

TACTICAL MISSILE STRUCTURES AND MATERIALS TECHNOLOGY

Significant advances have occurred in missile structures and materials technology since the late 1940's. Some of the advances were the result of improvements in design concepts and material properties; others were from the development of new materials and fabrication techniques. Rapid strides in computer technology and methods of structural analysis have also played an important role. This article discusses the structural design requirements for the next generation of missiles and reviews the technological advances that will make it possible to meet those requirements.

INTRODUCTION

The Applied Physics Laboratory's experience in tactical missile structural design began with the development of the rocket-powered Terrier and the ramjet-powered Talos in the late 1940's. The airframes for those missiles were designed with conventional aircraft materials, using slide rules, mechanical desk calculators, handbooks, and rule-of-thumb procedures. To ensure adequate safety margins, conservative analytical procedures were pursued, followed by extensive environmental testing. Simple aluminum and steel structures operating at temperatures below 300°F for Terrier and 600°F for Talos made up the major portion of the missiles. Terrier was constructed primarily from machined castings. Talos, shown in Fig. 1 with its central air duct configuration, used sheet metal construction. The early Talos production missiles contained no magnesium, but after alloys capable of withstanding aerodynamic heating temperatures to 600°F and suitable fire- and corrosion-preventive coatings were developed, magnesium was introduced into the primary and secondary structural components of Talos. The remaining structure, except for the aluminum tail fins and forward nonstructural covers, was made from sheet steel.

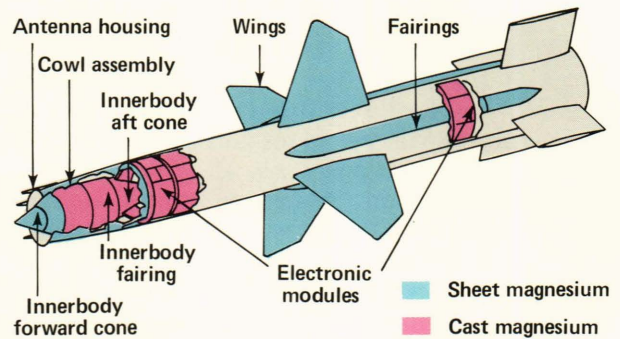


Figure 1 — Magnesium components in Talos represented 13% of the weight of the primary load-carrying structure and 10% of the gross launch weight.

The next major missile airframe development at APL was the Mach 4 Long Range Typhon (Typhon LR) in 1958-1960 (Fig. 2). The airframe was fabricated from superalloys of nickel and cobalt, using lightweight honeycomb and corrugation-stiffened panel construction. As with Talos, the Typhon's external airframe in the region of the combustor was protected from the hot combustion-chamber gases by means of an air-cooled shroud liner system that bypassed a portion of the relatively cool internal duct

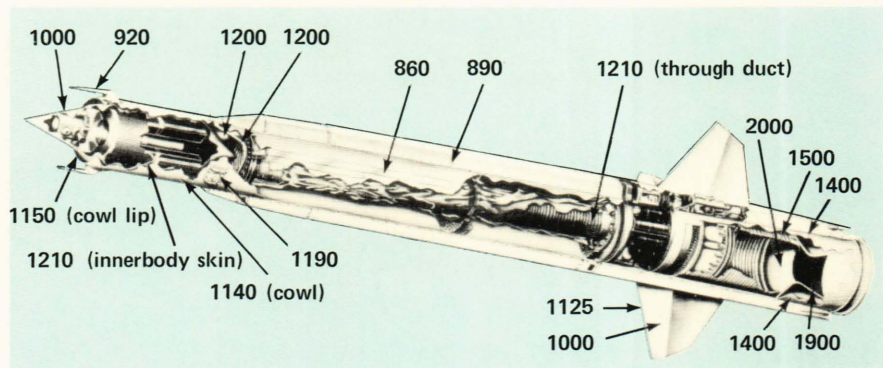


Figure 2 — The maximum temperatures (°F) of the structural components of the Long Range Typhon missile. Temperatures reached 1210°F as a result of aerodynamic heating and 2000°F from ramjet combustion.

air. Also, flame-sprayed zirconium dioxide coatings maintained the combustion-chamber components within acceptable temperature limits. The structural capability of the missile at speeds up to Mach 4.2 and at altitudes of 55,000 to 100,000 feet was demonstrated in successful flight tests at White Sands Proving Ground in 1960-1962. Studies conducted for NASA in 1963¹ indicated that the structure would be capable of operating at Mach 4.5 at high altitudes.

Subsequent advances in ramjet structural design technology have been in exploratory and advanced development programs that have not yet progressed to flight demonstration. However, structural improvements with flight demonstration have been incorporated into the successive upgrades of the rocket-powered Standard Missile. The improvements in the structure of Standard Missile, which provide higher speed, longer range, and higher temperature operating capability, have been achieved primarily by the use of external thermal protection. The Standard Missile-2 ER structural arrangement is illustrated in Fig. 3.

Proposed missile concepts for two classes of air-breathing engines are being examined in exploratory development programs. The first class uses subsonic combustion ramjets with more advanced technology than those of Talos and Typhon LR. The Advanced Surface-to-Air Ramjet (ASAR) shown in Fig. 4 is an example of this class. ASAR has an integral-rocket-ramjet engine in which a single combustion chamber is used for both the solid-rocket boost phase and the liquid-fuel ramjet sustain phase. The second class of engine uses supersonic combustion that extends the operating speed of ramjet missiles to above Mach 5. All of these missile concepts have guidance systems that require forward-viewing radar, infrared sensor windows, or electro-optical sensor windows instead of the interferometer antennas used on Talos and Typhon LR.

DESIGN REQUIREMENTS

Thermal Environment

The maximum speeds of future tactical missiles are expected to move from the supersonic (between Mach 1 and 5) to the hypersonic (Mach > 5) regime. Figure 5 shows the maximum temperatures that will occur in major components of missiles cruising at Mach 5 to 8 at 80,000 feet. Similar data for Mach 4 at

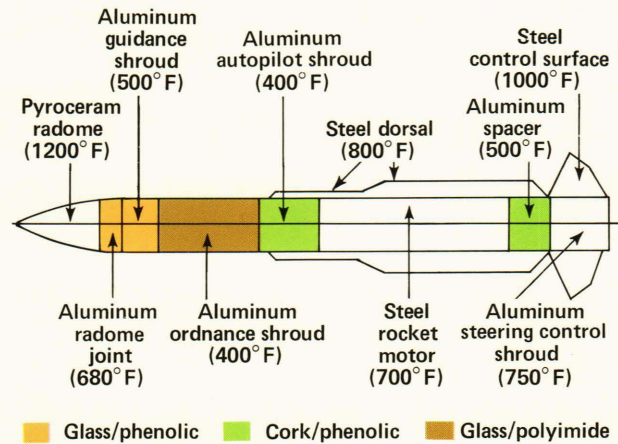


Figure 3 — The structural and thermal protection materials for Standard Missile-2 ER (without booster) and the maximum temperatures to which the structural materials are exposed.

sea level are given in Table 1. The radome, body, and swept leading edge temperatures apply to both rocket and airbreathing missiles, whereas the remaining temperatures are for airbreathing propulsion components.

In the hypersonic regime, a dual combustion ramjet propulsion system (DCR) is more efficient than a subsonic combustion ramjet. The DCR uses a supersonic combustion chamber in tandem with a much smaller subsonic dump combustion chamber that acts as a fuel-rich gas generator and flame holder (see the "Tactical Missile Design Concepts" article in this issue). As shown in Fig. 5 and Table 1, one of the structural advantages of the DCR over the pure subsonic ramjet is that the very high temperatures that

Table 1 — Typical maximum component temperatures (°F) for Mach 4 at sea level.

| | |
|------------------------------------|-----------|
| Radome | 1500 |
| Nose tip and unswept leading edges | 1550 |
| Swept leading edges | 1475-1520 |
| Forebody skin and control surfaces | 1460-1480 |
| Aft body skin | 1275-1300 |
| Subsonic inlet ducts | 1600 |
| Supersonic inlet ducts | 1470 |
| Gas generator | 4600 |
| Combustor | 3660 |
| Nozzle | 3500 |

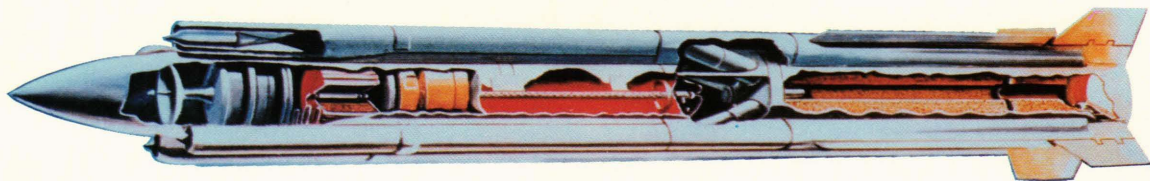


Figure 4 — Typical integral-rocket-ramjet concept. The aft section of the missile serves as both the solid-rocket booster case and the ramjet combustion chamber.

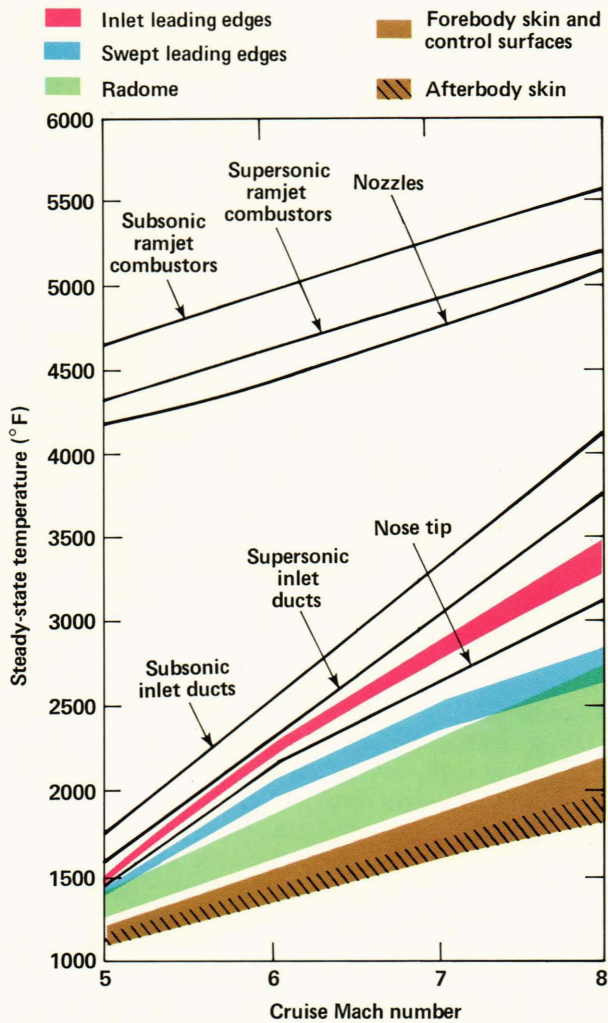


Figure 5 — Typical maximum temperatures at 80,000 feet altitude as a function of cruise Mach number for critical components.

occur in the subsonic inlet ducts and combustor of the DCR are restricted to small sections of the engine. Of even greater importance, at a given free-stream Mach number and altitude, the pressures and heat fluxes to walls in the supersonic inlet ducts and combustor are much lower than in the subsonic inlet ducts and combustor, so that both mechanical and thermal stresses are lower.

Loads Environment

Missile structures must be designed to withstand the loads imposed by shipboard shocks, launch, boost, and sustain flight conditions as well as the thermal stresses and distortions generated by aerodynamic and propulsion system heating. Little change is expected in the shipboard shock environment; however, newly developed vertical launching systems may include shock mitigation systems to alleviate the transmitted shock loads. Boost- and sustain-maneuver loads are likely to increase for advanced self-defense and area-defense missiles, but little change is expected for missiles in other mission categories.

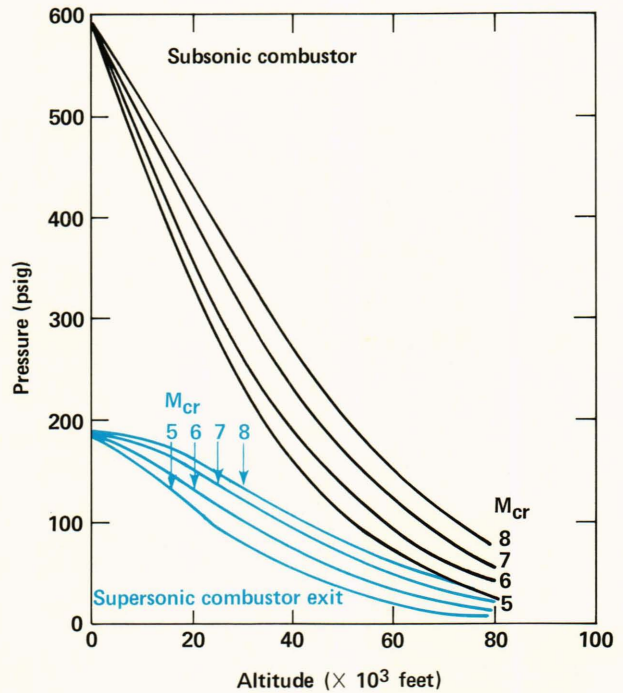


Figure 6 — Bursting pressures in subsonic and supersonic combustion chambers for a fuel equivalence ratio of 1.0 and Mach-altitude boundaries that vary linearly from Mach 4 at sea level to indicated cruise Mach number (M_{cr}) at 80,000 feet.

At a constant Mach number, the internal pressures in airbreathing engines decrease rapidly with increasing altitude. It is therefore common practice to establish Mach-altitude ($M-Z$) design boundaries that permit the Mach number to increase with altitude. Figure 6 shows typical maximum pressures in subsonic and supersonic combustion chambers for nominal $M-Z$ boundaries in which M varies linearly with altitude from Mach 4 at sea level to a prescribed cruise Mach number (M_{cr}) at $Z = 80,000$ feet. Pressures in subsonic and supersonic inlet ducts show similar reductions with altitude.

For the assumed $M-Z$ boundaries, the highest engine temperatures occur during high-altitude cruise; those temperatures dictate the choices of materials and insulation thicknesses for engine components. However, the highest internal pressures occur at sea level, and, depending upon the selected $M-Z$ boundary, they may determine the structural thicknesses of the engine components.

Range Requirements

Flight times for wide-area-defense missiles can exceed 10 minutes even though the average speed is hypersonic. When these flight times are combined with the temperatures shown in Fig. 5, the oxidation of structural materials, the cooling of avionics, and the thermal protection of fuel and missile subsystems become problems. Materials with high-temperature capability frequently have lower strength and higher density than those with lower-temperature capability.

Trade-off studies are required to determine whether structures using low-density, high-strength structural materials that require insulating or ablative materials to protect them are lighter or more volumetrically efficient than those using high-density, low-strength materials that do not require thermal protection.

Launching Requirements

Launching systems impose dimensional constraints on missile designs. For future surface-to-surface, area-defense, and wide-area-defense missiles, the principal launcher is the vertical launching system that is being developed for new ship construction. Unlike existing launching systems, the system is not trainable and is located below decks. The weapon, housed in the shipping container, is stowed in a cell and is launched vertically from its container. The internal dimensions of the container restrict the weapon size to approximately 21 inches in diameter and 256 inches in length. The current weight limit on the weapon is 4100 pounds. Relatively slow or short-range missiles will benefit from the vertical launching system because the designer will be allowed greater dimensional freedom and the installation of a shock mitigation system will be permitted. On the other hand, the system will constrain the size and weight of long-range high-speed missiles. This constraint will place increased importance on the development of minimum-weight structures and minimum-volume thermal protection systems for these weapons.

STRUCTURAL MATERIALS

Structural weight, which is a primary consideration in the selection of materials, is inversely proportional to a material weight efficiency parameter that depends upon the geometry of the structure, the type of loading, and the failure criteria. For pressure vessels such as rocket motor cases, combustion chambers, and inlet ducts with circular cross sections, this parameter is F_{tu}/ω , where F_{tu} is the ultimate tensile strength and ω is the density. The parameter for body skins that fail by buckling due to compressive or bending loads is $E^{1/2}/\omega$, where E is the modulus of elasticity. If the body skin design is dictated by vibration frequency requirements rather than buckling, E/ω is the appropriate parameter.

Homogeneous Metallic Materials

The parameters F_{tu}/ω and $E^{1/2}/\omega$ for several commercially available metal alloys are shown as functions of temperature in Figs. 7 and 8. Further advances in material technology will have the effect of moving the curves up and to the right. However, it is expected that the changes will be modest and will not affect the general trends that are displayed. The magnitudes of the material weight-efficiency parameters drop rapidly as temperature increases. As a result, the unprotected hypersonic missile structures made of commercially available metal alloys will be much heavier than the structures of current supersonic missiles.

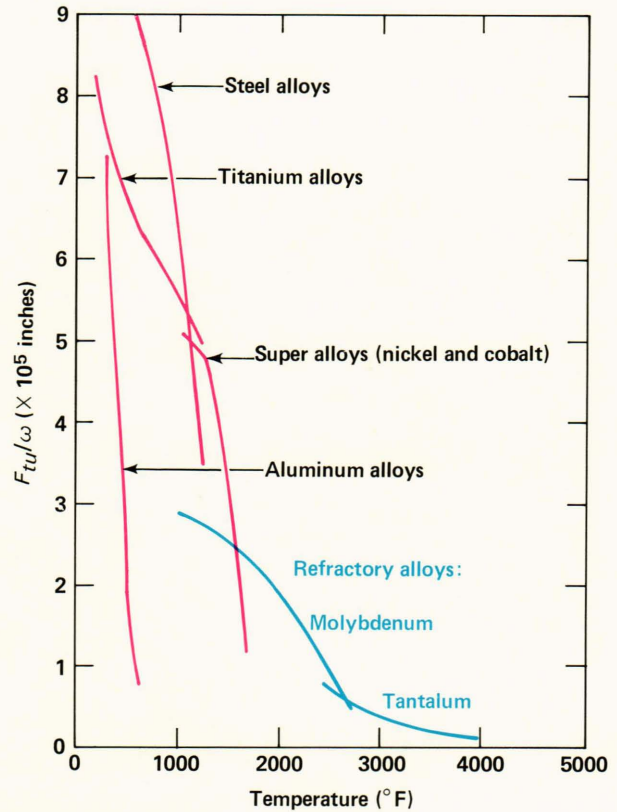


Figure 7 — Ultimate specific strength of commercially available metal alloy classes at elevated temperatures.

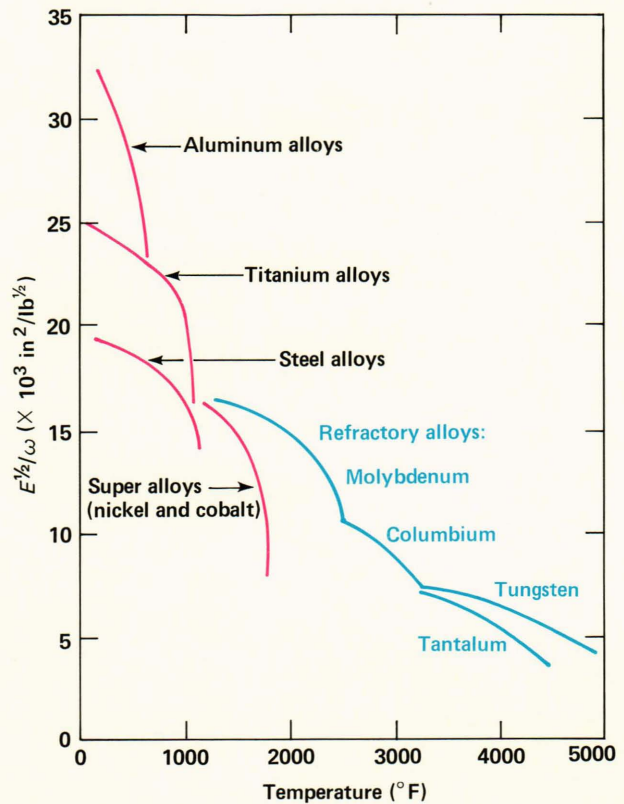


Figure 8 — Buckling efficiency parameter, $E^{1/2}/\omega$, of commercially available metal alloy classes at elevated temperatures.

It is seen from Figs. 5, 7, and 8 that superalloys (alloys of nickel and cobalt) may be used for external skins away from leading edges for speeds up to Mach 7 at 80,000 feet and for inlet ducts for speeds up to approximately Mach 5.5. At higher Mach numbers, refractory alloys (alloys of molybdenum, columbium, tantalum, and tungsten) will be required for those components. Refractory metals will also be required for unswept and swept leading edges at speeds above Mach 5.5 and 6, respectively, at 80,000 feet. The combustion chamber and nozzle temperatures are beyond the limits of metals and require thermal protection or construction from carbon materials.

Composite Materials

The concept of composite materials dates back several thousand years to the use of straw to reinforce dried-mud building blocks. The use of steel to reinforce concrete, or of carbon black and fabrics to reinforce rubber, has been commonplace for nearly a century. Fiberglass-reinforced plastics have had limited use in aircraft since the 1940's, but only now are advanced high-strength composites being used in aircraft structures. In missiles, composites were first used in fiberglass-epoxy radomes of subsonic missiles and filament-wound rocket motor cases of strategic missiles.

The high-strength reinforcing material in a composite can take the form of long filaments, short fibers or whiskers, or a finely divided particulate. Filament materials include glass, boron, graphite, organic fibers such as Kevlar, and silicon carbide (SiC), which is also used as a particulate material. Epoxy resin is the most commonly used matrix material for applications to 400°F. Polyimide and polybenzimidazole matrix resins are being developed for applications to 700 and 850°F, respectively, and it is predicted that future resin systems will have short-time capabilities to 1000°F.

Aluminum, magnesium, and titanium metal matrix materials are also undergoing development. The aluminum and titanium matrix composites will be useful to temperatures of approximately 600 and 1000°F, respectively. At the high end of the temperature scale, carbon-carbon composites consisting of graphite filaments in a pyrolyzed and graphitized pitch or resin matrix have been used in rocket nozzle applications where the temperatures approach 6500°F.

Two-dimensional filamentary composites are made by stacking and bonding together lamina composed of unidirectionally oriented filaments embedded in a matrix material. By selecting the filament and matrix materials, the volume fraction of the filaments, the stacking order of the lamina, and the directions of the filaments in each lamina, the designer can tailor the laminate to provide the necessary strengths and stiffnesses in all directions. The design procedures are complex, and computer programs are required to predict laminate properties. Three-dimensional filamentary composites are made by weav-

ing the filaments and infiltrating the resulting preform with the matrix material.

The ultimate tensile strength and stiffness of unidirectional filamentary lamina are lower than those of the filament; in turn, the ultimate stress and stiffness of the multidirectional laminate are less than those of the lamina. Multidirectional laminates usually are required in practical applications and one cannot directly compare the strengths of monolithic materials with those of filaments or unidirectional composites. A more rational approach, but one that does not take full advantage of the ability to tailor the composite to the loading, is to compare the properties of monolithic materials with those of "pseudoisotropic" composites. In these composite materials, the unidirectional lamina are oriented to provide the same properties in all directions lying in the plane of the laminate.

Figure 9 compares the room temperature values of F_{tu}/ω and E/ω of several unidirectional and pseudoisotropic composite materials with conventional metals. The results show the reason for the current interest in developing the composites to reduce weight in flight-vehicle structures.

The temperature limits on the composite materials exceed those of the matrix materials for short exposure times; e.g., the temperature limit of a graphite-aluminum composite is roughly 100°F greater than that of monolithic aluminum. Unprotected titanium matrix composites will be useful for skins away from leading edges at speeds to Mach 5.5 at 80,000 feet, and carbon-carbon has the ability to

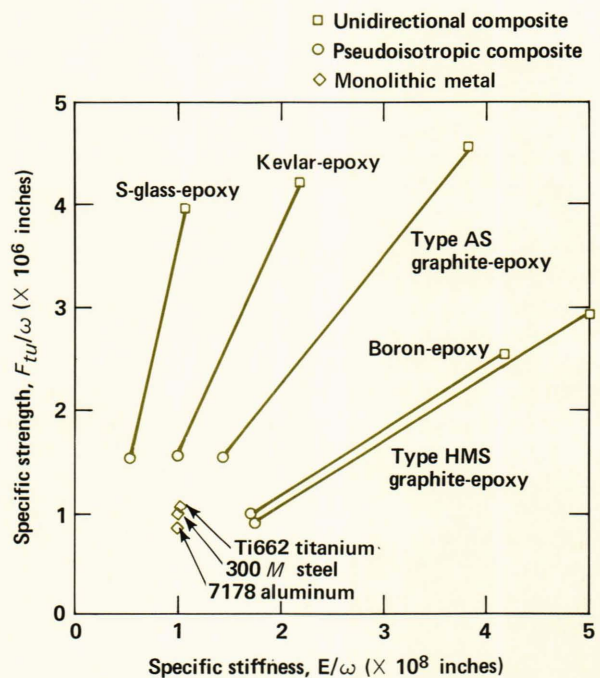


Figure 9 — Comparison of room temperature values of ultimate specific strength, F_{tu}/ω , and specific modulus, E/ω , of unidirectional and pseudoisotropic composite materials with commercially available metal alloys.

operate under the most severe temperatures shown in Fig. 5. However, most composites will require thermal protection for hypersonic missile applications. It remains to be determined whether the composites with thermal protection material will be more weight and volume efficient than the unprotected metal structures.

Filament-wound rocket motor cases of glass-epoxy and Kevlar-epoxy are operational, and graphite-epoxy cases are under development. For higher temperature applications, polyimide is expected to replace epoxy as the matrix material. While rocket nozzles of carbon-carbon are operational, there are extensive developmental efforts to improve the properties and reliability of those materials while reducing costs. Carbon-carbon appears to be a satisfactory material for combustion chambers and nozzles operating in a fuel-rich (reducing) environment, as is usually the case for solid-fuel rockets. For ramjet-powered hypersonic missiles, however, it will be necessary to develop ways to protect the material from oxidation by the hot combustion gases.

THERMAL PROTECTION SYSTEMS

The use of external insulation is generally restricted to the load-carrying structure; it can be either an ablating material or a low-conductivity material with sufficient structural integrity to withstand handling conditions and aerodynamic loads. Most ablators are resin compositions of Teflon (fluorinated ethylene propylene copolymer), silicone, or phenolic. Teflon is generally used in applications where a "clean" ablation is desired, such as over a radar transmission window. It converts from a solid directly to a gas (sublimation) when heated above about 700°F and, in so doing, does not attenuate the radar transmission. A good example of a Teflon ablator is Duroid 5650M. Reinforced silicone and phenolic ablators develop a char layer during the ablation process that provides additional thermal protection. Two charring ablators that have been considered for long-range hypersonic missiles are S-10 (a reinforced silicone resin) and HSHS, a heat-shrinkable heat shield that is a modified silicone rubber. External insulation is primarily used to keep the structure temperature below its allowable limit for strength and stiffness, but it also is used to reduce temperature gradients that could produce undesirable thermal stresses and to maintain a rocket case temperature below the pyrolyzation temperature of internal insulation.

Internal insulation materials must have low thermal conductivity and low density to minimize both weight and volume. Two attractive materials are Min-K and the high-purity quartz insulation Refrasil. These insulations are made in different forms for different temperature ranges. Min-K is a bonded structure reinforced with fibrous media; it contains appreciable quantities of very small particulate matter. Its maximum operating temperature is specified by the manufacturer as 2000°F, but it can be used up to

2600°F for times of less than 10 minutes. In the case of the insulation in a supersonic combustion chamber of a hypersonic ramjet, that temperature is exceeded, and an intermediate layer of zirconium dioxide felt insulation is proposed for short-time operation to 4600°F.

These materials are flexible and must be held in place by a secondary structure, usually made of 3-mil metal or rigidized refractory cloth. Assembly and accessibility requirements govern the design of the support material.

Internal rocket-case insulation must be able to protect the structure from gas temperatures as high as 6000 to 7000°F. For booster or short-range missile applications, the time of exposure may be only 5 to 10 seconds, and the liner material can be quite thin (about 0.1 inch). Ramjet combustors usually operate at lower temperatures, but the combustion gases are more oxidizing than the rocket gases and burn times may last for 5 to 10 minutes. Ramjet combustor insulation varies, depending on the missile speed and whether the missile uses a subsonic or a supersonic combustion chamber. For the Talos missile, whose speed does not exceed Mach 3 and which contains a subsonic combustion chamber, a coating of flame-sprayed zirconium dioxide on the air-cooled liner provided sufficient protection. For the supersonic combustion chamber of the proposed hypersonic dual combustion ramjet engine, flame temperatures up to 5000°F are predicted. Experimental investigations showed that a pyrolytic graphite coating over a carbon-carbon substrate is a good candidate system.

Considerable attention has been given in the past to the development of integral-rocket ramjets whose combustion chamber is used both for the rocket and ramjet (see Fig. 4). A special insulation material has been developed to provide thermal protection of the combustion chamber case during both phases of propulsion. The material (Dow Corning DC 93-104) is a silicone ablator containing a high percentage of silica that forms a low-conductivity char after the material has pyrolyzed. Considerable effort has been expended in developing a thermal model to predict the ablation and temperature response of this material.² An experimental demonstration of this material's thermal protection capability has been performed in both the Air Force's Advanced Strategic Air-Launched Missile ramjet flight test program and the Navy's Advanced Surface-to-Air Ramjet prototyping program. Future missile designs using a rocket-ramjet should be able to use DC 93-104 in the combustion chamber with reasonable confidence.

SENSOR WINDOWS

For radar-guided missiles, the window through which the radar must "look" is a radome; for vehicles guided by infrared radiation, the window is called an infrared (IR) dome. The windows are generally located in the missile's nose to provide a wide forward view, and they are shaped, as much as possible, to minimize drag. The frequency ranges for mi-

crowave (radar), infrared (including laser systems), and electro-optical systems are: 3×10^8 to 3×10^{11} , 3×10^{12} to 3×10^{14} , and 4.3×10^{14} to 7.5×10^{14} hertz, respectively, for which the protecting windows must have transmittances. Thus, materials with low dielectric constants (generally below 10) are used primarily. There are two important differences between radomes and infrared or optical windows: (1) the thickness-to-wavelength ratio for the radome is nearly always less than 1 (usually $\frac{1}{2}$), whereas for an infrared or optical window it can be in the hundreds; and (2) the electromagnetic signal passed by the radome is usually polarized and coherent, whereas the infrared or optical signal is not polarized and is non-coherent unless the energy emanates from a laser. Stated differently, if the radome design problem were scaled to infrared or optical frequencies, the radome wall thickness would be a thin film less than 10^{-6} inch thick, and it would be suitable for transmitting laser-type energy with minimum aberration and maximum efficiency.

Radomes

For the missile technologist, the word “radome” denotes a window of any shape and size that protects a radiating and/or receiving electromagnetic antenna with minimum aberration and maximum efficiency. The first airborne radomes were used in aircraft and were made of $\frac{1}{4}$ -inch plywood, but the problem of moisture absorption in the plywood resulted in a search for better materials. Glass-fiber-reinforced plastic (FRP) is now used for aircraft radomes and low-speed missiles. Some of the more widely used resins in RF radome materials are polyester, epoxy, phenolic, and silicone. Epoxies generally are superior for speeds less than Mach 2 at sea level. Their maximum use temperature is about 400°F. Phenolic resins have somewhat higher temperature resistance, low cost, high mechanical strength properties, and resistance to a large range of environmental conditions, but relatively poor electrical properties. Silicone resins yield good performance when electrical properties are of prime importance and temperatures to about 550°F are encountered.

Some investigations are in progress to determine the feasibility of using an ablative coating over an FRP radome for Mach 3 to 5 applications. Two candidate ablators are Duroid 5650M and Avcoat 8029, both of which provide good rain-erosion resistance and operation over a broad frequency band because of their relatively low dielectric constant.

An FRP radome can be attached to the main body structure by conventional shell structure methods. Joints may be made by using bolts, screws, rivets, latches, clamp rings, adhesive bonding, or straight threading of the radome attachment edge. The greatest departure from convention is found in the special techniques used to reinforce the attachment edge of the radome for the proper transfer of design loads. The reinforcement must be designed to overcome the “weak links” inherent in glass-reinforced

plastics: interlaminar shear, interlaminar tension, and bearing strength.

Ceramic radomes generally are used for missiles that fly faster than Mach 3. Ceramic materials have a higher temperature capability than FRP and high resistance to rain erosion at supersonic velocities. However, most of them are susceptible to thermal shock. Also, because of their brittleness and relatively low coefficient of expansion compared to those of metals, special care must be taken in the design of the radome attachment.

Ceramic radome design must take into account electrical, thermal, and mechanical conditions. Generally, the electrical design is addressed first and establishes the thickness of the wall – usually one-half the wavelength of the radar for which the radome is designed. For 10 gigahertz, this is about 0.20, 0.25, and 0.33 inch for alumina, Pyroceram 9606, and slip-cast fused silica, respectively. Good electrical performance can also be obtained with a thin-wall design whose thickness is about one-twentieth of the wavelength. Thin walls reduce the thermal stresses but are more susceptible to damage from handling and flight through rain.

For a number of years, APL worked on the development of a method to predict the environmental flight performance boundaries for ceramic radomes. That work was combined into a unified radome limitations (URLIM) computer program³ that can predict the flight limits of radome materials on the basis of electrical degradation, surface melting, thermal stress, and mechanical load. By means of this code, limitations were defined for alumina, fused silica, and Pyroceram 9606. The alumina and Pyroceram 9606 were found to be limited by thermal stress, and the fused silica was limited by radar performance degradation. Even though alumina has high strength and a high melting point, its high expansion coefficient causes thermal-stress limitations for nominal accelerations to velocities between 3000 and 4000 feet per second. The thermal-stress performance of several X-band radome materials was computed for a thermally severe trajectory (Fig. 10). The results indicate that Pyroceram 9606 experiences thermal stresses very near its design allowable stress, while the other materials demonstrate a comfortable margin.

To explore further the capabilities of the slip-cast fused silica, two experimental programs were carried out in which Standard Missile-size radomes were subjected to flow conditions simulating a hypersonic environment.^{4,5} Figure 11 is a photograph of a free jet test at the nominal flow conditions of Mach 2.8, total pressure of 200 pounds per square inch, absolute and total temperature of 4000°F. On the basis of an equivalent heat flux, these conditions simulate a Mach 8 flight at 86,000 feet altitude. An X-band radome and its attachment survived a 35 second test that included 10 seconds at 15° angle of attack. The fused silica withstood the hypersonic environment, but more development work is needed on the attachment. The low strength and thermal expansion of

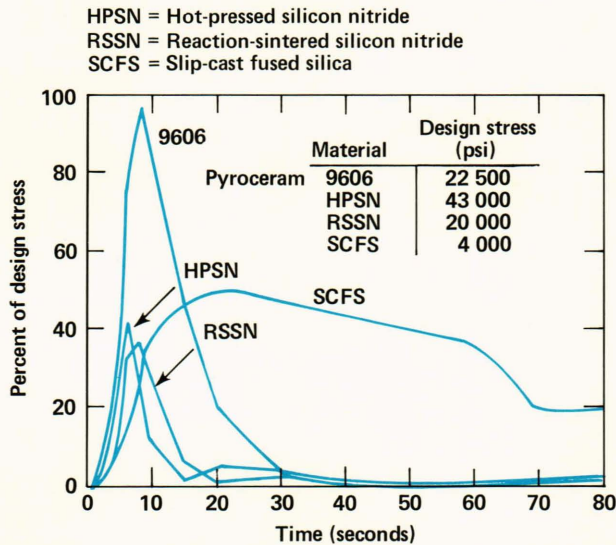


Figure 10 — Thermal stress performance of several radome materials.

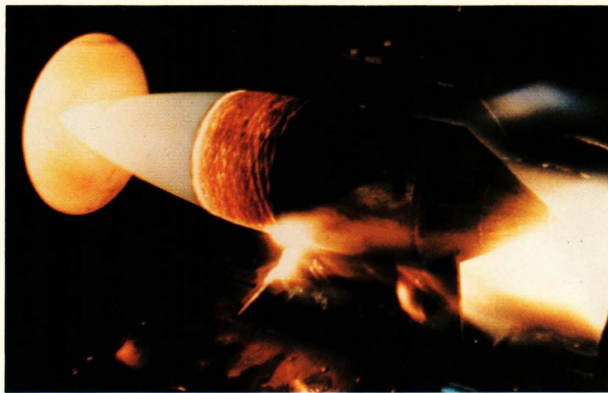


Figure 11 — Fused silica radome in free jet test. Conditions simulate flight at Mach 8 at 65,000 feet altitude.

fused silica combined with the high temperatures experienced in hypersonic flight make the attachment design difficult.

Figure 12 presents dielectric constants of several radome materials as functions of temperature. Of particular interest is the negative slope on ablator materials. Because these materials lose thickness during flight, it would be desirable to have positive slopes. Fused silica's small variation in dielectric constant with temperature is a very attractive characteristic.

Infrared and Electro-Optical Windows

In recent years, infrared and electro-optical guidance systems have become strong competitors of microwave guidance systems for tactical missiles; they offer greater tracking precision, smaller dimensions, and lower weight and cost in addition to passive homing, which is of primary importance in short-range encounters.

The wavelength range for electro-optical systems is the optical bandwidth (0.4 to 0.68 micrometer), while

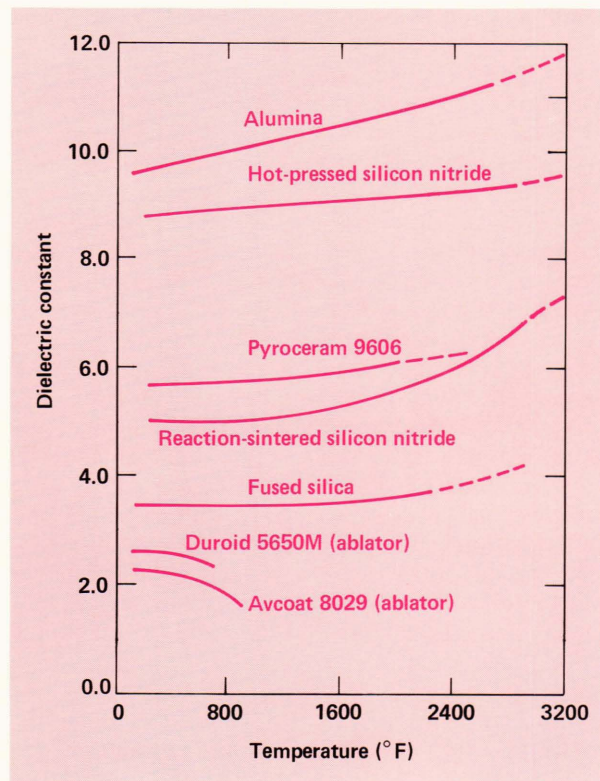


Figure 12 — Dielectric constant of several radome materials as a function of temperature.

infrared (heat-seeking) systems are restricted to regions in which there is good transmission through the atmosphere. The near-infrared windows are 0.75 to 1.3, 1.5 to 1.8, and 2.0 to 2.5 micrometers. The intermediate- and far-infrared windows are 3 to 5 and 8 to 13 micrometers, respectively. If the missile is to be used against targets having particularly hot areas (i.e., a tailpipe, nose, or leading edge), the guidance system can operate in the near-infrared window. If the missile is to have a broader capability in which detection of target plume radiation and hot surface radiation is required, the guidance system would have to operate in the intermediate-infrared window. A more advanced infrared system may use "cold body tracking," which would make use of the far-infrared atmospheric window and require infrared dome materials that transmit in the 8 to 13 micrometer range.

The previous discussion shows that the selection of an infrared dome for a particular missile application is intimately connected with the intended capability of the missile and the type of target for which it is designed. Current systems generally work in the near or intermediate windows. However, there is a trend toward the longer wavelengths (i.e., low-temperature targets), and future missiles may operate in the far-infrared window.

In considering the flight environment, there are three problem areas in infrared or electro-optical dome design: the hot window, rain erosion, and ther-

mal shock. Of the three, the last imposes the most critical requirements on the dome material.⁶

Sapphire, spinel ($MgAl_2O_4$), and IRG-11 ($Ca-Al_2O_3$) are the infrared dome materials that have the best thermal shock capabilities. Of the electro-optical materials, the silicas and VYCOR 7913 are very attractive. For most applications, the sapphire material cannot be considered because of cost, leaving spinel and IRG-11 as the best candidates for infrared applications where thermal shock is a design problem. Spinel is a magnesium aluminate glass having good thermal shock resistance and good infrared transmittance for wavelengths beyond 5 micrometers. IRG-11 is a calcium aluminate glass that has good thermal shock resistance because of its high tensile strength. A disadvantage is its slight solubility in water, which necessitates a protective coating, but it deserves special attention for infrared dome applications. For the near-infrared window, the VYCOR 7913 material may be most attractive since a 2-millimeter thick layer has a transmittance above 90% out to a wavelength of 2.5 micrometers. Thermal shock is not a major problem with most of the electro-optical materials; however, the low thermal expansion combined with the thin wall increases the difficulty of designing a suitable mount for the domes. Special compliant joints must be used to ensure that the dome is not overstressed by differential expansion at the mount.

For supersonic missile applications, the infrared dome temperatures will increase approximately in proportion to the flight velocity. These elevated temperatures generally reduce transmittance of the dome material as well as cause energy to radiate to the seeker, both of which affect the signal-to-noise ratio. Some data on transmissivity and emissivity as a function of temperature have been reported, but additional measurements must be made before a high-speed dome can be designed.

Because of reduced visibility, infrared and electro-optical systems generally are not expected to perform in bad weather conditions, but this does not fully remove the requirement that the domes must be able to withstand flights through rain. Missiles carried aboard aircraft are subject to rain, and a multimode guidance, long-range missile may have to fly through rain before reaching its terminal phase near the target. In the past, nearly all infrared and electro-optical

domes were hemispherical (future domes may take on a lower-drag aerodynamic shape), which presented a vulnerable region where frontal and near-frontal raindrop impacts were possible. Numerous reports (among them Ref. 7) have shown that rain erosion is proportional to the normal velocity component, and considerable damage to hemispherical domes exposed to rain at supersonic speeds might be expected. Efforts are in progress to develop suitable rain-erosion-resistant coatings for infrared and electro-optical domes and to use aerodynamic shapes that reduce the angle of incidence of the rain.

CONCLUDING REMARKS

Future missiles designed to meet the wide-area defense needs of the Navy's surface Fleet may fly at hypersonic speeds for up to 10 minutes. External aerodynamic and internal combustor heating will place stringent demands on structural materials, in many cases necessitating the use of external insulation to protect airframes and internal insulation to protect internal components. Flight environments also impose demanding requirements on guidance systems, necessitating the development of new and improved sensor window materials. New, high-firepower launching systems will impose weight and volume restrictions on missile designs. The incorporation of advanced resin, metal matrix, and carbon composites into missile airframes has the potential of increasing structural efficiency through improved weight and volume control.

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