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SNAP AEROSPACE SAFETY PROGRAM
PHASE I WIND TUNNEL TESTS

AEC Research and Development Report

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SNAP AEROSPACE SAFETY PROGRAM
PHASE I WIND TUNNEL TESTS

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ABSTRACT

The operation, parameter selection, and thermopaint technique are described for a series of tests conducted in a hotshot wind tunnel. These tests were performed to determine the distribution of aerodynamic heating intensities on the surfaces of the SNAP 10A reactor during orbital reentry. This project was undertaken in support of the Aerospace Safety Program.

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I. SUMMARY

Wind tunnel tests were used to derive comparative laminar, hypersonic boundary layer heat transfer data found on the irregular-shaped envelope of the SNAP 10A reactor. The data were accumulated at various angles of attack and simulated altitudes. The results of the experiments will be used in predicting the disintegration and ablation of SNAP-type reactors during a reentry trajectory.

This report is a description of the Rhodes and Bloxsom facility (where the tests were conducted) as applicable to the determination of hypersonic heat transfer rates through the use of the thermopaint technique. The understanding of this technique was necessary in the current Atomics International wind tunnel test program on present SNAP configurations. Restrictions as to application of the raw data are indicated.

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II. INTRODUCTION

The necessity for predicting the disintegration of SNAP units along their reentry trajectories makes it mandatory to establish the magnitude of the hypersonic heat transfer rates. This task cannot be done by theoretical work alone, since the available hypersonic boundary layer theory is limited mainly to relatively clean aerodynamic shapes. An experimental program is, therefore, required for determining the aerodynamic heating on complex shapes at arbitrary angles of attack as a function of proposed flight conditions encountered during the reentry trajectory. Because of the relatively high cost and the instrumentation problems involved in the remote measurements during an actual reentry, it is highly desirable to initially accumulate data from tunnels that simulate the desired flight conditions. The Aerospace Safety wind tunnel test program was designed to accomplish the following objectives:

- a) To gather data for the heat transfer rates from laminar, hypersonic boundary layers on the complex aerodynamic shapes of the SNAP 10A reactor at various flight conditions
- b) To provide aerodynamic heat transfer input data for calculating the temperature distribution on the reactor components during reentry
- c) To evaluate possible restrictions on the quality of the tunnel data.

The Rhodes and Bloxson facility was selected for the program because of its unique capability for determining aerodynamic heat transfer rates on the surface of complex shapes. Heat-sensitive paint color distribution is the technique used by the facility to provide the required information. This hotshot tunnel produces a simulated reentry environment for periods of 1 to 10 msec. The selection of the tunnel parameters involves (a) the calculation of the electrical energy input, which is used for matching the required enthalpy, (b) the initial mass of air, which controls the duration of the run, and (c) the geometry of the conical nozzle, which expands the flow to the desired simulation of flight conditions. Scale models of the SNAP 10A reactor were coated with the thermopaint for testing at simulated flight environments with relative surface heat transfer rates being indicated by the discoloration of the paint.

The scope of this report is limited to the description, operation, and possible restrictions of the data generated by the hotshot facility.

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III. DESCRIPTION AND OPERATION OF FACILITY

A. GENERAL

Figures 1 and 2 show the facility and models tested and a schematic diagram of the tunnel, respectively. The facility components depicted in Figure 2 are:

- 1) A bank of condensers which provides the necessary electrical energy for simulating reentry conditions for a period of time between 1 to 10 msec
- 2) The bomb or stagnation chamber provided with a Mylar diaphragm at its junction with the nozzle
- 3) A 45° total angle conical nozzle extended between its throat (or minimum diameter at the bomb) and its maximum diameter at the test section
- 4) A cylindrical test section, 60 in. in diameter, where models painted with thermopaint interact with the high-velocity stream of air
- 5) A diffusion pump that gives an initial back pressure lower than the value at the test section, which is required to establish the tunnel flow with a density corresponding to the simulated high altitude.

When the tunnel is put in operation, the test section is evacuated by the diffusion pump to a density below that of the previously calculated test condition. A Mylar diaphragm is inserted between the bomb and the nozzle throat, and the bomb is filled with air to a predetermined pressure. The bank of condensers is discharged by closing a switch that in turn starts a Fastax camera, which is used for visual recording of the flow in the test section. When the camera reaches its preset speed of 3750 frames/sec, the capacitor bank discharges its energy (up to 10^5 joules) into the stagnation chamber. The electrical energy is put into the air in the bomb by means of a tease wire which closes the firing circuit between the electrodes.

The Mylar diaphragm breaks when the heating process, at constant volume, raises the pressure in the bomb to the proper value. Expansion waves then propagate upstream into the bomb, while a shock moves downstream through the nozzle test section and far end of the tunnel.^{1*} Following the shock wave that

*Superscripts refer to the References at the end of this report.

Hypervelocity Wind Tunnel

Full-Scale Model

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Full- and Quarter- Scale Models

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Figure 1. Hypervelocity Wind Tunnel Test Facility and Scale Models

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Figure 2. Schematic Diagram of Wind Tunnel Test Facility

travels downstream, the air from the bomb expands in the nozzle in a manner controlled by the nozzle geometry. When this air reaches the model at the test section, a bow shock appears in front of the model and the hypersonic environment is established around the model. The expansion wave into the bomb drops the value of the stagnation pressure there. This effect, together with the continuous drainage of air from the bomb to sustain the flow in the nozzle, changes the flow conditions at the test section during the run. Finally, the flow in the test section is stopped when the downstream shock wave returns to this section after it has been reflected from the far end of the tunnel.

B. ELECTRICAL ENERGY AND INITIAL BOMB PRESSURE

The bomb is used to heat the air to an enthalpy value fixed by the simulated flight velocity during reentry. The density of the air in the bomb is set by the desired altitude or simulated density at the test section as well as the simulated flight velocity and nozzle throat area as will be explained. The actual magnitude of the density of the air in the bomb is attained by using the proper pressure of the air at ambient temperature.

The electrical energy, E_e , put into the bomb is given by the following expression:

$$E_e = \frac{1}{2} C V_o^2 , \quad \dots (1)$$

where C is the capacitance fixed at a constant value of 4.42×10^{-4} farads, and V_o is the applied voltage from the bank of condensers. The electrical energy is converted into thermal energy of the heated air in the bomb (E_g). The radiation losses to the surroundings of the bomb environment are neglected, and the efficiency of conversion is assumed to be 100%. Thus, one can write, with $\eta = 100\%$,

$$E_g = \eta E_e . \quad \dots (2)$$

The thermodynamic properties of the air in the bomb are obtained from the first law of thermodynamics,

$$E_e = E_g = \Delta \bar{U} + \Delta w , \quad \dots (3)$$

where $\Delta \bar{U}$ is the change in internal energy of the mass of air in the bomb during the heating process, and Δw is the work done on, or by, the air during this process. Since the diaphragm breaks after the heating cycle, it is reasonable to assume constant volume heating (i. e., $\Delta w = 0$). The increase in internal energy per unit mass, ΔU_b , is then given by the volume of the bomb as well as the air pressure and temperature in the bomb before heating,

$$\Delta U_b = \Delta \bar{U} \left[V_b \rho_o \frac{\rho_b}{\rho_o} \right]^{-1} , \quad \dots (4)$$

where V_b is the volume of the bomb, and ρ_b is the density of the air in the bomb, given by the expression

$$\frac{\rho_b}{\rho_o} = \frac{P_b}{P_o} \frac{T_o}{T_a} . \quad \dots (5)$$

The symbols P_b and T_a denote, respectively, the pressure and temperature of the unheated air in the bomb. The subscript zero denotes properties evaluated at a reference standard condition; i.e.,

$$P_o = 1 \text{ atm}$$

$$\rho_o = 1.288 \times 10^{-3} \text{ gm/cm}^3$$

$$T_o = 273.16^\circ\text{K.}$$

Since the enthalpy of the heated air in the bomb, H_b , should have a value set by the desired simulation of the flight velocity, V_∞ , the energy equation for flight conditions yields the following result:²

$$\frac{H_b}{RT_o} = \frac{1}{2} \frac{V_\infty^2}{RT_o} + \frac{H_\infty}{RT_o} \quad , \quad \dots (6)$$

where H_∞ is the ambient static enthalpy that depends on the temperature of the simulated altitude, and RT_o is a constant with the appropriate units (i.e., $RT_o = 33.86 \text{ Btu/lb}$ or $8.475 \times 10^5 \text{ ft}^2/\text{sec}^2$). It should be noted that the term, H_∞/RT_o , in Equation 6 is very small at altitudes of interest. The high temperatures produced in the bomb will cause dissociation and recombination processes that can be assumed to be in chemical equilibrium. Further, since the heating process takes place at constant volume, the density, ρ_b/ρ_o , remains constant. Thus, Equations 5 and 6, with a Mollier diagram for equilibrium air,² yield the temperature (T_b), pressure (P_{bH}), compressibility (Z_b), and degree of chemical dissociation of the heated air (α_b) in the bomb before the rupture of the diaphragm. These quantities are obtained as a function of the pressure of the unheated air and the simulated flight velocity. The voltage necessary to produce the stagnation enthalpy, H_b , is obtained as follows: by definition,³

$$H = U + \frac{P}{\rho} \quad . \quad \dots (7)$$

Since, in general,

$$P = (1 + \alpha)\rho RT = Z\rho RT \quad , \quad \dots (8)$$

then

$$\frac{\Delta H_b}{RT_o} = \frac{\Delta U_b}{RT_o} + Z_b \frac{T_b}{T_o} - Z_a \frac{T_a}{T_o} , \quad \dots (9)$$

where the compressibility factor at ambient conditions (Z_a) is unity. Since

$$\frac{H_b}{RT_o} = \frac{H_a}{RT_o} + \frac{\Delta H_b}{RT_o} , \quad \dots (10)$$

there follows that

$$\frac{\Delta U_b}{RT_o} = \frac{H_b}{RT_o} - Z_b \frac{T_b}{T_o} - \frac{1}{\gamma - 1} \frac{T_a}{T_o} , \quad \dots (11)$$

where the following relationships have been used in Equation 11:

$$H_a = C_p T_a$$

$$R = C_p - C_v$$

$$\gamma = C_p / C_v ,$$

where γ is the ratio of specific heats for the unheated air (i. e., $\gamma = 1.4$). Using Equations 1 to 6 and 11, one gets the following result for the required voltage:

$$\frac{1}{2} \frac{CE^2}{RT_o \rho_o V_b} = \frac{P_b T_o}{P_o T_a} \left[\frac{1}{2} \frac{V_\infty^2}{RT_o} - Z_b \frac{T_b}{T_o} - \frac{1}{\gamma - 1} \frac{T_a}{T_o} + \frac{\gamma}{\gamma - 1} \frac{T_\infty}{T_o} \right] . \quad \dots (12)$$

Equation 12 is an expression for the energy per unit mass at standard conditions. It shows that the voltage is a function of the simulated flight velocity, the pressure of the unheated air in the bomb, and the size of the bomb.

C. FLOW IN THE NOZZLE

When the diaphragm breaks as a consequence of the pressure rise in the bomb, the pressure ratio across the conical nozzle sets up a flow that is unsteady because of the time variation of both the pressure in the bomb and also the back pressure downstream from the test section. The density and velocity at the test section can be calculated by using the assumptions of one-dimensional,

quasi-steady, isentropic flow. These calculations require the use of the equations for the chemical kinetics of high-temperature air.⁴ Further, the determination of the isentropic flow in the core of the nozzle requires knowledge of the boundary layer displacement thickness at the walls of the nozzle, an effect that becomes important at the low densities of interest. The condition of isentropic flow requires the absence of any shock waves in the flow along the supersonic and hypersonic section of the nozzle.⁵ At the tunnel facility, the density and velocity at the test section are calculated as described in the following paragraph.

From the continuity equation, as obtained from the principle of conservation of mass in the nozzle, one obtains the following result:³

$$\frac{A_T}{A^*} = \frac{1}{M_T} \left[\left(\frac{2}{\gamma_b + 1} \right) \left(1 + \frac{\gamma_b - 1}{2} M_T^2 \right) \right]^{\frac{\gamma_b + 1}{2(\gamma_b - 1)}}, \quad \dots (13)$$

where A denotes the physical area of the conical nozzle, and A^* the nozzle throat area. It should be noted that in Equation 13, the area ratio denotes actual rather than geometrical flow areas, which are identical only when the boundary layer is neglected. For a conical nozzle with a 45° included angle, the area ratio, A/A^* , is given by

$$\frac{A_T}{A^*} = \left(1 + \frac{2X_T}{d^*} \tan 22.5^\circ \right)^2. \quad \dots (14)$$

The symbol, X , is the distance along the longitudinal axis of the nozzle, A^* is the throat diameter, and the subscript T refers to the test section. The right-hand side of Equation 13 involves two parameters: (1) the ratio of specific heats, γ_b , and (2) the Mach number, M . The former one is estimated from the assumption that the chemistry of the air in the bomb freezes as soon as the expanding gas enters the nozzle; i. e., $\gamma_f = \gamma_b$. The ratio of specific heats in the bomb can be calculated from the fraction of dissociation in the gas, i. e.,

$$\gamma_b = \frac{7 + 3\alpha_b}{5 + \alpha_b} = \frac{4 + 3Z_b}{4 + Z_b} , \quad \dots(15)$$

as is derived in Reference 6. Equation 15 is based on the assumption that the vibration degrees of freedom are in the frozen state. Therefore, the ratio of specific heats, γ_f , for the flow in the nozzle is fixed arbitrarily by the initial bomb conditions before the rupture of the diaphragm. The Mach number is obtained from its definition as follows:

$$M = \frac{V}{a} , \quad \dots(16)$$

where V denotes the air velocity in the test section of the nozzle, and a is the speed of sound. For the foregoing assumption of frozen chemistry at the bomb conditions, the speed of sound is given by:

$$a_T = 1089 \left[\frac{Z_b \gamma_b T_T}{1.4 T_o} \right]^{\frac{1}{2}} \text{ ft/sec} , \quad \dots(17)$$

where T is the static temperature in the test section, which can be obtained from the energy equation for one-dimensional flow as follows:

$$\frac{T_T}{T_b} = \left[1 + \frac{\gamma_b - 1}{2} M_T^2 \right]^{-1} . \quad \dots(18)$$

Finally, the isentropic relationship, $P\rho^{-\gamma}$, can be put in the following form:

$$\frac{\rho_T}{\rho_b} = \left(1 + \frac{\gamma_b - 1}{2} M_T^2 \right)^{-\left(\frac{1}{\gamma_b - 1} \right)} . \quad \dots(19)$$

The foregoing system of equations can be used to determine the density, velocity, Mach number, and temperature distributions along the longitudinal axis of the

conical nozzle. Flow conditions at time $t > t_0$, where t_0 is the initial time of the run, are not considered.

D. SELECTION OF WIND TUNNEL PARAMETERS

There are five basic tunnel variables that can be controlled in order to obtain a simulated range of flight velocity and altitude conditions. These variables are: (1) voltage, (2) bomb size, (3) pressure of the unheated air in the bomb, (4) nozzle throat area, and (5) nozzle area at the test section. The first three fix all the stagnation conditions in the bomb for a given maximum value of the simulated flight velocity; while the latter two, with the pressure of the unheated air, fix the minimum value of the simulated altitude. For example, for a maximum value of the simulated flight velocity of 25,500 ft/sec, Equation 6, plotted in Figure 3, fixes the dimensionless stagnation enthalpy, H/RT_0 , at 385. For a given value of the pressure of the unheated air in the bomb, Equation 5, plotted in Figure 4, sets a corresponding value for the density in the bomb, which remains constant during the heating process at constant volume. For the foregoing stagnation enthalpy and initial bomb pressure, the use of the Mollier diagram for equilibrium air² yields the ratio of specific heats (see Figure 5), the temperature (see Figure 6), and pressure (see Figure 7) of the heated air in the bomb. The foregoing stagnation enthalpy and initial bomb pressure, together with the bomb size, determine the required voltage, as illustrated in Equation 12 and plotted in Figure 8.

For a desirable minimum value of the simulated altitude during the run, i. e., 250,000 ft, Figure 9 yields the required density at the test section. Since the initial bomb pressure fixes the density of the air in the bomb, the altitude and initial bomb pressure fixes the value for the density ratio, ρ_T/ρ_b . Hence, Equation 19, plotted in Figure 10, fixes the flow Mach number. Equation 13, plotted in Figure 11, yields, then, the required area ratio. Equation 14, plotted in Figure 12, gives a set of values for both the throat diameter and the location of the test section along the longitudinal axis of the nozzle. Thus, for a given flight velocity (or stagnation enthalpy in the bomb) and altitude (density in the test section), each value of the assumed initial bomb pressure sets a corresponding value for the nozzle area ratio. Further, for a given location along the axis of the conical nozzle, i. e., a test section near, or at the cylindrical section of the tunnel, there is one value of throat area for the assumed value of the bomb initial pressure and flight conditions.

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Figure 3. Flight Velocity vs Stagnation Enthalpy

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Figure 4. Initial Bomb Pressure vs Bomb Density

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Figure 5. Ratio of Specific Heats, γ , for
Equilibrium Air at Bomb Conditions
 $H/RT_0 = 385$

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Figure 6. Initial Bomb Pressure vs Initial
Bomb Temperature

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Figure 7. Initial Bomb Pressure vs Initial Bomb Pressure of Heated Air

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Figure 8. Initial Bomb Pressure vs Voltage for the 1623 cm³ Bomb and the 980 cm³ Bomb

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Figure 9. Ambient Density vs Altitude

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Figure 10. Ratio of Density at Test Section
to Initial Bomb Density

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Figure 11. Ratio of Area at Test Section to Throat Area vs Mach Number at Test Section

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Figure 12. Geometric Area Ratio Distributions Along the Longitudinal Axis of the 60-in. Nozzle for Various Nozzle Diameters

E. PAINT TECHNIQUE

The heat transfer experiments were performed using the tunnel facility's proprietary thermopaint technique. The model surface is sprayed with a thin film of this paint. When a heat pulse is applied to the model, its surface changes color, which is related to heat transfer rates through the use of a relative calibration technique. This method employs the use of a calibration sphere coated with the thermopaint and run simultaneously with the model. The stagnation point on the hemisphere shows a characteristic color for each set of tunnel conditions. Since the heat transfer rates on the sphere can be estimated by using laminar, equilibrium boundary layer theory, the stagnation color on the sphere can then be related by calculation to heat transfer rates. With this colormeter, one can read colors on the surface of the reactor and, hence, the heat transfer that corresponds to the respective color.

The colors on the models and calibration sphere, which appear after the completion of the tunnel run, are read by using the proper wave length of incident light on the surface of the model. It is well known that the thickness of a thin layer, which is transparent to light, is proportional to the wave length of the incident light and inversely proportional to the index of refraction, μ . The expression for the thickness, $\bar{X}$, is as follows:⁷

$$\bar{X} = \frac{n\lambda}{4\mu} \quad , \quad \dots (20)$$

where n is an integer number with values such as 3, 5, and 7.

The thermochemical mechanisms in the paint are not understood theoretically. It has been found experimentally, that a thin layer of paint with opaque optical properties becomes transparent after it has been heated. Rhodes and Bloxsom state that calibrations between color and heat transfer rates indicate that colors are always proportional to the estimated heat transfer rates, and thus establish the conversion technique of color to heat rate.

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IV. HEAT TRANSFER RATES

The usefulness of the paint technique rests on the validity of an experimental result reported by the tunnel facility, which relates the heat transfer rate to the color of the paint on the surface of the model. This relationship has the following form:

$$q = KC_f \quad , \quad \dots (21)$$

where K is a constant for each experiment, q is the local heat transfer rate, and C_f is the corresponding color number obtained at the end of the run. This equation should obviously hold above a threshold value for the heating rate, q , since it should otherwise break down for very low heat transfer rates when there is no change in the color of the paint. It should be noted that the constant, K , does not have an absolute value. Hence, Equation 21 can yield, at best, only relative measurements of the heat transfer rates. The absolute level in the data must be estimated theoretically by using the theory for either thin laminar boundary layers, or thick shock layers at the stagnation point of the calibration sphere. The tunnel facility calculation of the constant, K , is described in the following paragraphs.

From the empirical Equation 21, the constant, K , can be evaluated as follows:

$$q = \frac{C_f}{C_{f_s}^*} q_s^* \quad , \quad \dots (22)$$

where $C_{f_s}^*$ and q_s^* denote, respectively, the color number and heat transfer rate at the stagnation point of a reference sphere. The tunnel experiment yields the relative quantity $C_f/C_{f_s}^*$ in Equation 22, but the absolute value of q is introduced by using a q_s^* determined from theory for the desired flight rather than tunnel conditions. Thus, q_s^* is calculated from the theory for thin laminar boundary layers in chemical equilibrium at the stagnation point of a sphere without dissociation ahead of the bow shock. This can be represented as a function of flight conditions by the following expression:⁸

$$q_s^* = \frac{865}{(R_n)^{1/2}} \left(\frac{\rho_\infty}{\rho_o} \right)^{1/2} \left(\frac{V_\infty}{10^4} \right)^{3.15} \text{ Btu/ft}^2\text{-sec} , \quad \dots(23)$$

where R is the radius of the sphere in feet; ρ_∞/ρ_o is the density at the altitude of interest as made dimensionless by the sea level density, ρ_o ; and V_∞ is the flight velocity in ft/sec. Since the calibration sphere has a radius of 0.74 in., Equation 23 yields the following result:

$$q_s^* = \frac{865}{\left(\frac{0.74}{12} \right)^{1/2}} \left(\frac{\rho_\infty}{\rho_o} \right)^{1/2} \left(\frac{V_\infty}{10^4} \right)^{3.15} \quad \dots(24)$$

Introducing Equation 24 into Equation 22 for the tunnel, one gets the final result for the interpretation of the tunnel facility color data as follows:

$$q = \frac{C_f}{C_{f_s}^*} \left[\frac{865}{\left(\frac{0.74}{12} \right)^{1/2}} \left(\frac{\rho_\infty}{\rho_o} \right)^{1/2} \left(\frac{V_\infty}{10^4} \right)^{3.15} \right] \text{ Btu/ft}^2\text{-sec} , \quad \dots(25)$$

where $C_{f_s}^*$ is the color number at the stagnation point of the calibration sphere, and C_f is the final color number measured at any point on the reactor surface. The ratio of the quantity in the bracket of Equation 25 to $C_{f_s}^*$ represents the heat transfer rate value assigned to a color number equal to unity on the stagnation point of the calibration sphere, and matches exactly the value reported for a color number of unity.

A partial but direct check of Equation 21 can be obtained by applying it to the calibration sphere, for which case, Equation 22 yields the result:

$$\frac{q^*}{q_s^*} = \frac{C_f^*}{C_{f_s}^*} , \quad \dots(26)$$

where the asterisk (*) denotes conditions for the sphere. The laminar boundary layer in this case remains attached to the surface of the sphere. Therefore,

the use of thin laminar boundary layer theory for equilibrium air yields the following results:⁹

$$\frac{q^*}{q_s^*} = \frac{(\gamma \rho_w \mu_w u_e)^* (1 + 0.096\beta)^*}{1.068 \left\{ \left[2 \int_0^X \rho_w \mu_w u_e r^2 dX \right] \left[2 \rho_{ws} \mu_{ws} \left(\frac{du_e}{dX} \right)_s \right] \right\}^{1/2}}, \quad \dots(27)$$

where r is a distance perpendicular to the radius of the sphere that is aligned with the free-stream flow direction, V_∞ . The symbols ρ and μ denote, respectively, density and viscosity; u_e is the air velocity at the external edge of the boundary layer; X is a dimension along the surface of the sphere measured from the stagnation point; $(du_e/dX)_s$ is the velocity gradient at the stagnation point; and β is a pressure gradient parameter. Since the air next to the body surface is undissociated for the case of equilibrium chemistry, Equation 27 may be rewritten as follows:

$$\frac{q^*}{q_s^*} = \frac{\left(r \frac{\mu_w}{RT_w} \frac{P}{P_s} u_e \right)^* (1 + 0.096 \sqrt{B^*})}{1.068 \left\{ \left[2 \int_0^X \frac{\mu_w}{RT_w} \frac{P}{P_s} u_e r^2 dX \right] \left[2 \frac{\mu_{ws}}{RT_{ws}} \frac{1}{R_n} \left(\frac{2P_s}{\rho_s} \right)^{1/2} \right] \right\}^{1/2}}, \quad \dots(28)$$

where, from the energy equation as applied along the stream line around the edge of the boundary,

$$u_e = \left[(2)(0.8475 \times 10^6) \frac{H_s}{RT_o} \left(1 - \frac{H_e}{H_s} \right) \right]^{1/2}, \quad \dots(29)$$

where H_e denotes the static enthalpy. For equilibrium air, the ratio, H_e/H_s , can be expressed by:¹⁰

$$\frac{H_e}{H_s} = \left\{ 1 - \frac{1.43(P_s/P_o)^{0.035}}{(H_s/RT_o/250)^{0.3877}} \left[1 - \left(\frac{P}{P_s} \right)^{0.035} \right] \right\}^{1/0.3877} \quad \dots(30)^\dagger$$

[†]The development of this expression requires use of the equation, $dh = \rho - 1 dp$, along the streamline next to the body.

Using the results from Reference 11, the pressure behind a normal shock can be expressed by:

$$\frac{P_s}{P_o} = \frac{\left[\frac{518.69}{T_\infty (^{\circ}R)} \right] \left(\frac{V_\infty^2}{0.8475 \times 10^6} \right) \frac{P_\infty}{P_o}}{(1.1258 - 1.365 \times 10^{-6}) V_\infty - 5.3 \times 10^{-8} H} \quad \dots(31)$$

while the density behind a shock with arbitrary curvature is given by:¹⁰

$$\frac{\rho_s}{\rho_o} = \frac{3.09 \times 10^{-2} \left(\frac{P_s}{P_o} \right)^{0.965}}{\left(1.0477 \frac{H_s}{250 R T_o} \right)^{0.6123} - 0.0477} \quad \dots(32)$$

The subscript, ∞ , in Equation 31 denotes free-stream conditions, while h is the altitude in feet, and V_∞ is the flight velocity in ft/sec. Because of the weak dependence of H_e/H_s on P_s/P_o , as indicated in Equation 30, it is seen that the velocity, u_e , in Equation 28 is mainly a function of the pressure ratio P/P_s along the surface of the sphere. Also, since q^*/q_s^* varies inversely as the one-fourth power of (P_s/ρ_s) , as shown in Equation 28, and P_s/ρ_s itself varies approximately as the 0.6 power of the stagnation enthalpy, as seen from Equation 32, q^*/q_s^* varies only as the 0.15 power of the stagnation enthalpy. Therefore, the parameter q^*/q_s^* in Equation 28 depends mainly on the body geometry, r , the wall temperature, T_w , and pressure distribution, P/P_s . Further, since for hypersonic flow the pressure is related to the geometry, r , the parameter q^*/q_s^* is fixed only by the body geometry, r , and the wall temperature, T_w . Thus, for the case of a sphere with arbitrary radius and wall temperature equal to standard ambient conditions, Equation 28 gives results that can be represented empirically by the following expression:⁹

$$\frac{q^*}{q_s^*} = (\cos \theta)^{1.1} \quad \dots(33)$$

where θ is an angle, less than 90° , generated by the rotation of the radius away from the stagnation point. Equation 33 indicates that the heat transfer distribution

on the forward hemisphere of any sphere is always the same regardless of the flight velocity and altitude or the flow conditions in the tunnel.

Applying Equation 33 to the calibration sphere and using Equation 26, yields:

$$\frac{C_f^*}{C_{fs}^*} = (\cos \theta)^{1.1}, \quad \dots(34)$$

which also holds regardless of the simulated altitude and velocity in the tunnel. Thus, the distribution of color numbers along the surface of the sphere is unique, regardless of the simulated flow conditions in the tunnel, including the shape of the heating pulse and the duration of the run. Since Equation 34 is based on Equation 21, the latter equation can be verified by measuring the colors along the calibration sphere (colormeter) and fitting them into Equation 34.[†]

It is worthwhile to note that the foregoing results concerning the universality of the heat transfer or color distribution on blunt bodies of revolution is quite general, and applies to any such body, provided that the thin, equilibrium laminar boundary layer remains attached to its surface. Therefore, any other body of revolution besides the sphere can be used to establish the relative color distribution. A generalization of the results presented in Equation 33 are indeed available for other geometries, and they are given in Figure 1 of Reference 8. It should be emphasized, however, that this universal result cannot be theoretically extended to the complicated flow geometry of the SNAP 10A reactor (see Figure 1). The reason is that the basic Equation 27 for q/q_s is restricted only to hypersonic, laminar boundary layers that have similar velocity and enthalpy profiles while they remain attached to the body surface. Since the boundary layer on a given reactor component separates from the body surface as the flow enters the base flow region and wake of that component, it is not probable that the boundary layer on other reactor components submerged in these wakes will remain always similar. The complex flow field that exists around the SNAP 10A reactor prevents the tracing of some boundary layer histories, which is necessary for defining the

[†]Another check of Equation 21 consisted of obtaining color numbers, C_{fs}^* , for spheres of different radii, R_n , for the same tunnel conditions at high densities. Equation 21 predicts a linear relationship between C_{fs}^* and $R_n^{-1/2}$ which has been verified in the tunnel.

parameter q/q_s . While a modified q/q_s^* may be defined by using the stagnation point of another boundary layer on the calibration sphere, a possible universality of this latter parameter would rest on empirical grounds. Such a universality condition requires knowledge of the appropriate tunnel flow regime, because empirical results cannot be extrapolated reliably outside their range. Should q/q_s^* be found to be independent of tunnel or flight conditions, a theoretical implication from such universal condition would be as follows (see Figure 13):[†] (a) the boundary layer on each blunt component should have similar profiles as discussed previously, and (b) the heat transfer rates at the stagnation points of every component must be at a fixed ratio to the heat transfer rate of the stagnation point of the calibration sphere.

Since the color variation along the surface of the reactor model is independent of either the size or shape of the calibrating geometry, the selection of the latter is restricted only to the requirements that it should give two conditions: (a) proper coverage of color spectrum in the reactor surface, and (b) a thin boundary layer at the stagnation point for the required altitude of interest. The measurement of the color number at the stagnation point of the calibration sphere yields C_{fs}^* . The color number scale along the surface of the ball is readily obtained from Equation 34. The final colors on the reactor are then given numbers by matching them to the respective color on the sphere. This procedure yields the local color number, C_f , required for determining the local heat transfer rate from Equation 25.

It should be pointed out that q_s^* must be calculated from the general Fay and Riddell equation for the assumed conditions of equilibrium flow behind the bow shock of the calibration sphere rather than from Equation 24. The Fay-Riddell equation has the following form:¹⁴

$$q_s = \frac{0.76}{\sigma^{0.6}} \left\{ (\rho\mu)_{ws}^{0.1} (\rho\mu)_s^{0.4} \left[1 + (L^{0.5} - 1) \frac{H_{Ds}}{H_s} \right] (H_s - h_w) \left(\frac{du_e}{dX_s} \right)^{1/2} \right\}, \dots (35)$$

[†] The ratios in Figure 13 are based on the following three assumptions: (a) similar laminar boundary layers with equilibrium chemistry, (b) the dimensionless hypersonic flow field nearly independent of flight velocity and altitude (i.e., at $L = 3d$, $P_{oon}/P_{sn} = 10^{-2} = \text{constant}$),^{12,13} and (c) negligible velocity defect in wake immediately upstream of normal shocks (i.e., isentropic expansion to P_{oon} and constant H_g). Note that the successive shocks produce entropy increases that lower the stagnation pressure and density and, therefore, the heat transfer rates.

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Figure 13. Parameters for Spheres Submerged in Successive Hypersonic Wakes Along a Typical Reentry Trajectory

where σ is the Prandtl number, L is the Lewis number, H_{D_s} is the enthalpy of dissociation, and the other terms are as defined in Equation 27. For the assumption of equilibrium flow behind the shock, the Mollier diagram for equilibrium air must be used together with the pertinent flow conditions upstream of the shock or test section of the tunnel (these calculations will be treated in more detail in a subsequent document). Their result indicates a shift or correction of the magnitude of the density at the test section for the higher densities. At low densities in the test section of the tunnel, the nonequilibrium¹⁵ and vorticity¹⁶ effects must be incorporated in the calculation of q_s^* . It should also be mentioned that proper simulation requires matching¹⁷ of two flow parameters (enthalpy and pressure) at the stagnation point of the calibration sphere for both the desired flight and actual tunnel conditions. This requirement leads to a definition of at least a mean value for q_s^* in the tunnel, i.e.,

$$q_s^* = \Delta t^{-1} \int_0^t q_s^* dt ,$$

where Δt denotes the time for significant color change of the calibration sphere.

It should be emphasized that Equation 35 is a general expression and, for the heat transfer rate at the stagnation point, holds even for the case where the air in the free stream is dissociated. Equation 24, on the other hand, is a special case of Equation 35, and holds only when the latter equation is applied to atmospheric conditions with zero free-stream dissociation. To illustrate this fact, Equation 24 can be written in the form:

$$q_s^* = \frac{865}{(R_n)^{1/2}} \left(\frac{\rho_\infty}{\rho_o} \right)^{1/2} \left(\frac{V_\infty}{10^4} \right)^{1.15} \left(\frac{V}{10^4} \right)^2 , \quad \dots(36)$$

or with the aid of Equation 6:

$$q_s^* = \frac{(865)(2 \times 8.475 \times 10^{-3})}{(R_n)^{1/2}} \left(\frac{\rho_\infty}{\rho_o} \right)^{1/2} \frac{H_s}{RT_o} \left(\frac{V_\infty}{10^4} \right)^{1.15} . \quad \dots(37)$$

Thus, Equation 37 indicates that in the case when the density at the test section of the tunnel and the stagnation enthalpy are simulated correctly, errors can result when the velocity in the test section fails to match the flight value. These errors are due to the energy that is tied up in the dissociation of the air at the test section. (The significance of these errors cannot be established without a thorough investigation of the phenomena. Such an investigation is being undertaken.) The stagnation enthalpy of the air is expressed by:

$$\frac{H_b}{RT_o} = \frac{1}{2} \frac{v^2}{RT_o} + \frac{H_D}{RT_o} + \frac{9}{2} \frac{T}{T_o}, \quad \dots(38)$$

where

$$\frac{H_D}{RT_o} = \frac{0.2 \alpha_b (5 \text{ ev}) + 0.8 \alpha_b (9 \text{ ev})}{2.354 \times 10^{-2} \text{ ev/molecule}}. \quad \dots(39)$$

The static enthalpy $(9/2)(T/T_o)$ includes the seven degrees of freedom for translation, rotation, and vibration. Equation 39 takes into account the dissociation of oxygen (5 ev) and nitrogen (9 ev). Table 1 compares the simulated conditions and the actual flight conditions. The ratio of kinetic energy to RT_o and the ratio of energy of dissociation to RT_o are presented to allow a dimensionless comparison of the conditions. These values are calculated using the quoted value of $\alpha = 60\%$ at the test section with a corresponding velocity of 18,200 ft/sec.

TABLE 1
DISSOCIATION IN FREE STREAM FOR SIMULATED AND
ACTUAL FLIGHT CONDITIONS

Equation No.	Equation	Simulated Conditions	Actual Flight Conditions
	$\frac{1}{2} \frac{v^2}{RT_o}$	$\frac{1/2 \times 1.82 \times 10^8}{8.475 \times 10^5} = 191$	385
(39)	$\frac{H_D}{RT_o}$	$\frac{8.2\alpha}{2.354 \times 10^{-2}} = 209$	-0-
(38)	$\frac{9}{2} \frac{T}{T_o}$	$\frac{9}{2} \times \frac{10}{273} = 0.16$	4.5
(Total Enthalpy)	H_b/RT_o	400.16	389.5

Table 1 indicates that the energy tied up in dissociation in the freestream section of the tunnel is larger than the kinetic energy. This should be contrasted with the conditions that will exist during actual atmospheric reentry, where $H_D/RT_0 = 0$, and $1/2 V/RT_0 = 385$ (refer to Equation 6), and for which case Equation 24, of course, does hold. The magnitude of the variance between the simulated and actual flight conditions of the individual terms of Equation 38 is not uncommon in hotshot tunnels. A significant aspect of the simulation is the comparison of the total enthalpy, which shows good agreement with that of actual flight conditions.

Finally, from the physical conditions of the experiment, it is expected that the final color, C_f , should depend on the total heat transferred to the paint during the time of the run, i. e.,

$$Q = \int_{t_i}^{t_f} q \, dt,$$

rather than on the instantaneous heat transfer rate, q , as expressed by Equation 21. This is so because C_f should be given by the following integral:

$$C_f = \int_{t_i}^{t_f} C \, dt \propto \int_{t_i}^{t_f} \bar{X} \, dt, \quad \dots(40)$$

where C is the instantaneous color during the time of the experiment, which should be proportional to the instantaneous depth, $\bar{X}$, of the transparent film of paint. This latter parameter is expected to be a function of the temperature in the film of paint, which depends on the total heat transferred to the body. Although the level of the color C_f depends on the time history of the instantaneous heat transfer rate, as given by Equation 40, the distribution of the final color number along the surface of the calibration sphere is independent of the shape of the heating pulse. This can be seen by using Equation 33, which yields the following result:

$$\frac{Q^*}{Q_s^*} = \frac{\int q \, dt}{\int q_s \, dt} = (\cos \theta)^{1.1}, \quad \dots(41)$$

or with Equations 33 and 34:

$$\frac{Q_s^*}{Q_s^*} = \frac{q_s^*}{q_s^*} = \frac{C_f^*}{C_{fs}^*} = (\cos \theta)^{1.1}, \quad \dots(42)$$

which indicates that Equation 21 can be written for the calibration sphere in the following forms:

$$q_s^* = KC_f^* \quad \dots(43)$$

or

$$Q_s^* = K' C_f^*, \quad \dots(44)$$

where $K = q_s^*/C_{fs}^*$ and $K' = Q_s^*/C_{fs}^* = K \int q_s^* dt / q_s^*$. Since C_{fs}^* depends on the time of the run, K is also a function of time. Further, the time dependence of K must be such as to yield a product KC_{fs}^* that is independent of time for the case of a constant heat transfer rate, q_s^* . The fact that K is a function of time makes the use of the calibration sphere a permanent requirement for each run.

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V. TEST REQUIREMENTS AND PRELIMINARY DATA

Atomics International submitted the test schedule in Table 2 to the operators of the tunnel facility. The purpose of this test plan was to obtain detailed heat transfer data for the irregular surface shapes of the SNAP 10A configuration. Full- and quarter-scale models were tested in the facility's hotshot tunnel. The first three experiments were conducted to investigate whether or not altitude scaling could be applied in the tunnel. Subsequent simulated altitudes were selected to facilitate data collection with respect to optimum tunnel operation.

TABLE 2
WIND TUNNEL TEST SCHEDULE

Run Number	Configuration	Scale	Angle of Attack	Tunnel Altitude (ft)	Velocity (ft/sec)
1	A	Full	0°	250,000	25,500
2	A	Full	0°	275,000	25,500
3	A	Full	0°	200,000	25,500
4	A	Full	90° A [†]	250,000	25,500
5A	Mod A	1/4	0°	250,000	25,500
5B	Mod A	1/4	30° A	250,000	25,500
5C	Mod A	1/4	30° B [§]	250,000	25,500
6A	Mod A	1/4	90° A	250,000	25,500
6B	Mod A	1/4	90° B	250,000	25,500
6C	Mod A	1/4	60° A	250,000	25,500

[†]Thermal switches into flowstream (see Figure 14, A Orientation)

[§]Vertical NaK pipes into flowstream (see Figure 14, B Orientation)

The A configuration consisted of the reactor with pump and fins but without the support legs, reflectors, and wiring harness (see Figure 2, full-scale model). The Modified A configuration (Mod A in Table 2) included the support legs down to the top manifold on the converter and the neutron shield (see Figure 2, quarter-scale model).

A ORIENTATION

B ORIENTATION

α ANGLE OF ATTACK

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Figure 14. Angle-of-Attack Orientation

Runs No. 1 through 4 were performed using full-scale models of the SNAP 10A reactor. Their purpose was to investigate briefly the effect of altitude on the laminar heat transfer rates for zero angle of attack, as well as the effect of the angle of attack itself. Runs No. 5 and 6 were utilized to investigate further the effect of angle of attack on the parameter C_f/C_{fs}^* . In order to maximize the data obtained, three 1/4-scale models were used in each of the last two experiments.

As discussed previously, the proper combination of five variables is needed to set the tunnel to match the desired simulated flight conditions. These variables are the bomb size (V_b), the voltage (V_o), the initial gage pressure for the bomb (P_b), the nozzle throat area (A^*), and the area of the test section (A). The variables A^* , A , and V_b , are sequentially adjusted before the bomb is ignited; therefore, the initial gage pressure, P_b , and voltage, V_o , are the final parameter settings before the run and the only ones recorded during the test.

All the calculations for the foregoing parameter selection are performed as a function of the pressure gage parameter since this parameter is recorded. In Figure 8, the ignition voltage is given as a function of the gage pressure. For a given gage pressure, it is also possible to find the bomb density, the bomb stagnation temperature, T_b , and the bomb stagnation pressure, P_b . The parameter selection is an iterative procedure. A final answer to the iteration procedure is obtained when a value is found for both the ratio of the initial bomb density to density for the desired altitude at the test section and a compatible location in the test section. The following sample calculation will illustrate the method of selecting the tunnel parameters.

- a) Consult Figure 3 with the reentry velocity of 25,500 ft/sec. The resulting H/RT of 385 is one of the entering parameters for the Mollier Diagram.
- b) A gage pressure is assumed. For this example, let this pressure be equal to 3.3 psig. From Figure 4, this pressure yields a corresponding bomb density of $1.4 \times 10^{-3} \text{ gm/cm}^3$, or a dimensionless value of $\rho_b/\rho_o = 1.09$. This is the second parameter for entering into the Mollier Diagram. It should be mentioned that the same ambient corrections to the bomb; i. e., the barometric pressure is always 14.3 lb/in.², and the cold bomb temperature is always 300°K.

TABLE 3

RUN PARAMETERS USED IN PHASE I WIND TUNNEL TESTS

Parameter	1	2	3	4	5	6
1. Simulated altitude (kft)	250	275	200	258	250	250
2. Simulated velocity (ft/sec)	25,500	25,500	25,500	25,500	25,500	25,500
3. Stagnation enthalpy ($\text{ft}^2/\text{sec}^2 \times 10^8$)	3.25	3.25	3.25	3.25	3.25	3.25
4. Voltage (kv)	16.1	16.1	19.0	16.1	16.1	16.1
5. Bomb pressure (psig)	3.3	3.3	25.1	3.3	3.3	3.3
6. Bomb density (gm/cm^3)	1.40×10^{-3}	1.40×10^{-3}	3.15×10^{-3}	1.40×10^{-3}	1.40×10^{-3}	1.40×10^{-3}
7. Area ratio test section	9.2×10^3	2.7×10^4	2.3×10^3	1.2×10^4	9.2×10^3	9.2×10^3
8. Model distance from throat (in.)	53.5	53.5	53.5	58.5	53.5	53.5
9. Throat diameter (in.)	0.450	0.260	0.905	0.450	0.450	0.450
10. Bomb volume (cm^3)	1623	1623	980	1623	1623	1623
11. Frozen bomb (γ_b)	1.57	1.57	1.57	1.57	1.57	1.57
12. Density ratio at test section (ρ_T/ρ_o)	3.6×10^{-5}	10^{-5}	2.5×10^{-4}	3.6×10^{-5}	3.6×10^{-5}	3.6×10^{-5}
13. Mach number at test section	36.0	50.0	25.0	37.5	36.0	36.0
14. Temperature at test section ($^{\circ}\text{K}$)	27	12.9	70	27	27	27
15. Flow velocity at test section (ft/sec)	18,700	18,700	19,200	18,700	18,700	18,700
16. Stagnation color number on calibration sphere	1.8	1.5	2.0	1.8	1.9	1.6

- c) Consult Figure 5 with the gage pressure of 3.3 psig; read a γ_b of 1.57. This parameter is necessary in Equations 13 and 19.
- d) Using the gage pressure of 3.3 psig, Figure 6 gives the initial bomb temperature of 9190°K, and Figure 7 gives a pressure of the heated gas, prior to diaphragm rupture, of 62 atm.
- e) Consult Figure 8 to find the required voltage across the capacitors to match the flight enthalpy. The 1623 cm³ bomb is chosen in order to give a maximum run duration. The required voltage is then 16.5 kv.

(The next steps in the iterative process is to check the tunnel geometry to see if the model location is compatible with existing nozzles and available areas in the tunnel test section. Because of the size of the full-scale model, the 60-in. nozzle was used throughout the experiment.)

- f) Consult Figure 9 with the altitude of 250,000 ft, which gives a $\rho_T/\rho_o = 3.5 \times 10^{-5}$, and a ρ_T of 4.5×10^{-8} gm/cm³.
- g) Form the ratios of ρ_T/ρ_b . For Run No. 1, it is found to be 3.20×10^{-5} . Consult Figure 10 and read a Mach number of 36.0.
- h) Consult Figure 11 with the Mach number of 36.0. An area ratio of 9.2×10^3 is then required.
- i) Consult Figure 12 to establish a necessary nozzle throat diameter and a position along the longitudinal axis of the nozzle. Note at this point that the throats of 0.260 in., or 0.450 in. will be adequate.[†] The 0.260-in. nozzle is discarded because the model will not fit into the test section at the 28-in. station. Therefore, the 0.45-in. nozzle is used with the model at the 53.5-in. station. At this point, the parameter selection is complete. Using the 1623 cm³ bomb, a nozzle throat area of 0.450 in. with the model located at the 53.5-in. station, a bomb initial pressure of 3.3 psig and a voltage of 16.5 kv will simulate in the tunnel test section an initial altitude of 250,000 ft and an initial velocity of 25,500 ft/sec. See Table 3 for the run parameters that were used in the six heat transfer experiments of Phase I of the wind tunnel test program.

[†]A value obtained from the final iteration has been used for the gage pressure in this sample calculation. Otherwise, new values of gage pressure are used until the throat areas fall in the available range of values.

PHASE I, EXPERIMENT NO. 3
 0° ANGLE OF ATTACK
 200,000 ft ALTITUDE
 25,500 ft/sec VELOCITY

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Figure 15. Top Half of Reactor Vessel

The foregoing sample calculation and the results of similar calculations appearing in Table 3 have been obtained at Atomics International in a manner independent of that done at the tunnel facility, and the agreement was found to be within 5%.

There are estimated curves of the decay in the pressure, temperature, and enthalpy of the gas in the bomb during the time of the run. The drop in pressure controls the change in the density of the air in the test section.

The heat transfer data from the tunnel were presented to Atomics International in the form of color number contours for several components of the reactor. These were drawn on a surface development of the geometry of each component. These drawings were too large and could not be reduced to size accurately for presentation in this report. Certain sections of the drawings have been reduced to size, however, and these are shown in Figures 15 and 16. Figure 15 is a half view of the reactor vessel top and shows the color number "C" on the lip and top dimples. Figure 16 includes the leading edge of a fin and pump magnet. The quantity in the bracket of Equation 24 divided by the reference color are also given in the drawings for each test.

The tunnel facility's tunnel test procedure incorporates the use of the parameters that were calculated previously and the correction, if any, needed to compensate for the location of the calibration sphere. This sphere is sometimes located downstream from the reactor in order to avoid flow interferences on the model. Therefore, correction must be applied to the color number in order to compensate for this difference in location. Before the tunnel is ignited, the heat rate on the calibration sphere is obtained from a family of curves that are plotted from Equation 24. For the 0.74-in. radius of the calibration sphere, the heat rate on the stagnation surface of the sphere is $363 \text{ Btu/ft}^2\text{-sec}$ for a velocity of 25,500 ft/sec, at 250,000-ft altitude. After the run, a corresponding color number is read, $C_{fs}^* = 1.8$. The tunnel facility makes the correction for this density gradient as described in the following paragraph.

For Runs No. 1 through 4, the reactor is located at the 50-in. station and the calibration sphere at the 65-in. station; the operators of the tunnel facility then write the following statements:

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Figure 16. Top of Fin and Pump Magnet

$$\begin{aligned} (q_s^*)_{50} &= K_1 (\rho_{50} \mu_{50} V_{50})^{\frac{1}{2}} H \\ (q_s^*)_{65} &= K_1 (\rho_{65} \mu_{65} V_{65})^{\frac{1}{2}} H , \end{aligned}$$

where H denotes the stagnation enthalpy, while ρ , μ , and V are, respectively, the density, viscosity, and flow velocity at the edge of the boundary layer. Therefore,

$$\frac{(q_s^*)_{50}}{(q_s^*)_{65}} = \left[\frac{\rho_{50} V_{50} \mu_{50}}{\rho_{65} V_{65} \mu_{65}} \right]^{\frac{1}{2}}$$

From the conservation of mass in the nozzle,

$$\rho_{50} V_{50} A_{50} = \rho_{65} V_{65} A_{65}$$

or

$$\frac{\rho_{50} V_{50}}{\rho_{65} V_{65}} = \frac{A_{65}}{A_{50}} ,$$

where A denotes the flow cross sectional area; i.e., $A = 1/4 \pi D^2$. In the short distance in the tunnel, the tunnel facility assumes that

$$\mu_{50} = \mu_{65}$$

Therefore,

$$\frac{(q_s^*)_{50}}{(q_s^*)_{65}} = \frac{D_{65}}{D_{50}} = 1.14$$

from the tunnel geometry.

Hence, Equation 21 yields:

$$\left(C_{fs}^*\right)_{50} = \left(C_{fs}^*\right)_{65} \frac{\left(q_s^*\right)_{50}}{\left(q_s^*\right)_{65}} .$$

Therefore,

$$C_{fs}^* 50 = (1.8) (1.14) = 2.05$$

Inserting the foregoing results in Equation 25, the local heat rates are given by:

$$q = (363/2.05)C_f = 176 C_f \text{ Btu/ft}^2\text{-sec} ,$$

where C_f is the local color number for the reactor.

A more consistent approach for relating the two sets of color numbers for calibration spheres placed at two different stations is obtained by using Equation 34 as follows:

$$\frac{\left(C_f^*\right)_{50}}{\left(C_{fs}^*\right)_{50}} = \frac{\left(C_f^*\right)_{65}}{\left(C_{fs}^*\right)_{65}} = (\cos \theta)^{1.1} \quad \dots (45)$$

or

$$\left(C_f^*\right)_{50} = \left(C_f^*\right)_{65} \frac{\left(C_{fs}^*\right)_{50}}{\left(C_{fs}^*\right)_{65}} , \quad \dots (46)$$

where, from Equation 21:

$$\frac{\left(C_{fs}^*\right)_{50}}{\left(C_{fs}^*\right)_{65}} = \frac{\left(q_s^*\right)_{50}}{\left(q_s^*\right)_{65}} . \quad \dots (47)$$

If Equation 24 is used at $x = 65$, as it was at $x = 50$, one gets from Equation 47:

$$\left(C_{fs}^*\right)_{50} = \left(C_{fs}^*\right)_{65} \left[\frac{(\rho_{\infty}/\rho_o)_{50}}{(\rho_{\infty}/\rho_o)_{65}} \right]^{\frac{1}{2}}, \quad \dots (48)$$

where the quantity in the bracket of Equation 48 is obtained from Figure 9. It should be mentioned that the velocity at the section $x = 50$ and $x = 65$ are the same, a result that is obtained from the following expressions:

$$\frac{V_{50}}{V_{65}} = \frac{M_{50}}{M_{65}} \left[\frac{T_{50}}{T_{65}} \right]^{\frac{1}{2}} = \frac{M_{50}}{M_{65}} \left[\frac{T_{50}/T_b}{T_{65}/T_b} \right]^{\frac{1}{2}} \quad \dots (49)$$

which are obtained from Equations 16 and 17. Using Equation 18 in this result, one gets:

$$\frac{V_{50}}{V_{65}} = \frac{M_{50}}{M_{65}} \left[\frac{1 + \left(\frac{\gamma_b - 1}{2}\right) M_{65}^2}{1 + \left(\frac{\gamma_b - 1}{2}\right) M_{50}^2} \right]^{\frac{1}{2}} \quad \dots (50)$$

Substitution of $M_{50} = 34.5$ and $M_{65} = 40.5$ in Equation 50 yields $V_{50} = V_{65}$. The value of $(C_{fs}^*)_{50}$, as given by Equation 48, is at least a consistent correction for the color number at the stagnation point of the calibration sphere, which is required in Equation 25. However, it should be emphasized again that the right-hand side of Equation 47 must be calculated properly by using Equation 35 rather than Equation 24.

A potential advantage to this thermopaint technique is that the experimental data, within a limited range, are applicable to different altitudes. Using Equations 21 and 24, it can be shown that the predicted color numbers for the altitude of interest are related to the color numbers from the original experiment in the following manner:

$$C_{f_{\text{predicted}}} = C_{f_{\text{experimental}}} \left[\frac{\left(\frac{\rho_{\infty}}{\rho_o} \right)_{\text{predicted}}}{\left(\frac{\rho_{\infty}}{\rho_o} \right)_{\text{experimental}}} \right]^{\frac{1}{2}} \quad \dots (51)$$

This result is based on the assumption that the ratio, q/q_s^* , is constant for a given geometry. This correlation in experimental data showed up in Runs No. 1 and 2, in which the predicted and the measured values agreed.

The data from all the heat transfer experiments were recorded on 35-mm film. These photographs give accurate reproduction of the colors on the stagnation surfaces.

As previously mentioned in Section I, the foregoing discussions have been restricted to a description of the testing facility. A subsequent document is planned that will present a more detailed analysis of the wind tunnel characteristics and will develop corrections required for improved interpretation of the raw data. These corrections are expected to include the following effects:

- a) Nonequilibrium chemistry of the flow in the tunnel
- b) Boundary layer displacement thickness in the tunnel wall
- c) Decay of flow in the nozzle
- d) Shock wave phenomena in the core of the nozzle
- e) Dissociation in the free stream
- f) Vorticity in the shock layer of the calibration sphere
- g) Nonequilibrium chemistry of the inviscid flow at the edge of the boundary layer
- h) Density gradients on a calibration sphere.

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