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Thermal Structures: Four Decades of Progress

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Since the first supersonic flight in October 1947, the United States has designed, developed, and flown flight vehicles within increasingly severe aerothermal environments. Over this period, major advances in engineering capabilities have occurred that will enable the design of thermal structures for high-speed flight vehicles in the 21st century. This paper surveys progress in thermal structures for the last four decades to provide a historical perspective for future efforts.

Introduction

THE design of structures for winged flight vehicles that fly through the earth's atmosphere, either to and from space or in sustained flight, poses severe challenges to structural designers. Major components of the challenge are to select materials and design structures that can withstand the aerothermal loads of high-speed flight. Aerothermal loads exerted on the external surfaces of the flight vehicle consist of pressure, skin friction (shearing stress), and aerodynamic heating (heat flux). Pressure and skin friction have important roles in aerodynamic lift and drag, but aerodynamic heating is the predominant structural load. Aerodynamic heating is extremely important because induced elevated temperatures can affect the structural behavior in several detrimental ways. First of all, elevated temperatures degrade a material's ability to withstand loads, because elastic properties such as Young's modulus are significantly reduced. Moreover, allowable stresses are reduced and time-dependent material behavior such as creep come into play. In addition, of course, thermal stresses are introduced because of restrained local or global thermal expansions or contractions. Such stresses increase deformation, change buckling loads, and alter flutter behavior.

Today, the advent of the National Aerospace Plane offers structural engineers new challenges for the design of thermal structures for high-speed flight. A theme of this paper is that the progress in research and development of thermal structures and related technologies over the last four decades provides the foundations upon which to meet the new challenges.

One justification for a survey paper is expressed very well by the opening of the "House divided" speech given by Abraham Lincoln in 1858, "If we could first know where we are, and whither we are tending we could better judge what to do, and how to do it." The speech, given to a political convention in Springfield, Illinois in June of that year, referred to an altogether different subject, but surely the words apply in the present context. As our nation looks to high-speed flight vehicles for the next century, we review past efforts to provide a historical perspective for the future.

Aerospace technology has advanced far in a very short time. These advances are particularly notable for high-speed flight vehicles. This paper provides a historical perspective for thermal structures by describing the evolution of thermal structure technology from the early 1950s to the technology of the early 1990s. The early 1950s was selected as the reference point because the first manned supersonic flight in 1947 stimulated a period of intense research on high-temperature structures. We are fortunate today to have access to the papers, reports, and books that describe research of that era. Unfortunately, although many researchers of that period are alive today, much of their personal experience is not available to younger researchers because of the "generation gap" that currently exists in the aerospace establishment. Thus for all of us interested in thermal structures, there are lessons to be learned from surveying the progress of the last 40 years.

Evolution of Thermal Structures

From World War II to Sputnik

The need to understand aerothermal loads and the design of thermal structures have their origins in the late 1940s. In World War II, airplane speeds had become high enough for compressibility phenomena to have a significant role in performance. Transonic phenomena were not well understood, and over a period of years the phrase "sound barrier" came into use. The need for a transonic research airplane was recognized during the war, and in 1944 the design development of the Bell X-1 program was initiated.[1] The X-1 proved to be enormously successful, and the flight of Captain Charles E. Yeager on October 14, 1947 proved beyond doubt that manned aircraft could fly faster than the speed of sound. An advanced version of the aircraft, the X-1B shown in Fig. 1, flew several research missions for NACA to study aerodynamic heating effects. The original X-1 aircraft, as well as the advanced version, used aluminum construction throughout. Measured skin temperatures are shown for a NACA research mission flown in January 1957 at Mach 1.94. Note that skin temperatures are low, less than 200°F. Thereafter, supersonic flight speeds increased rapidly, and the need for considering aerodynamic heating became evident. The difficulties presented by high temperatures accompanying flight at supersonic speeds became known as the "thermal barrier." For about 10 years, the "thermal barrier" caused concern that large structural weight increases would be required to keep material temperatures within allowable values. Subsequently researchers[2,3] found that these concerns did not materialize because the problems were overcome through research and development of effective thermal structures.

After the first supersonic flight, research and development of high-speed aircraft intensified. A contract for the design, development, and construction of two X-2 swept-wing, supersonic research aircraft was awarded to the Bell Aircraft Corporation in 1947. The X-2 was the first aircraft structure designed for aerodynamic heating.[4] Until the X-2, speeds had not been high enough for the structure to be affected adversely by aerodynamic heating. For increased strength at elevated temperatures, the fuselage was constructed from K-Monel, and aerodynamic skins used stainless steel. A drawing shown

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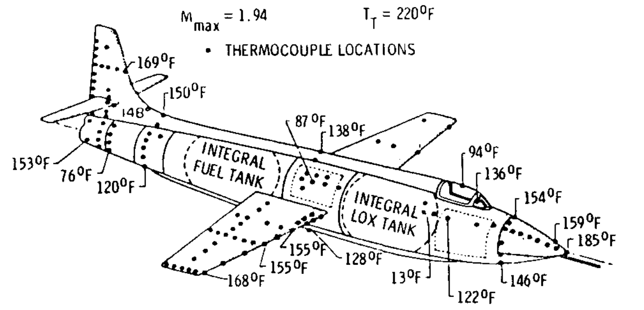

Fig. 1 Maximum measured temperatures on X-1B airplane, Mach 1.94, 1957.[12]

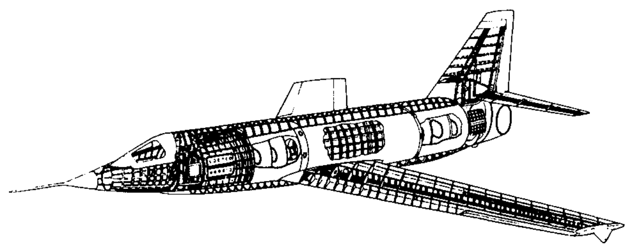

Fig. 2 X-2, K Monel internal structure and stainless steel skin, Mach 3.2, 1956.

in Fig. 2 illustrates the X-2 structure. The X-2 became the first research aircraft to explore the "thermal barrier" with speeds above Mach 2.5. On September 27, 1956 the X-2 achieved its maximum speed of Mach 3.2, but unfortunately the aircraft went out of control and the pilot, Captain Milburn G. Apt, was killed.

During this period, theoretical and experimental studies were in progress to develop the technologies needed for high-speed aircraft. Considerable research was underway in compressible flow, thermal structures and materials. Excellent descriptions of these efforts are available through survey articles and collections of papers in books describing thermal structures conferences. The *Applied Mechanics Reviews* article[5] of November 1955 by Professor N. J. Hoff of Polytechnic Institute of Brooklyn describes high temperatures effects in aircraft structures with emphasis on the effects of temperature on buckling and creep. In December 1955 Professor R. L. Bisplinghoff presented the Nineteenth Wright Brothers Lecture to the Institute of Aeronautical Sciences in Washington, D.C. The paper,[6] including discussion by several prominent researchers, was published in April 1956. Professor Bisplinghoff's paper gives a comprehensive review of structural considerations for high-speed flight and an excellent account of design and analysis practices. A book[7] published by the Advisory Group for Aeronautical Research and Development (AGARD) of the North Atlantic Treaty Organization (NATO) in 1958 describes effects of high temperatures on aircraft structures caused by aerodynamic heating. The book, edited by Professor Hoff, contains 16 articles written by U.S. and European authors.

The next major flight program that stimulated thermal structural research was the X-15. The X-15 had complex origins including the prewar and postwar work of German scientists, Eugene Sanger and Irene Bredt, who, in 1944, outlined a hypersonic, rocket-propelled aircraft. The evolution of their ideas, which contributed to the development of the X-15, is described in the award-winning paper by Richard P. Hallion.[8] Further descriptions of the X-15 program are given by Hallion[9] and by NASA Langley research scientist John V. Becker.[10] The paper written by John Becker was presented in Bonn, West Germany in December 1968. At this meeting he accepted the Eugene Sanger medal awarded by the German Society of Aeronautics and Astronautics to honor the success of the X-15 program.

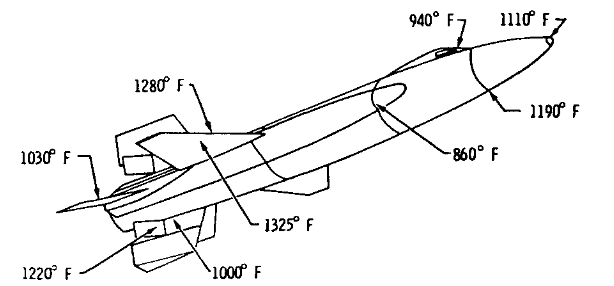

Fig. 3 Measured temperatures on a X-15 flight, circa 1965, Mach 5.0.[11]

On June 24, 1952 the prestigious NACA Committee on Aeronautics charged the agency to study problems of manned and unmanned flight at altitudes between 12 and 50 miles and speeds of Mach 4 to Mach 10. By 1954 the NACA Langley Laboratory (now NASA's Langley Research Center) had formed a hypersonic study team with John Becker as the chairman. According to Becker[10] the unprecedented problems of aerodynamic heating and high-temperature structures appeared to be so formidable that they were viewed as "barriers" to hypersonic flight. Nevertheless, the NASA group evolved a baseline design that closely resembled the ultimate X-15 configuration. Three X-15 aircraft were built by North-American Rockwell.

A thick-skinned, heat-sink approach was adopted to suit the short duration missions of the X-15. A typical research mission lasted 10–12 minutes.[11] Surfaces exposed to aerodynamic heating were made of Inconel X, a nickel alloy. Internal structures not subject to high temperatures were made of titanium. Skin temperatures were designed for a maximum of 1200°F. Figure 3 shows maximum temperatures experienced in an X-15 flight.

On October 4, 1957 the Soviet Union orbited Sputnik 1, the world's first artificial satellite. This event changed the nation's priorities for high-speed, high-altitude flight, making the X-15 program vital to America's national prestige. The X-15 program served the nation well, accomplishing 199 missions between 1959 and 1968. The X-15 was the first, and to date, the only manned vehicle capable of flying atmospheric missions at Mach 5 for altitudes fo 100,000 feet or higher. It made many contributions to the understanding of hypersonic flight, including the design of thermal structures. Later sections refer to some of these contributions.

From World War II to Sputnik, the nation's research programs were focused on high-speed flight of aircraft. In thermal structures, the NACA Langley laboratory made significant contributions. Many of the Langley research efforts were led by Richard R. Heldenfels. After his retirement, he described research[12] conducted at Langley from 1948 to 1958 on structural problems caused by aerodynamic heating.

After Sputnik

After Sputnik, the nation's high-speed flight research program was broadened to include a major emphasis on manned space flight. The National Aeronautics and Space Act of 1958 created NASA, and NACA became NASA on October 1, 1958.

The expanded scope of the research is reflected in the 1960 paper by Heldenfels.[3] The paper describes proposed re-entry structures as well as space vehicles and space structures. The proceedings[13] of a conference held in Cambridge, Massachusetts on July 25, 1961 contains 13 papers describing research on thermal structures for manned and unmanned re-entry vehicles. The conference was concerned primarily with thermal protection systems for lifting vehicles. Two types of thermal protection systems are described: "cool-structure" and "hot-structure" approaches. In the cool-structure approach, the load-bearing structure is insulated from high temperatures by an external heat shield. In the hot-structure approach, the load-bearing structure operates at nearly skin temperatures.

A NASA-University Conference on the Science and Technology of Space Explorations was held in Chicago, Illinois on November 1–3, 1962. The conference proceedings describe research on structure for launch vehicles, winged aerospace vehicles, and planetary entry vehicles. The paper[14] by Mathauser gives a good account of Langley research on thermal-structural problems for winged vehicles. Types of winged vehicles under consideration by NASA include a research airplane (the X-15), a re-entry glider, and a large hypersonic aircraft. The re-entry glider is representative of a X-20 Dyna-Soar vehicle that was to be launched with a ground-based booster. The hypersonic aircraft was to possess horizontal takeoff capability and be capable of sustained hypersonic flight. All three vehicles utilize radiation-cooled hot structures. Radiation equilibrium temperatures [see Eq. (13)] estimated for a re-entry glider are shown in Fig. 4.

The re-entry glider and hypersonic aircraft of the Mathauser paper are hypothetical vehicles used in NASA's fundamental research studies. However, the Dyna-Soar project sponsored by the Air Force did lead to the final design of the Boeing X-20. The events leading to the Dyna-Soar program, its evolution, and subsequent demise are described by Miller[4] and Hallion.[8] The Boeing development program began with a contract award in November 1959. The X-20 was designed to provide a piloted, maneuverable vehicle for conducting experiments in the hypersonic and orbital flight regime. However, the X-20 program was canceled in December 1963 before the first vehicle was completed. The Dyna-Soar program accelerated progress in several technologies that ultimately were applicable to the space shuttle. A review of the Dyna-Soar winged spacecraft technology appears in the 1961 paper by Yoler.[15] The structural design utilized a Rene 41 nickel super alloy primary structure, a columbian alloy heat shield, a graphite and zirconia nose cap, and molybdenum alloy leading edges.

Concurrent with these activities, the U.S.'s manned space flight program accelerated rapidly. The effort began with the Mercury program, including the Alan B. Shepard, Jr. suborbital flight on May 5, 1961 and America's first orbital flight by John H. Glenn, Jr. on February 20, 1962. The effort continued with the Gemini program and the first two-man flight by John W. Young and Virgil I. Grissom on March 23, 1965. The Gemini program achieved the first rendezvous and docking in space and the first American "space walk." The Apollo program began with the October 11–22, 1968 flight by Walter M. Shirra, Jr., Donn F. Eisele, and R. Walter Cunningham. The program reached its zenith with the historic flight of Apollo 11 by Neil A. Armstrong, Edwin E. Aldrin, Jr., and Michael Collins and the first lunar landing on July 20, 1969. The lunar program concluded with Apollo 17 making the sixth and last lunar landing in December 1972. America's first Earth-orbiting space station, Skylab, was launched atop a Saturn V booster on May 14, 1973. Three, three-manned crews visited the space station with the last mission returning to Earth in February 1974. The rendezvous and docking of an Apollo spacecraft with a Russian Soyuz craft in Earth orbit on July 18, 1975 closed out the Apollo program.

All of the manned spacecraft missions used blunt-body re-entry vehicles and ablative heat shields to dissipate aerodynamic heating. These ballistic or semiballistic vehicles had small lift to drag ratios permitting only limited maneuverability.

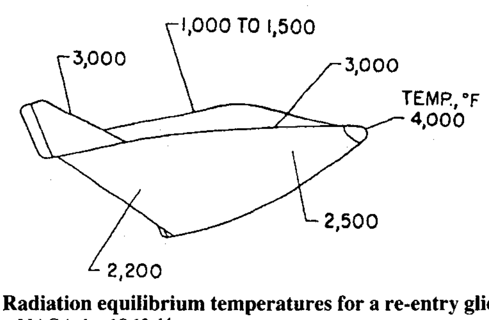

Fig. 4 Radiation equilibrium temperatures for a re-entry glider computed by NASA in 1962.[14]

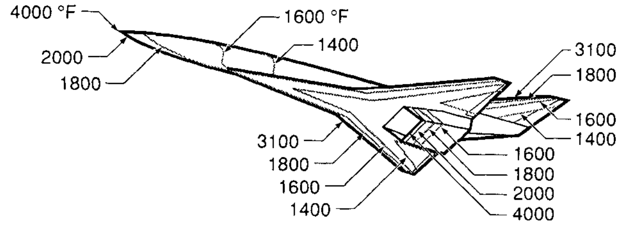

Fig. 5 Equilibrium surface temperatures for a NASA hypersonic vehicle concept for sustained flight at Mach 8 at 88,000 feet, 1966.[16]

The lack of a follow-on winged research flight vehicle to succeed the X-15 forced thermal structures research in the 1960s into a period of more fundamental rather than applied research. The 1962 paper[14] by Mathauser indicates that research had begun at Langley on hypersonic structures. A 1966 paper[16] by Heldenfels discusses structural prospects for hypersonic vehicles in more detail. Figure 5 shows isotherms on a hypersonic vehicle assumed to be cruising at Mach 8 at 88,000 feet. Temperatures shown are radiation equilibrium temperatures expected during a typical flight of 1–2 h duration. The paper discusses thermal-structural designs for fuselage tanks for liquid hydrogen, wings, and engine structures. Hydrogen fuel-cooled structures for engines and passive hot structures of high temperature materials for airframes are described. The proceedings of a conference held at Langley in November, 1971 contains several papers describing basic hypersonic vehicle research. A paper by Anderson[17] and Kelly reviews the technology base for propulsion structures, primary structures and liquid hydrogen tanks. A recent paper by Shore[18] reviews research on convectively-cooled structures in the 1960s and 1970s.

Space Shuttle

The Space Shuttle resulted from a perceived need in the 1960s for a logistical spacecraft to support orbital space stations. However, after the lunar landing in 1969, NASA recognized that funds would not be available to support both the Shuttle and a space station. Justification for the Shuttle shifted from space station support to its use as a substitute for expendable launch systems. With strong support from the Department of Defense, the preliminary analysis phase began in February 1969 with contracts to Lockheed, General Dynamics, McDonnell Douglas, and North-American Rockwell. Rockwell eventually became the prime contractor, and construction of the actual Shuttle orbiter began in June 1974. Rockwell completed this vehicle, the Enterprise, in September 1976. Difficulties with the main engines and thermal protection system delayed the second shuttle, the Columbia, until 1981. Piloted by astronauts John W. Young and Robert L. Crippen, Columbia completed the Shuttle's first orbital flight on April 14, 1981. Further details of its history are given by Hallion.[8]

The orbiter basically has a conventional skin-stringer aluminum aircraft structure. The design of the thermal protection system (TPS) had the requirement of keeping structural temperatures less than 350°F. The thermal protection is composed of two types of reusable surface insulation (RSI) tiles. The RSI tiles covering the orbiter are made of coated silica fiber. The two types differ only in surface coating to provide protection for different temperature environments. The low-temperature insulation (LRSI) consists of 8-in.-square silica tiles and covers the top of the vehicle where temperatures are less than 1200°F. The high-temperature insulation (HRSI) consists of 6-in.-square tiles that cover the bottom and some leading edges of the orbiter where temperatures are below 2300°F. Reinforced carbon-carbon (RCC) is used for the nose cap and wing leading edges where temperatures are above 2300°F.

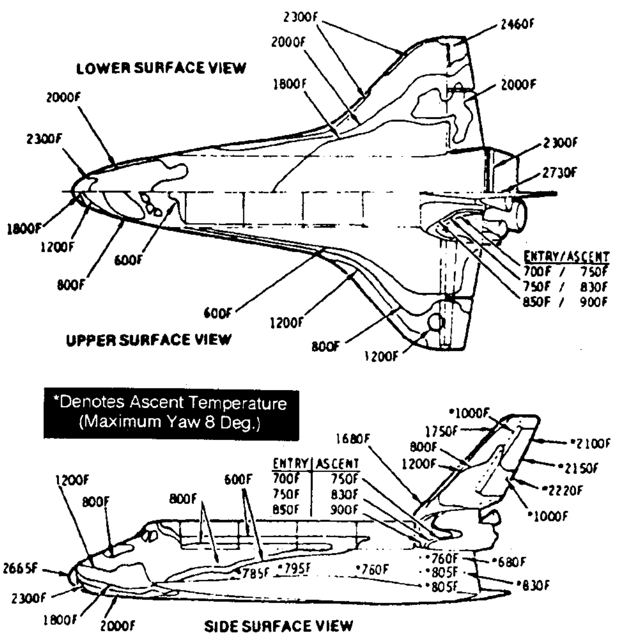

Fig. 6 Space shuttle expected surface temperatures, 1976.[19]

Flexible reusable surface insulation (FRSI) is used at locations where temperatures are less than 700°F. Orbiter isotherms for a normal flight are shown in Fig. 6. Further details of the orbiter are given in Ref. 19. The launch of the first shuttle flight was delayed by problems with the TPS, and the article[20] by Langley researchers Paul A. Cooper and Paul F. Holloway explains the problems and their resolution.

National Aerospace Plane (X-30)

In his State of the Union address in February 1986, President Reagan made the initial announcement of research on development of the National Aerospace Plane (NASP). The November 1986 article[21] by Robert L. Williams describes the program objective "to develop, and then demonstrate in an experimental flight vehicle (the X-30), the requisite technologies to permit the nation to develop both military and civil vehicles capable of operating at sustained hypersonic speeds within the atmosphere and/or operating as space launch vehicles for delivering payloads into orbit." Under joint sponsorship by NASA and DOD, technology development began in 1986, and by 1988 McDonnell Douglas, General Dynamics, and Rockwell had won separate contracts to proceed with development of the airframe. Pratt and Whitney and Rocketdyne had won contracts for propulsion system development. A consortium of these contractors is participating in a 30-month materials and structures augmentation project.[22]

The competitive nature of the development program and security issues have limited the number of publications describing thermal structures for the X-30. However, at the 30th SDM conference, David A. Ellis of the McDonnell Douglas Corporation presented an overview paper[23] describing the concepts under consideration as well as the types of trade studies being conducted. Figure 7 shows representative surface temperatures for a typical NASP hypersonic condition. Candidate primary structure design concepts include hot structures, actively cooled structures, and thermally protected structures. The design of the center fuselage is particularly challenging because the external skin is subjected to aerodynamic heating, and the internal structure is exposed to cryogenic temperature due to the hydrogen fuel. The paper stresses the importance of a highly interactive, multidisciplinary design/analysis procedure.

Aerodynamic Heating

The prediction of aerodynamic heating is a problem that has challenged thermal analysts since before World War II.

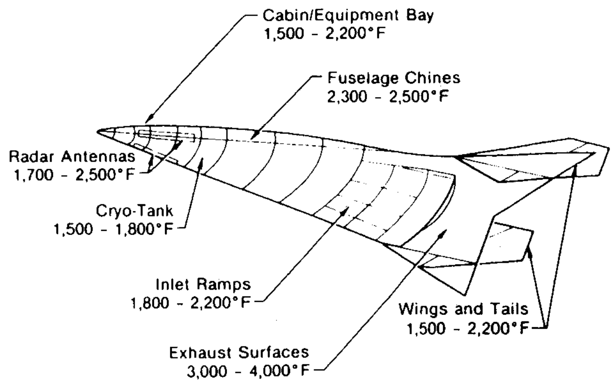

Fig. 7 Representative surface temperatures for the NASP, 1989.[23]

The problem involves solving the equations that describe the conservation of mass, momentum, and energy for fluid flow. For altitudes less than 300,000 ft the atmosphere may be assumed to be a continuum, and the set of conservation equations are known as the Navier-Stokes equations. For two-dimensional flow the Navier-Stokes equations governing viscous, compressible flow are a set of four nonlinear, partial differential equations with mixed hyperbolic, parabolic, and elliptic behavior. A solution of these equations with appropriate boundary conditions and constitutive equations for the fluid provides distributions for the density ρ, the velocity components u and v, the pressure P, and the temperature T throughout the fluid. At the fluid/aerodynamic surface interface, the aerodynamic heating is computed from Fourier's law using the fluid thermal conductivity and the derivative of temperature normal to the wall.

Prior to the digital computer, there was no hope for solving the complete Navier-Stokes equations. Instead, analysts devoted substantial effort to obtaining approximate solutions, and with the help of experimental data they developed engineering relations to predict aerodynamic heating as well as skin friction. The engineering relations typically are based on boundary layer or other approximations. The relations are restricted by a number of assumptions, and the application of computers to the problem of aerodynamic heating has contributed significantly to a better understanding of the phenomena and improved reliability of predictions.

There is substantial literature describing aerodynamic heating, and the esoteric details will be left for thermal analysts. However, because of the interdisciplinary nature of thermal-structural problems for high-speed vehicles, some fundamental aspects are presented. Early approaches for the prediction of aerodynamic heating are described in the 1956 paper[24] by Van Driest, and in the 1960 text[25] by Robert W. Truitt. A recent discussion appears in the 1989 text[26] by John D. Anderson, Jr.

The preceding section mentioned that aerodynamic heating effects become significant at Mach numbers above 2.5. For Mach numbers up to about five, the flow is considered supersonic, and above Mach five the flow is termed hypersonic. The NASP will experience aerodynamic heating through the supersonic and hypersonic regimes with a peak Mach number of about 25. As the Mach number increases above five, new physical phenomena become progressively of greater importance, making hypersonic flow more complex than supersonic flow. These phenomena include: 1) fluid dynamics effects that limit the validity of boundary-layer approximations, and 2) high-temperature effects that introduce chemical reactions. In addition, at altitudes above 300,000 ft low-density effects restrict the application of continuum models.

Classical Boundary-Layer Predictions

The concept of the boundary layer was introduced by Prandtl in 1903 for incompressible flows. The extension of the concept to compressible flows began in the 1930s, and analytical work

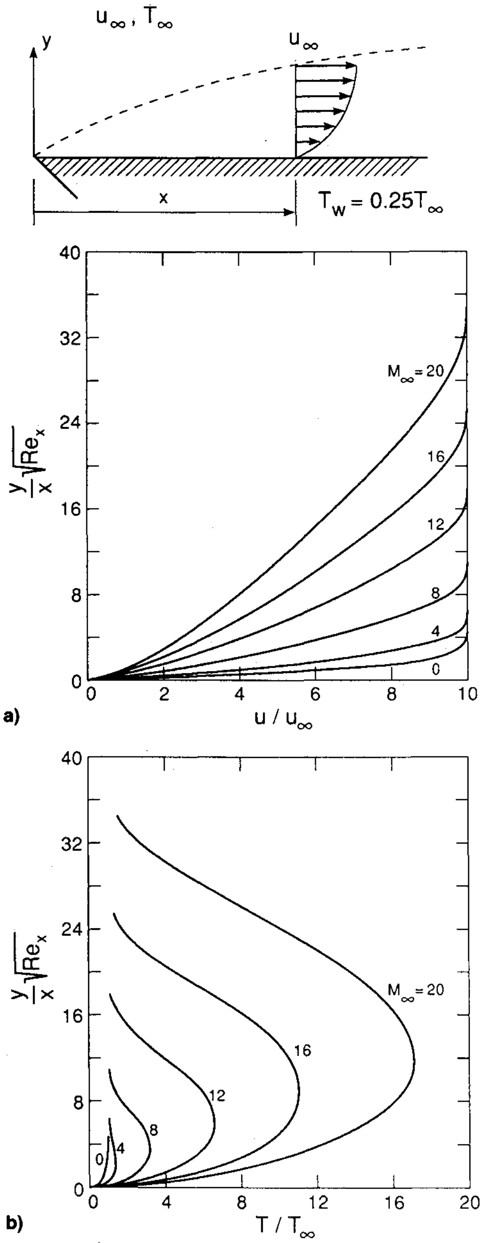

Fig. 8 Compressibility boundary-layer flow, 1952[27]: a) velocity profiles; b) temperature profiles.

continued until the mid 1950s. Thereafter, most research turned to the development of computational methods for solving aerodynamic heating problems.

The boundary-layer equations are obtained from the Navier-Stokes equations by an order-of-magnitude analysis for the thin fluid layer next to the body where viscous effects dominate. Outside of the boundary-layer the fluid is assumed inviscid, and the flow is described by the Euler equations. For steady, two-dimensional, laminar flow, the boundary-layer equations are

$$
\begin{gathered}
\frac{\partial(\rho u)}{\partial x} + \frac{\partial(\rho v)}{\partial y} = 0 \\
\frac{\partial u}{\partial x} + \rho v \frac{\partial v}{\partial y} = \frac{\partial}{\partial y}\left(\mu \frac{\partial u}{\partial y}\right) \\
\frac{\partial p}{\partial y} = 0 \\
\rho u \frac{\partial(c_p T)}{\partial x} + \rho v \frac{\partial(c_p T)}{\partial y} = \frac{\partial}{\partial y}\left(k \frac{\partial T}{\partial y}\right) + u \frac{\partial P_e}{\partial x} + \mu \left(\frac{\partial u}{\partial y}\right)^2
\end{gathered} \tag{1}
$$

In the above, P_e denotes the pressure at the edge of the boundary layer, ρ is density, and T is temperature. The fluid specific heat is c_p, the thermal conductivity is k, and the viscosity is μ. In general, these fluid properties are temperature dependent. The boundary-layer equations are nonlinear but exhibit parabolic behavior. The perfect gas law is used to relate pressure, density, and temperature. For further details, including boundary conditions, see Ref. 26.

For some simple but important cases, analytical solutions to the boundary-layer equations have been obtained. The approach involves making changes of independent and dependent variables to yield simpler, nonlinear, nondimensional equations. These transformations lead to the concept of self-similar solutions for problems such as a flat plate. Typically, the nondimensional equations can be solved numerically for various values for the Mach number M, the Reynolds number Re, and the Prandtl number Pr.

The solution[27] for the flow over a flat plate obtained by Van Driest in 1952 demonstrates typical features. The problem consists of a flat plate with specified uniform temperature T_w, Fig. 8. The inviscid flow outside of the boundary layer has constant velocity u_∞ and temperature T_∞. Within the boundary layer the viscosity varies with temperature according to Sutherland's law,[26] and the Prandtl number $Pr = \mu c_p / k$ is assumed constant, $Pr = 0.7$. The ratio of the wall to freestream temperature is taken as $T_w/T_\infty = 0.25$. Figure 8a shows typical u velocity profiles, and Fig. 8b shows typical temperature profiles. In these figures, the Reynolds number, $Re_x = \rho_\infty u_\infty x / \mu_\infty$. The flat plate boundary-layer solution illustrates results representative of high-speed flows. The velocity profiles show that the boundary-layer thickness increases rapidly with increasing Mach number. In fact, Anderson[26] shows that a hypersonic boundary-layer thickness increases approximately as Mach number squared. One consequence of a large hypersonic boundary-layer thickness is that the viscous boundary layer alters the outer inviscid flow limiting the accuracy of boundary-layer predictions. This behavior is called viscous interaction.

The temperature profiles show the fluid behavior that is of great importance to the structure. The important point is that the peak temperature within the boundary layer is higher than the freestream temperature. Moreover, this peak temperature increases rapidly with increasing Mach number. The high fluid temperature within the boundary layer is due to viscous dissipation. Viscous dissipation is the process where the large kinetic energy of the high speed flow is converted to thermal energy by boundary-layer shearing stresses. The aerodynamic heating rate is proportional, by Fourier's law, to the slope of the temperature profile at the wall ($y = 0$). Figure 8b shows that fluid temperatures within the boundary layer rapidly increase with increasing Mach number. This is the basic reason for the need of special thermal structures for high-speed flight.

Classical, self-similar boundary solutions have also been derived for the stagnation region of a blunt body. An important result of this analysis was to show for hypersonic flows that the stagnation heating rate varies inversely with the square root of the blunt body radius. Thus to reduce aerodynamic heating the vehicle nose and leading edge regions need to be as blunt as possible.

Convective Boundary Condition

For determining structural temperatures, aerodynamic heating is often represented as a convective boundary condition. Boundary-layer analysis of compressible flow shows that the heat transfer between the boundary layer and the wall can be expressed as

$$q_w = h(T_{aw} - T_w) \tag{2}$$

where q_w is the local heating rate (e.g. Btu/ft^2-s) at the wall, h is the convection coefficient, T_{aw} is the adiabatic wall temperature, and T_w is the wall temperature. The adiabatic wall

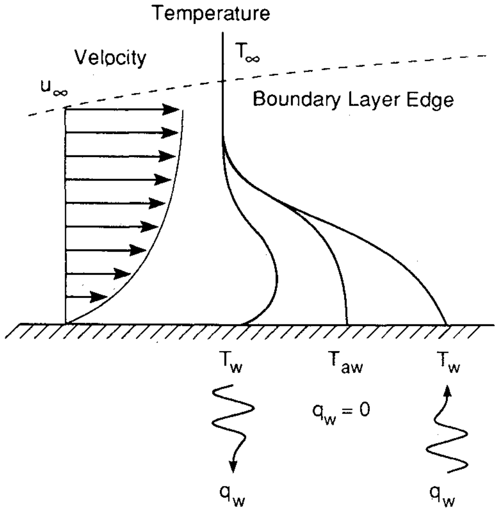

Fig. 9 Velocity and temperature profiles in high-speed flow.

temperature is the temperature the fluid attains for no heat transfer between the fluid and wall. Temperature profiles illustrating the heat transfer possibilities for high-speed flow are shown in Fig. 9. The figure shows that for wall temperatures less than T_{aw}, the fluid flow heats the wall; when the temperature at the wall is T_{aw}, there is zero heat transfer; and, for wall temperatures greater than T_{aw}, the wall heats the fluid. Equation (2) shows that aerodynamic heating is directly proportional to the difference between the adiabatic wall temperature and the actual wall temperature. The adiabatic wall temperature is always higher than the free stream temperature T_∞ and "drives" the heat transfer.

The heat transfer coefficient is often written in nondimensional form

$$C_H = \frac{h}{\rho_e u_e c_p} \tag{3}$$

or using Eq. (2)

$$C_H = \frac{q_w}{\rho_e u_e c_p (T_{aw} - T_w)} \tag{4}$$

where C_H is the Stanton number. Another alternative is to define the Stanton number in terms of fluid enthalpies, i.e.

$$C_H = \frac{q_w}{\rho_e u_e (H_{aw} - H_w)} \tag{5}$$

where H_{aw} is the adiabatic wall enthalpy, and H_w is the wall enthalpy. In this approach, enthalpy replaces temperature as an unknown in the flow analysis. For a calorically perfect gas (constant specific heats), $h = c_p T$, otherwise, $dh = c_p(T)\,\mathrm{d}T$.

Thus, to use a convective boundary condition it is necessary to know the Stanton number and the adiabatic wall temperature. Both, in general, are local quantities; that is, they vary with position along the wall. The classical flat plate boundary-layer solution will serve as an illustration. Extensive studies[24] of the flat plate problem show that the adiabatic wall temperature can be calculated with good accuracy by

$$T_{aw} = T_\infty \left[1 + r\frac{\gamma - 1}{2} M_\infty^2\right] \tag{6}$$

where γ is the ratio of fluid specific heats $\gamma = C_p/C_v$, and r is the recovery factor. Analytical and experimental results show that the recovery factor can be computed as $r = Pr^{1/2}$ for laminar flow and $r = Pr^{1/3}$ for turbulent flow. The boundary layer analysis for the flat-plate flow shows that the Stanton number has the form

$$C_H = \frac{f(M_e, Pr, \delta, T_w, T_e)}{\sqrt{Re_x}} \tag{7}$$

where f denotes a functional relationship. Eckert[28] in 1956 showed that by using the concept of a reference temperature, a simple formula for C_H could be developed from the corresponding result for low-speed, incompressible flow. This approach, called the reference temperature method, computes the Stanton number from

$$C_H = \frac{0.332}{\sqrt{Re_x^*}} (Pr^*)^{-2/3} \tag{8}$$

where Re_x^* and Pr^* are evaluated at a reference temperature T^*. That is

$$Re_x^* = \frac{\rho^* u_e x}{\mu^*} \qquad Pr^* = \frac{\mu^* c_p^*}{k^*} \tag{9}$$

where P^*, c_p^* and K^* are evaluated at a reference temperature T^*. The reference temperature is computed as

$$T^* = T_e[1 + 0.32M_e^2 + 0.58(T_w/T_e - 1)] \tag{10}$$

where as before the subscript e denotes quantities at the edge of the boundary layer. Further details including expressions for skin friction and equations for turbulent flow are given in the Eckert paper,[28] and text[26] by Anderson, and other convective best transfer texts.

Limitations of Classical Boundary-Layer Theory

Classical boundary-layer theory provides basic insight to understand fundamentals of aerodynamic heating. The boundary-layer solution in some cases yields very practical engineering results that are frequently used. But boundary layer theory has limitations that deserve mentioning.

Classical self-similar solutions are limited to a few problems with simple geometries (e.g., the flat plate) and simple boundary conditions (e.g., constant wall temperature). However, in the age of the computer, this limitation is not very serious because the boundary-layer equations, Eq. (1), are solved numerically. In fact, since the equations are parabolic, marching methods may be used, and very efficient computer programs have been developed. When combined with inviscid flow programs to determine variables at the edge of the boundary layer, very effective analysis procedures have been developed. Some of these methods are described in the 1987 paper[29] by DeJarnette et al. which reviews approximate methods for aerodynamic heating analysis.

There are several flow situations where the boundary-layer approximation of uncoupling the inviscid and viscous analysis does not apply. Two such situations are where viscous interaction or flow separation occur. As mentioned earlier, viscous interaction refers to flows where the viscous boundary layer interacts with the inviscid flow. Such a situation may occur for hypersonic flows with relatively low Reynolds numbers. Boundary-layer thicknesses vary inversely with the square root of the Reynolds number, so low Reynolds number flows have thick boundary layers. Flow separation refers to the case where the flow turns away from the wall into itself, introducing a local recirculation region next to the wall. Flow separation does not occur when the pressure decreases in the flow direction. However, when the pressure increases in the flow direction (called an adverse pressure gradient) flow separation occurs, and boundary layer theory does not apply. Flow separation may occur in several problems of importance in high-speed flight. X-15 experience provides two such examples: 1) flow over protuberances and 2) shock/boundary-layer interactions. In both examples an adverse pressure gradient occurs, and flow separation develops. A consequence of flow separation in both instances is to increase significantly the local aerodynamic heating. For these problems, boundary-layer theory is not applicable, and the phenomena is governed by the Navier-Stokes equations.

Navier-Stokes Solutions

One of the major advances in computational mechanics in the last four decades is the development of capability to solve the Navier-Stokes equations computationally. The survey article[30] by Douglas L. Dwoyer et al. describes current capability of computational fluid dynamics (CFD) for hypersonic aircraft. The Navier-Stokes equations written in conservation form are

$$\frac{\partial}{\partial t}\{\mathbf{U}\} + \frac{\partial}{\partial x}\{\mathbf{E}\} + \frac{\partial}{\partial y}\{\mathbf{F}\} = 0 \qquad (11)$$

where $\{\mathbf{U}\}$ is a vector of conservation variables; $\{\mathbf{E}\}$ and $\{\mathbf{F}\}$ are vectors of the flux components in the x and y directions. These vectors are

$$\begin{aligned} \{\mathbf{U}\}^T &= [\rho \;\; \rho u \;\; \rho v \;\; \rho \mathscr{E}] \\ \{\mathbf{E}\}^T &= [\rho u \;\; \rho u^2 + P \;\; \rho uv \;\; \rho u \mathscr{E} + Pu] \\ &\quad - [0 \;\; \sigma_x \;\; \tau_{xy} \;\; u\sigma_x + v\tau_{xy} - q_x] \qquad (12) \\ \{\mathbf{F}\}^T &= [\rho v \;\; \rho uv \;\; \rho v^2 + P \;\; \rho v \mathscr{E} + Pv] \\ &\quad - [0 \;\; \tau_{xy} \;\; \sigma_y \;\; u\tau_{xy} + v\sigma_y - q_y] \end{aligned}$$

where $\mathscr{E}$ is the total energy. Each of the flux vectors contains two vectors of components representing inviscid and viscous flux contributions. The conservation equations are supplemented by a thermodynamic equation of state relating pressure, temperature, and density (typically the perfect gas law). In the viscous flux components, the stresses σ_x, σ_y and τ_{xy} are related to the velocity gradients assuming Stoke's hypothesis. The heat fluxes q_x and q_y are related to temperature gradients by Fourier's law. For air, the temperature-dependent viscosity is computed from Sutherland's law, and the thermal conductivity is computed assuming a constant Prandtl number of 0.72. Considerable success has been achieved in numerical solutions for the two-dimensional Navier-Stokes equations with supercomputers. Numerical solution schemes include finite difference, finite volume, and, more recently, finite element methods. Three-dimensional flows remain a challenge although a viscous solution [31] around an X-24c lifting body has been obtained. Such solutions, even with the largest supercomputers, are not capable of resolving three-dimensional flow details, particularly aerodynamic heating rates. An important problem for which Navier-Stokes solutions are providing valuable insight is shock interaction phenomenon for the NASP.

In 1967 NASA conducted a series of X-15 flights with a dummy hypersonic ramjet engine mounted on a pylon under the rear of the fuselage. On the third flight with the dummy engine on October 3, 1967 the X-15 reached a maximum Mach number of 6.7 at an altitude of 99,000 feet. During the flight[32] severe pylon structural damage was experienced due to complex shock impingement and interference effects on local aerodynamic heating. Since then, shock interference heating has been recognized as a critical problem for high-speed vehicles because extreme pressure and heat transfer rates can occur in highly localized regions where the interference pattern impinges on the surface.

Shock interference heating is an important consideration in the design of the cowl leading edge of the engine structure

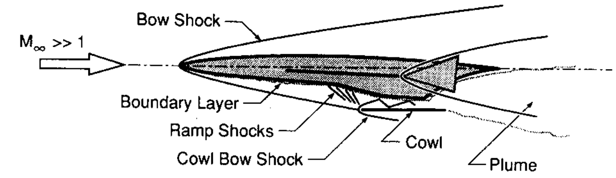

Fig. 10 Overall flowfield for the NASP.[34]

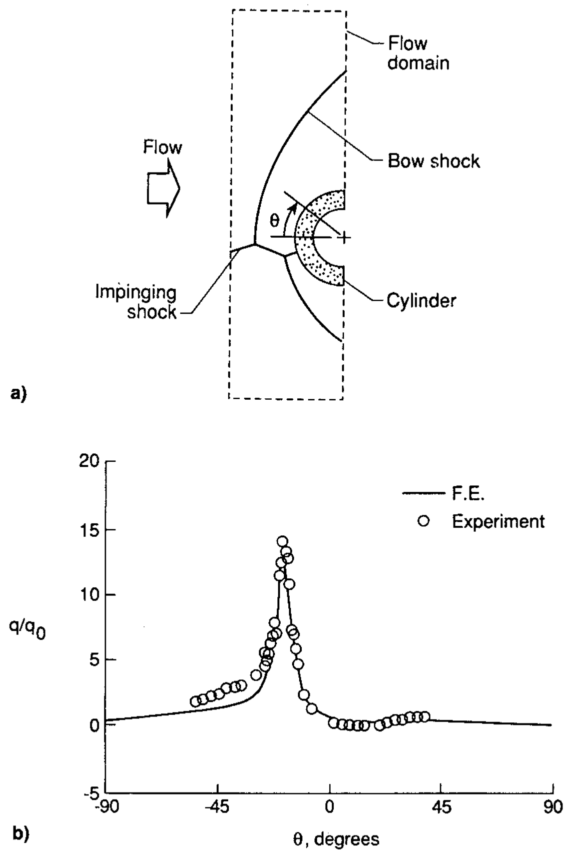

Fig. 11 Shock interaction problem for cylinder[37]: a) shock interaction; b) comparative surface heating rate distribution.

of the NASP. The problem has been strong motivation for recent studies of shock interference heating on leading edges. Figure 10 shows the overall flowfield for the NASP. The interaction between the aircraft bow shock and the cowl bow shock can cause severe local aerothermal loads on the cowl leading edge. Shock interference heating on leading edges has been studied experimentally by Wieting[33] and Holden and computationally by Klopfer[34,35] and Yee as well as Thareja[36] et al. Dechaumphai[37] et al. studied the flow-thermal-structural behavior of leading edges. Figure 11 shows the problem statement for a Navier-Stokes solution and a comparison of experimental and predicted heating rate distributions. The comparative surface heating rates (Fig. 11b) show very good agreement, but the paper points out there was some difference in prediction of the undistributed heating rate q_o that normalizes the curves. Note that the results show the interference heating rate is almost 15 times the undisturbed level. Peak interference heating rates for NASP flight conditions and geometries can be as high as 70,000 Btu/ft^2-s. To put this value in perspective, a typical 1-kW portable hair dryer produces about 30 Btu/ft^2-s; solar heating is 0.12 Btu/ft^2-s. Such high local heating presents a formidable challenge to the structural designer.

A discussion of aerothermodynamics of transatmospheric vehicles from the design perspective appears in the papers of Tauber et al.[38,39]

Aerothermal Load Effects on Flight Structures

In his classic 1956 paper,[6] Professor R. L. Bisplinghoff identified the basic structural and aeroelastic considerations for high-speed flight. From a structural perspective the considerations include: 1) deterioration of mechanical properties at elevated temperatures, 2) thermal stresses introduced by temperature gradients, 3) modification of stiffness and vibration properties, and 4) aeroelastic instabilities. These considerations remain important today. The general problem consists of determining the behavior of the flight structure under aerothermal loads, primarily aerodynamic pressures and heating. The behavior of a flight structure can be classified according to whether the response is quasistatic or dynamic. In a quasistatic reponse inertia forces are negligible, and the structure responds to the aerothermal loads slowly. In a dynamic response inertia forces have a significant role, and the structure responds with vibratory motions. The determination of the quasistatic response of structures has traditionally been called thermal stress analysis. The determination of the structural dynamic response considering the interaction of the deforming structure and the aerodynamic pressure in the presence of aerodynamic heating is called aerothermoelasticity. An area of recent concern is the response of structures to acoustic loads. The determination of the dynamic response of a structure to random fluctuating pressures in the presence of aerodynamic heating can be called aerothermoacoustics.

Quasistatic Interactions

To determine the quasistatic response, a logical approach is to separate the aerothermal-structural problem into distinct uncoupled problems by assuming weak coupling between the external aerodynamic flow and the structural response. This assumption is permissible when structural deformations are too small to alter the external flow. During X-15 flights[9,10] several unexpected thermal problems were encountered due to intense local aerodynamic heating because of quasistatic interactions. Windshield damage occurred when thermal buckling of the retainer frame caused intense local heating in the glass. These problems were identified and solved during flight tests. However, according to Ref. 10, "the really important lesson here is that what are minor and unimportant features of a subsonic or supersonic aircraft must be dealt with as prime design problems in a hypersonic airplane."

More recent examples[40] of quasistatic flow/structural interaction are thermal protection systems tested in the Mach 7 8-ft high-temperature tunnel at the NASA Langley Research Center. The tests show that panels "bowed-up" into the flow to produce heating rates that are up to 1.5 times greater than flat plate predictions. Thornton and Dechaumphai[41] used a finite element approach to study coupled flow, thermal and structural behavior of aerodynamically heated panels. For the Mach 6.6 conditions studied, panel deformations introduce shocks, expansions, and recirculation regions in the flow. The effect of convex panel deformation is to increase local heating rates on windward surfaces and decrease them on leeward surfaces.

In the overview paper[23] describing the design of an airframe structure for NASP, the interaction between the external flow, and the thermoelastic deflections of a movable wing is described. This paper notes that consideration of the interaction resulted in the prediction of lower deflections and stresses which translated into reduced structural weight.

The recent analyses of quasistatic flow/structural interactions illustrate some of the important effects of this behavior on flight structures. However, because of their multidisciplinary nature, the analyses are difficult and expensive. In addition, the analyses have not been validated by experimental data.

Dynamic Interactions

In a 1963 paper[42] I. E. Garrick of NASA Langley surveyed developments in aerothermoelasticity. The classical aeroelastic triangle representing interactions between the fields of aerodynamics, elasticity, and inertia was extended to include thermal effects. A new figure, the aerothermoelastic tetrahedron, illustrated the interdisciplinary aspects of aerothermoelasticity. The paper also discusses aeroelastic consideration of the X-15, effects of transient heating on vibration frequencies and panel flutter. Early in the X-15 flight program the pilot reported a rumbling noise at high dynamic pressures. This turned out to be panel flutter of large areas of the skin of the side fairings and tail. The problem was strong motivation for NASA studies in the 1960s and 1970s of panel flutter at elevated temperatures. A recent assessment[43] of flutter model testing relating to NASP provides an excellent summary of flutter literature for supersonic and hypersonic flight.

The response of aerodynamically heated structures to acoustic loads is an important consideration because of acoustic fatigue. Reference 44 notes that operational experience for a variety of aircraft has demonstrated that intense acoustic pressures can cause fatigue failures of lighweight structures. Preliminary estimates of acoustic loadings for the NASP indicate pressure levels that are well into the range where acoustic fatigue failures have occurred in the past. Thermal effects are important because: 1) there are little acoustic data for high-temperature structures and materials, 2) thermal prestrain and buckling can affect strain levels significantly, and 3) temperature can change material properties and the fatigue lifetime as expressed in the S-N diagram. Recent efforts to develop computational methods for aerothermoacoustics are described in the dissertation by Locke[45] and the paper[46] by Locke and Mei.

Design of Thermal Structures

The design of thermal structures is a complex process that involves consideration of the flight vehicle trajectory, aerothermal loads, thermal structural concepts, and materials. A near-optimum design involves tradeoffs among these and other factors. The design of thermal structures is too complex to be discussed in detail in this paper, but fundamentals will be cited.

Flight Regime

To determine aerothermal loads and heat transfer to the vehicle, the flight trajectory must be determined. The trajectory is determined by mission requirements. The basic equations of flight mechanics, hypersonic aerodynamics and reentry heating appear in the text[47] by Wilbur L. Hankey. Recent design studies[48] for hypervelocity vehicles and for NASP[23] describe relationships between mission requirements and thermal-structural design concepts.

A parameter used in the design of thermal structures is radiation equilibrium temperature. The radiation equilibrium temperature is the upper level that the surface of a structure can reach. An energy balance at the surface of a structure at radiation equilibrium states that the aerodynamic heat flux given by Eq. (2) is equal to the heat flux emitted by radiation, i.e.

$$q_w = h(T_{aw} - T_r) = \sigma\epsilon T_r^4 \tag{13}$$

where T_r is the radiation equilibrium temperature, ϵ is the surface emissivitiy, and σ is the Stefan-Boltzman constant. Eq. (13) states that all of the incident aerodynamic heat flux is emitted by radiation; none of the incident flux is conducted into the structure. Actual surface temperatures may be lower than radiation equilibrium temperatures because of conduction heat transfer into the surface.

Two key factors of the flight regime for thermal structural design are radiation equilibrium temperatures and exposure times. Radiation equilibrium temperatures for the X-15, the Space Shuttle, and the X-30 can be compared in Figs. 3, 6, and 7, respectively. The flight regime for NASP vis-a-vis other flight vehicles is shown in Fig. 12. The important point is that

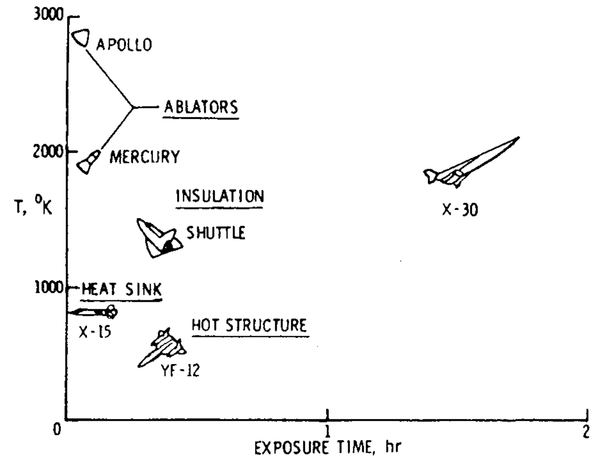

Fig. 12 Flight regimes for hypersonic vehicles.[17]

the design of hypersonic cruise vehicles like NASP differ from previous designs because of much longer flight durations. A third important factor of hypersonic flight is use of liquid hydrogen fuel. Among other considerations, cryogenic hydrogen introduces significant structural temperature gradients. Both maximum temperatures and gradients influence structural design concepts and material selection.

Thermal Protection Systems

Hypersonic flight vehicles require thermal protection systems to withstand sustained aerothermal loads. The thermal protection system, the supporting airframe, and the engine structure are examples of thermal structures. The design of a thermal protection system is based upon the principle that the energy transmitted by the hot boundary-layer flow must be absorbed or rejected by the thermal protection system. Figure 12 identifies thermal protection systems used by hypersonic flight vehicles. For relatively short missions the Mercury and Apollo spacecrafts absorbed the thermal energy through ablative heat shields. The Space Shuttle absorbs reentry heating and thermally insulates the airframe by a very effective, but fragile, tile system. For relatively short missions, the X-15 absorbed the aerodynamic heating by using skin and airframe of high temperature metallic materials. For long duration but lower temperature flights the YF-12 (or SR-71) used a high-temperature titanium structure with high-emissivity surface coating at radiation equilibrium to reject the aerodynamic heating. For sustained, very high temperature flights a combination of several thermal protection concepts is likely to be used. Candidate thermal protection systems for NASP include hot structures, insulated structures, and convectively cooled structures. Convectively cooled airframe structures are relatively new with no previous flight experience. Design and experimental studies[18] conducted over the last 20 years indicate that these systems will be effective in absorbing thermal energy in regions of intense local heating, e.g., shock interference heating on engine structures.

Materials

The evolution of thermal structures for high-speed flight vehicles has placed ever-increasing demands on material performance. Structural designs typically require high-stiffness, thin-gauge materials that can be fabricated into complex, built-up structures. Thermal structural designs typically require high-strength, low-density materials that retain their desirable properties at elevated temperatures. There is a substantial research effort currently underway to develop new aerospace materials to meet these challenges. References 49–56 are recent papers describing progress in material development for thermal structures.

The performance of the NASP will depend on the development of new, lightweight materials that can perform at temperatures higher than today's materials can stand. Today's high-temperature materials such as nickel-based alloys cannot be used because structural weight limits mandate lighter-weight materials. The structural weight limits require the use of thin-gauge, low-density alloys and composites. Figure 13 compares specific strength and stiffness of several advanced materials. Below 2000°F the structural designer has several alternatives, but above 2000°F, choices are limited. The NASP materials and structure augmentation project[22] has the specific goal of advancing the readiness-for-use data of several advanced materials that are needed for the vehicle. The contractor and materials under development in this program are listed in Table 1.

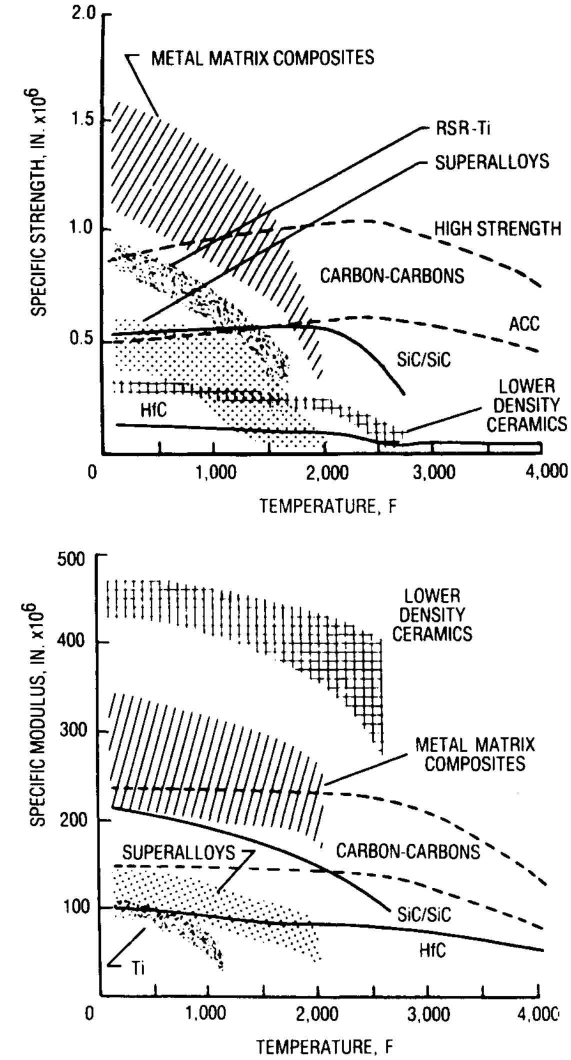

Fig. 13 Specific strength and stiffness of advanced materials.[52]

Table 1 NASP materials and structures augmentation program

Contractor	Material
General Dynamics	Refractory composites Carbon-carbon composites Ceramic composites
McDonnell Douglas	Titanium metal matrix composites
Pratt and Whitney	High creep strength materials Titanium aluminide alloys Titanium aluminide composites
Rockwell North American	Titanium aluminide alloys
Rockwell Rocketdyne	High thermal conductivity composites Copper matrix composites Beryllium alloys

In addition to good thermal-structural performance, a material must perform well in its chemical service environment. The chemical service environment normally encountered in hot structures is an oxidizing environment. The effects of oxidation are addressed by the use of oxidation barrier coatings. However, when the chemical environment includes hydrogen, such as encountered in NASP engine structures, there are significant unresolved problems. Little experience exists on the effects of hydrogen on the thermal-structural behavior of most aerospace structural materials, but there is evidence that indicates exposure to hydrogen at elevated temperature seriously degrades material performance. Hydrogen environment effects on advanced alloys and composites are described by Howard G. Nelson in Ref. 57. The hydrogen compatibility problem is an emerging challenge for the development of the next generation of high-temperature materials.

Heat Transfer and Thermal Stresses

The design of thermal structures requires the determination of temperatures, displacements, stresses, and strains throughout the structures. A structural heat transfer analysis is needed to determine maximum operating temperatures that guide material selection. Temperature distributions are also needed to compute the "thermal-loads" for a structural thermal-stress analysis. The heat transfer and structural analyses rest on the conservation equations of continuum mechanics, constitutive models of material behavior, and computational methods implemented on modern computers.

Conservation Equations

For continuum formulations of solid mechanics, the conservation equations that must be considered are conservation of linear momentum, conservation of angular momentum, and conservation of energy. The conservation of linear momentum produces three equations of motion

$$\frac{\partial \sigma_{ij}}{\partial x_j} + B_i = \rho \frac{\partial^2 u_i}{\partial t^2} \tag{14}$$

where σ_{ij} are components of the stress tensor, B_i denote body force components per unit volume, ρ is the material density, and u_i are the displacement components. Conservation of angular momentum shows that the stress tensor is symmetric. Conservation of energy considers the work done by the stresses, thermal energy transported across surfaces by conduction, thermal, and mechanical energies stored within the material and kinetic energy due to material motion. The conservation of energy equation for a continuum model of a deformable body is

$$\frac{\partial q_i}{\partial x_i} - \sigma_{ij} \frac{\partial \epsilon_{ij}}{\partial t} + \rho \frac{\partial u}{\partial t} = Q \tag{15}$$

where q_i denotes components of heat flux, ϵ_{ij} are components of the strain tensor, u is the internal energy per unit mass, and Q is the rate of internal heat generation per unit volume. The internal energy of the solid depends on the strains and temperature, that is $u = u(\epsilon_{ij}, T)$. The strain components at a point are related to the displacement components by

$$\epsilon_{ij} = \frac{1}{2}\left[\frac{\partial u_i}{\partial x_j} + \frac{\partial u_j}{\partial x_i} + \frac{\partial u_k}{\partial x_i}\frac{\partial u_k}{\partial x_j}\right] \tag{16}$$

Often displacement gradients are assumed small, and the last term in Eq. (16) that involves products and powers of displacement gradients is neglected in comparison to the first two terms. The result is the linear strain-displacement relations

$$\epsilon_{ij} = \frac{1}{2}\left[\frac{\partial u_i}{\partial x_j} + \frac{\partial u_j}{\partial x_i}\right] \tag{17}$$

Equation (16) and (17) show that the strain tensor is symmetric. Thermal-structural problems are often formulated in terms of the linear strain-displacement relations. However, under severe conditions, structures may experience large deformations, and the nonlinear strain-displacement relations may be required. In these circumstances, the definitions of the stress components have to be interpreted relative to the undeformed and deformed configurations of the structure. The conservation equations, in the context of thermal stresses, are derived in the classic text[58] by Boley and Weiner, and in the recent volumes[59,60] on thermal stresses edited by Hetnarski.

Conservation of energy, Eq. (15), states that there is a relationship between stresses, strains and temperature in a deformable body. In one interpretation, conservation of energy indicates that variations of stresses and strains within the solid alter the heat flow and thermal energy. The equation expresses a conversion of mechanical to thermal energy. The relationship between stresses, strains, and temperature in the energy equation is known as thermal-mechanical coupling. The conversion of mechanical energy to thermal energy according to conservation of energy is a well-known phenomenon which has been studied[61] extensively for elastic behavior. Studies of coupled thermoelasticity have shown that for metallic materials within the elastic range, the conversion of mechanical energy to thermal energy may be neglected for aerospace applications. While studies[62–63] of thermal-mechanical coupling for nonlinear, inelastic behavior suggest that coupling may be important under some circumstances, in analyses of flight structures thermal-mechanical coupling is usually neglected. One argument for this assumption is that the external energy supplied to the structure by aerodynamic heating is so large that the thermal energy converted from mechanical energy is negligible in comparison. This means that the energy equation can be simplified by assuming the solid is undeformable, that is $\epsilon_{ij} = 0$, which is the usual approach in heat transfer texts. Under this circumstance the internal energy is regarded as a function of temperature alone, and it is customary to take

$$\frac{\partial u}{\partial t} = c(T)\frac{\partial T}{\partial t}$$

where $c(T)$ is the material's specific heat which is temperature dependent. With these simplifications, the conservation of energy equation reduces to

$$\frac{\partial q_i}{\partial x_i} + \rho c \frac{\partial T}{\partial t} = Q \tag{18}$$

which is the equation customarily found in heat transfer texts. The heat flux components are normally related to the temperature gradients by Fourier's law. For an anistropic material, Fourier's law states

$$q_i = -k_{ij}\frac{\partial T}{\partial x_j} \tag{19}$$

where k_{ij} are components of a thermal conductivity tensor. In general, the material thermal conductivities are temperature dependent.

Structural Heat Transfer

For conduction heat transfer in aerospace structures, the classical heat conduction equation is used. Substituting Eq. (19) into Eq. (18) yields

$$-\frac{\partial}{\partial x_i}\left[k_{ij}\frac{\partial T}{\partial x_j}\right] + \rho c \frac{\partial T}{\partial t} = Q \tag{20}$$

which is a parabolic partial differential equation. This means that thermal disturbances propagate at infinite speeds through the body. To address this anomaly, some authors[61,65,66] have used a modified form of Fourier's law to derive a hyperbolic energy equation. In the hyperbolic energy equation, thermal disturbances propagate at a very high, finite wave speed which is called "second sound." The application of the hyperbolic energy equation to solids is controversial because the phenomena of finite wave speeds of thermal disturbances has never been demonstrated for structural materials, although it has been for gases. The parabolic energy equation is customarily used for structural heat transfer.

The heat conduction equation is solved subject to an initial condition and boundary conditions on all portions of the surface. The initial condition specifies the temperature distribution at time zero. The boundary conditions may consist of specified surface temperature, specified heat flow, convective heat exchange, and radiation heat exchange. These may be written as

$$\begin{aligned} T_s &= T_1(x_1, x_2, x_3, t) && \text{on } S_1 \\ q_i n_i &= -q_s && \text{on } S_2 \\ q_i n_i &= h(T_s - T_e) && \text{on } S_3 \\ q_i n_i &= \sigma \epsilon T_s^4 - \alpha q_r && \text{on } S_4 \end{aligned} \tag{21}$$

where n_i denote components of a unit outward normal, and S_i (i = 1, 4) denote portions of the surface. The specified surface temperature is T_1, and the specified surface heat flux is q_s (positive into the surface). In the convective boundary condition the convective exchange temperature T_e is T_{aw} [see Eq. (2)] for aerodynamic heating. In the radiation boundary condition α is the surface absorptivity, and q_r is the incident radiation heat flux. For structural heat transfer with surfaces at significantly different, elevated temperatures, radiation exchanges between surfaces can occur. The determination of radiation exchanges between surfaces is complicated because: 1) radiation emitted by a typical surface depends on its surface temperature which is unknown, and 2) the geometrical relationsihp affects the exchange. Often, the radiation problem is handled by discretizing the radiation boundary into N discrete surfaces which are assumed isothermal. If a radiation heat flux on the ith surface is called H_i, a set of N simultaneous equations may be developed to determine H_i in terms of the temperatures T_i. In matrix form these equations may be written as

$$[[I] - [F][1 - \epsilon]]\,\{H\} = [F]\{\epsilon\sigma T^4\} \tag{22}$$

where the components of the matrix $[F]$ are the viewfactors F_{ij}. $[I]$ denotes the identity matrix, and F_{ij} is the fraction of the radiation energy leaving surface i that arrives at surface j. The determination of viewfactors for complex three-dimensional structures is a formidable computational task.

An excellent example of heat transfer for a complex structure with aerodynamic heating is the space shuttle wing. William L. Ko and coauthors have documented several studies[67-69] of heat transfer analyses with comparisons to flight-measured temperatures. The effects of internal radiation and internal convection were found to be significant.

For convectively cooled structures the heat transfer in coolant passages must be considered. The dominant mode of heat transfer in the coolant flow is forced convection. An engineering model of flow in the coolant passage is typically used. The engineering formulation is based on assumptions that produce a one-dimensional energy equation with the bulk temperature $T_f(x, t)$ of the coolant as the fundamental unknown. The coolant energy equation takes the form

$$-\frac{\partial}{\partial x}\left(k_f A \frac{\partial T_f}{\partial x}\right) + \dot{m} c_f \frac{\partial T_f}{\partial x} - hp(T_w - T_f) + \rho_f c_f \frac{\partial T_f}{\partial t} = 0 \tag{23}$$

where the subscript f denotes fluid quantities. In the above, A_f is the cross-sectional area of the coolant passage, $\dot{m}$ is the coolant mass flow rate, h is a convective coefficient describing heat exchange between the wall of the coolant passage and the coolant, and p is the coolant passage perimeter. Structural temperatures and coolant temperatures are determined by solving the energy equations, Eqs. (20) and (23), simultaneously. Further details of the formulations and numerical examples are described by Thornton and Wieting.[70-72]

Over the last four decades there have been significant advances in computational methods for structural heat transfer. The use of computers and computer graphics has made analysis of complex thermal structures a routine step in the design process. The proceedings[73] of a conference on computational aspects of heat transfer in structures held at NASA Langley in 1981 describes capability to compute temperatures of flight vehicles. Computer hardware and software have made and continue to make significant advances. Software is based on the finite difference/humped-parameter method or the finite element method. Programs based on the former method include TRASYS, MITAS and SINDA. TRASYS has been used extensively for U.S. spacecraft radiation heat transfer analysis since 1972. There are several widely-available finite element programs with heat transfer capability including ABAQUS, ANSYS, EAL, MARC, MSC/NASTRAN, and PATRAN. Current computer hardware trends include improving computational speeds by vector and/or parallel processing.

Thermal Stresses

Thermal stresses in a structure are determined by solving the equations of motion, Eq. (14) and the strain-displacement relations, Eq. (16) simultaneously with constitutive equations relating the stresses and strains. One of the basic assumptions in thermal stress analysis of flight structures is that the inertia forces in Eq. (14) can be neglected. In most practical applications, the thermal response time of a structure is relatively long in duration compared to characteristic times of the structural response. Under these circumstances, the structure responds to a time-varying thermal load in a quasistatic manner proceeding through a succession of equilibrium positions without oscillations. Thermally induced oscillations occur only when the thermal response time and the structural response time are about the same. Suddenly applied thermal loads of high intensity to thin beams, plates, or shells can induce oscillations,[58] but typical flight structures respond to aerodynamic heating quasistatically. Neglecting inertia forces means that in a quasistatic thermal stress analysis the equations of motion reduce to the equilibrium equations:

$$\frac{\partial \sigma_{ij}}{\partial x_j} + B_i = 0 \tag{24}$$

The equilibrium equations are solved subject to boundary conditions specifying either the displacement components or surface tractions on all external surfaces of the structure. The initial conditions include specifying values of the structure's displacements, and velocities and if the structure has been subject to previous loads, initial values of stresses and strains may be required.

For linear, elastic behavior the stress components are related to the strain components by generalized Hooke's law. For a homogeneous, isotropic material this constitutive relation may be written as

$$\sigma_{ij} = \lambda \delta_{ij} \epsilon_{kk} + 2G\epsilon_{ij} - (3\lambda + 2G)\delta_{ij}\alpha(T - T_0) \tag{25}$$

where δ_{ij} is the Kronecker delta; λ and G are the Lame' constants; α is the coefficient of thermal expansion, and T_0 is the reference temperature for zero thermal stress. The Lame' constants are related to more familiar engineering constants by

$$\lambda = \frac{\nu E}{(1 + \nu)(1 - 2\nu)}$$
$$G = \frac{E}{2(1 + \nu)} \tag{26}$$

where E is the modulus of elasticity, and ν is Poisson's ratio. For small temperature changes the elastic properties are constant, but for large temperature changes the elastic properties are temperature dependent. Thus if the temperature varies throughout the structure, the properties vary from point to point. For small stresses, strains and/or temperature changes, the behavior of structural members is elastic, and the solution of boundary/initial value problems makes up the field of thermoelasticity.[58-61]

One of the most significant developments of the last four decades is the finite element method. With finite elements, virtually any complex structure can be modeled and analyzed with a high degree of accuracy. Literally dozens of finite element books exist, and a vast literature comprising thousands of papers describes finite element methodology. Although the method originated for aircraft structural analysis, finite elements enjoy success in related thermal-structural disciplines including heat transfer and compressible flow analysis. Previous sections have mentioned these applications and given selected references. Finite element thermal stress analysis capability exists in a variety of commercial software including the codes mentioned in the preceding section on structural heat transfer.

Under high rates of loading with the material at elevated temperatures, flight structures will experience inelastic behavior that includes rate-dependent plastic (viscoplastic) deformations. Most metals exhibit viscoplastic behavior at temperatures above 40% of their melting temperature. One of the significant advances in the last 20 years is the development of constitutive models that include both plasticity and creep in a single set of equations called unified constitutive equations. Several investigators have developed unified constitutive models. The models are based on microphysical behavior of materials, are guided by phenomenological considerations, and employ concepts of continuum mechanics. Typically, the equations use the concept of internal state variables to represent the evolution of material behavior. The equations also involve a number of material parameters (some of which are temperature dependent) that must be determined experimentally. Reference 74 contains articles describing several constitutive models.

One of the most well known of the constitutive models was introduced by S. R. Bodner in 1968. A survey article[74] by Bodner describes the evolution of his unified constitutive model that has become known as the Bodner-Parton constitutive model. In contrast to classical elasticity and plasticity, the Bodner-Parton constitutive model assumes that for all load levels there is inelastic strain. In addition, the model does not employ a yield criterion. In applications, the inelastic strain component is small in comparison to the elastic component at low load levels and becomes significant only when inelastic phenomena become prominent. A NASA-Lewis-sponsored research program (HOST) conducted by the Southwest Research Institute recently concluded a 4-year effort[75] to further develop unified constitutive models for isotropic materials and to demonstrate their usefulness for analysis of high-temperature gas turbine engines. One result of this study is material property data for nickel-based alloys over a wide temperature range.

Unified constitutive models have been implemented into finite element analysis by a number of researchers and in the commercially available MARC program. The paper by Thornton et al.[76] contains references to these finite element viscoplastic analyses and describes viscoplastic analysis of hypersonic structures subjected to severe aerodynamic heating. A thermoviscoplastic analysis was performed for a convectively cooled segment of the scramjet engine structure for the NASP. Recent thermal-structural analyses of hydrogen-cooled leading edge concepts for hypersonic flight vehicles are described in Refs. 77 and 78.

Thermal-Structural Testing

Thermal-structural testing remains an important step in the development of structures for high-speed flight. Test facilities exist at several NASA and Air Force installations as well as in private industry, but the principal U.S. government facilities are at the Air Force Flight Dynamics Laboratory[79] and the NASA Dryden Flight Research Facility.[80] Two recent studies related to the NASP are good sources of information about past test programs and current capabilities. Inger P. Friedman et al.,[81] provide an assessment of thermoelastic analysis and testing applicable to NASP; H. A. Hanson and J. J. Casey[82] describe a study to determine high-temperature (1000–3000°F) capability for testing full-scale aerospace vehicle structures. Over the years NASA Dryden has been involved with extensive flight and laboratory test programs for the X-15, the YF-12, the Space Shuttle, and hypersonic structural components.

An excellent review of hot-structures test and analysis technology is the proceedings[83] of a workshop held at NASA Dryden in November 1988. In summary, V. Michael DeAngelis, the workshop coordinator, made the following observations: 1) hot-structure testing is expensive, requiring a sophisticated computer control system, large amounts of instrumentation, and high power, 2) it is time consuming because of the need for extensive instrumentation checkout and calibration, and 3) test procedures are in an early stage of development for new, high-temperature structures. He noted that correlation of test data with analysis is becoming more difficult as Mach number increases because: 1) structures are becoming more complex with new materials and active cooling; 2) computational complexity is increasing with the need for finer models to capture thermal gradients and high local stresses; 3) test requirements are increasing, e.g., more instrumentation is needed; 4) measurement capability decreases with increasing temperature, e.g., accurate strain measurements are particularly critical; and 5) test capability decreases significantly at high temperatures.

A conclusion that may be drawn from reviewing this recent literature is that there is a significant, real need to develop new high-temperature test technology. This technology is needed to assess the performance of new materials and design concepts as well as to validate the analysis tools required to develop thermal structures for high-speed flight.

Concluding Remarks

This paper surveys progress in thermal structures from the early days of supersonic flight to the current research and development for the National Aerospace Plane. Fundamental concepts of aerodynamic heating, aerothermal load effects on flight structures, design of thermal structures, as well as heat transfer and thermal stress analysis and testing are reviewed. Major advances in technology have occurred that provide the foundations for the design of thermal structures for flight vehicles in the 21st century. Much progress has been accomplished, yet there are a number of research needs that must be addressed:

1) The X-15 was the first and, to date, the only manned vehicle capable of flying atmospheric missions at Mach 5 for altitudes of 100,000 or higher. The last X-15 flight was in 1968. The X-15 flights were enormously successful, making many significant contributions to the understanding of hypersonic flight, including the design of thermal structures. Although the Space Shuttle has contributed significantly to the nation's space program, there is a very strong need for

an experimental, hypersonic flight vehicle. To remain the world leader in high-speed flight, the national must not falter in efforts to develop the NASP.

2) Significant advances have occurred in computational fluid dynamics and computer hardware that permit high-quality solutions for the Navier-Stokes equations. Accurate prediction of aerodynamic heating for two-dimensional flows is possible, but the prediction of aerodynamic heating in three-dimensional flows remains a challenge.

3) The importance of interactions between high-speed flows and hot, deforming structures has been recognized, but analyses of coupled flow/thermal-structural interactions is in an early stage of development. Computational studies of interaction effects should continue. There is also a need for experimental data to support computational studies of interaction effects.

4) Substantial progress has been made in the design and development of convectively cooled structures for high-speed flight, but there is a clear need for a flight test program to validate the designs under realistic conditions.

5) The need for new, lightweight materials with well-understood behavior for structural applications above 2000°F is critical. High material costs currently limit experimental studies, particularly in university research programs. Substantial basic research studies of new materials such as metal matrix composites are needed.

6) Thermal-structural analysis capability with the finite element method has reached an advanced stage of development. New developments in constitutive modeling permit analysis of highly nonlinear material behavior. For complex flight structures, analysis capability has exceeded the available experimental data base. There is a need for high-temperature experimental data to validate analysis capability.

7) New high-temperature test technology is needed to support experimental studies of new materials and design concepts at elevated temperatures. Development of methods for accurate measurement of strains at elevated temperatures is a high priority.

Many of these recommendations should and are being pursued to support the development of the NASP. In addition, the nation must maintain a broad-based fundamental research program. Richard R. Heldenfels, looking back in 1982 on NACA thermal-structural research from 1948 to 1958, made the argument for basic research very well: "A healthy research program must provide freedom to explore new ideas that have no obvious applications at the time. These ideas may generate the technology that makes important, unanticipated flight or vehicle opportunities possible."

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