield changes in the base caused by variations in the main engine pitch position.

Radiative and convective heating components of the total base heating environment prediction methodology have different methods. They are computed independently and are summarized in the following paragraphs.

RADIATIVE PLUME HEATING

Solid Rocket Motor

The sea level math model for the SRM plume radiation was originally based on experimental data taken on sea level firings of the Titan IIIC solid motor[3]. Then it was scaled to the SRM motor size. This sea level model was subsequently updated based on data obtained during static firing tests of the SRM. Narrow view angle radiometer data were obtained along the plume centerline. This was done to characterize the plume emissive power. Wide angle radiometer data were obtained at positions that simulated locations on the Shuttle vehicle. From these data, a new sea level plume emissive power radiation model was developed[4]. Subsequent testing of the SRM (QM-3) provided narrow view angle radiometer measurements near the nozzle exit plane. These were slightly higher than measurements taken earlier and the data resulted in an update of the SRM plume emissive power radiation model[5].

With the math model for sea level plume emissive power defined, radiative heating rates to various design points on the Shuttle were calculated with a radiation view factor computer program[6]. Initial predictions assumed no altitude variation. Later predictions (before flight data became available) considered altitude effects with a Monte Carlo radiation code[7]. The predictions coupled with detailed, two-phase plume flowfield calculations resulted in the plume model discussed in reference 8.

Space Shuttle Main Engine

Radiative heating rates from the SSME plumes were initially calculated using the basic NASA band model gaseous radiation program[9]. An extensive effort was made to correctly model the Mach disk region and the viscous shear layer of the plume. To calculate SSME radiation to the large number of design points required for a design environment, a geometrical representation of the SSME plume radiation model was constructed. This allowed for view factor calculations to be made[2]. At low altitudes, the plumes do not interact. Therefore, detailed radiation calculations were made for each plume. The environment was generated at a given design location by adding the contributions from each plume. The complex three-dimensional flowfield which occurs at high altitudes was approximated using two-dimensional techniques[2].

CONVECTIVE BASE HEATING

Convective base heating predictions were based almost entirely on short duration, hot firing model test data. Eight separate base heating test programs were conducted to support the convective environment analysis.[10,11] The model used throughout these tests for first stage conditions was a 2.25% scale model of the fully integrated launch vehicle. These tests had short duration techniques that included hot firing hydrogen-oxygen simulation of the SSME, hot firing simulation of the booster SRM, and simulated external air flow over the model. The model used for second stage test conditions was a 4% scale model of the Orbiter base region, vertical fin, OMS pod, and body flap which included hot firing hydrogen-oxygen simulation of only the SSMEs. These tests were conducted in altitude chambers with no external flow, only a variable chamber back pressure.

During these tests, model heating rates and gas temperatures were measured over a range of simulated altitudes. All factors affecting convective base heating were parametrically varied to provide a detailed base heating data base. When the flight conditions were established, this data base was used to extract the model heat transfer coefficient corresponding to the specific flight condition. The techniques used to scale from model to full scale were based on the Colburn Turbulent Scaling Law. Analytical predictions for the mass average base gas recovery temperature were made by estimating the mass flowrate of exhaust products into the base region. Then integrating the total energy flowrate in the nozzle boundary layer from the nozzle wall to this mass flowrate. Details of these analytical techniques are provided in reference 2.

FLIGHT RESULTS

Flight instrumentation to monitor ascent base heating consisted of total calorimeters, radiometers, and gas temperature probes. With the exception of the gas temperature measurements, the data were generally good, consistent from component to component, and were of significant value in understanding the base heating environments. No valid gas temperature measurements were obtained. Complete presentations of all base heating data for STS-1 through 5 are presented in references 12 through 16. An overview of all the base heating flight data is given in reference 10.

Close examination of the flight data indicates that two changes were necessary in the basic SRM plume radiation prediction methodology. One change involved the sea level radiation model modified to account for the combustion zone between the SRMs from the outgassing ET base TPS material combusting as it flows downstream between the SRM plumes. The other change was a correction factor to account for altitude changes developed from the flight experience. The altitude correction factor eliminated the launch stand correction factor that was present in the earlier methodology. It also accounted for the SRM shutdown spike at the end of the SRM burn. These methodology changes are discussed in reference 11.

The Shuttle flight data generally validated the convective methodology. For most base surfaces, the good agreement between prediction and flight data indicated that the scaling methods were correct. However, at three distinct base locations, the prediction methodology was obviously incorrect. These locations were the upper interior region of the Orbiter base heatshield, the upper ET aft dome surface, and the outboard SRB skirt. At the upper heatshield location, the preflight methodology overpredicted convective heating during second stage. Conversly, the methodology underpredicted ET dome and outboard SRB skirt convective heating during the intense recirculation period at the end of first stage boost. Final operational flight environments will account for these discrepancies.

A comparison of actual flight data with the DCR environments and the current operational environments is shown in figure 5. This comparison is representative of the complete data base obtained on

Shuttle flights STS-1 through 5. There was generally good agreement overall between flight data and preflight prediction.[17]

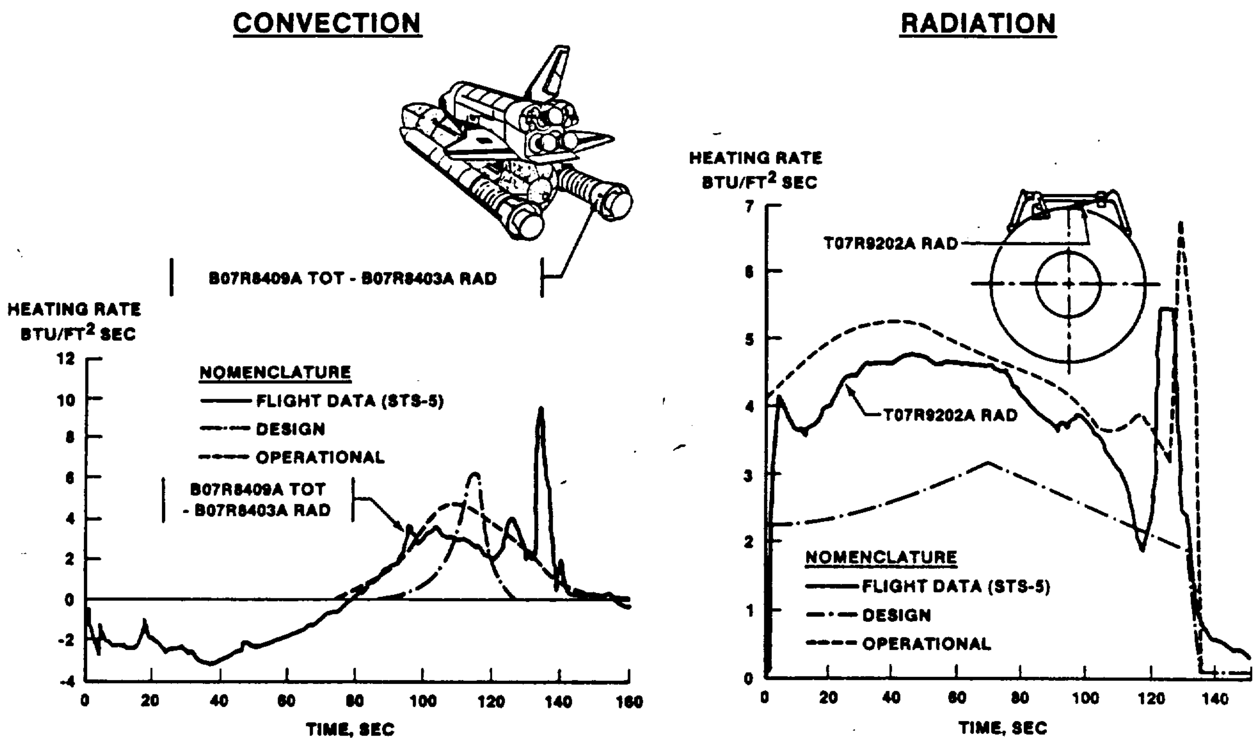

FIGURE 5.- COMPARISON OF PREDICTION WITH FLIGHT DATA.

AERODYNAMIC HEATING

The choice of aerodynamic heating methodology and the technique of generating the thermal design criteria proved to be the most important challenge faced during the Shuttle ascent design effort. While heating to the Orbiter was assessed for ascent flight, the majority of the Orbiter thermal protection system (TPS) was designed by the entry environment. The base region was designed by ascent plume heating. To develop a comprehensive ET and SRB TPS design, the thermal analysts required ascent heating prediction environments at many locations (body points) on the elements. The ET has approximately 1600 body points, the SRB approximately 600, and the Orbiter approximately 2000. Math models and computer programs had to be developed that were capable of accurate but rapid calculations. These calculations would accommodate the large number of body points and trajectory time steps. Simplified flowfield and heat transfer models were derived and checked with wind tunnel test data and results from large and cumbersome exact analytical solutions.

The wind tunnel test program[1] for the baseline Space Shuttle configuration began in 1973 and continued through late 1982. Thousands of hours of wind tunnel facility time were used to test the Orbiter, ET and SRBs as individual elements and as integrated vehicles for the first and second stage flight configurations. There are a limited number of aerothermal test facilities and those used were Arnold Engineering and Development Center (AEDC), NASA Ames, Langley, and Cornell Aeronautical Laboratory (CAL). Testing was performed on small scale models mainly with the use of thin-skin thermocouple-instrumented models. Additional testing was done with the phase change paint technique where stycast models are coated with paint which melts at known temperatures. One of the challenges involving the wind tunnel testing was associated with small scale models. Very little data were obtained for protuberances. A scaled Shuttle vehicle limits the size of a protuberance which precludes obtaining meaningful data. Therefore, data from the literature were used to develop heating predictions for most protuberances. A few tests were conducted with large scale protuberances mounted on flat plates. Another challenge was that the flowfield over the models differed from flight because of size limitations in the wind tunnels, facility flight simulation constraints and lack of plume simulation for the SRM and SSME.

The basic methodology for the calculations of the design heating rates involved analytically calculating undisturbed heating histories for each element and modifying these histories for interference effects of the integrated elements measured in the wind tunnel. Undisturbed heating is defined as the heat flux to the single element without protuberances. The analytical predictions were verified with wind-tunnel data obtained on each element without protuberances.

Early in the program a conservative design philosophy was adopted to assure safety of flight. This philosophy consisted of the worst possible vehicle orientation for the undisturbed heating calculations coupled with the worst vehicle attitude for the interference heating. This conservatism was identified as the envelope technique which used the maximum values of interference over undisturbed heat transfer coefficients (h_i/h_u).

FLIGHT RESULTS

The first six Space Shuttle flights were instrumented with heat flux sensors (calorimeters) on the ET and SRBs to measure heating rates to verify or modify the prediction methods. Instrumentation installation of the SRBs was relatively straightforward. However, installing sensors on the cryogenic ET proved to be a challenge. The sensors had to be placed in islands to insulate them from the cold structure and to preclude ice formation. Since this concept protruded above the surrounding TPS, shallow angle ramps were designed to surround the islands to minimize local flow disturbances. Surface thermocouples and pressure taps were installed in the Orbiter TPS tiles to measure surface temperatures and local static pressure. The surface temperatures were used to obtain heating rates for comparison with preflight predictions.

Examination of ET flight data became a new challenge with the necessity to "correct" the flight measurements. Calorimeters on the forward ogive of the ET were affected by wall temperature mismatch. This occurs when the air flows from the hot Spray on Foam Insulator (SOFI) surface to the cold calorimeter surface resulting in an erroneously high heat transfer measurement. Once these corrections and other data manipulations were made, such as subtracting out the radiation heating from those sensors which measure both convective and radiative aerodynamic heating, the flight data were then compared with the predictions.

The flight data evaluation has revealed that on the ET nose cap, the flight data were higher than the predictions. This is because the predictions were based on laminar flow wind tunnel data whereas during flight the flow was turbulent. This was a very localized effect and the predictions were updated to reflect this higher testing.

For most of the Space Shuttle areas the flight data are in reasonable agreement with the predictions when they are based on the actual flown trajectory. This gives confidence in the calculation of the undisturbed heating and in the applicability of the wind tunnel derived interference factors. The design values, however, are significantly higher than the flight data. This is because worst-on-worst trajectory parameters were used in the design environments. Removing the undue conservatism caused by the worst-on-worst envelope approach is the biggest remaining challenge. New approaches using exact vehicle attitudes have been developed resulting in optimized environments when the upcoming operational flight design heating values are published.

CONCLUSIONS

The Space Shuttle Program provided the opportunity to develop a complex two-phase plume flowfield calculation which will contribute immeasurably to future predictions of plume radiative heating. Flight results revealed the combustion affects of TPS outgassing on the sea level plume radiation model. Also they provided corrections for altitude effects near sea level and effects of the shutdown spike at the end of SRM burn.

The plume convective methodology was generally validated with flight data. However, the preflight methodology overpredicted the plume heating at the upper exterior region of the Orbiter base heat shield during second stage. And it underpredicted the ET dome and outboard SRB skirt during the recirculation period at the end of first stage.

The aerodynamic heating predictions are in reasonable agreement with the flight data everywhere except the ET nose cap. There the measurement exceeded the prediction because of an assumed laminar flow for an actual turbulent flow environment. The design values greatly exceeded the measured flight values since a conservative dispersed trajectory was used for predicting the design heating environment. An exact vehicle attitude method will reduce the preflight envelope technique. Reduced trajectory dispersions and updated aerodynamic heating math models should lower the design operational values sufficiently to reduce the ET and SRB TPS requirements.

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