Introductory Aerothermodynamics of Advanced Space Transportation Systems

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The aerothermodynamic problems associated with hypervelocity flight at high altitudes (low ambient density) are considered. The physical phenomena are described and their severity in terms of aerodynamic heating for advanced space transportation systems are estimated. Flight domains corresponding to various physical phenomena are mapped. Approximate methods of analysis for conceptual studies are noted and preliminary results and estimates are presented. Problem areas that need both theoretical and experimental research are highlighted. The development of advanced computational codes that need to be prepared for use when approximate conceptual studies have defined advantageous configurations is discussed. Two advanced space transportation systems are considered: the aeroassisted orbital transfer vehicle concept and the small, rapid-response, manned, maneuverable concept that takes off from Earth or a conventional aircraft, enters a near-Earth orbit, and finally re-enters the atmosphere with lift and cross-range capability to land on an airstrip.

Nomenclature

g = gravitational constant
GO = geosynchronous orbit
k = catalycity coefficient
p = pressure
q = heating rate
r = distance from the center of the Earth
R = body nose radius, "equivalent body radius," surface reflectivity
T = temperature
U = velocity relative to the Earth's atmosphere
V = inertial velocity or speed
Γ = ratio of radiative flux to freestream energy flux
Δ = shock standoff distance
ϵ = surface emissivity
λ = mean free path
ρ = gas density
σ = Stephan-Boltzmann constant

Subscripts

b = body
c = convective
e = at the surface of the Earth
eq = chemical equilibrium condition
go = geosynchronous orbit
o = circular orbit, reference value at Earth's surface
r = radiative
s = conditions behind the shock
w = conditions at the wall or surface
ν = spectral frequency
∞ = ambient freestream conditions
1 = body 1
2 = body2

Introduction

THE aerothermodynamic problems of hypervelocity flight in the Earth's atmosphere appear to be at the threshold of a major change of direction from those of past Earth atmosphere entry. In the past, entry has been characterized by a rather abrupt passage through the upper atmosphere, such that the entry vehicle decelerates in the lower relatively dense regions of the Earth's atmosphere. However, new concepts of space vehicles are being considered that will perform entry maneuvers mostly in the very-low-density region of the upper atmosphere. This is a region that has been considered as a secondary feature of the entry problem and as such has received relatively little attention from supporting research and technology, the major problem in the past having been the severe heating pulse deeper in the atmosphere. Aside from Space Shuttle, previous vehicles have had primary ablative heat shields. However, new concepts will have heat shields that do not ablate and cannot withstand severe temperatures. Thus, the high-altitude (say 80 km) flight regime that was a fascinating curiosity has become the primary problem. Hypervelocity flight through this low-density upper atmosphere is where we will encounter the significant heat pulse; therefore, we must come to understand the physical and chemical gasdynamics of high-temperature, low-density flow over both drag and maneuvering configurations.

Two generic concepts have been introduced wherein the significant aerodynamic and aerothermodynamic problems occur in very-low-density environments at high altitude. One of these concepts is the aeroassisted orbital transfer vehicle (A0TV).[1-4] Such a vehicle extends the utility of the Shuttle to higher altitudes. It carries a payload to or from an orbit higher than the Shuttle and returns to be retrieved by the Shuttle. Austin et al.[5] have shown that great economy of payload is achieved if the returning vehicle decelerates during a pass through the upper atmosphere before rendezvous with the Shuttle, eliminating the need to carry the fuel necessary for the deceleration.

A second concept is that of a spacecraft which maneuvers from one low Earth orbit to another. A concept with closely related aerothermodynamic problems is a sortie vehicle that takes off from the ground, goes into a near-Earth orbit for a short period of time, and returns to Earth to land horizontally. That vehicle would have a short turnaround time and would have to be very durable.

In order to enhance durability, various thermal protection schemes have been examined, including the hot-structures concept by Kelly et. al.[6] These metallic structures are limited to temperatures of about 1033 K (1400°F). Thus, the allowable heating rates are low and the sortie vehicle would

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also decelerate in the low-density regions of very high altitudes. A hybrid thermal protection system has been examined by Wurster[7] that incorporates carbon-carbon, reusable surface insulation and a standoff metallic surface over a microquartz felt insulation system to enhance durability. But the maximum use temperature of the insulation is 1811 K (2800°) and the metal temperatures are 1922, 1589, 1256, and 1144 K (3000, 2400, 1800, and 1600°F) for tantalum, coated columbium, a cobalt-based superalloy, and a nickel-based superalloy, respectively. The hybrid system appears to have weight advantages, but nonablating metallic skin temperatures are a major limitation. We should note that convection was the only heating mechanism examined which may be sufficient if the entry velocity is limited.

Thus, at the outset of conceptual studies of such vehicles, we must re-examine what we think we know about the aerothermodynamics of low-density, hypervelocity flight in order to define the technological problems that must be addressed and solved. We will do this with the simplest means available, but will learn more than we might expect at the outset. During conceptual studies, this is the proper way to proceed—learn what problems must be solved with a minimum expenditure of effort, manpower, and cost. Various concepts should be examined as simply as possible. Once a promising concept has been defined, the detailed design procedures and costly robust flowfield and material codes can be used to accurately predict the environment and the required thermal protection. The present paper will demonstrate, in a preliminary manner, some of the technological problems, how they might be addressed, and their severity and will address some of the limitations of these new concepts.

Aeroassisted Orbital Transfer Vehicle (AOTV) Aerothermodynamics

The aerothermodynamic flight domain of an orbital transfer vehicle is shown in Fig. 1. The coordinates are altitude and speed because these correspond to both ambient thermodynamic variables and important energy-related variables including the freestream energy flux. For an orbiting object, the circular orbital velocity is

$$V_0 = \sqrt{\frac{r_e^2}{r} g_e} \tag{1}$$

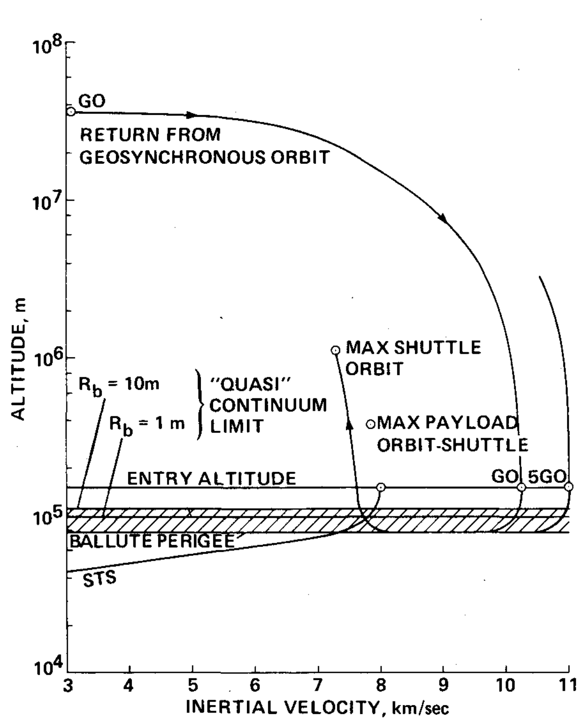

Fig. 1 Flight domain.

At the top of Fig. 1, the symbol that represents an object in geosynchronous orbit is shown at about 35,000 km altitude and at an orbital velocity of about 3 km/s. The line to the right of the symbol is illustrative of the path of an orbital transfer vehicle whose tangential velocity has been diminished so that it is falling to Earth at a near/vertical speed of about

$$V = r_e \sqrt{2g_e\left(\frac{1}{r} - \frac{1}{r_0}\right)} \tag{2}$$

This AOTV encounters the "top" of the Earth's atmosphere at 150 km at an inertial speed of 10.2 km/s at a shallow angle relative to the atmosphere. The vehicle decelerates at high altitude by aerodynamic drag and exits the atmosphere at about 7.5 km/s to rendezvous with the Shuttle at some altitude between the two symbols shown in Fig. 1. The perigee of the AOTV is just below 80 km altitude, which happens to correspond to that of a "ballute" concept (an inflatable drag device).[8] At the far right of Fig. 1 is the entry of an AOTV from an orbit that is five times geosynchronous radius at a velocity of almost 11 km/s.

A trajectory for the STS (Shuttle or Space Transportation System) is shown for comparison. Two features are significant. The AOTV decelerates at a much higher altitude than the STS and the passage of the AOTV through the upper atmosphere is very energetic. Thus, the AOTV resides in a very energetic rarefied environment if it returns from a geosynchronous orbit or beyond. We will examine some physical features of that environment; first, however, we should consider how to treat the rarefied regime.

At one time, the upper atmosphere was characterized by free molecular flow, a transitional region, and a continuum regime[9,10] defined by the Knudsen number. More recently, Liepmann and Roshko[11] have clarified the concept of rarefied flow, while Adams and Probestein[12] have simplified the treatment of rarefied flow by the use of the rigid-sphere molecule concept of kinetic theory. It was shown that the Knudsen number λ_∞/R_b is comparable to λ_s/Δ. It was also shown that a regime exists for a blunt body where continuum theory applies, although a slip condition may occur at the wall for both velocity and temperature (moreover, a slip condition probably occurs behind the bow shock wave). But for a cold wall, slip does not affect either skin friction or heat transfer significantly. The limit of applicability of continuum theory is

$$\lambda_\infty/R_b \approx \lambda_s/\Delta \ll 1 \tag{3}$$

(We will employ corresponding approximate expressions to estimate convective heating). Thus, the upper boundary of the shaded area is termed the "quasi"-continuum limit for a blunt body with an equivalent radius of 10 m (which corresponds roughly to a ballute), while the line below that corresponds to a 1 m radius.

Radiative Transfer

Let us now ask what heating mechanisms may be important and, because of the energetic nature of the AOTV flight regime, first question gaseous radiative transfer. The heavy line labeled $R_b = 10$ m on the left side of Fig. 2 is a line of Γ of 0.5% for a blunt body having a 10 m nose radius. The symbol Γ is the ratio of the stagnation region radiative heating rate to the freestream energy flux, where molecular radiation (N_2) is dominant (to a first order it also corresponds to a radiation/convection ratio, after Goulard[13-15]). Initially, the line curves toward the right with increasing altitude because density diminishes and as a result the velocity must increase to maintain constant Γ. However, a further increase in altitude produces a knee in the curve where the density is sufficiently low so that the gas behind the bow shock begins to relax chemically. Thus, high "nonequilibrium" temperatures exist behind the shock and undissociated molecules that radiate by vibrational transitions in the infrared reside in that high-temperature bath. The tendency is to enhance radiation from this chemically relaxing mixture, so the line must turn back to lower velocities to maintain Γ constant. The computation of equilibrium radiation was from Kivel[16] and others and was correlated in a useful form by Howe.[17,18] The radiation from chemically relaxing air is derived from the work of Camm et al.[19]

The significance of the heavy line described in the above paragraph is that it is the threshold where radiation becomes important.[14] To the right of the line, Γ exceeds 0.5%. If Γ is 1%, radiation affects the flowfield to the extent that radiation predictions at the wall are altered by 10% because radiative emission cools the flowfield. Thus, the coupling between the radiative transfer and the flowfield needs to be accounted for to the right of the heavy line.

At very high altitudes, there are insufficient molecular collisions to excite radiative mechanisms. The phenomenon is not well defined and is simply noted as collision limited on the figure. In Fig. 2, a second heavy line is shown for a nose radius of 1 m ($R_b = 1$ m). The discussion above is applicable. On the far right of Fig. 2, air behind the bow shock is not only dissociated, but it is ionized. The third line on the lower right represents the ionization threshold[20] and corresponds to an electron mole concentration of about 1%. The line is not well defined in the upper atmosphere where finite rate chemical relaxation of air occurs behind a bow shock wave. It needs better definition. Preliminary estimates of ionization rates at the elevated nonequilibrium temperature associated with energetic flight in the low-density regime indicate that the line curves toward the left as shown.

The importance of the ionization region to the right of the threshold is that additional radiative transfer mechanisms become important. In addition to radiation from molecules mentioned previously, radiation from the interaction of free electrons with atoms and molecules takes place and actually becomes dominant. These are free-free and free-bound transitions that give rise to continuum radiation in the visible and ultraviolet part of the spectrum. Park[21] discusses these phenomena in chemically relaxing air corresponding to the low-density regime of interest.

Thus far, we have determined that the AOTV is subject to both radiative and convective heating. Let us now estimate the magnitude of the heating rates. On the right side of Fig. 2, lines of constant freestream energy flux ($\frac{1}{2}\rho_\infty U_\infty^3$) in watts per square centimeter are drawn in the upper atmosphere. Note that the freestream energy flux increases very rapidly (exponentially) as a vehicle dips deeper into the atmosphere. This is an index of the necessary thermal protection. In general, to avoid excessive thermal protection systems, the vehicle should decelerate as high in the atmosphere as possible. For the AOTV flight path, the maximum value is of the order of 500 W/cm^2. The heating rate is a small fraction of that, which we will now estimate.

Radiative heating is composed of the component from that portion which is in chemical equilibrium plus that portion is a function of velocity, density, and body size. It can be estimated from the correlation of Ref. 17 or 20 and is found to be small in the low-density regime of interest. However, the nonequilibrium contribution has been considered to be a function of velocity only because of "binary scaling" and may be quite significant.

The concept of binary scaling was postulated by Camm et al.[19] It was argued that two body collisions determine the excitation rate of radiating species behind the shock, which leads to an excitation distance behind the shock that varies inversely with density. The chemical relaxation distance behind the shock also varies inversely with density. Thus, the length of the nonequilibrium region varies inversely with density. However, the peak radiation emitted from this region varies directly with density. The product of the peak emission and the size of the nonequilibrium region is a measure of the total radiation emitted from the nonequilibrium region and is thus independent of density (according to this argument). Aside from boundary-layer effects, if the shock standoff distance is greater than the nonequilibrium region, the total nonequilibrium radiation is also independent of the shock standoff distance (or body size) as well. Thus, the radiation from the chemically relaxing part of the flowfield is thought

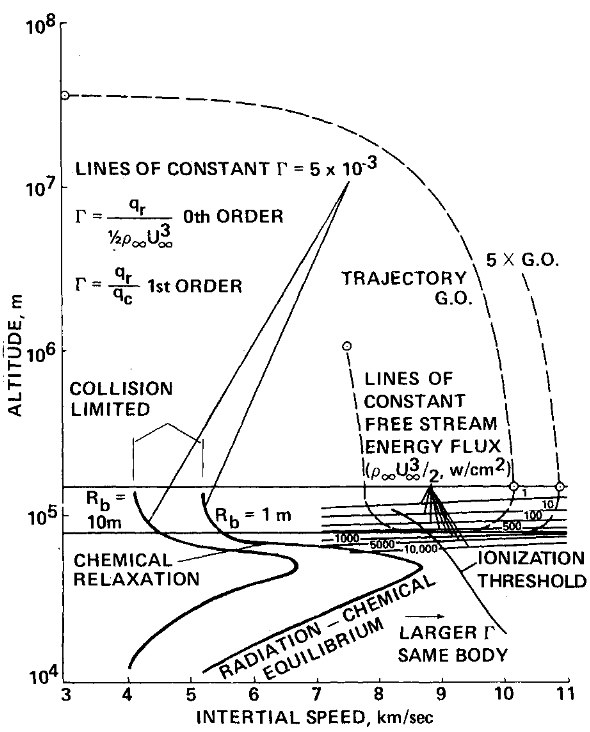

Fig. 2 Stagnation radiative heating domain.

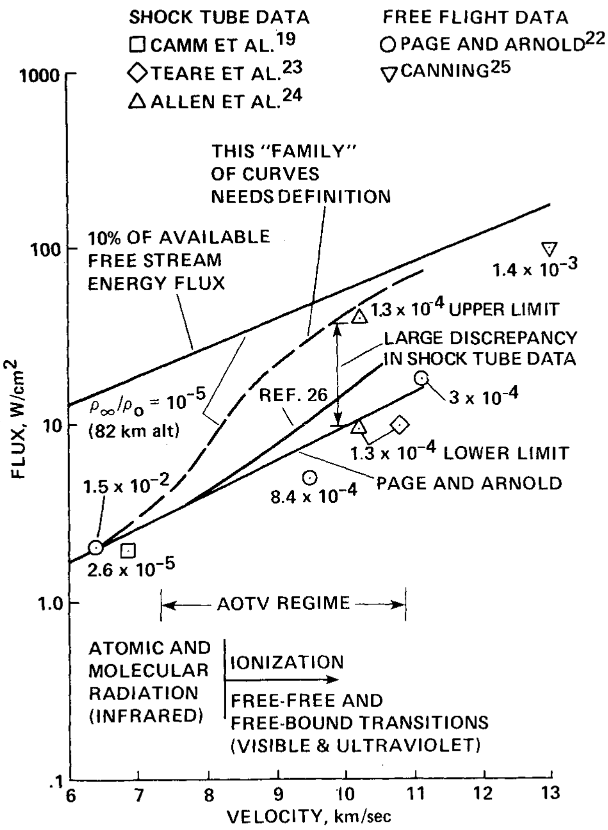

Fig. 3 Radiation from chemically relaxing air.

to be independent of density (or altitude) and body size, depending only on velocity.

Free-flight experiments by Page and Arnold[22] appeared to support the binary scaling concept. Figure 3 has been adapted from Ref. 22. The straight line representing the circular symbols shows nonequilibrium radiation as a function of velocity alone. Note that for velocities of 8-11 km/s, nonequilibrium radiation corresponding to that line is 3-16 W/cm^2. Results are also shown for shock-tube data[19,23,24] and a high-speed free-flight result by Canning.[25] There is considerable spread in the data. The data of Allen indicate that nonequilibrium radiation could be 10-40 W/cm^2 at a velocity of 10 km/s. The implications for AOTVs designed for very low convective heating rates are conceptually and quantitatively of major significance.

Although it is not shown in Fig. 3, the Project Fire flight data[26] indicated that binary scaling does not apply for nonequilibrium radiation. Reference 26 correlated it with pressure up to a stagnation pressure of 0.24 atm (ground-based data scatter widely about the Fire correlation). As a convenience, however, Ref. 26 applied binary scaling that was a mean of ground facility data of Refs. 25 and 27 (not shown) to compare with Apollo 4 radiometer data (which cut off the ultraviolet radiation because of a quartz window). The resulting curve from Ref. 26 has been multiplied by 2π sr and is shown in Fig. 3. That curve is thought to represent ground-based data within approximately a factor of two, although there are questions of spectral mismatching that need to be resolved more closely. Although nonequilibrium radiation for Apollo was a small part of the radiative heating, for the present applications where most of the flight is in the low-density regime that is not the case. Clearly, new analytical techniques, including viscious interactions, slip, and improved thermodynamics and rate processes, should be devised and used to interpret existing radiation data more carefully.

Moreover, the concept of binary scaling was a useful notion in that it simplified out thinking about a very complicated set of phenomena. As such, it can be used for purposes of estimates in conceptual studies. But it is not a physical principle. At high altitudes where we approach collision limiting, nonequilibrium radiation must diminish as ambient density diminishes, so that nonequilibrium radiation varies with both velocity and density. So, at sufficiently low densities, we have a family of such lines corresponding to ambient density levels.

Moreover, at higher flight velocities where ionization phenomena provide additional radiative mechanisms, the line for a given density should trend sharply upward. In Fig. 3 we have sketched a dashed line that we estimate to be a reasonable line for an ambient density of $\rho_\infty/\rho_o = 10^{-5}$ (corresponding to 82 km altitude). Note that our estimate approaches 10% of the freestream energy flux at very high velocity for that density. Nonequilibrium radiation is estimated to vary 9-65 W/cm^2 at velocities of 8-11 km/s. Such curves need to be defined by very sophisticated experiment and analysis. Menees[28] and Park[21] are beginning such work. Our ability to compute radiation from chemically relaxing high-temperature air has improved greatly since Ref. 22 was published.

The dashed curve for $\rho_\infty/\rho_o = 10^{-5}$ in Fig. 3 was estimated by assuming that the value of Page and Arnold was approximately correct at 6.7 km/s and that the binary scaling notion is useful for our purposes at that low velocity. The nonequilibrium flux is fairly small, 2 W/cm^2. At high velocity we assumed that the curve approached 10% of the freestream energy flux, which for the equilibrium radiation computations for air appear to be an approximate limit (see Fig. 4). At the ionization threshold in Fig. 3 (which is actually not a sharp line), we simply drew the curve about midway between the Page-Arnold line and our 10% freestream energy flux line.

With respect to a radiative flux limitation of about 10% of the freestream energy flux, this was our experience with equilibrium radiation as shown by the shaded area in Fig. 4, which was adapted from Ref. 29. The threshold for significant equilibrium radiation appears to be about 10 km/s for $0.1 \leq p_s/p_o \leq 1$ (or $0.013 \leq \rho_\infty/\rho_o \leq 0.134$ approximately). In principle, it can be shown that the radiation from chemically relaxing air at a given velocity (and lower density) can be much larger than if eqiulibrium prevailed. Briefly, the reason is that the nonequilibrium radiative emission is proportional to the ratio of the nonequilibrium to the equilibrium temperature behind the shock raised to a power estimated to be of the order 10, depending on the emitting species. Nevertheless, for nonequilibrium radiation we limit the flux estimate (dashed line) to 10% of the freestream energy flux for present purposes so as not to exceed our equilibrium experience (this may not be conservative). Accordingly, the dashed line has been sketched in Fig. 4 to illustrate the present estimated fraction of the freestream energy flux to which the vehicle may be exposed for chemical nonequilibrium air for a density ratio $\rho_\infty/\rho_o = 10^{-5}$. This again introduces a family of curves that needs to be defined. Note that the fraction of freestream energy flux which is nonequilibrium radiation must decrease at lower altitudes (higher density) at a given velocity, as more available energy is associated with equilibrium radiation and convective heating.

Convective Heating

In order to estimate convective heating rates for hypersonic flight in the low-density environment under consideration, we return to the "quasi"-continuum concepts presented previously. There the rigid sphere molecule of kinetic theory leads to $\lambda\rho$ being constant. Thus, one can estimate the stagnation convective heating rate for a blunt object as[30]

$$\frac{-q_c}{\frac{1}{2}\rho_\infty U_\infty^3} \approx \sqrt{\frac{\lambda_s}{\Delta}} \approx \sqrt{\frac{\lambda_\infty}{R_b}} \tag{4}$$

Figure 5 shows the heating rate regime through which an object of 10 m nose radius (such as a ballute) would pass. The figure shows a (subjective) quasicontinuum boundary $\lambda_\infty/R_b \leq 0.1$, a continuum boundary $\lambda_\infty/R_b \leq 0.01$, and lines of constant convective heating according to the estimate of Eq. (4). The ballute trajectory and configuration taken from Andrews and Bloetscher[8] are shown. Most of the deceleration and heating occur in the fully continuum regime.

Our estimate of peak heating of 15 W/cm^2 is higher than that of Ref. 8 because of external vorticity effects intrinsic in Eq. (4) for low-density flight and also because our R_b is 10 m, where the ballute is 12.2 m. The convective heating estimated

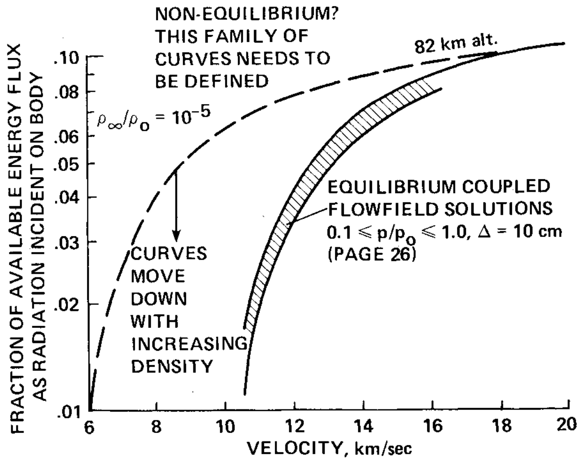

Fig. 4 Radiative flux limitation.

for ballutes in Ref. 8 was calculated by an expression given as

$$q_c \propto \rho_\infty^{0.5} U_\infty^3 \tag{5}$$

which would be equivalent to our estimate [Eq. (4)] if the proportionality constant is $\frac{1}{2}\sqrt{(\rho_\infty \lambda_\infty / R_b)}$. In Ref. 8 convective heating would be controlled to the levels cited previously by running the retrorocket engine at a low thrust level to envelope the vehicle in the low-enthalpy exhaust gases to partially block the convective flux from the high-enthalpy atmospheric gases.

It should be mentioned in passing that the convective heating of Ref. 8 appears to have been estimated by the correlation of results of Fay and Riddell[31] for dissociated equilibrium air. In the ionization regime, the equilibrium convective heating rate is higher than Fay and Riddell's as shown by Howe and Viegas.[20]

Figure 6 shows the convective heating regime for a body having a nose radius of 1 m and about the same perigee as shown in Fig. 5. More of the convective heating occurs in the quasicontinuum regime and is significantly higher (about 50 W/cm²) in the stagnation region according to our approximation because of the smaller nose radius.

Aerothermodynamics of Orbit on Demand Concepts

Here we are considering small (e.g., 16 m long), control-configured vehicles that will take off from Earth, orbit the Earth for a short period of time (e.g., 24 h), reenter the atmosphere, and land horizontally. The payload will be small, the vehicle may be configured for two persons in a "space suit" environment, will be durable, have a short turnaround time (e.g., 24 h), be control configured to have cross-range capability (e.g., 3000 km), and have no special ground handling requirements other than fuel (which would quite likely be something other than L_H/L_O because of volume requirements). One such concept is depicted in Fig. 7.

The entry trajectory is determined by the planform loading and the thermal protection system must be able to survive the heating associated with that trajectory. However, for present purposes of assessing aerothermal issues, let us examine trajectories that correspond to the thermal limitations, realizing that the planform loadings are not the same.

Because of the durability requirement, much of the thermal protection will probably have a metallic outer skin, such as was mentioned in the introduction. Allowable metal use temperatures will limit the altitude at which the vehicle decelerates through its heat pulse—the lower the temperature, the higher the altitude.

Zoby[32] has shown that the convective heating on the centerline of the underside of the Space Shuttle at angle of attack can be approximated by that heating derived from the flow over an equivalent axisymmetric blunt body. Thus, if the convective heating to a vehicle of one size (such as the Space Shuttle) is known, we can estimate the heating to a vehicle of another size and, if the second vehicle has a known use temperature, we can estimate the altitude limitation for the heat pulse of the second vehicle. We revert to quasicontinuum theory [Eq. (4)] and consider that the nose radii of the two vehicles characterize their relative sizes. Thus, at a given body location (for the rigid sphere molecular model kinetic theory),

$$\frac{q_{c1}}{q_{c2}} = \frac{k_1 U_{\infty_1}^3}{k_2 U_{\infty_2}^3} \sqrt{\frac{R_{b_2} \rho_{\infty_1}}{R_{b_1} \rho_{\infty_2}}} \tag{6}$$

where k is a simple adjustment for catalycity of the surface. For surfaces that are convectively heated and radiatively cooled, wherein the materials are in thermal equilibrium with the heating environment and for a common velocity, this estimate becomes

$$\frac{\epsilon_1 T_{w_1}^4}{\epsilon_2 T_{w_2}^4} = \frac{k_1}{k_2} \sqrt{\frac{R_{b_2} \rho_{\infty_1}}{R_{b_1} \rho_{\infty_2}}} \tag{7}$$

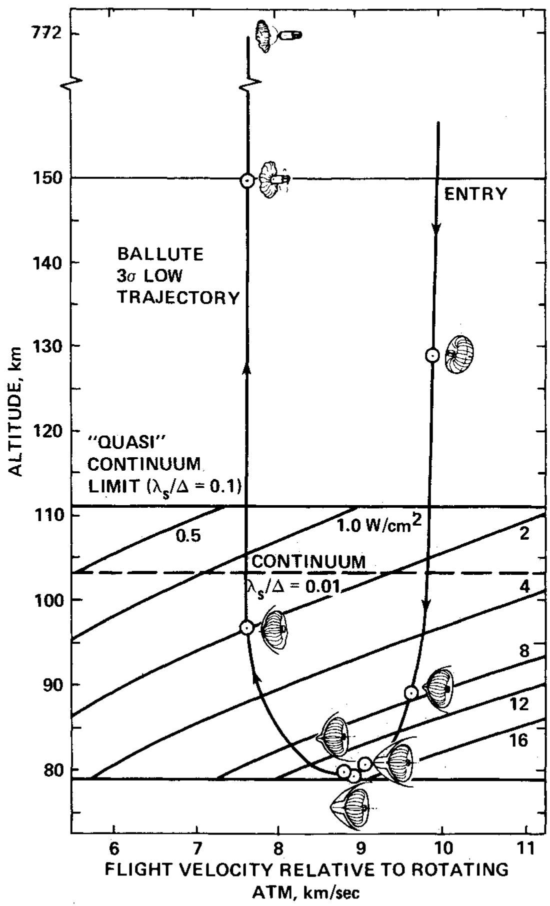

Fig. 5 AOTV stagnation region convective heating rate, W/cm², $R_b = 10$ m.

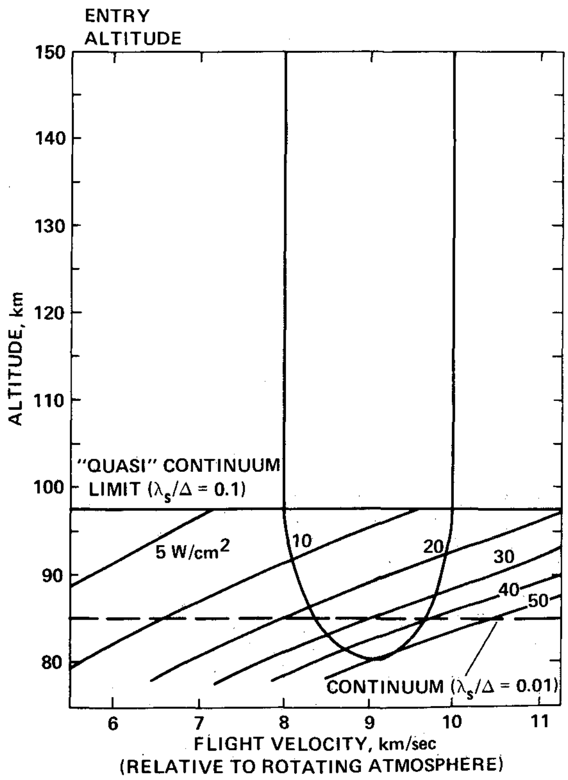

Fig. 6 AOTV stagnation region convective heating rate, W/cm², $R_b = 1$ m.

If the emissivity and catalycity of the two surfaces are the same and one vehicle is shuttle size (37.2 m long) and the second vehicle is smaller (15.25 m), and if Shuttle use temperature is 1589 K (2400°F) and the second vehicle's is 1033 K (1400°F) (René 41 for example), then $\rho_{\infty 2} = 0.0133\rho_{\infty 1}$. Figure 8 shows a Shuttle trajectory as a solid line. The heating pulse, the location of the infrared image of the Shuttle (IRIS), and transition regime from laminar to turbulent flow are shown. The upper dashed line is our estimate of the trajectory of the small second vehicle; its heating pulse is at about 27 km higher altitude than that of the Shuttle. If the Shuttle is noncatalytic and the small vehicle catalytic, the altitude would be higher proportional to the $(k_2)^{-1/2}$. Figure 8 also shows a trajectory for a small vehicle with a use temperature of 1256 K (1800°F) (cobalt-based superalloy). The tantalum vehicle trajectory could be that of a Shuttle vehicle if a noncatalytic coating could be applied.

This figure indicates some aerothermodynamic technology problems to be considered in conceptual tradeoff studies. The aerodynamic heating occurs at high altitudes where the effects of a chemically relaxing gas must be accounted for. A small hybrid vehicle[7] could probably fly anywhere in those domains, given the proper surface coatings. It is noteworthy that turbulent convective heating need not be a problem for any of these small vehicle concepts because the Reynolds number would be lower than that for the Shuttle due to size and air density. Note also that we have limited entry velocity to about 7.5 km/s relative to the atmosphere. This lower velocity avoids the onset of significant ionization (to the right of the vertical dashed line at the right of the figure) and the concomitant radiative heating mechanisms discussed in the section on AOTV aerothermodynamics.

The entry of maneuverable, small, orbit-on-demand vehicles may be limited to less energetic conditions than those of the AOTVs to avoid the use of heat shields that add to launch weight. This would limit the use of small orbit-on-demand vehicles to near-Earth space—similar to the Space Shuttle. However, rendezvous could be made with orbital transfer vehicles to perform tasks required at higher orbits.

More energetic entries of small maneuverable vehicles could be considered and tradeoffs would have to be made with reflecting or backscattering radiation. Numerous studies of noncatalytic, backscattering heat shield have been performed, even for ablating materials.[33-38] It is shown that for a heated surface in thermal equilibrium with its surroundings

$$T_w = \left[\frac{kq_{c_{eq}} + \sum_{\nu=0}^{\infty} (1-R_\nu) q_{r_\nu}}{\sigma \sum_{\nu=0}^{\infty} \epsilon_\nu} \right]^{1/4} \tag{8}$$

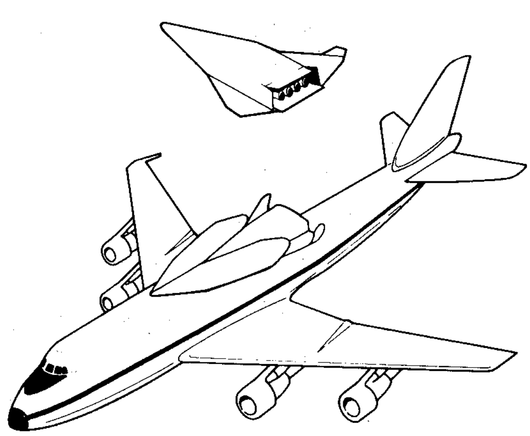

Fig. 7 Maneuverable launch, orbit, re-entry concept.

Thus for a given flight condition, the use temperature could be maintained for a surface exposed to both convective and radiative heating from a chemically relaxing flowfield if the catalytic efficiency k is low, q_{r_ν} is in the visible part of the spectrum in which R_ν is large, and ϵ_ν (which is in the infrared part of the spectrum for the surface temperatures under consideration) is large. This tells us the nature of our surface coating. Dielectric surfaces such as silica have such properties. This too is a major area for innovative supporting research and technology in which Goldstein et al.[39] are heavily involved.

Supporting Research and Technology Issues

Some of the immediate aerothermodynamic problems for conceptual studies of advanced space transportation systems have been identified in this paper. Our next step is to see what aerothermodynamic efforts should proceed in conjunction with conceptual studies that will result in our ability to design such vehicles. We must develop the ability to compute three-dimensional real-gas flowfields, including the effects of the transport of mass, momentum, energy, and chemical species; gaseous radiative transfer; finite excitation and chemical relaxation rates; and the appropriate boundary conditions for hypervelocity flight in low-density flow. Modern computational algorithms and gridding techniques need to be developed. Moreover, inputs to such codes need to be defined and radiative and chemical rate coefficients must be identified and obtained. These data should include theoretical computations of atomic and molecular properties from first principles and experimental measurements. Flow visualization experiments over unusual and complicated shapes must be performed to identify the principal features of the flow in order to aid computational grid-generation techniques. Experimental heating simulations should be performed to validate the results derived from the flowfield computations. Low-density hypervelocity aerodynamic characteristics of candidate configurations must be obtained by theory and experiment, probably in free-flight facilities. Finally, chemical, structural, thermal, and optical properties of materials and their surfaces must be obtained, both as

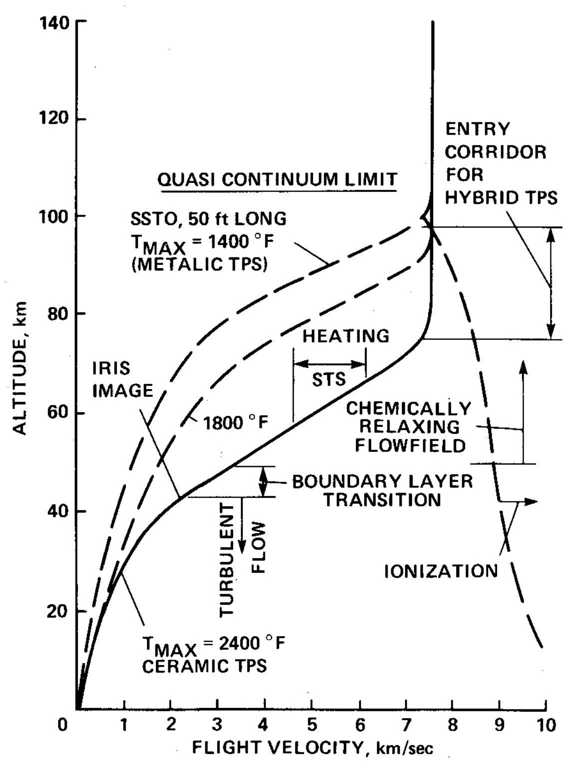

Fig. 8 Flight domain, manned maneuverable entry vehicle.

inputs to computational codes and to size vehicle components adequately.

Conclusions and Comments

At the outset of conceptual studies of advanced space transportation systems, we have defined some of the dominant aerothermodynamic features associated with two generic concepts. Aeroassisted orbital transfer vehicles that operate from the Space Shuttle out to geosynchronous orbit (or beyond) and return for a shallow pass through the upper atmosphere will be exposed to hypervelocity, low-density environments. This is a very energetic flight regime and radiation from chemically relaxing air will dominate the vehicle heating. It may be advantageous to use deployable drag devices with surface properties that are tailored to the environment, are noncatalytic with respect to recombination of dissociated air, reflect or backscatter radiation in the visible and ultraviolet part of the spectrum, and are highly emissive in the infrared. Dielectric materials such as silica appear to be candidates for surface coatings or as actual structural materials—either flexible or rigid.

Small, advanced maneuverable spacecraft that are ground launched or launched from the back of a conventional aircraft, that operate in a low Earth orbit, that re-enter the atmosphere with both lift and cross-range capability, and that land on a runway were also considered. It was noted that it may be desirable to limit the entry velocity of these spacecraft relative to the atmosphere to about 7.5 km/s to avoid a massive thermal protection system for radiative heating. The thermal protection for these vehicles should, for the most part, be nonablating so that they are reusable with rapid turnaround time. The vehicles should be durable. Thus, where metallic surfaces are considered (such as for standoff TPS systems or hot structures), flight is limited by a metal use temperature. If the use temperature is 1033 K (1400°F), it was estimated that the vehicles will fly almost 30 km higher than the Space Shuttle during entry heating. (It should be noted that, for standoff TPS, insulation weight may be a problem because of long-duration heat pulses.) If the allowable use temperature is 1256 K (1800°F), such vehicles will decelerate almost 20 km above the Shuttle entry path. A more reasonable approach is the hybrid TPS approach, which employs small areas of graphitic and reusable surface insulation (RSI) materials and larger areas of various tailored standoff metallic systems. Such a hybrid has been shown by others to have weight advantages. Because the vehicle is small, the flow will be laminar and such a maneuverable vehicle would have a broad flight envelope.

Aerothermodynamic technology problems associated with energetic flight in low-density air must be studied and more adequately defined. Such problems as those concerned with unusual flowfield structure over complicated configurations, aerodynamic characteristics, radiative excitation and de-excitation rates, as well as chemical relaxation rates of the air species that will reside in very high-temperature environments must be determined. Robust three-dimensional, real-gas flowfield codes should be developed. These supporting research and technology efforts should proceed in parallel with more approximate conceptual studies so that reliable predictions of the behavior of promising configurations can be performed. Fortunately, our theoretical and experimental capability to perform such research has improved dramatically over the past decade or two.

Finally, as a boundary condition for our aerothermodynamic studies, we must devise and characterize the properties of advanced thermal protection materials.

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