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Westinghouse Astronuclear Laboratory

A PROGRESS REPORT
ON THE DEVELOPMENT OF
REENTRY BODY ANALYSIS TECHNIQUES

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REENTRY BODY ANALYSIS TECHNIQUES

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I. INTRODUCTION

Recent activity on technical proposals associated with isotopic power supply units for use in near earth orbits has emphasized the need for WANL capability to analyze reentry vehicle and heat shield requirements. Consequently, the Systems Engineering Department has been chartered under Product Development funding to assemble the basic analytical "tools" necessary to carry out some limited-scope heat shield evaluations. This report is a summary of the status of this effort.

The purpose of a heat shield is to protect the payload from thermal damage during high speed atmospheric reentry. As a thermal protection device, the heat shield specifications are influenced by:

- 1) The thermal and mechanical properties of the payload being protected.
- 2) The properties of the heat shield material.
- 3) The trajectory of the re-entering body through the atmosphere.
- 4) The motion of the body itself, i.e., pitching, yawing, and rotary movement about its center of mass during reentry.

It is obvious from the above considerations that heat shield evaluation is a complex problem, embodying at least the following basic computational blocks:

- a) A trajectory simulation program taking into account both the payload's motion about the earth's mass center and its motion about its own mass center.
- b) An aerodynamic heating program to determine the convective and radiative heat flux to the skin of the heat shield from the air stream.
- c) Heat transfer programs to determine the thermal history of pre-selected points within the heat shield and/or payload.

These computational blocks are not necessarily independent of one another. For example, the trajectory of a body and the body's motion can be radically altered as an ablative heat shield loses mass to the airstream. Such mass loss effects could, depending on the body geometry and initial reentry parameters, render the payload into a dynamically unstable condition in which the entire reentry package would be subsequently destroyed from reentry heating occurring over the entire body surface.

Generally, it is not practical to assemble all three computational blocks into one computer program. The reason is that each block is, in itself, a reasonably large program and computer installation limitations must be considered as well as the difficulties always attendant to fusing large programs together.

One acceptable alternative is to use an iterative procedure wherein the appropriate results from the trajectory and aero-heating programs are used as input into the heat transfer programs. The findings obtained from the latter are then used to modify the initial input to the trajectory program (if necessary) and the cycle repeated until a convergent solution is obtained. It was felt that for the present at least, the iterative procedure would be sufficient, and in the time available, the best course of action would be to get in operational form the necessary analytical tools to handle reentry shield calculations for the non-ablative (heat sink) type of shield. Thus the effort expended was directed toward:

- 1) Modifying and converting the NASA Manned Spacecraft Center's generalized 3 and 6 degree of freedom trajectory program (SGGEM) to operate on the Westinghouse CDC-6600 computer system.
- 2) Constructing a thermal model to analyze the transient heat transfer within the heat shield as a function of the SGGEM output.
- 3) Performing a literature search and compiling information on the behavior of typical ablative heat shield materials to obtain performance data and working knowledge of these types of shields.

- 4) Performing a study on the theories and approaches to calculate the aerodynamic force, moment and stability coefficients in hypersonic continuum flow for typical blunt reentry bodies (for use with the 3 or 6 degree trajectory program).
- 5) Carrying out a trial run in an attempt to duplicate known results for a beryllium shield on a Discoverer shape body.

In the following sections, a resume will be given describing the progress, status, current capability and results as appropriate to each of the first four items listed above. Item 5 is partially covered here and will be more completely documented in a supplemental report.

II. THE SGGEM PROGRAM

The SGGEM Program is a generalized trajectory computational system that consists of four simulations or packages. Each of the packages in turn contains one or more trajectory programs. The user specifies, by means of input data, which trajectory program within the chosen package is to be executed. The four packages available are:

- Two-Dimensional Space Simulation
- Three-Dimensional Space Simulation
- Multi-vehicle Simulation
- Interplanetary Simulation

The trajectory programs contained in each of the packages are assigned unique input call numbers. A brief description and the corresponding number of each of the trajectory programs follows:

1. Two-Dimensional Space Simulation

- a) Subprogram 21: Three Degrees of Freedom Trajectory Program.
This subprogram computes translational and rotational motion of a rigid body in two-dimensional space with the angular velocity of the body computed from the moment on the vehicle.
- b) Subprogram 22: Quasi-Three Degrees of Trajectory Program.
This subprogram computes translational and rotational motion of a rigid body in two-dimensional space with the angular velocity of the body dictated by a control system.
- c) Subprogram 23: Two-Dimensional Commanded Attitude Trajectory Program.
This subprogram computes translational motion of a rigid body in two-dimensional space with the body orientation dictated by a control system.

- d) Subprogram 24: Two Degrees of Freedom Trajectory Program.
This subprogram computes the translational motion of a point in two-dimensional space.

2. Three-Dimensional Space Simulation

- a) Subprogram 31: Six Degrees of Freedom Trajectory Program.
This subprogram computes translational and rotational motion of a rigid body in three-dimensional space with the angular velocity of the body computed from the moments on the vehicle.
- b) Subprogram 32: Quasi-Six Degrees of Freedom Trajectory Program.
This subprogram computes the translational and rotational motion of a rigid body in three-dimensional space with the angular velocity of the body dictated by a control system.
- c) Subprogram 33: Three-Dimensional Commanded Attitude Trajectory Program.
This subprogram computes the translational motion of a rigid body in three-dimensional space with the body orientation dictated by a control system.
- d) Subprogram 34: Three Degrees of Freedom Particle Trajectory Program.
This subprogram computes the translational motion of a point in three-dimensional space.

3. Multi-vehicle Simulation

- a) Subprogram 35: Three-Dimensional Commanded Attitude Trajectory Program for up to Four Vehicles.
This subprogram operates in the same manner as Subprogram 33.

4. Interplanetary Simulation

a) Subprogram 44: Three Degrees of Freedom Particle Trajectory Program.

This subprogram computes the translational motion of a point in three-dimensional space with the perturbations resulting from the planets, sun, and moon taken into account.

The SGGEM Program permits the running of simulations with any of the above subprograms in any of three available modes viz. normal mode, parameter mode, and boundary value mode. In addition, various output formats are optional in all modes.

The normal mode is by far the one most frequently used. A program operating in the normal mode reads data from the input device, integrates the equations of motion to a specified end condition, and returns to read more data. This sequence of reading data and integrating to termination is called a phase. A case consists of one or more phases, and a run consists of one or more cases. The importance of the concept of phases is discussed below.

The parameter mode allows the program to return to an arbitrary point in the normal trajectory, adjust any single input parameter to one or more preset values, and repeat the remaining phases in a case for each value. An option is available to allow linear interpolation and/or extrapolation on an input parameter to reach a given end condition. Two choices are available for the interpolation; either the program will run a series of cases with the input parameter set to given values until the end condition is bracketed, or one step-off value of the input parameter may be run with interpolation and/or extrapolation starting immediately.

The boundary value mode allows the program to return to an arbitrary point in the normal trajectory and adjust up to six initial boundary parameters in order to satisfy the required end conditions for a given trajectory.

Most trajectory simulations involve the numerical integration of functions which are, at most, piece-wise continuous. The discontinuities are produced by such physical requirements as abrupt mass changes produced by release of spent motor cases, rocket motor ignition or shut-off, initiation of guidance mechanisms, etc. Other less obvious discontinuities are introduced where tabular data for aerodynamic coefficients, rocket parameters, etc., are not presented as smooth functions. Unless all such discontinuities are properly handled in the simulation, large errors may be introduced whenever these discontinuities occur within a single integration step.

The presence of these discontinuities (i.e., those associated with abrupt mass changes, etc.) requires that the program user subdivide the computation into a number of "phases" such that no discontinuities occur within any given subdivision. Thus, a phase separates discontinuities (or other desirable breakpoints) in the mathematical model and a run will denote a trajectory computation beginning at same initial point and may encompass one or more phases. SGEM permits the user to make any number of runs in sequence on the machine.

Although most computations will normally use only one of the major subprograms in SGEM, the program user may, via input cards, shift control from one subprogram to another at will during one run. For example, one could use six degrees of freedom subprogram from launch up to the top of the sensible atmosphere, temporarily store orientation and angular rate data at this point, shift to a particle subprogram "over the top", reset the orientation and angular rate data at reentry, and shift back to a six degrees of freedom computation for the rest of the flight. Such manipulations are controlled entirely by appropriate integers on input cards. In each of the subprograms, several phases of computation could, if need be, have been used as noted above.

At the beginning of each run or phase within a run the program user must specify the necessary initial conditions for that part of the computation as well as the condition which

determines when that phase ends. In general, a complete list of initial conditions is required only at the start of the first phase; for later phases only those quantities which the user desires to change need be entered.

Termination of a phase may be accomplished when any quantity computed by the machine reaches a pre-specified value. Thus, one can terminate any phase on the basis of the attainment of a velocity, altitude, time, or any of the quantities given in the list of variable indices which accompanies the operating instructions for each subprogram. This termination is accomplished by specifying that the machine automatically monitor the "termination quantity" specified by the user and continue the integration until the desired termination point is encountered or passed.

The basic input scheme to start a run, in addition to the mode, simulation and initializations control cards, involves specifying the (1) initial velocity, altitude and flight path angle, (2) position in space, (3) body geometry, (4) body orientation and angular orientation rates, (5) reference body weights and aerodynamic areas and lengths, (6) option control to select the types of aerodynamic calculations, atmosphere models, etc., (7) output quantities and output format, (8) frequency of output, (9) error control parameters, and (10) various control cards to handle the tabular input data noted below.

The evaluation of aerodynamic coefficients, rocket parameters, and any physical parameter such as hypersonic shock wave characteristics is performed by numerical interpolation in tables. Each table is controlled by an input "table control" card as noted above. Tabular data is entered with a prescribed format. Any tabular quantity computed by various interpolation schemes within the SGGEM program can be a function of up to three independent variables as specified on the table control card for that particular table. The importance of tabular data lies in the fact that through their judicious use, the simulation of the motion of an arbitrary body through a planetary atmosphere can be described in terms of either wind tunnel data or theoretical calculations as a function of the body geometry, angle of attack,

etc. The ability to incorporate such features into a program is necessary to properly simulate six degree of freedom motion*.

As a typical indication of the program flexibility, on input, the position of the body in space can be specified by any of the following three ways:

- a) Rectangular coordinates relative to the inertial frame.
- b) Altitude, geodetic longitude and geodetic latitude.
- c) Radial distance from the center of the earth, geocentric longitude and geocentric latitude.

The velocity of the body can be expressed upon input in six different systems:

- a) Inertial system components of inertial velocity.
- b) Geocentric system components of inertial velocity.
- c) Geodetic system components of inertial velocity.
- d) Inertial velocity magnitude and associated path angle and azimuth angle.
- e) Magnitude of velocity relative to the earth and associated path angle and azimuth angle.
- f) Magnitude of velocity relative to the air (local wind) and associated path angle and azimuth angle.

The body orientation can be represented by any one of the following:

- a) Direction cosines (relative to the inertial frame).
- b) Body attitude angle (similar to path angle of velocity), azimuth angle and bank angle.
- c) Angle of attack, angle of sideslip and bank angle.
- d) Euler angles.

* Six degree of freedom motion refers to following the change of position of a body's center of mass in a spatial coordinate system and at the same time following the motion of any point on the body about the body's center of mass. Thus for general motion of a rigid body moving in space, there are 3 degrees of translational motion (i.e., changes in the X,Y,Z position of the body in space-fixed coordinate system and 3 degrees of rotation motion about the body's own mass center.

Output is completely arbitrary; any quantity computed by the subprogram can be printed. Hence, the trajectory and vehicle motion may be determined relative to the assumed inertial frame, relative to an oblate spheroidal earth, etc.

It is noted that the desired simulation complexity dictates the complexity of the program representation. Also, additions and modifications can be made to any subprogram if, in its present form, it does not represent the desired simulation.

Historically speaking, the SGGEM program is a descendent of the MASS (Missile and Satellite Systems) program. The MASS program was used by the Manned Spacecraft Center in the Mercury Astronaut Program and an updated version of MASS in the Gemini Program. WANL was given access to the original MASS program during safety studies on the intact reactor reentry problem. The original MASS program was supplied to WANL in binary form (machine language) only and was run on the IBM-7094 in somewhat of a "black box" fashion. The conversion to the CDC 6600 system necessitated a FORTRAN IV source language version of the MASS program which was in due time supplied by NASA-Houston as the SGGEM program. The SGGEM program was written for the IBM 360 and Univac systems and in converting the program to operate on the CDC 6600 a number of problems cropped up. Some of these were due to fundamental differences between computer hardware, and others were due to differences in local system software (compiler, etc.) options. Once the troubles had been diagnosed, extensive reworking of the SGGEM routines for input/output, tabular data handling, and tape manipulation was required.

Previous experience with the MASS program had indicated that because of the desired flexibility in operating the program, the input data scheme was inherently somewhat complicated, and thus, the opportunities for input data errors were numerous. To aid in checking out and denoting any errors in the input, a new subprogram was written. This program, named AUDIT, reads all tabular and non-tabular input data, checks and denotes the occurrence of any errors, and as part of the output lists all the input data together with appropriate error messages. If input data errors are detected the AUDIT program will pass over that particular

problem run and go to the next problem (if there are stacked problems). This latter feature was not available on the original MASS or SGGEM programs. Since such a read/checkout system had to be woven into the existing fabric of the SGGEM program, a considerable number of problems naturally arose in a modification of this type. In fact the bulk of the effort on this project was devoted to the conversion of the SGGEM program to operate on the Westinghouse CDC 6600 and the subsequent AUDIT modification.

III. EVALUATION OF AERODYNAMIC COEFFICIENTS

In order to determine the motion and heating of a body passing through the atmosphere, it is necessary to know the forces applied to the body from the airstream. In this section a brief treatment of the analysis performed to evaluate the drag coefficients of a Discoverer-shaped body is given. The results obtained are compared with wind tunnel data obtained from the literature subsequent to the calculations.

There are several theories which are available that can be employed to determine the lift, drag, normal and axial force coefficients and the various stability parameters for a body in hypersonic continuum flow*. The best of these theories for use with the Discoverer body geometry is the hypersonic blunt body theory in which the pressure at any point on the surface of the body that "sees" the flow is a function of the angle between the local surface and the pressure on the most forward (or stagnation) point in the airstream. This has been expressed mathematically as:⁽¹⁾

$$C_{P_L} = C_{P_s} \cos^2 \theta \quad (1)$$

where θ = angle between local surface and stagnation point.

Equation (1) is often called the modified Newtonian impact flow theory. The basic Newtonian hypersonic impact flow theory simply states that the shock wave lies on (or very close to) the body surface and that only the normal component of the free stream gas momentum passing into the shock will give rise to a surface pressure on the body. It can be shown⁽²⁾ that this requires high Mach Numbers and specific heat ratios approaching the

* Continuum flow refers to the fact that the atmosphere behaves as a continuous medium and any characteristics of individual molecules are masked. This is in contrast to more rarefied flow regimes (e.g., free molecule) in which molecules can act independently.

value of one for the gas in the shock layer (the region between the leading edge of the shock and the body surface). A brief discussion will be given below to demonstrate the validity and application of the modified Newtonian theory for a Discoverer shaped body in hypersonic flow.

The theory of oblique and curved shocks specifies that upon passage of the airstream through the shock wave, the law of conservation of mass normal to the shock is maintained and that the momentum of the gas flow parallel to the shock is conserved. Thus, referring to Figure 3-1, the pressure balance across the shock (the normal component of the flow) is:

$$P_{\infty} + \rho_{\infty} U_{\infty}^2 = P_s + \rho_s U_s^2 \quad (2)$$

where P_{∞} = free stream static pressure
 ρ_{∞} = free stream density
 U_{∞} = normal component of free stream velocity at shock
 P_s = static pressure just behind shock wave
 ρ_s = density just behind shock wave
 U_s = normal component of velocity behind shock wave

Equation (2) can be rewritten as follows:

$$P_s = P_{\infty} + \rho_{\infty} V_{\infty}^2 (1 - k) \sin^2 B \quad (3)$$

where $k = \rho_{\infty} / \rho_s$ density ratio across shock
 B = local angle between shock and free stream direction

At the stagnation point the shock wave is normal to the flow stream (for blunt bodies) and equation (3) reduces to:

$$P_s = P_{\infty} + \rho_{\infty} V_{\infty}^2 (1 - k) \quad (4)$$

Figure 3-1. Pictorial Representation of Blunt-Body and a Hypersonic Shock Wave.

Assuming that the flow from inside the shock wave to the body surface is incompressible, then the Bernoulli equation can be applied. That is to say, the relation between the pressure on a local surface of the body and the pressure just behind the shock is:

$$P_s + \frac{\rho_s V_s^2}{2} = P + \frac{\rho_s V^2}{2} \quad (5)$$

where P_s = pressure at body surface

V = velocity of gas in shock layer at stagnation point

V_s = velocity of gas just behind shock layer

Substituting equation (4) into (5) gives the following:

$$P - P_\infty = P_\infty V_\infty^2 (1-k) + \frac{P_s V_s^2}{2} - \frac{P_s V^2}{2} \quad (6)$$

The local pressure coefficient relating the local pressure on a body surface to the free stream pressure is, by definition:

$$C_p = 2(P - P_\infty) / \rho_\infty V_\infty^2 \quad (7)$$

Therefore, by rearranging equation (6) and comparing the result with equation (7), it can be shown that the local pressure coefficient, C_p can be expressed as:

$$C_p = 2(1-k) + \frac{P_s}{P_\infty} \left[\left(\frac{V_s}{V_\infty} \right)^2 - \left(\frac{V}{V_\infty} \right)^2 \right] \quad (8)$$

By recognizing from the continuity of mass flow normal to the shock that $\frac{V_s}{V_\infty} = k$, and from the shock layer theory of Li and Geiger⁽³⁾:

$$\left(\frac{V}{V_\infty} \right)^2 = \frac{X^2}{R^2} (k) (2-k) \quad (9)$$

where X = arc length along shock wave

R = radius of shock wave (refer to Figure 3-1)

Then equation (8) can be reduced with the aid of (9) to:

$$C_p = (2-k) \left[1 - \frac{X^2}{R^2} \right] \quad (10)$$

Referring to Figure 3-1, the quantity X/R is equivalent to θ and, thus for small θ , equation (10) becomes:

$$C_p = (2-k) (1 - \theta) \rightarrow (2-k) (1 - \sin^2 \theta) = (2-k) \cos^2 \theta \left\{ \theta \rightarrow 0 \right\} \quad (11)$$

By comparing equations (1) and (11), it is seen the two are identical if the stagnation pressure coefficient, C_{ps} , is taken to be equal to $2-k$. Moreover, equation (11) stipulates that the modified Newtonian theory is applicable primarily around the stagnation point, or where θ is small.

It is surprising, though, that the modified Newtonian theory is reasonably valid for points on the body behind the shock wave considerably removed from the stagnation line. This is particularly true for bodies having a larger radius of curvature (the blunt body). Examining equation (10) with a large R , it becomes apparent that a greater latitude in values of X are allowable without violating the bounds of the modified Newtonian equation (eq. 11); i.e., $\theta < 1$.

Examining equation (8) shows that the last term in the brackets, $(V/V_\infty)^2$ is, in effect, a "pressure relieving" term which, from equation (9), gets larger with increasing distance (X) from the stagnation point. The result is that the modified Newtonian theory does in a sense take into consideration the increase in the resultant velocity in the shock layer with increasing angular distance from the stagnation point. The increase in shock layer velocity will effectively decrease the local pressure on the body surface.

The modified Newtonian theory predicts that the surfaces that "see" the airstream will experience some pressure greater than the freestream pressure. In other words, at no point on any surface exposed to the flow will the pressure coefficient, C_p , vanish. There are theories⁽⁵⁾ which do predict for bodies of revolution a point where the pressure coefficient does go to zero and consequently where the shock layer flow pulls away from the body surface. Wind tunnel data, however, indicates that for a blunt body, the shock layer does not separate from the body surface. Flow separation, however, is observed on slender bodies (Large L/D) in moderate hypersonic velocities; i.e., Mach 3 to 5.⁽²⁾

Finally, more exhaustive analysis indicates that Newtonian-type flow will exist over spherical bodies down to Mach No. as low as 1.4,^(1,2) provided that the detached shock wave has the same shape as the body. For a blunt body of revolution having a large radius of curvature on the leading to surface, the shock curvature will follow closely the surface curvature. Thus for the Discoverer body shape, it could be expected that the pressure coefficient follows reasonably well the modified Newtonian impact theory down to low Mach numbers, viz., to sonic velocity.

The foregoing discussion is admittedly somewhat tedious. But it cannot be over-emphasized that a properly evaluated pressure coefficient is absolutely necessary in order to evaluate the axial and chord forces and moment coefficients on a rigid body during hypersonic reentry. The previous discussion gives only a partial insight to the problem of evaluating the pressure coefficient.

The determination of the axial and normal forces coefficients on a body will be by-passed in this report. However, these force coefficients can be shown to be a function of the local pressure coefficient, the angle of attack, angle of yaw (sideslip) and the body geometry.⁽⁶⁾ The lift and drag forces as well as the pitch and yaw moment can then be determined by resolving the normal and axial forces. The angular accelerations about the body's center of mass and can be determined from the moment of the resultant forces. The rate of change in the pressure coefficient due to the change angle of attack and angle of sideslip can also be theoretically predicted.

It becomes apparent that the analytical determination of the moments, forces, and body motion as a function of the airstream characteristics is indeed a difficult problem. For the problem at hand, it was assumed that the Discoverer shaped body was flying completely trimmed, or in other words, had a zero angle-of-attack and no angular motions. A real situation with trimmed flight would be the reentry into the earth's atmosphere following orbital decay. For a zero angle-of-attack trimmed reentry, the axial force coefficient is equivalent to the drag coefficient. The lift coefficient is zero for a symmetrical body of revolution in trimmed flight. Therefore, the problem of predicting aerodynamic forces acting on the Discoverer body in trimmed flight becomes one of evaluating the drag coefficient. Then the subsequent reentry trajectory becomes a function of only the body's mass (gravitational acceleration) and the acceleration produced by drag force.

Referring to Figure 3-2, the elemental drag per unit area, $\frac{dD}{dA}$, is:

$$\frac{dD}{dA} = P_L \cos \theta \quad (12)$$

where P_L = local pressure

θ = angular displacement from body center line of symmetry

Now the elemental surface area on a spherical shape can be shown to be $2 \pi R^2 \sin \theta d\theta$, and recalling the definition of the local pressure coefficient (Equation 7), the above relationship can be expressed in integral form:

$$D = 2 \pi R^2 \int C_{p_L} q_{\infty} \cos \theta \sin \theta d\theta \quad (13)$$

Substituting into equation (13) the modified Newtonian theory expression for C_{p_L} results in the following:

$$D = 2 \pi R^2 q_{\infty} (2 - k) \int_{\theta_i}^{\theta_k} \cos^3 \theta \sin \theta d\theta \quad (14)$$

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Figure 3-2. Pictorial of the Elemental Drag Area.

Equation (14) is directly integrable for a given radius of curvature, R , and between any angular limits θ_i to θ_k . The Discoverer body has two radii of curvature as is shown in Figure 3-3. Hence, equation (14) must be evaluated for each of the two frontal area regions corresponding to the two radii. Thus the total drag on the Discoverer body can be shown to be:

$$D_T = D_1 + D_2 = \frac{\pi R_c^2}{2} q_\infty (2-K) \left[(1.09)^2 (1 - \cos^4 \theta_i) + (0.726)^2 (\cos^4 \theta_k - \cos^4 \theta_i) \right] \quad (15)$$

where R_c = base radius of the cone

$$\begin{aligned} \theta_i &= 16.5^\circ \\ \theta_k &= 80.0^\circ \end{aligned}$$

The drag coefficient is commonly expressed as follows:

$$C_D = D_T / A_R q_\infty \quad (16)$$

where D = total drag

A_R = reference area

$$q_\infty = \rho_\infty V_\infty^2 / 2$$

Thus by comparing equations (15) and (16), and noting that the reference area is πR_c^2 , then upon substituting the appropriate angular values into equation (16) the drag coefficient becomes:

$$C_D = \left(\frac{2-K}{2} \right) \quad (0.7)$$

It can be shown that the density ratio across the shock is a function of the freestream Mach Number or more specifically:

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Figure 3-3. Discoverer Geometry

$$k = \frac{(2/M_1^2) + \gamma - 1}{\gamma + 1} \quad (18)$$

In hypersonic flow, i.e., $5 \leq M_1 \leq \infty$, k values for a perfect diatomic gas lies between 0.2 and 0.167. Thus it is to be expected that C_D values would be approximately equal to 0.64 throughout the hypersonic continuum flow regime.

On the other hand, if the gas dissociates ($\gamma \rightarrow 1$) then the high Mach Numbers ($M_1 \rightarrow \infty$) the density ratio would tend toward zero and C_D would approach the value of 0.7. The Newtonian impact theory of aerodynamics implies that $\gamma = 1$. Hence, if the modified Newtonian theory is used, then it could be expected that the drag coefficient would vary between 0.63 and 0.7 for a Discoverer body in hypersonic flow. In the work cited in Section V, the upper limit, $C_D = 0.7$, was used throughout the continuum flow regime.

It is interesting to point out that in Reference 4, ballistic-range tests revealed experimental C_D values of approximately 0.7 for a Discoverer-shaped body at an angle of attack near zero. Figure 3-4, which has been taken from that reference shows the agreement between the modified Newtonian theory at Mach = 3.5 and 8.5 and the experimental C_D values. Note that the spread of C_D as calculated by theoretical considerations lies in the range of 0.66 to approximately 0.69 (at zero angle of attack) which is in agreement with the findings above. It is believed that the experimental results substantiate the validity of the modified Newtonian impact theory and furthermore it can be shown that the impact theory yields quantitatively acceptable results for the aerodynamic stability derivatives. However, for first designs and evaluations of heat shields, the use of trimmed flight is perfectly permissible and proper, and the rigorous mathematical treatment necessary to evaluate the stability, etc., can be dispensed with, thus treating the problem primarily as a three-degree trimmed flight.

Figure 3-4. Drag as Function of Angle-of-Attack and Mach Number. (4)

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IV. THERMAL ANALYSIS OF REENTRY VEHICLE HEAT SHIELDS

The specific problem is to determine whether a particular heat shield design is capable of preventing excess heat flow to the various members of the payload structure or subsystem. Solution of the problem entails setting up a detailed model of the heat shield and, perhaps, the payload. Boundary conditions are established from previously generated shock layer heating rates from the SGGEM program and knowledge of the payload. If any limits are exceeded, heat shield thicknesses are adjusted and the aerodynamic heating calculations are re-run. Alternately, it may be determined that a change in the type of thermal protection system or material is warranted.

A. The CINDA Program

The primary tool chosen to handle the heat transfer problems within the shield is the Chrysler Improved Numerical Differencing Analyzer (CINDA). This program was developed by the Chrysler Corporation Space Division at the NASA Michoud Assembly Facility⁽¹²⁾ and is currently in use throughout the aerospace industry.

CINDA is a dimensionless, multi-option computer program. It constructs and analyzes a mathematical model of any arbitrary one-, two-, or three-dimensional lumped parameter representation of a physical system governed by a set of diffusion equations, e.g., the Fourier equation with an additional source term. Utilization of the program involves the construction of a thermal analog network representation of the physical system. Material properties and boundary conditions may be calculated simultaneously as a function of one or more independent variables. The program options offer the user a variety of differencing methods for the solutions of the set of the resulting simultaneous equations, each differing in machine speed, core space required and solution accuracy.

In developing the CINDA program, many of the thermal analyzer type programs used both in government and private industry were studied. The development of a single computer program was sought which made available all of the advantages of the various programs studied with as few disadvantages as possible. In order to include the various individual programs as components of CINDA, it became necessary to develop a systems compiler program that automatically optimized the utilization of computer core space. The end result is that once a particular problem has been processed by the compiler portion of CINDA, only the data and instructions necessary for its solution are brought into the core from the master program tape. Thus, this minimization technique make virtually the entire core available for problem solution.

The master program tape makes available many general subroutines that can be sequenced to perform a more complex operation. Additional computational and complex manipulative operations may be programmed and added to the master program tape as required by the user.

To utilize a thermal analyzer type program, a user must construct a lumped-parameter representation of the physical problem. To do this, the physical system is represented as a finite number of small elemental volumes. The geometry and physical properties of each volume and its relationship to all other volumes is an input to the program. Thermal analyzer programs solve transient non-linear problems by quasi-linearization of the non-linear set; i.e., all variables other than the state variable (temperature) are held constant over a small computer interval. The program performs a large number of integrations on the quasi-linear set of equations and allows material property and boundary condition changes (non-linearities) to be performed between the time steps. The program features means for calculating boundary conditions (aeroheating, etc.) or non-linear material properties which are not actually a basic part of the network problem. They are add-on features which do increase the program's capability but should in no way interfere with the network solution.

The program user combines the volume elemental properties of the lumped parameter representation into a thermal capacitance term (capacitance = volume*density*specific heat) which is considered to be concentrated at a point or node in the center of the elemental volume. This procedure is performed for all the elemental volumes. When two elemental volumes are in physical contact with each other, the user specifies that an energy flow path (thermal conductor) exists between them (conductance = thermal conductivity times path cross-sectional area/path length between nodes). An internal node in a one-dimensional problem will have two attached conductors; for a two-dimensional problem, there will be four attached conductors, etc.

Hence, a simple one-dimensional network problem such as is depicted in the following analog scheme

yields the following set of energy balance equations (T1 is a boundary temperature node):

$$C2(T2' - T2) = (G1(T1 - T2) + G2(T3 - T2)) \Delta t$$

$$C3(T3' - T3) = (G2(T2 - T3) + G3(T4 - T3)) \Delta t$$

$$C4(T4' - T4) = G3(T3 - T4) \Delta t$$

The prime indicates future temperatures after time passage Δt , and the C's and G's represent the thermal capacitance and conductance of the elemental volumes.

The standard mathematical approach for solving a network would be to construct the set as a matrix equation. Thus, in matrix notation:

$$(A) T = B \tag{1}$$

The term (A) is the coefficient matrix as follows:

$$(A) = \begin{bmatrix} G_1 + G_2 + C_2/\Delta t & -G_2 & 0 \\ -G_2 & G_2 + G_3 + C_3/\Delta t & -G_3 \\ 0 & -G_3 & G_3 + C_4/\Delta t \end{bmatrix} \quad (2)$$

T is the temperature column matrix and B is termed the boundary column matrix and includes all of the initial and boundary conditions. The above example is sufficient to show that the coefficient matrix for a one-dimensional problem (regardless of the number of nodes) consists of not more than three finite coefficients in any row or column. A three-dimensional problem would normally be limited to seven finite coefficients in any row or column. Thus for a 1000-node 3-D problem, the coefficient matrix would contain about 993 zeroes in every row and column for a total of approximately 99300 zero values, which greatly exceeds the number of core locations available in many present day computers. What is necessary is a method of storing and referring to only the finite coefficients in the coefficient matrix.

Hence, the basis or foundation of the thermal analyzer type programs is the method by which they store and manipulate only the finite coefficients. The CINDA program reads all node data in three groups: diffusion nodes, arithmetic nodes, and boundary nodes.

Diffusion nodes reserve core locations for temperature, capacitance and source values, arithmetic nodes reserve temperature and source locations and boundary nodes reserve space for temperature values only. Conductor data is read in as two groups: linear conductors and non-linear (radiation) conductors and the adjoining node pairs for each conductor.

The program compiler now constructs the data addressing information into a pseudo-compute sequence. The programming instructions necessary to construct the compute sequence require a large amount of core space, approximately 10,000 locations. However, once the operations have been performed, the instructions for producing the computing sequence are not needed for

the problem solution. For this reason, CINDA operates in a segmented fashion. The operations for forming the computing sequence are done by a compiler phase which outputs all of the data values and pseudo-compute sequence onto a working problem tape. Core location information is retained and the subroutine calls and arguments are processed and also output onto the working problem tape. When the above processing is completed, the program monitor takes control and reads the working problem data back into core over the core space which was used by the compiler. As a result, the 1,000 node 3-D problem would require, at most, only 10,500 core locations and the program can handle a problem consisting of more than 3,000 linear 3-D nodes.

Another outstanding feature of the CINDA program is that the user can explicitly specify a radiation conductor. This is most significant because radiation heat transfer plays an important role in determining the total heat transfer to a reentry vehicle.

A radiation conductor value must consist of:

$$\sigma A \mathcal{F} \quad (1)$$

where: σ = Boltzmann's constant
 A = Surface area of the radiator
 $\mathcal{F} = f(F, \epsilon)$
 F = Geometrical view factor
 ϵ = Emissivity

Since ϵ is a function of temperature then $\mathcal{F}$ is also temperature dependent. There are several available programs presently at WANL for determining view factors.

Because $\mathcal{F}$ is a temperature dependent variable another distinct advantage of the CINDA program is that it contains a special subroutine which calculates $\mathcal{F}$, via a matrix solution, before the next temperature iteration, and as a function of emissivities at the previous iteration temperatures.

As it now exists, the CINDA program can be used to calculate vehicle temperature distributions for any of the types of heat protection systems discussed in the next section with the exception of char-forming ablators. To examine char-forming ablating shields, a one-dimensional ablation program (STAB II) has been obtained.

It should be noted that analyses need not be limited to heat shields only. With the tools currently available it is possible to perform an analysis of the entire reentry vehicle including payload and structure with appropriate data input.

B. The STAB II Program⁽¹³⁾

STAB II has been written to analyze the transient thermal performance of a charring ablator heat protection system. The program considers one ablating material and up to twelve different backup materials with or without air gaps. Pure conduction or radiation and/or convection between backup materials is allowed. The ablation material may be divided into a maximum of 50 nodes, and each backup material may be subdivided into a maximum of 10 nodes. The thermal properties of the materials are in tabular form and are temperature dependent. Ablation material properties are also dependent upon the material's state, that is, whether the ablator is fully charred, partially charred, etc.

The following surface boundary condition options are provided:

- 1) Cold-wall convective and radiative heat flux tables as a function of time. These components are specified separately, since mass transfer at the surface blocks part of the convective heating but, in general, has no effect on the radiant heating.
- 2) Surface temperature as a function of time.
- 3) Surface recession as a function of temperature or time. Surface recession as a function of temperature and pressure is also available.

Heat loss to the interior environment for the last node of the backup structure can be specified by two methods:

- 1) Conduction into the node and radiation and/or convection loss to the interior environment.
- 2) Conduction into the node and adiabatic wall.

The tabular output presents the computed results in block type form for each computation step as controlled by the print count control number. Both the computational time step and print control can be varied throughout the running of a problem. Therefore, excessive printed output is avoided, and there is a considerable savings in actual machine computation time. The program prints a listing of the data input parameters for identification of the problem and ease in identifying any input mistakes. For stacked problems, the program prints only that input information that is changed from the previous problem. The following calculated problem output is printed:

- 1) Time, sec.
- 2) Cold wall convective heating rate without blowing, $\text{Btu/ft}^2\text{-sec.}$
- 3) Radiative heating rate, $\text{Btu/ft}^2\text{-sec.}$
- 4) Velocity, ft/sec.
- 5) Gas ablation rate, $\text{lbm/ft}^2\text{-hr.}$
- 6) Char ablation rate, $\text{lbm/ft}^2\text{-hr.}$
- 7) Total ablation rate, $\text{lbm/ft}^2\text{-hr.}$
- 8) Surface recession depth from original surface, in.
- 9) Hot wall convective heating rate without blowing, $\text{Btu/ft}^2\text{-sec.}$
- 10) Temperature distribution in ablation material, $^{\circ}\text{R.}$
- 11) Temperature distribution in backup structure, $^{\circ}\text{R.}$

The temperatures printed for the ablation material are for fixed distances from the original surface. These distances are calculated from the initial ablation material thickness and number of nodes in ablation material.

V. SUMMARY OF HEAT SHIELD MATERIALS

The design of the reentry vehicle (R/V) is predicated on the basis of preventing excessive thermal and mechanical loads to the payload during flight through a planetary atmosphere. The R/V design is, of course, also strongly influenced by overall system constraints on weight, dimensions, and the desired R/V flight characteristics. However, the single most important factor in R/V design is the mission profile which will determine the heating rates and aerodynamic loadings on the reentering body.

Since the heat shield is an integral part of the R/V, it becomes clear that in the final analysis a detailed heat shield design cannot altogether be treated separately from the remainder of the R/V. However, the fundamental behavior of various heat shield materials will govern their selection on the basis of being best suited for a given R/V design and reference mission. In this section a brief compendium of heat shield material characteristics is presented. This information actually forms the cornerstone for preliminary heat shield evaluations in support of R/V design studies. Since there are four major types of shield materials, viz., the heat sink, subliming ablator, charring ablator, and sublimar-filled ceramic, the following text is divided into a discussion section for each material.

1. Heat Sink or Hot-Structure

A thermal protection system which uses the high heat capacity and low thermal conductivity of a heat shield material to distribute incident heat within itself and retard heat flow to the payload is considered a heat sink type. In addition, rather high surface temperatures are used to reradiate a substantial part of the incident heat away from the body. This type of heat shield is best used where reentry time is considerable as would be for a manned, lifting vehicle. Ceramics are typically used in this application. A cross-sectional view of a ceramic heat shield configuration is shown in Figure 5-1.

- ① Porous Ceramic Module
- ② Low Density Feltmetal
- ③ Honeycomb Sandwich
- ④ Support Reinforcing Pad
- ⑤ Support Leg Attached to Drilled Reinforcing Pad
- ⑥ Lightweight Insulation
- ⑦ Primary Structure, water cooled
- ⑧ Coolant Passages

Figure 5-1. Ceramic Heat Shield and Support Panel Configurations.⁽¹⁾

Basically, the potential advantage of a ceramic heat shield system are as follows:

- a) The ability to maintain smooth and stable vehicle contours during reentry in order to facilitate vehicle control and maneuverability.
- b) Use of materials physically and chemically stable in the flight environment.
- c) Relatively lightweight in certain applications since heat protection is based on radiation rejection of a very large proportion of the incident heat.

Disadvantages which have been associated with this type of system are:

- a) Inability to analyze and accurately predict mechanical and thermal stress performance.
- b) Lack of well-established reliability of brittle materials.
- c) Difficulty in assuring uniform quality during manufacture.

2. Subliming Ablator

This type of heat protection system takes advantage of the heat of sublimation of the ablating material. In addition, heat transfer to the surface of a vehicle during reentry may be significantly reduced by injection of the sublimed material (gas) into the boundary layer. Typical materials used in this application are Teflon, Fluorogreen, Graphite, and Boron Nitride.

The useful areas of application for the plastics are rather limited. They are useful only in low temperature applications because of their low subliming temperature. They cannot be used with bodies experiencing radiative heat input from the shock layer because they are transparent to the radiation and, as a result, direct heating of the heat shield backface occurs.

Although graphite sublimates at a much higher temperature, its usefulness is limited by a comparatively high thermal conductivity. It is also subject to parasitic mass loss by oxidation.

Boron nitride has a sublimation temperature ($\sim 3000^{\circ}\text{K}$) which is nearly as high as that of graphite but its oxidation rate is much lower. Other desirable properties include good strength at high temperatures and excellent machinability. Outgassing of residual water prior to use is necessary to reduce spalling, however. Composites of boron nitride with phenolics are used to lower its thermal conductivity, but the effective heat of ablation (heat transfer rate/mass loss rate) is lowered because boron nitride particles are expelled as the phenolic decomposes⁽²⁾.

Total mass loss rate for several of these materials has been correlated well with heating rate and stagnation pressure⁽³⁾ so that these predictions are easily made.

3. Charring Ablator

This type of thermal protection system makes use of the reradiating advantage of the heat sink type as well as the sublimation and gas injection features of subliming ablaters. The materials used are usually binary composites. One component is a sublimer and the other a material which decomposes with heat to form a char or cinder. Typical materials are phenolic-nylon, silicone-silica, silicon-phenolic, filled silicone in honeycomb, and carbon fiber reinforced phenolic. The thermophysical properties of these materials have been studied extensively⁽⁴⁾ but analysis of the heat and mass transfer mechanisms are extremely complex and difficult as indicated below.

Figure 5-2 shows a schematic model of a charring ablator. Heat is transferred to the char surface by convection and radiation from the shock boundary layer and by oxidation of the char and pyrolysis products at the char surface. Heat is lost by (1) sublimation, (2) conduction to the backface, and (3) reradiation from the char surface. A small amount of heat is lost in the thermal cracking of the pyrolysis gases and the subsequent deposition of carbon in the char layer.

Self absorption of radiant energy in gas reaction products significantly reduces the radiation input to the char. Reradiation tends to decrease and self-absorption increases the convective heating input to the virgin material. The combined effect significantly reduces radiant heating but has little effect on convection heating, as shown in Figure 5-3.

Studies have shown⁽⁷⁾ that for steady state ablation and diffusion rate controlled char oxidation the linear rate of pyrolysis is equal to the char removal rate. The mechanism involved is that of oxygen diffusion through a laminal boundary layer and the resultant competing oxidations of both the char surface and pyrolysis products gas. The reaction rate is determined by the rate at which oxygen can be supplied by diffusion through the boundary layer plus pyrolysis of the plastic. Because of this, the char oxidation rate, the steady-state ablation rate, the boundary layer composition at the char surface and the heat released by chemical reactions are found to be dependent on the composition of the virgin plastic and the pyrolysis gases. In general, the rate of ablation mass loss decreases with either an increasing carbon content in the virgin material or a decreasing oxygen and nitrogen content in the airstream.

Oxidation of the char and pyrolysis gases with turbulent boundary layers is reaction rate limited since the turbulence will carry oxygen in excess of that needed to the char surface. As a result, if there is a transition from laminar to turbulent flow at the char surface, non-uniform surface regression occurs with the higher mass loss rates occurring in the turbulent flow areas. This necessitates the careful analysis of boundary layer flow regimes.

A typical assembly for a lifting energy vehicle is shown in Figure 5-4. Honeycomb support structures are used for ease of manufacture (injection of ablator material into cells) as well as for structural support.

4. Sublimar Filled Ceramic

It is possible to achieve the same advantages of the char-forming ablators without the disadvantage of their high and fairly unpredictable weight losses by use of a porous ceramic

SCHEMATIC OF A PHENOLIC-NYLON CHAR FORMING ABLATOR

Figure 5-3. Comparison of Uncorrected Heat Input with Actual Heat Input.⁽⁵⁾

Figure 5-4. Design Concepts for Char-Forming Ablative Heat Shield.⁽⁹⁾

material which has been filled with a subliming ablator. Some composites which have been studied⁽⁸⁾ are foamed zirconia and foamed silicon carbide filled with phenyl-silane, epoxy resin or phenolic resin. Other types may use an inorganic salt sublimer. Although these materials have the disadvantage of higher thermal conductivities than the char-forming ablators, some of them can be applied by spraying. This method of application is quite useful in applying thin layers and is cheap.

A general survey of heat shield material effective thermal capacity (total cold wall heat load/material unit weight — lb/sq.ft.) is given in Tables 5-1 through 5-3.

In addition, an extensive bibliography on heat shield materials has been prepared and is listed in the Appendix of this report.

TABLE 5-1
SUMMARY OF TEST RESULTS FOR CHARRING COMPOSITE MATERIALS⁽⁸⁾

Material	Composition		Density, ρ , lb/ft ³	Unit weight, w , lb/ft ²	Weight loss, w_a/w	Back-surface-temperature history						Heating rate (average cold wall), $\dot{q}$, Btu/ft ² -sec	Total heat load (cold wall), Q_0 , Btu/ft ²	Effective heat capacity, C_p/w , Btu/lb	
	Component materials	Percent weight				Exposure time		Termination of exposure			Maximum temperature rise				
						t , sec for $\Delta T = 50^\circ F$	t , sec for $\Delta T = 500^\circ F$	t_r , sec	ΔT_r , $^\circ F$	$(\frac{d\Delta T}{dt})_r$, $^\circ F/sec$	t_m , sec	ΔT_m , $^\circ F$			
FR-4 ^a	Phenolic (BPP-5549)	15.8	36	3.03	96.7×10^{-2}	149	254	257	350	14.0	320	660	116.6	30,000	9,900
	Nylon powder (Zytel)	65.4		3.00	95.6	110	262	265	358	16.0	325	598	110.8	29,400	9,800
	Phenolic Microballoons Eccospheres	15.8 5.0													
FR-5 ^a	Phenolic (GTL-9118)	25.0	36	2.99	94.5×10^{-2}	125	258	262	377	26.4	320	675	108.2	28,500	9,470
	Nylon powder (Zytel)	47.0		3.06	92.6	153	263	267	450	33.2	305	747	105.2	28,100	9,180
	Phenolic Microballoons Eccospheres	25.0 5.0													
Chance Vought phenolic nylon	Phenolic Nylon Phenolic Microballoons Quartz		36	3.10	94.8×10^{-2}	170	291	296	375	14.9	339	609	98.8	29,200	9,420
Avcoat 25055	Epoxy Phenolic Microballoons Silica		37	3.00	79.2×10^{-2}	185	249	252	350	14.0	310	676	108.0	27,200	9,070
Barmco 3168	Epoxy Phenolic Silica		63	3.00	68.4×10^{-2}	88	215	220	315	3.8	300	475	114.0	25,100	8,370
				3.02	54.0	104	258	255	290	8.7	327	424	106.5	25,000	8,280
FR-2 ^a	Phenolic (BPP-5549) Nylon powder (Zytel) Phenolic Microballoons	25.0 50.0 25.0	36	3.00	95.1×10^{-2}	140	256	258	340	26.0	308	703	96.4	24,900	8,500
FR ^a	Phenolic (BPP-5549) Phenolic Microballoons	75 25	49	3.00	89.6×10^{-2}	151	227	230	355	15.6	290	700	105.0	23,700	7,900
				5.00	100.0	178	210	215	576	29.8	270	730	110.0	25,400	7,808
Barmco 4505			60	3.00	80.0×10^{-2}	94	194	198	315	3.3	288	558	119.0	25,600	7,870
				2.99	79.0	95	195	196	307	2.6	300	572	108.6	21,500	7,128
Chance Vought phenolic melamine	Phenolic Melamine Phenolic Microballoons Quartz		61	3.07	95.8×10^{-2}	147	258	259	318	14.5	275	515	100.0	25,900	7,790
Barmco 4016	Phenolic Nylon		74	2.98	92.4×10^{-2}	127	209	211	316	10.6	254	480	108.0	22,800	7,690
				2.98	99.5	115	195	199	398	45.0	250	528	108.1	21,500	7,280
Chance Vought phenolic melamine	Phenolic Melamine Phenolic Microballoons Quartz		36	3.06	95.7×10^{-2}	130	258	259	316	16.0	287	575	96.5	23,100	7,590
Emerson Electric Therm-Lag T-500 E-167			86	3.00	67.8×10^{-2}	110	206	208	324	5.7	246	480	108.0	22,500	7,500
				3.62	96.8	127	241	247	351	9.5	305	528	106.5	26,500	7,270
FAB ^a	Phenolic (BPP-5549) Ammonium chloride Phenolic Microballoons	25 50 25	45	3.00	100.0×10^{-2}	158	198	195	365	87.0	---	---	110.0	21,500	7,150
Avcoat 25052	Epoxy Phenolic Microballoons Silica		66	3.04	81.0×10^{-2}	126	195	198	346	14.8	250	680	107.0	21,200	6,970
FR ^a	Phenolic (BPP-5549) Nylon powder (Zytel) Glass fiber	38.5 50.0 17.5	78	2.99	90.9×10^{-2}	114	189	188	346	16.4	221	558	118.5	20,800	6,990
Avcoat 25054	Epoxy Phenolic Microballoons Silica		65	3.10	92.5×10^{-2}	122	187	192	358	9.5	240	595	111.8	21,500	6,960

^aMaterials fabricated at the Langley Research Center. Designations used for convenience in identifying materials.

TABLE 5-1 (Continued)

Material	Composition		Density, ρ , lb/ft ³	Unit weight, W , lb/ft ²	Weight loss W_0/m	Back-surface-temperature history							Heating rate (average cold wall), $\dot{q}$, Btu/ft ² -sec	Total heat load (cold wall), Q_0 , Btu/ft ²	Effective heat capacity, Q_0/m , Btu/lb
	Component materials	Percent weight				Exposure time		Termination of exposure			Maximum temperature rise				
						t , sec for $\Delta T = 500^\circ F$	t , sec for $\Delta T = 500^\circ F$	t_f , sec	ΔT_f , $^\circ F$	$(\frac{dT}{dt})_f$, $^\circ F/sec$	t_m , sec	ΔT_m , $^\circ F$			
Harmco 4503			80	3.00	92.9×10^{-2}	107	182	180	284	6.0	223	453	113.0	80,300	6,770
				3.02	92.6	106	178	182	372	17.3	240	615	107.9	19,700	6,380
FW 1*	Phenolic (BPP-5549) Nylon powder (Zytel)	50.0 50.0	75	2.96	88.5×10^{-2}	116	172	174	342	21.2	250	724	114.0	19,800	6,690
				2.99	87.9	111	167	172	478	39.4	250	752	116.6	20,100	6,720
GE Phenolic nylon glass	Phenolic Nylon Glass		82	3.04	85.6×10^{-2}	110	188	192	334	9.6	231	541	105.0	20,200	6,650
GE Phenolic nylon	Phenolic Nylon		75	3.02	92.5×10^{-2}	130	183	186	358	23.5	210	464	107	19,900	6,590
Harmco 4501			79	3.00	90.8×10^{-2}	82	170	176	346	6.4	215	461	111.0	19,500	6,500
GE 524-A	Epoxy Phenolic Microballoons		42	3.06	98.0×10^{-2}	168	184	186	458	171.0	---	---	106.0	19,700	6,440
GE 525-C	Epoxy Phenolic Microballoons		50	3.00	98.5×10^{-2}	155	176	178	444	89.0	---	---	108.5	19,500	6,430
GE 125-A	Epoxy		76	3.04	98.9×10^{-2}	136	174	176	410	61.9	---	---	109.5	19,300	6,350
GE 124-A	Epoxy		75	2.98	98.6×10^{-2}	142	175	178	1,120	36.2	---	---	106.0	18,900	6,340
PA*	Phenolic (BPP-5549) Ammonium chloride	50 50	85	3.00	91.5×10^{-2}	85	164	165	329	7.2	200	456	111.0	18,500	6,100
Avcoat X5026	Epoxy Phenolic Microballoons Silica		76	3.08	57.6×10^{-2}	97	153	162	390	12.1	230	732	111.4	18,000	5,840
				3.14	56.9	99	156	160	353	14.5	210	620	110.7	17,700	5,640
Harmco 4016			73	3.00	91.2×10^{-2}	115	151	155	359	25.9	185	585	111.0	17,800	5,730
Harmco 4504			85	3.00	96.4×10^{-2}	90	141	145	346	10.3	158	407	117.0	17,000	5,670
Harmco 4506			95	3.00	84.0×10^{-2}	80	147	148	310	6.8	186	458	111.0	16,400	5,470
Harmco 4502			93	3.00	85.6×10^{-2}	69	145	147	309	4.6	204	455	110.5	16,200	5,400
GE Phenolic refrasil	Phenolic Silica fiber		100	3.00	20.2×10^{-2}	57	142	149	334	5.3	240	630	107.0	15,900	5,300
GE 525-C	Epoxy Glass		96	3.02	85.5×10^{-2}	81	136	140	350	13.0	180	602	108.5	15,200	5,050
PAI*	Phenolic (BPP-5549) Ammonium chloride Glass fiber	52.5 50.0 17.5	96	2.98	90.8×10^{-2}	59	129	131	320	10.6	158	412	109.5	14,500	4,800
Harmco 4012	Phenolic Silica		105	3.00	16.5×10^{-2}	42	107	109	300	5.3	205	598	115.0	12,500	4,170
Phenolic* glass	Phenolic (BPP-5549) Glass fiber	70 30	90	3.04	59.7×10^{-2}	66	100	101	315	17.7	---	---	108	10,900	3,580
Harmco 4014			106	3.00	12.9×10^{-2}	39	89	91	308	5.3	184	615	109.0	9,980	3,310
Harmco 525-T4-VA			111	3.00	13.2×10^{-2}	30	72	74	316	8.2	150	665	114.5	8,470	2,880

* Materials fabricated at the Ames Research Center. Designations used for convenience in identifying material.

TABLE 5-2
SUMMARY OF TEST RESULTS FOR CERAMIC MATERIALS⁽⁸⁾

Material	Composition		Density, ρ , lb/ft ³	Unit weight, w , lb/ft ²	Weight loss, w_a/m	Back-surface-temperature history								Heating rate (average cold wall), $\dot{q}$, Btu/rs ² -sec	Total heat load (cold wall), Q_0 , Btu/rs ²	Effective heat capacity, C_p/m , Btu/lb
	Component materials	Exposure time				Termination of exposure			Maximum temperature rise							
		t , sec for $\Delta T = 300^\circ F$				t , sec for $\Delta T = 300^\circ F$	t_T , sec	ΔT_T , $^\circ F$	$(\frac{dT}{dt})_T$, $^\circ F/sec$	t_m , sec	ΔT_m , $^\circ F$					
Martin E-13	ErO ₂ Epoxy		97	3.30	75.7×10^{-2}	99	169	167	314	6.4	240	340	119.9	19,400	5,880	
Martin E-14	ErO ₂ Epoxy		93	3.32	73.9×10^{-2}	98	173	176	380	5.2	235	340	108.6	17,700	5,180	
Martin E-1	ErO ₂ Phenyl-silane		72	3.14	99.1×10^{-2}	68	173	198	305	5.7	230	620	113.5	15,700	5,000	
Martin E-11	ErO ₂ Epoxy		97	3.20	76.0×10^{-2}	92	148	151	342	9.0	222	640	103.8	15,700	4,910	
Martin A-6	SiC Phenyl-silane ErO ₂ coating on SiC		73	3.14	98.0×10^{-2}	49	140	145	306	5.4	245	642	103.0	14,900	4,750	
Martin E-12	ErO ₂ Epoxy		98	3.32	80.1×10^{-2}	88	144	149	353	8.2	225	605	104.3	15,500	4,670	
Martin E-8	ErO ₂ Phenolic		90	3.3	41.3×10^{-2}	76	126	132	340	5.2	202	550	115.0	15,200	4,610	
Avco RAD 50 B	Si Phenolic		80	3.02	6.1×10^{-2}	58	122	127	305	4.8	220	580	108.7	13,800	4,570	
Martin E-7	ErO ₂ Phenolic		93	3.29	44.5×10^{-2}	91	142	144	312	6.3	219	570	104.0	15,000	4,560	
Martin E-10	ErO ₂ Phenolic		99	3.3	49.2×10^{-2}	88	141	141	300	5.5	227	545	106.5	15,000	4,550	
Avco RAD 50 B (extended)	Si Phenolic Voids on back surface		57	2.98	2.7×10^{-2}	28	106	108	315	5.0	190	544	109.0	11,800	3,960	
Martin E-9	ErO ₂ Phenolic		60	3.31	47.3×10^{-2}	72	119	124	333	5.8	233	651	105.5	13,100	3,960	
Martin A-5 (resin back)	SiC Phenolic ($\frac{1}{2}$ filled) ErO ₂ coating on SiC		60	3.02	23.0×10^{-2}	45	95	104	352	5.1	212	640	107.5	11,800	3,710	
Martin E-5	ErO ₂ Phenolic		91	3.29	34.8×10^{-2}	62	112	113	340	7.6	212	676	104.6	12,000	3,650	
Martin A-15	Al ₂ O ₃ Phenyl-silane		63	3.27	38.9×10^{-2}	37	102	102	300	6.7	200	728	116.0	11,800	3,610	
Martin E-15	ErO ₂ Phenolic		105	3.36	91.2×10^{-2}	68	100	102	341	10.8	149	555	105.5	10,800	3,210	
Martin A-8 (resin back)	SiC Phenyl-silane ($\frac{1}{2}$ filled) ErO ₂ coating on SiC		57	2.96	21.4×10^{-2}	30	88	87	308	4.2	165	622	106.0	9,220	3,110	
Martin A-13	Al ₂ O ₃ Phenyl-silane		84	3.25	35.2×10^{-2}	35	92	89	292	5.6	185	647	111.0	9,080	3,040	
Martin A-1	SiC		17	2.75	57.4×10^{-2}	38	80	81	305	8.2	155	520	103.0	8,340	3,030	
Martin A-7	SiC Phenyl-silane ErO ₂ coating on SiC		54	3.27	36.2×10^{-2}	24	89	95	326	4.6	126	655	99.0	9,410	2,880	
Martin A-5 (resin front)	SiC Phenolic ($\frac{1}{2}$ filled) ErO ₂ coating on SiC		61	3.04	24.8×10^{-2}	31	77	80	306	6.4	158	620	105.0	8,400	2,760	
Martin A-12	Al ₂ O ₃		36	2.75	73.2×10^{-2}	33	64	65	320	10.5	131	710	111.0	7,220	2,630	
Martin A-6 (resin front)	SiC Phenyl-silane ($\frac{1}{2}$ filled) ErO ₂ coating on SiC		57	3.25	21.3×10^{-2}	25	71	77	349	6.5	152	667	106.0	8,120	2,510	
Martin A-2	SiC ErO ₂ coating		59	3.04	22.2×10^{-2}	28	61	65	328	7.6	116	613	114.0	7,120	2,360	
Martin A-4	SiC Phenolic ErO ₂ coating on SiC		50	3.16	33.2×10^{-2}	19	62	68	345	7.8	130	790	104.5	7,110	2,250	
Martin A-10	SiC ErO ₂ coating		84	2.80	3.9×10^{-2}	30	49	50	308	7.4	159	546	113.5	5,780	2,060	
Martin A-11	Al ₂ O ₃		34	3.17	78.0×10^{-2}	25	52	54	322	8.5	126	575	115.0	6,220	1,960	
Avcoet I	Si Metallic additives Inconel honeycomb reinforced		146	3.02	0.7×10^{-2}	15	32	36	360	14.8	40	---	104.0	3,740	1,440	

TABLE 5-3
SUMMARY OF TEST RESULTS FOR LOW TEMPERATURE MATERIALS⁽⁸⁾

Material	Density, ρ , lb/ft ³	Unit weight, w , lb/ft ²	Weight loss, w_0/w	Back-surface-temperature history						Heating rate (average cold wall), $\dot{q}$, Btu/ft ² -sec	Total heat load (cold wall), Q_0 , Btu/ft ²	Effective heat capacity Q_0/w , Btu/lb	
				Exposure time		Termination of exposure			Maximum temperature rise				
				t , sec for $\Delta T = 50^\circ F$	t , sec for $\Delta T = 500^\circ F$	t_f , sec	ΔT_f , OF	$(\frac{dT}{dt})_f$, OF/sec	t_m , sec	Δt_m , OF			
Polyethylene	61	3.03	100.0×10^{-2}	66	72	74.0	1,120	371			111.0	8,200	2,700
Nylon	72	2.96	100.0×10^{-2}	50	57	60.0	1,000	248			108.5	6,500	2,200
Avcoat 5019	68	3.06	99.3×10^{-2}	53	54	58.0	1,880	397			110.0	6,580	2,090
Teflon	135	3.00	100.0×10^{-2}	23	43	49.0	-----	---			112.5	5,500	1,830
Teflon (low density)	87	3.00	100.0×10^{-2}	44	46	49.0	1,345	476			110.5	5,400	1,800

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APPENDIX - I

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