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TEMPERATURE AND HEAT FLUX MEASUREMENTS - CHALLENGES FOR HIGH TEMPERATURE AEROSPACE APPLICATION

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Thesis Statement

The fundamental problem is that you and I would like to hear broad, sweeping universal truths about this subject, thermal sensors, but it is dominated by the details of the particular application.

These details involve the materials system complexity; the characteristics of those materials in which measurements are made; the presence, if any, of sub-surface cooling and the comparative use of "add-on" or "build-in" thermal sensors which infer heating rates by either understanding the thermal capacity of a material or the rate at which heat conducts through that material.

INTRODUCTION AND BACKGROUND

The measurement of high temperatures and the inference of heat transfer data is not strictly a problem of either the high temperatures involved or the level of the heating rates to be measured at those high temperatures. It is a problem of duration during which measurements are made and the nature of the materials in which the measurements are made. Thermal measurement techniques for each application must respect and work with the unique features of that application.

In the past 30 years, high heat flux has successfully been measured on a number of high temperature flight test systems. Among these programs were (1) the NASA "FIRE" program, (2) the NASA "RE-ENTRY F" program and the (3) the NASA SPACE SHUTTLE program. The table below lists the peak measured heating rates on these vehicles, the flight duration of the experiment, and the peak wall temperature achieved on the sensor.

Flight Vehicle	Peak Heating Rate (Btu/Ft^2Sec)	Peak Measured Temperature $(°F)$	Test Duration (Sec)
Space Shuttle Forebody	16.2	2015 $°F$	1200
Space Shuttle Flap	26.9	2311 $°F$	1200
FIRE	1100.0	2440 $°F$	25
Re-entry F	545		3.2

Each of these was a unique experiment with quite different test goals. As a result, many of the flight-measurement features differed in detail from experiment to experiment.

The Space Shuttle, a lifting body, generated data for the longest duration with the substantial surface wall temperature. The other experiments, while measuring higher heating rates at comparable surface temperatures, obtained these data during far shorter test durations.

Matthews et al [1] discussing the need for high temperature heat flux gages stated that...

> ...Reliable heat flux gages are currently limited to relatively low operating temperatures, and gages that can operate at temperatures of 2000$°F$ and above are required to measure the heat delivered to a structure during ground and flight testing.

In spite of this, flight tests have generated reliable heat flux data at surface temperatures substantially greater than 2000$°F$. Admittedly, in each case the thermal instrumentation was well integrated into the thermal protection system in a manner which allows the efficient inference of heat transfer from temperature measurements. The basic thermal instrumentation on the Space Shuttle, for example, was a single platinum thermocouple placed near the surface of the tile, as shown in figure 1, with the surface temperature and the heat flux inferred from this thermocouple through the use of a thermal model. In both FIRE and RE-ENTRY "F" in-depth temperature plug gages, shown in figures 2 and 3, were used to define heat transfer. This instrumentation initially inferred heating through the temperature drop between adjacent thermocouples in the plug although post test analyses suggested that for the short test durations, multiple gages placed in depth were not required and a single thermocouple defining the heat capacity of the surface material would suffice.

Why would a quote in 1991 argue with successes of the past 30 years? The problem is NOT that heat transfer cannot be deduced at high operating temperatures but that either heat transfer cannot be deduced at high temperatures for more general applications or the problem lies in the definition of the term "heat flux gage". Each of the cases just discussed was carefully crafted from the materials viewpoint, from the test duration viewpoint and from the instrumentation viewpoint to facilitate measurements.

Matthews continues by saying that:

> ... (Heat flux) gage design emphasis is placed on COMPATIBILITY ISSUES associated with integrating the gage within the structure...and minimizing the gage's influence on the surrounding material response, particularly in actively-cooled structures.

This statement is a key. For each example case, the instrumentation and materials in which it was placed were thermally compatible. Compatibility issues define whether a commercially-available add-on thermal sensor or a built-in thermal sensor should be used in a specific application and whether accurate heat flux can be deduced at all. The corollary is that NOT ALL STRUCTURES CAN BE INSTRUMENTED TO DEDUCE HEAT TRANSFER today. That is the basis of Matthew's comments and a substantial challenge for the measurement community. A more correct interpretation of Matthew's statements, which I might offer, would be...

> Significant deficiencies exist in deducing heat transfer from temperature measurements on ARBITRARY MATERIALS and DURING ARBITRARY TEST PERIODS and these problems are made worse by the selection of "add-on" rather than "design-in" thermal sensors. Examples of successful high temperature, high heat flux measurements are the result of careful, integrated designs where the instrument, the flight vehicle and the test acquisition phase of the flight have been properly selected to achieve successful data.

Several concepts have been briefly introduced and these concepts require further discussion.

THERMAL GAGES AND HEAT FLUX GAGES: AN OPERATIONAL DEFINITION

In this paper, thermal gages are instruments that measure temperature. They include thermocouples, resistance thermometers and fiber-optic based temperature measurements. Heat flux gages are self-contained instruments that use thermal measurements, with a thermal model, to infer causal heat flux.

ADD-ON VS DESIGN-IN HEAT FLUX GAGES

Heat flux is deduced through thermal models which interpret the measured temperature on or in the structure in terms of the heating that caused those measured temperatures. In so doing, the process must not alter the thermal character of the surface being measured. Heat flux gages are highly localized embodiments of those thermal models, concepts committed to hardware. A cross-section of heat flux gages for low and high temperature applications are defined in the tables below.

Dimensions of Low Temperature Heat Flux Sensors

Heat Flux Sensor Type	Typical Diameter (Ins.)
Thin Film resistance Thermometer *	0.0030 x 0.200
Schmidt-Boelter Gage	0.250
Coax Gage *	0.0150
Plated Thermocouple Gage *	0.008 x 0.008

* Requires proper materials to form a valid thermal model

Dimensions of High Temperature Heat Flux Sensors

Heat Flux Sensor Type	Typical Diameter (Ins.)
Water-cooled Gardon Gage *	0.0625
High Temperature Gardon Gage *	0.250
Vatell Gage	0.100 x 0.125
In-depth Thermocouple Gage *	0.125

* No surface thermocouple measurement

It is possible that heat transfer can be deduced through add-on, commercial heat flux sensors but it is also possible that heat flux must be deduced by distributing thermal sensors within the structure of the hardware being measured and using that hardware as the "heat flux gage" itself. In the Space Shuttle example the TPS material had thermal sensors imbedded in it and the material with the imbedded thermal sensors became the heat transfer gage. For the FIRE and RE-ENTRY "F" examples either the beryllium structure could be thought of as having thermal sensors imbedded in it or the thermocouple plug could be thought of as a thermally transparent add-on heat flux gage.

There were also add-on heat flux gages used on the Space Shuttle in the external tank[2] and added to the orbiter heat shield[3]. These examples are discussed in appendix A. In both cases, the metallic add-on heat flux gages were poorly integrated into the insulative TPS material of the Space Shuttle and the resulting heat transfer "measurements" were of an unacceptable quality.

Thermally incompatible heat flux sensors present two very different problems which must be addressed: 1) the sub-mold-line transfer of heat between the parent structure and the sensor which may destroy the "assumed" thermal model and 2) surface boundary layer distortion of CONVECTIVE flow over the structure housing the heat transfer gage. This distortion is caused by the step function change in the wall boundary condition with no corresponding change at the "edge" of the boundary layer. Since the surface is heated by the gradient of the static temperature distribution through the boundary layer, the sensed heat flux is sensitive to wall temperature disturbances. This is a FLOW FIELD-INDUCED phenomena present in a convective flow which has a developed boundary layer but not in a radiative calibration rig or on the stagnation point of a model placed in a convective flow. The aerodynamic problems of temperature mismatch were discussed both by Carnahan[4] and Praharaj[5]. Both of these problems can be addressed through well-defined numerical techniques which correctly model the flow. The material mismatch problem is found in all high temperature or long time applications of thermal instrumentation.

Both adding-on thermal instruments or designing-in those instruments are potentially acceptable techniques within limitations defined by the specifics of the experiment. The details of the specific application may dictate one technique over the other. In either case, the basis of heat flux measurement must be respected. Clearly, in the case of the add-on gages for the Space Shuttle application, the gages were not well thermally integrated into the TPS system and the measured results were, as a consequence, not representative of the structure with no instrument installed.

Thermal Gage Location Within the Structure:

The inference of heat transfer requires the precise placement of thermal gages with the structure to be measured. Large gradients of temperature are present within the structure and imprecisely located thermal sensors reflect inaccuracy in the inferred heating rates. Knowledge of thermal sensor location is far more precise in add-on heat gages than in design-in heat gages. The instrument vendor precisely locates these sensors and calibrates the heat gage to assure they are properly located. This is not the case for design-in heat sensors for which thermal sensors are placed within a structure either during or after the fabrication process. Examples abound of poor data generated as a result of thermal gages imprecisely located within the structure or of shifting within that structure during the process of a test. One such example is, again, the Space Shuttle instrumentation shown in figure 1. In this case, the platinum thermocouple was placed within the soft, insulative tiles during the fabrication process without strict quality controls on the installation process [6]. The result was some ambiguity concerning the actual in-depth placement of the thermocouple wire during the installation and later during the flight-to-flight refurbishment of the tiles containing the thermal gage. Hodge initially identified the problem which is most acute in short duration flight maneuvers and defined a very sophisticated software tool to understand it [7]. Similar studies were conducted by Jones et al [8]. Figure 4 from their paper indicates that the accurate location of thermocouples was required to match heating rate data deduced from thermocouples with heat transfer data deduced from calorimeter gages. In-situ calibration of test hardware containing design-in heat gages is extremely important as well as the collateral use of parameter estimation techniques to generalize that calibration information. The paper by Kipp and Eiswith [9] proposed in situ calibration of instrumentation that is as true today as it was a decade ago.

THERMAL CAPACITY VS CONDUCTION RATE HEAT FLUX INSTRUMENTS

Although aerodynamic heating is numerically computed from the slope of the static temperature in the boundary layer near the wall boundary condition, aerodynamic heating is experimentally inferred though its influence on the structure as shown in figure 5. The numerist worries only about fluids assuming a perfectly responding structure but the experimentalist worries about both questions of materials response as well as questions of the boundary layer fluid.

As noted earlier by example, successful instruments can be based either on the premise of capturing the heat pulse within the structure or determining the rate of heat transfer through the structure. The Shuttle, because of its insulative tile thermal protection system, essentially captured the heat pulse within the tiles; whereas, the FIRE and RE-ENTRY "F" programs were instrumented with gages capable of either capturing the heat pulse or measuring the rate of heat transfer through the structure. Heat capacity gages are indirect measures of heat transfer requiring a sometimes complicated thermal model to interpret measured temperature in terms of causal heating rates. Heat conduction rate gages are direct reading gages creating a temperature-difference signal proportional to causal heat transfer rates. This is, of course, an idealized description of the problem assuming only a direct relationship between causal convective heating to the surface and conductive dissipation of that heating within the structure, a classic low temperature application.

At high temperatures, the causal convective heating may be dissipated either through internal dissipation, ideally one-dimensional conduction into the skin of the flight vehicle or re-radiated back to space (or other structural elements if you are really unlucky). As the surface temperature of the structure increases, more of the heat radiates (as the 4th power of the wall temperature) and correspondingly less conducts inward. At some point during the flight operation of an uncooled surface, almost all of the heat is re-radiated away from the surface. In the limit, rate gages define zero heat transfer THROUGH the surface although substantial surface heating to the surface is still occurring. In this limiting case, the sole index of heat flux is through a surface temperature measurement that is used to infer heating through calibration of the emissivity of the surface material. On the Space Shuttle, this situation occurred rapidly as shown in figure 6.

Rate gages, while direct reading, may not always measure the total heat load to the surface. All high temperature rate gages must measure not only the rate of heat transfer through the structure but also the absolute level of temperature at the surface. Some do not and others accomplish this only through calibration. Measuring both rate and level requires additional channels of information compounding the difficulty of the measurement process. Finally, heat capacity gages require a larger volume of structure for dissipation as the test duration increases and/or the thermal conductivity of the structural material increases. It is not always possible to capture the heat pulse within a structure and for those cases where it is not, rate gages may be a better choice.

SUB-SURFACE COOLING

Actively cooled materials are feasible and of increasing interest. While cooling reduces the material surface temperature (and the heat re-radiated to space from an instrumentation perspective), cooling also provides an additional heat loss mechanism. Heat capacity measurements, in this environment, require that the heat loss to the coolant be measured. Direct reading heat-rate measurements applied in this environment require that all possible heat paths be measured and that direct measurements be made within the thickness of the panel being cooled and within the portion of that cooled panel at which the temperature gradient (due to conduction) is linear. These thickness dimensions can be very small, challenging the design and integration of thermal instruments.

WHAT IS MEASURED AND WHAT IS CALIBRATED OUT

Re-radiation:

Not all the elements of aerodynamic heating can be measured with either thermal capacity or heat rate sensors. One element that requires calibration is surface re-radiation. The heating due to radiation follows the equation

$$\dot{q}_{Radiation} = \epsilon \sigma T_{Wall}^{4}$$

Evaluating radiation from the wall is inherently inaccurate resulting in an error of +/- 10% or more in heating rate. The emittance measurement has experimental scatter and the surface temperature measurement converted to a digital signal has experimental scatter whose magnitude depends on the type of thermal sensor used and the quality of the data train (from sensor to digital output). Further, the measured temperature is raised to the 4th power compounding the error and finally, the measured temperature may not be the actual surface temperature but a sub-surface measurement that must be related to surface conditions either numerically or through calibration techniques.

Chemical Energy:

Another increment of heating that normally requires calibration is that associated with incomplete recombination of dissociated boundary layer flow. Static temperature increases through the shock system about the vehicle and the deceleration of the flow in the boundary layer may dissociate the air. Low density flows, which limit energy transfer in the gas, may lock in the stagnation region chemical activity that sweeps over the surface. If this occurs, the catalytic character of the surface material and that of the thermal gage placed in that material may recombine the dissociated flow. This catalytic character must be calibrated. Upstream history effects are important as well as the local conditions at the instrument in question. Calibration of the surface catalysticity is possible to accomplish but surface aging as the flight surface is repeatedly flown is also important.

Our flight experience with catalytic surfaces is through the Space Shuttle flights. The non-catalytic nature of the Shuttle tile coating (a glassy material) was noted to reduce the measured heating rate up to 60% of a fully catalytic wall [10]. It has been noted that the sudden release of chemical energy over a catalytic gage placed in a non-catalytic surface results in a temperature rise greater than the equilibrium value that would be present for an entirely catalytic surface and gage [11]. Evidence of aging phenomena was also noted and reported by Jones [12] who noted a 20% increase in heating from flight 2 to flight 5 of the Space Shuttle due to either catalysis or emissivity changes.

This situation argues for in-situ calibration of thermal sensors on flight vehicles, yet another measurement challenge. In-situ calibration is more important for thermal sensors which are designed-in to the structure rather than added-on to the structure.

THERMAL MODELS AND THEIR IMPORTANCE:

Heat transfer is inferred by strategically placed temperature measurements through the use of a thermal model of the temperature dissipation. We implicitly use thermal models while we tend to forget the approximations that limit their application. Whether the causal relationship is seemingly trivial, a 1D conductive flow through the structure defining the flow within a so-called heat gage, or whether it is complicated, full 3D time varying dissipation through a built-up structure, the process is conceptually the same. This paper will not discuss thermal models and their limitations but even a cursory review of the literature demonstrates the problems which are continuing today.

THE SIMILARITIES AND DIFFERENCES BETWEEN GROUND TEST AND FLIGHT APPLICATIONS

While this paper deals with flight measurements. The reader might deduce that either high temperature measurements are not made on the ground or that the instrumentation and techniques for ground-based measurements are solved or are the same as for flight. None of this is the case.

Ground test applications cover a spectrum of thermal and heat transfer needs and goals. Aeromechanic testing is generally at low temperatures because of experiment design. High temperature ground test goals are structural and propulsion development. While the structural testing uses materials and structural concepts in common with flight goals, the propulsion testing may present ground test unique instrumentation problems. Propulsion tests focus on defining propulsion efficiency without the added cost and complexity of using flight-weight structural materials. This is accomplished by fabricating the model structure out of heat-sink copper. The challenge is to "measure" the highly non-uniform surface heat transfer within highly-conductive copper. A particular concern is the measurement of localized heating peaks within copper structures. These peaks create lateral heat conduction paths that may well invalidate the installed instrumentation.

Another application of ground-test experimentation that has been often proposed is the need for a pre-flight operational validation of flight instrumentation in ground test facilities. While on the surface this appears to be useful, the practical aspects of the problem for high temperature instrumentation make such a test of questionable technical value. Some of these technical issues involve:

1. Defining the radiation environment between the "model" surface and the test-peculiar surroundings.

2. Incorporating sufficient structure about the instrument to correctly simulate the flight structure at flight-simulated test conditions.

3. Generating a clean, simulating test flow without the presence of test-induced contaminants.

Other technical solutions - primarily analysis techniques- should first be investigated and discarded as inappropriate.

The Complexity of the Materials System:

In past examples the instrumented materials system was homogeneous and thick enough to capture the thermal pulse for the duration of the flight experiment. The instrumentation of such systems presents no significant technological instrumentation problems. Instrumentation problems arise when systems goals of weight-efficiency conflict with the test goals to measure temperature and to infer heat flux on those materials. To illustrate, Figure 7 from a paper by Grallert and Keller [13] shows the unit weight of various TPS materials as a function of the wall temperature of that material. Representative metallic TPS candidates are lighter than the ceramic tiles of the Space Shuttle program as the surface temperature increases. The metallic concepts are superior at high surface temperatures but the challenge is to install thermal sensors within or onto these "...thin metallic foils..." and to infer heat transfer rates from those thermal measurements.

Figure 8 shows both the stiffened heat shield (the ceramic shingle design) that is being considered for the Hermes system and the metallic multiwell concept. The material of the structures as well as the thickness of the typical surface layer are shown. Recalling the dimensions of thermal sensors previously presented, installing any thermal sensor is difficult enough but inferring heat flux through thermal sensor measurements while not altering the thermal character of the surface being measured is certainly a challenge for high temperature aerospace application.

Apart from the thicknesses of the materials, the materials selected conduct heat across the section and radiate that heat both inward and back to space. Finally, the material fabrication process, particularly for coated molybdenum will make bonding of gages to coated materials impossible.

Seven advantages are noted for these advanced material systems which are indicative of advanced materials thinking; instrumentation installation is not one of them. Complex material systems are certainly a challenge to instrument if they can be instrumented at all.

CHALLENGES FOR HIGH TEMPERATURE AEROSPACE APPLICATIONS

I see several contemporary challenges in the development of measurement technology. These challenges will now be discussed in order.

Challenge 1: To capture the character and localized peak values within highly nonuniform heating regions

The characteristics of highly non-uniform heating regions are (1) localized peaks in the imposed heating to the surface, (2) incomplete knowledge of the location of heating peaks due to real gas effects and (3) high thermal gradients along the surface of the material driven by the gradients in imposed heating.

Both ground test and flight test measurements are concerned first with identifying the level of the peak and secondarily defining the structure of the interaction. Either fields of individual point sensors are applied to define peaks by capturing the heating pulse whose location is guided by our incomplete knowledge of the distribution, or survey testing is applied to map the entire interaction process in sufficient detail. Errors in the measurement process are not random but of a bias which produces lower data than actually present. These errors are due to (1) the size of the heat sensor being substantially larger than the spike in heating being measured, (2) the thermal model relating measured temperature to the imposed heating is incomplete and ignores characteristic temperature gradients along the surface or (3) the measurement span of the temperature sensor is inadequate to the measurement (significant for survey testing techniques).

The challenges that I observe are to: (1) develop a compatible system between the types and locations of physical sensors (thermal or heat transfer sensors) and the thermal model that uses these data to define a heat transfer distribution. In the case of highly peaked distributions, the resulting thermal model may be applied to a single sensor or to a sensor field as a group. The need is to define temperature gradients along the surface as well as through the material system; (2) to create thermal and heat transfer sensors that are properly scaled to the characteristic dimensions of the peaked interaction, which implies very small sensors as well as tightly packed thermal sensors; (3) to efficiently manage the volume of data extracted from a field of either isolated sensors or temperature maps produced by various survey techniques; and (4) to broaden the temperature span of survey techniques. The last two items will be discussed in later sections of this paper.

Challenge 2: To manage large volumes of thermal instrumentation in order to efficiently derive critical information

On the ground and in flight substantial amounts of raw data are generated that must be managed in acquisition, storage and manipulation. Both the volume and complexity of test data are increasing today. Single test runs can acquire over 100 million data elements. These large volumes of data must be reduced in an efficient but complete manner. This challenge considers the efficient transformation of data into information.

Ground Test:

Survey sensing techniques now produce digital data that both increase the effective numbers of sensed points and multiply the volume of data to be manipulated. Each "frame" of data can deliver 262 thousand data values, and frames of data must be taken rapidly and sequentially to accurately map the test surface. Video refresh rates of 30 to 60 frames per second are commercially available and the test duration of 1 to 10 seconds is typical. Each run then can generate as many as 100 million data values. Admittedly, a small percentage of these values merit full reduction and much can be gleaned from even a partial reduction of these data but the information contained in these data may require a rather sophisticated reduction for complex interacting flows. Consider, for example, the evaluation of conduction losses along the surface caused by highly non-uniform heating of less than perfect insulators; the required information is captured in the data but the computational effort to reduce the data is not trivial. Numerical analyses by Dorignac and Vulliemre [14] demonstrate that these problems exist even for ideal test surfaces and must be considered in the data reduction and analysis.

Initial, zero order, data reduction of ideal insulative surfaces approximates the actual data reduction equation with a 1D closed form relationship. Using this technique, each data run requires about 4 minutes of computation on an Intel 80386 based machine. Balageas [15], who has written extensively on survey techniques at ONERA, points out that:

> " The snare to be avoided, and it is not a small one, is to keep from being swamped by the flood of data generated by this technique. Processing methods will have to be developed that are thrifty in computational time and memory space, but sophisticated enough that what is of interest can be distinguished from what is secondary or even useless, with user-friendly graphic postprocessing".

Flight Test:

Managing large volumes of high-frequency flight data with reasonable downlinks requires on-board information processing. This was observed over a decade ago by Galleher [13] who stated that

> "...as flight performance measurements become more demanding...more and more on-board processing and data compression techniques will have to be devised

that are acceptable to both system designers and experimenters..."

This paper quote referred to the flight test of BMO's **A**dvanced **M**aneuvering **R**eentry **V**ehicle, AMARV, on which 60, in-depth thermocouple plugs (similar to those on the FIRE and RE-ENTRY F programs were used. In each of these plugs 3 to 4 high-temperature Tungsten 5% Rhenium/ Tungsten 26% Rhenium thermocouples were placed.

Challenge 3: To accommodate thermal sensors into practical flight structures

Our history is rooted in the thermocouple measurement of temperatures within blocks of material. Our future requires the use of whisker-like sensors to define heat transfer within shim stock. Wind tunnel heat transfer, in the past, was inferred using "thin skin models" having backface mounted thermocouples. Today we can directly measure temperature gradients within that thin skin.

The challenge is to develop more highly integrated sensors into increasingly complex and non-uniform structures without disturbing the natural heat paths of these material systems. The challenge is to transform measurements from add-on to designed-in and, in so doing, to approach the concept of a smart skin.

Challenge 4: To broaden the capabilities of thermal survey techniques to replace discrete gages in flight and on the ground

Thermal survey techniques have been used for 30 years to observe heat patterns on the ground and in flight. Initial wind tunnel applications of temperature paint were made in the mid 1960's. Flight test examples are documented on the X-7A[16] and the X-15 flight programs (figure 9) where shock interactions and boundary layer transition were observed. The reasons for those survey techniques of the 1960's still exist and although the quality and sophistication of techniques have improved, there are several challenges which remain.

Survey techniques emcompass older irreversible temperature sensitive paints and newer reversible surface coatings such as thermographic phosphors and infrared (IR) measurements. The newer survey techniques generate large amounts of data that must be acquired, stored and reduced. A challenge is the ability to handle those data efficiently. On the positive side, the effective gage density s increased by a factor of 500 and all that data is "recorded" even though only a fraction may initially be reduced.

Recall that these survey techniques measure surface temperature and not heat flux. Heat flux is derived by a thermal model which relates measured surface temperatures to the causal aerodynamic heating. Ideal thermal models are designed on the basis of one-dimensional dissipation of aerodynamic heating with the structure. More general applications of this technique will require multi-dimensional inverse analysis which is more computationally intensive.

GROUND TEST APPLICATIONS:

There are several un-resolved needs for survey testing in ground tests facilities. Two examples are: (1) the need to define the complex boundary layer transition front on 3D models during both aerothermodynamic studies and the longer duration studies of overall forces and moments; (2) the need to more-fully understand the complexities of internal flows dominated by shock interactions. The challenges are to broaden the range of temperature response, eliminate false signals and manage the volume of derived test data.

The range of heat transfer measurable with survey techniques is limited. These techniques are only partially useful in hypersonic facilities. The limitations of the techniques can be observed in the measurement of shock interaction regions where high gradients and an order of magnitude variation occurs between the peak heating and the undisturbed heating level. Survey techniques with limited band-width bias the heating in interaction regions to lower peak heating than are actually present. [17*]

Transition Front Measurements:

While the accuracy of CFD techniques at high Reynolds numbers critically depends on defining the transition front as an experimental input and while that front is difficult to predict for highly three-dimensional flows, few, if any experimental studies map transition fronts as an input to CFD validation. Most force and moment studies measure only gross vehicle forces and never provide the instrumentation to state whether the boundary layer is laminar, turbulent or, most likely, some of each. Matthews et al [1] observes that:

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*To be published

"...unfortunately, the ability to predict the boundary layer state accurately - laminar versus turbulent - continues to elude the aerodynamicist".

The fundamental technology exists to observe the state of the boundary layer. The challenge is to implement that technology in the required production test facilities.

Internal, Shock Interaction Flows:

There are internal flows dominated by shock interaction processes. In none of these can the thermal effects be adequately defined through any practical number of discrete thermal sensors. Survey techniques are required to satisfactorily measure the thermal loads. This poses the challenge of placing the survey sensor within the restricted passage being measured and the challenge of managing the difficult, technique-peculiar errors involved in these measurements.

FLIGHT APPLICATIONS:

The most striking, modern application of survey techniques applied to flight vehicles is/was the use of an infrared (IR) sensor installed on the vertical tail of the Shuttle vehicle, Columbia. The overall features of this technique, its placement on the Shuttle and the views from the infrared camera are shown in figure 10.

Originally conceived in the 1970's and conceptually simple in design, this experiment has been difficult to install in the orbiter and successfully use to generate complete data. The specific technical difficulties encountered have been defined as the following:

- A protective plug over the windows that wouldn't jettison when required
- Inadequate cooling of the windows as heating levels increased during re-entry into the sensible atmosphere.
- Erratic operation of the camera scanning mechanism.
- Massive amounts of raw data that require efficient reduction

Figure 11 demonstrates the high temperatures that occur on the observation windows, temperatures that can only be numerically deduced from an extrapolation of inner pane measurements through the use of thermal modelling techniques, a supplementary challenge in thermal measurements.

Apart from these technical issues, there is a more difficult and fundamental challenge: the interaction between the inevitable "tweaking" of a complex experiment such as this and the operational imperatives of a schedule-driven, multi-goal flight vehicle such as the Space Shuttle.

The problems are not fundamental in nature but true engineering problems of creating a flight test measurement system that works.

Challenge 5: To provide supporting instrumentation conduits which connect the measurement points to the thermally controlled data acquisition system

Data acquisition requires hardware from the surface measurement point back to an environmentally stabilized location on the vehicle where data transmission or on-board recording can be accomplished.

A significant challenge in making flight measurements is in implementing this chain of hardware. The material systems of the flight vehicle as well as the duration of the flight are important considerations. The Space Shuttle, by virtue of its cold structure concept, attained a benign thermal environment within inches of the point of measurement, but other possible flight configurations may not. Hot structures would create a far more difficult thermal instrumentation situation.

The X-20A (Dyna Soar), a hot structure flight vehicle concept, required 3700 feet of 1800 deg F wire and connectors to connect the 750 sensors to be placed on each flight test system. The wire of that day consisted of inconel tubing 0.090 ins OD, magnesium oxide electrical insulation inside the sheath and two electrical connectors.

These high temperature conduits can include simple thermocouple leads, regulated power lines and, possibly, fiber-optic lines. Hellbaum [18] proposed "...platinum films laid down on a substrate of alumina..." to define transition for flights at lower Mach numbers. Platinum films are powered resistance thermometers that require a constant current flow through the measurement film. Similarly, high temperature microphones and photodiodes were suggested for this function. Each of these gage types is electrically-powered and the difficulty of delivering that power increases as the temperature of these conduits increases.

In the 1990's these lines may include fiber-optic bundles connected to surface temperature sensors or fiber optic conduits to interrogate or observe the thermal characteristics of a remotely located surface of interest.

Apart from conduits there is also a need for high temperature connectors to facilitate instrument replacement as well as in-line amplifiers to boost signal strength, regulators to provide precise power to surface measurements and even cooled lens systems to direct the images remotely sensed.

All of these problems are accentuated if flight duration increases and/or thermal conductors are selected as material systems. The Space Shuttle was one model of TPS and not the unique technology demonstration for future systems which must be instrumented.

Challenge 6: **To develop a class of "vehicle tending" thermal sensors to assure the integrity of flight vehicles in an efficient manner**

There have been consistent thermal problems over the past 30 years caused by uncontrolled internal flow due to leakage of boundary layer air through the flight structure. Flight configurations are not the homogeneous structures but are mechanically built-up of many separate elements held firmly in place with Sauerizen (R) or similar indispensable materials. They present many possible internal flow paths which are driven by large hypersonic pressure differences. McBride, 1983 [19] and 1986 [20], termed this flow "sneak flow" as he outlined Space Shuttle experience. He concluded (1) sneak flow is important for any reusable thermal protection system (TPS) large or complex enough to require interfaces and (2) it is difficult to make quantitative predictions of sneak flow effects. Because the problem is severe and the analysis is complex, because every hypersonic flight system has demonstrated sub-surface heating and the best "sensor" to date is a discolored surface, the challenge is to create a new class of vehicle tending thermal sensors.

Vehicle-tending thermal "gages" are a new class of sensors which do not produce a point measurement of either temperature or heating rate but develop a sense of leakage through regions which, ideally, would have none. These developmental sensors monitor the health of joints and gaps determining the severity of imperfect seals.

The X-15 nose wheel door created a gap that allowed hot, boundary layer air to enter the wheel well and destroy the instrument lines located there. Figure 12 is a photograph of that situation. The X-15 only flew at Mach 5. Figure 13 from McBride indicates the Space Shuttle penetrations including doors, gaps and coves that all could produce potentially serious sub-mold-line flows. These regions are caused by the aero-thermo-elastic effects of hypersonic flight which cannot be simulated in ground tests. They are driven by pressure differences across the produced gap and require an understanding and modeling of the sub-surface flow paths.

McBride, 1983 [19] observed that "...penetration thermal instrumentation (on the Space Shuttle) was only adequate. More sensors were required at difficult-to-predict environment locations. Available DFI (developmental flight instrumentation) should have been more concentrated. MORE EXTENSIVE USE OF PASSIVE TEMPERATURE-SENSITIVE DEVICES COULD HAVE BEEN MADE." Of the approximately 6000 recorded measurements (about 2000 of them temperature) available on the orbital flight test orbiter, 627 were dedicated to TPS elements BUT ONLY 90 OF THOSE ARE STRICTLY RELATED TO PENETRATIONS. Table I of AIAA 83-1486 outlines in greater detail the TPS penetrations and their instrumentation.

These regions are characterized by a large seam on the vehicle which could leak at any location on that line or in that region. While point sensors could be placed in such regions, the question is where to place them and what kinds of data are required from them to define leakage.

What we require is an overall impression of whether leakage occurred and whether that leakage was significant. The challenge is to develop a sensor that achieves these goals rather than to measure temperature. The challenge is to produce better coverage with fewer sensor assignments.

One conceptual recommendation is to use ablative (or phase change) overcoat surrounding a distributed sensor. The sensing surface would create a signal proportional to the amount of the overcoat removed as a result of the temperature exceeding a defined threshold level. Perhaps an ablative coating could be applied over a fiber optic bundle through which light is being transmitted and received. In a sense, this is another application of a smart skin. This technique was also discussed by Measures, 1989 [21].

CONCLUSIONS:

The conclusions I draw from this material are:

1. The discussion of high temperature, high heat flux

measurements is contingent upon the details of the experiment proposed. Those details are (1) the type, complexity and dimensions of the material system (2) the duration of the flight experiment and (3) the selection of "add-on" or "design-in" heat gages for those measurements.

2. The technical challenges in those heat flux measurements are fundamentally two: (1) the construction of ever smaller physical thermal sensors and (2) the development of efficient inverse thermal models that relate those thermal measurements to causal heat transfer.

3. Six areas of technical challenge have been postulated. These treat the heat flux measurement problem in a broader context, the delivery of information on thermal questions to a customer. They are intended to start a discussion concerning thermal measurement technology.

Finally, a conclusion implicit in this review paper is that thermal instrumentation is an enabling technology that makes possible both flight test and ground test programs. Instrumentation should not be an after-thought of a larger systems-related program or relegated to the catalog purchase of "proven" devices (proven on the last flight vehicle). If considered early and funded adequately, instrumentaton can enable tests otherwise impossible and/or reduce the cost of even routine tests substantially. More and more temperature measurements must be designed into the developing structural component and must be considered as an integral part of that development process.

Appendix A
Add-On Thermal Gages Applied to the Space Shuttle Hardware

A. Space Shuttle External Tank

Commercially available Schmidt-Boelter and "pill-type" heat flux sensors were installed in the insulative foam surface of the external tank. These metallic gages present a non-uniform surface temperature to the boundary layer, the gages being relatively cold and the surrounding insulator being hotter. Praharaj [1], reviewing the experience, noted that severe temperature mismatch was present producing "...a large measurement error in a convective flux environment...". "...The underprediction of these island measurements was 100% or more in the peak heating region...". He further notes that these effects can be "...successfully factored out of the flight data in undisturbed regions..." but that "...temperature mismatch effects in the interference regions are not dealt with in the existing literature and consequently were not factored out of the flight measurements...". Praharaj further notes that "...the choice of sensors for future space vehicles must consider this effect (temperature mismatch) and efforts must be made to reduce the temperature mismatch effects on the measurements...". Finally, Praharaj notes that "...temperature along with heat flux should be measured so that one can be derived from the other. This would help eliminate erroneous readings in a much easier fashion..."

The analysis of undisturbed temperature mismatch used by Praharaj was due to Westkaemper [22] following the expression:

$$\frac{\bar{h}(W,L)}{h(W,0)} = F(L/W)\,\frac{(T_{w_1}-T_0)}{(T_{w_2}-T_0)} + H(L/W)\,\frac{(T_{w_2}-T_{w_1})}{(T_{w_2}-T_0)}$$

where: $F(L/W) = \frac{5}{4}\,\frac{[1-(L/W)^{0.8}]}{(1-L/W)}$

and: $H(L/W) = \frac{5}{4}\,\frac{(L/W)^{0.8}}{(1-L/W)}\,[(W/L)^{0.9}-1]^{8/9}$

where: L is the running boundary layer distance to the gage
W-L is the gage width dimension
T_{w_1} is the wall temperature of the surrounding material
T_{w_2} is the gage surface temperature
T_0 is the recovery temperature

B. Space Shuttle Orbiter

Little has been written concerning the use of Schmidt-Boelter gages on the orbiter. Figure A-1 is a sketch of the installation. The sketch pre-dates the flight and may not represent actual flight hardware. Figure 4 of this paper shows data from these gages and indicates that problems exist with these gages relative to imbedded thermocouples to which they are compared.

C. General Comments:

Hornbaker and Rall [23] presented an excellent review article on this phenomenon as on many aspects of thermal instrumentation. The reader would do well to review this and the several other review papers which were written by these authors.

APPENDIX B

A REVIEW OF THERMOSENSING ELEMENTS AND THEIR USE IN AEROSPACE APPLICATIONS

The standard thermo-sensing elements used to measure temperatures were/are wire thermocouples. There are a series of these thermocouple material combinations available with which to measure temperatures at different temperature levels. Table II from Moffat [24] shows representative material pairs:

Material Designation	Temperature Limit, deg R	Output, mv/ 100 deg F
Chromel-Alumel	2290	2.20
Platinel	2650	2.20
Platinum-Rhodium	3730	0.43
Tungsten-Rhenium	4630	0.76

Note that Tungsten/Rhenium thermocouples must be placed in an inert atmosphere. "The main problems were centered around embrittlement of the Tungsten leg (of the thermocouple) and oxidation".
Tungsten thermocouples have been successfully used in high temperature flight tests by applying sheathed configurations.

High temperature gages also suffer from a progressive de-calibration (of the thermocouple) with time due either to changes in the composition of the material or changes associated with grain growth and the annealing out of the residual cold work from fabrication".

The progressive oxidation problems are eliminated through sheathing the thermocouple in a protective material. Very small sheathed thermocouples are currently available with outside diameters as small as 0.008 ins (0.2mm). One concern with thermocouple sheathing is understanding precisely the location of the thermocouple junction within the sheath. This problem, annoying for the measurement of temperature, is a critical deficiency when sheathed thermocouples are a part of a built-in heat transfer measurement system.

As small as these devices have become, the general rule of thumb is that the thermocouple assembly should have an outer diameter roughly 20% of the thickness of the material into which it is placed to avoid excessive conduction down the thermocouple wires. That would place the minimum material thickness at 0.040 inches or greater, far thicker than anticipated applications shown by Grallert and Keller.

This problem can be circumvented by: (1) avoiding the problem by using a non-conductive thermo-sensor material (such as a fiber-optic-based thermal sensor) (2) integrating the sensor into the thermal analysis of the material, either directly through an inverse technique containing the actual structure elements that are approximately through correction factors developed to account for conduction losses down the wire.

As the temperature to be measured increases, the selection of thermocouple materials decrease as well as the output sensitivity of available thermocouple materials.

Newer thermocouple configurations are plated rather than wire. The output sensitivity of these plated thermocouples is about half that of the corresponding wires. Techniques are, in principle, available to plate extremely small and thin single and multiple thermocouples on "selected" substrate. Plated thermocouples have been studied by several groups; the Vatell heat flux gage is one attempt to use this technology in a fabricated heat gage. The major challenge is broadening the domain of applicability of these plated thermo-sensors.

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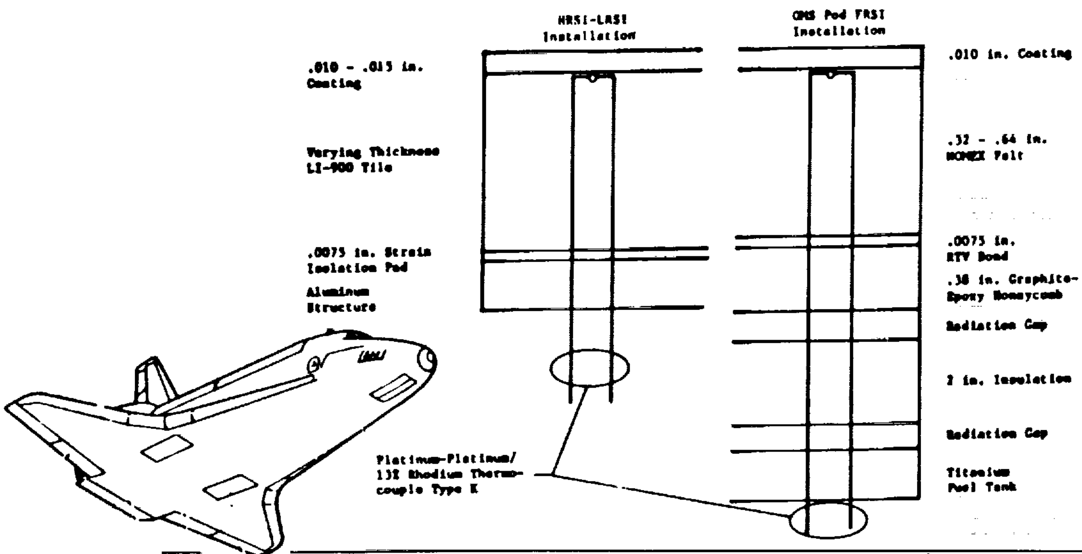


Figure 1, Space Shuttle Thermal Instrumentation

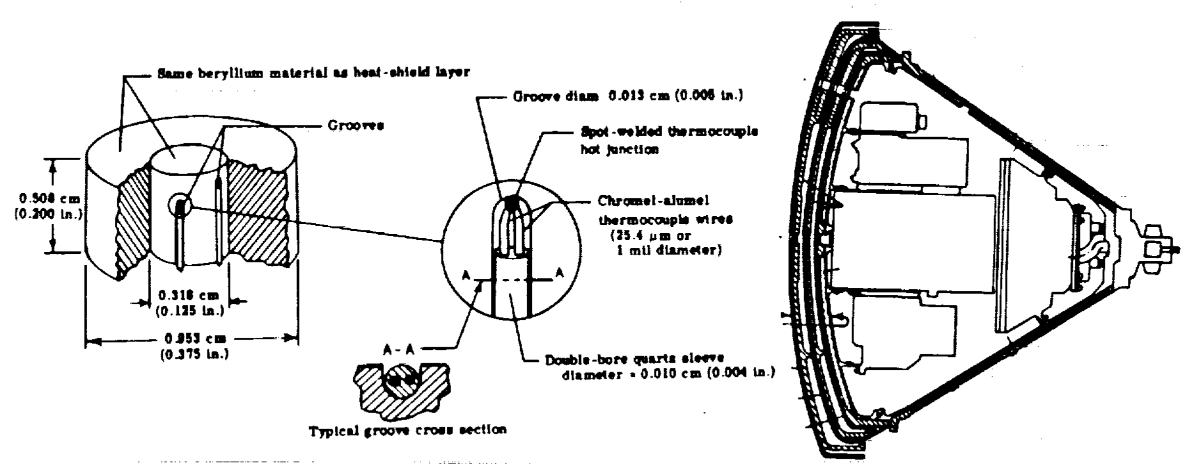


Figure 2, NASA FIRE Project Thermal Instrumentation

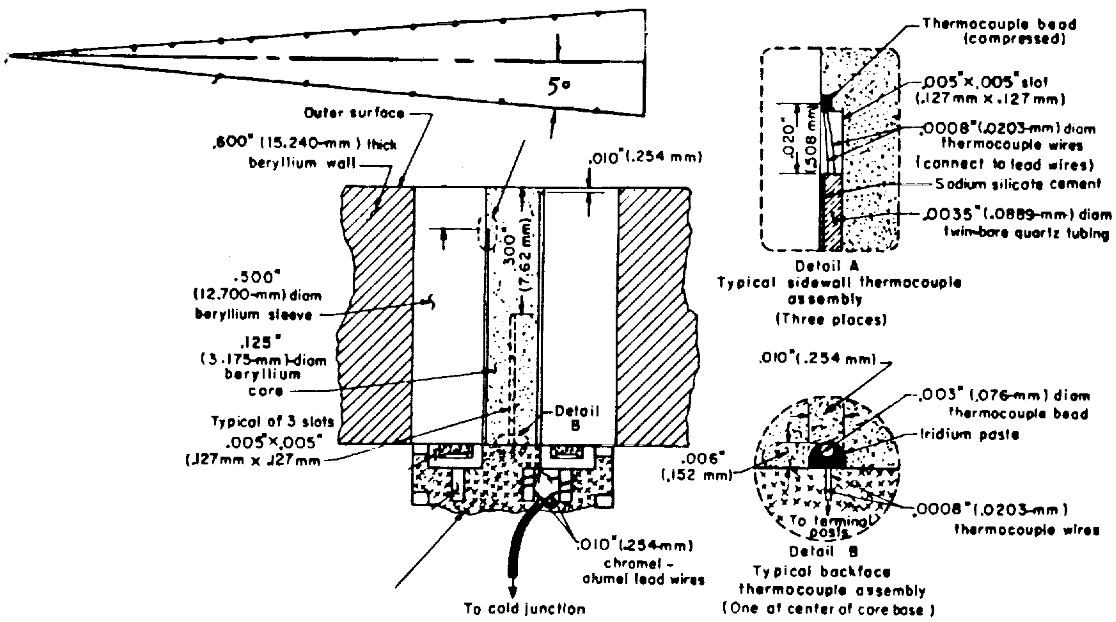


Figure 3, NASA Re-entry "F" Project Thermal Instrumentation

Gross difference is due to uncertain depth of thermocouple below the outer mold line.

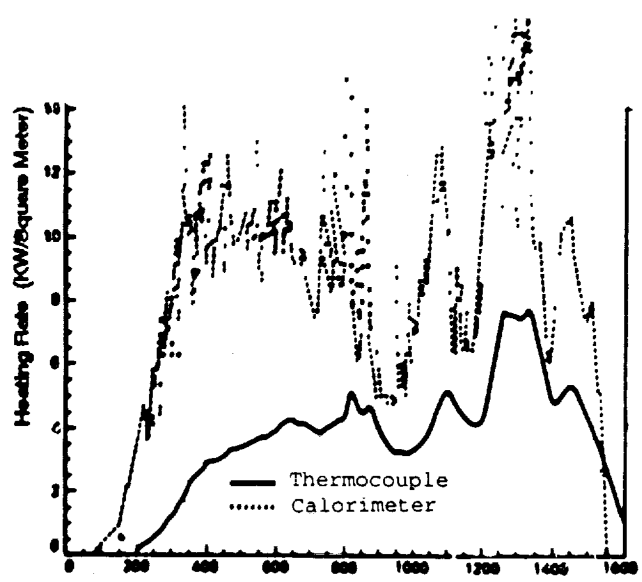


Figure 4, Difference Between Calorimeter and Thermocouple-Based Heat Transfer Inference

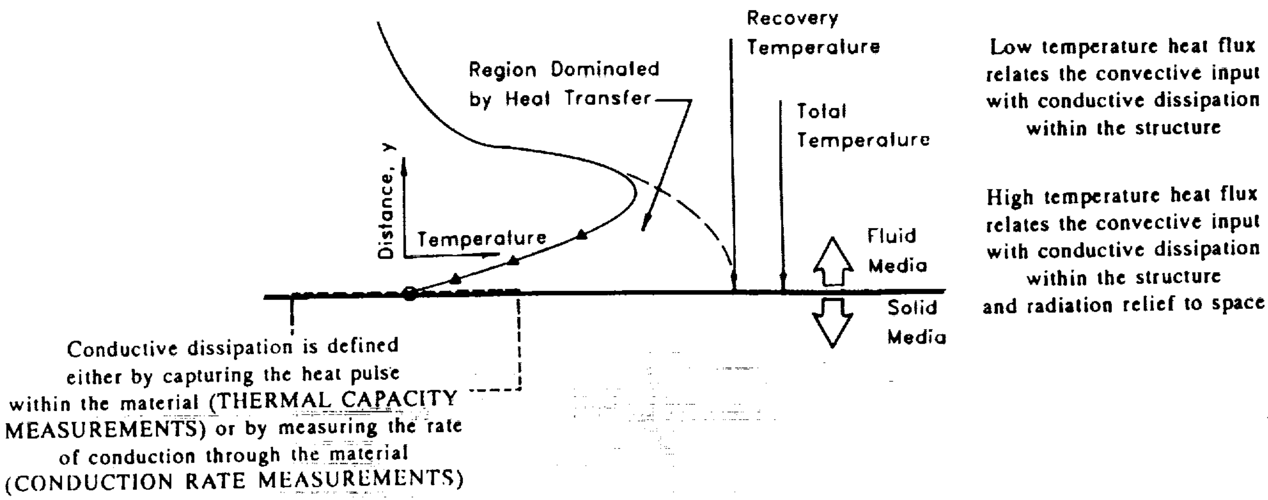


Figure 5, The Process of Heat Transfer Rate Inference

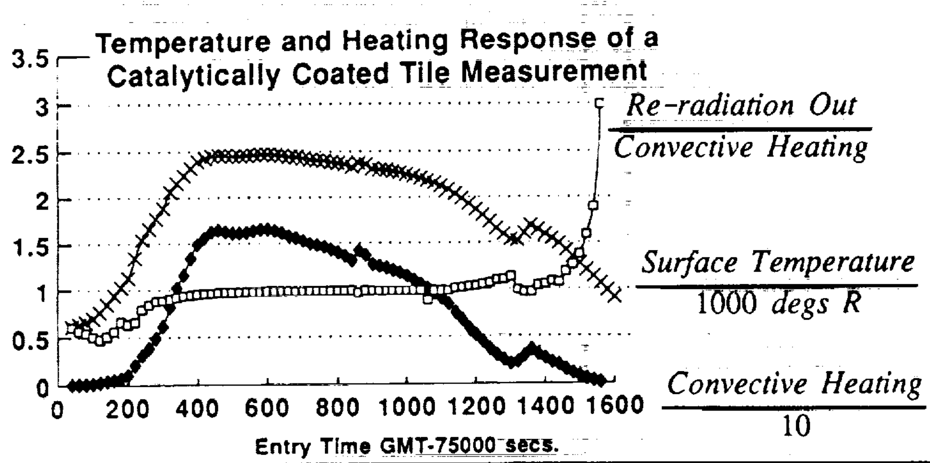


Figure 6, Temperature and Heating Response of a Catalytically Coated Tile Measurement on the Space Shuttle

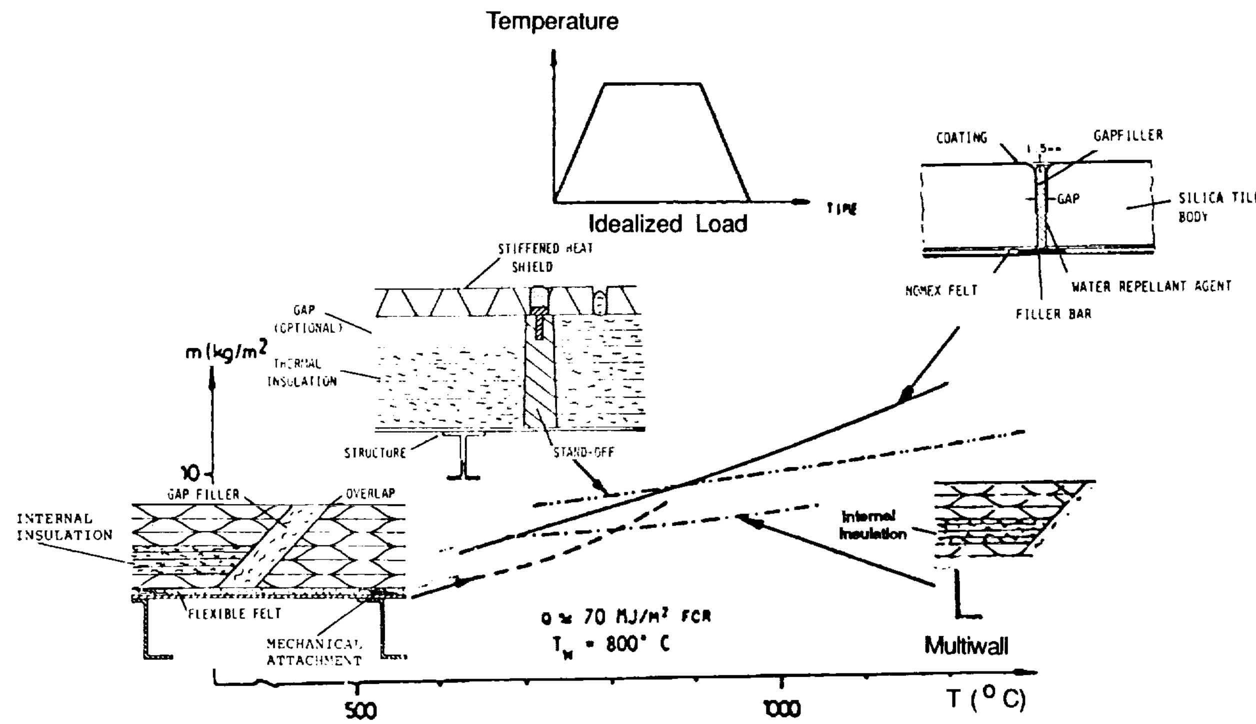
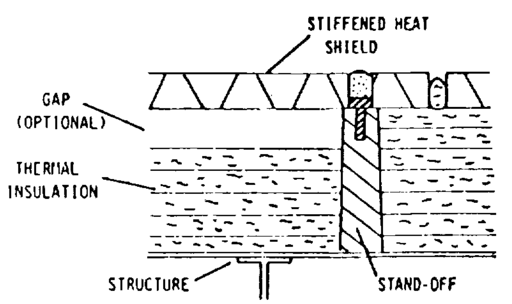
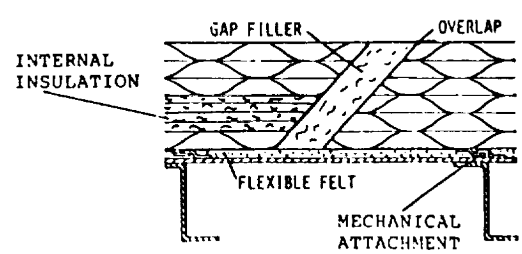


Figure 7, Trend of TPS Mass Per Area vs Temperature for Representative TPS Candidates from Gallert and Keller



Stifferend heat shield is a rigid ceramic shingle of carbon fibers in a silicon carbide matrix or silicon carbide fibers in a silicon carbide matrix. Shield surfaces are, typically 0.020 ins thick.



Metallic multiwall concept is either nickel or cobalt based alloy for use up to 1000 degs C and molybdenium based alloy for use up to 1300 degrees C. Typical thickness is 0.002 to 0.004 ins.

Figure 8, Details of the Scale of Proposed TPS Systems from Grallert and Keller

Figure 9, Early Use of Temperature Sensitive Paint on the X-15

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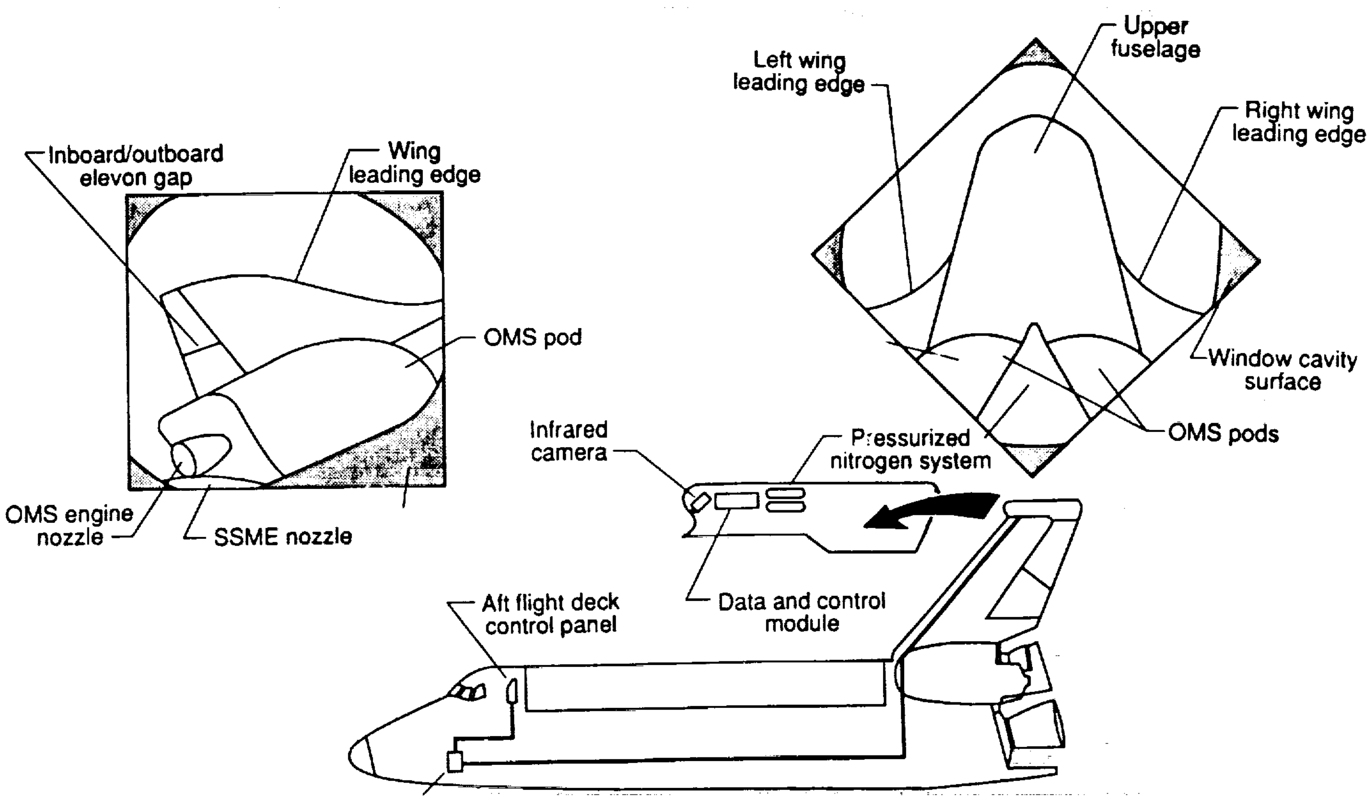

Figure 10, Overall View of the IR Flight Experiment on the Space Shuttle Flight Vehicle, Columbia

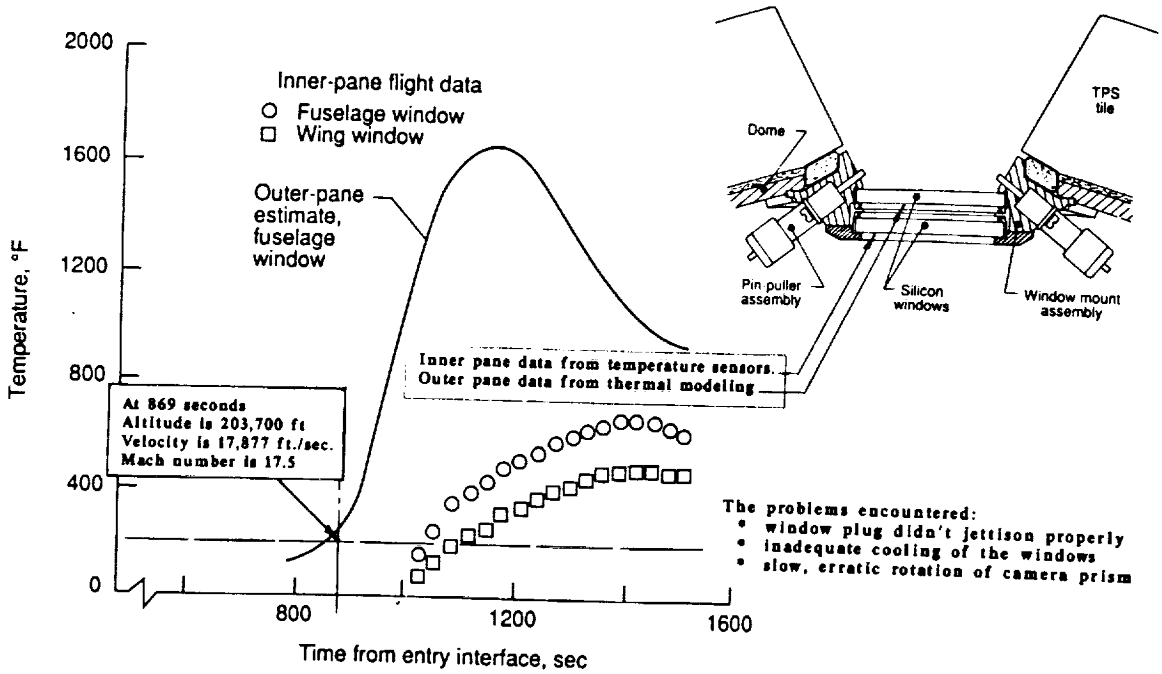

Figure 11, Discussion of the Problems in the IR Flight Experiment

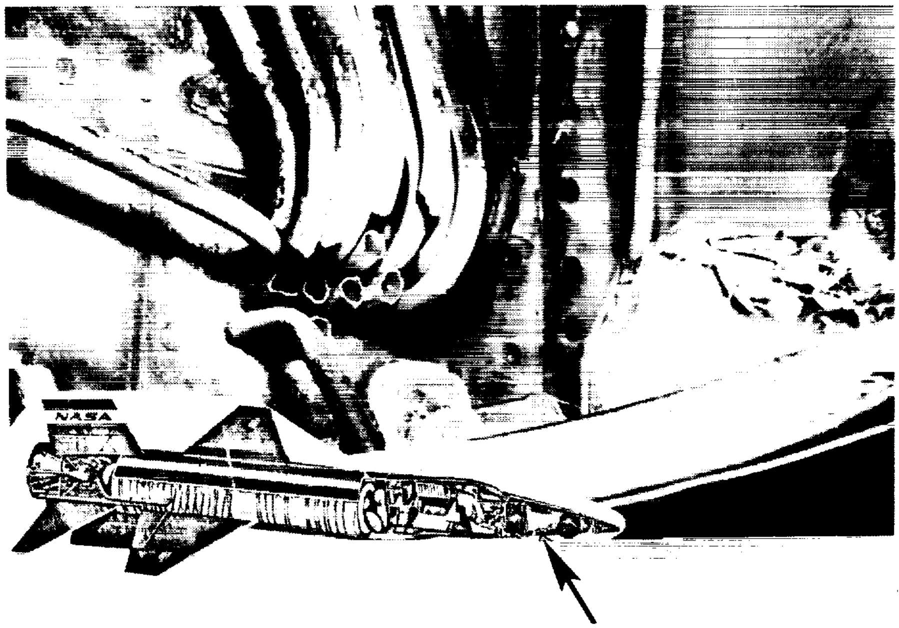

Figure 12, Example of "Sneak Flow" in the
X-15 Nose Wheel Cavity

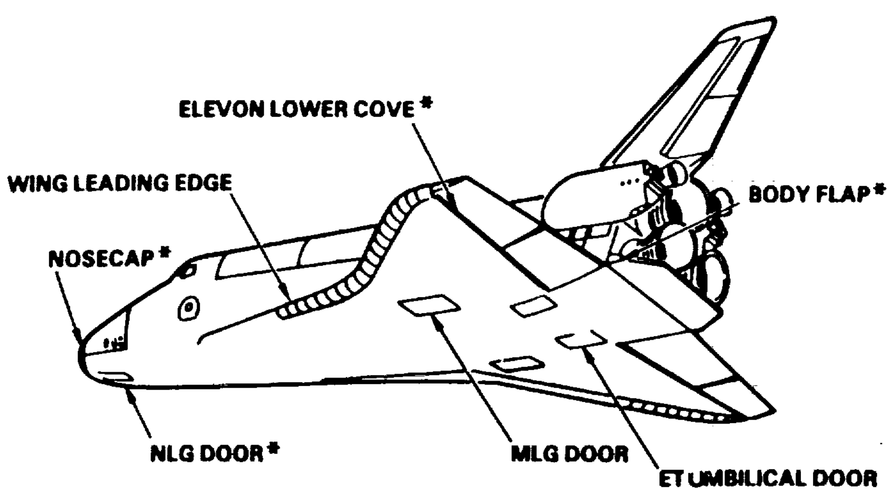

Figure 13, Penetrations on the Space Shuttle Orbiter