In this study we consider two configurations of bomblets. The first is a sphere with a total mass of 10 kilograms and a diameter of 20 centimeters. The second is a conical bomblet with a length of 20 centimeters, and also with a mass of about 10 kilograms. We show that heatshield requirements for both of these bomblets on intercontinental-range trajectories can easily be met using materials that were developed over 30 years ago.

We assume the bomblets follow a 10,000 kilometer-range minimum-energy trajectory, and have a speed of 7 km/s and a reentry angle of 24 degrees (with respect to the local horizontal) at an altitude of 150 kilometers as they begin to reenter the atmosphere.\(^1\)

We calculate the trajectory of the bomblet by integrating the equations of motion under the influence of gravity and atmospheric drag.\(^2\) The key parameter governing the behavior of the bomblets as they reenter through the atmosphere is their ballistic coefficient (\(\beta\)), which is given by

\[
\beta = \frac{W}{C_D A} \quad (F-1)
\]

where \(W\) is the weight of the bomblet, \(A\) is the cross-sectional area perpendicular to the direction of motion, and \(C_D\) is its drag coefficient. The higher the value of \(\beta\), the less the object is slowed by air resistance and the faster it falls through the atmosphere. Early reentry bodies were made to have small ballistic coefficients so that they would slow down relatively high in the atmosphere where a smaller fraction of the heat generated during reentry is transferred to the body. As heatshields improved and could withstand higher heating rates, the United States and the Soviet Union increased \(\beta\) by shaping the ballistic missile reentry vehicles as narrow cones. Faster reentry increases the accuracy of a reentry vehicle since it spends less time in the atmosphere being subjected to winds and other forces. Modern warheads have values of \(\beta\) in the range of 100,000–150,000 N/m\(^2\) (2,000–3,000 lb/ft\(^2\)).

The trajectory is then used to calculate the heat transferred to the bomblet by using empirically derived equations for the heat transfer to bodies in hypersonic flow.\(^3\) These equations give the heat absorbed per area per unit time for the stagnation point (the point at the front of the reentry vehicle, where the air flow is brought to rest), for laminar boundary layer flow across a flat plate, and for turbulent flow across a flat plate.

**Stagnation point:**

\[
\left( \frac{dq}{dt} \right)_{SP} = 1.83 \times 10^{-4} \left( \frac{R}{\rho} \right) \left( 1 - \frac{h_w}{h_0} \right) \rho^{0.5} V^3 \quad (F-2)
\]

**Laminar flow:**

\[
\left( \frac{dq}{dt} \right)_{L} = 2.53 \times 10^{-5} \frac{(\cos \phi)^0.5 \sin \phi}{x^{0.5}} \left( 1 - \frac{h_w}{h_0} \right) \rho^{0.5} V^{3.2} \quad (F-3)
\]

**Turbulent flow (for \(V \leq 4\) km/s):**

\[
\left( \frac{dq}{dt} \right)_{T} = 3.89 \times 10^{-4} \frac{(\cos \phi)^{1.78} (\sin \phi)^{1.6}}{x^{0.2}} \left( 1 - \frac{h_w}{h_0} \right) \rho^{0.5} \left( \frac{556}{T_w} \right)^{0.25} \rho^{0.8} V^{3.37} \quad (F-4)
\]

\(^1\) We begin our heating calculation at 150 km altitude since heating is negligible at (and above) this altitude.

\(^2\) The program that calculates the trajectory is described in Lisbeth Gronlund and David Wright “Depressed Trajectory SLBMs,” *Science and Global Security*, Vol. 3 (1992), pp. 101–159.

Turbulent flow (for $V > 4$ km/s):

$$\left( \frac{dq}{dt} \right)_{T>0} = 2.2 \times 10^{-5} \left( \cos \phi \right)^{2.08} \left( \sin \phi \right)^{1.6} \times \left( 1 - 1.11 \frac{h_w}{h_0} \right)^{0.8} V^{3.7}$$  

(F-5)

Here $dq/dt$ is the heat flux (in J/m²s), $\rho$ is the atmospheric density (in kg/m³), $V$ is the speed of the body relative to the air (in m/s), $x$ is the distance along the surface of the body measured from the nose (in m), $\phi$ is the angle between the surface of the body and the freestream airflow, $R$ is the radius of the nose (in m), $T_w$ is the wall temperature (in K), $h_0$ is the stagnation enthalpy per unit mass (in J/kg), and $h_w$ is the surface or “wall” enthalpy per unit mass.

Notice that the stagnation heating rate varies inversely with the square root of the radius of curvature of the nosetip.

The factor in these equations containing the ratio of the wall enthalpy and the stagnation enthalpy can be interpreted as modifying the equations to give the “hot-wall heating rate,” that is, it takes into account the fact that the heat transfer to the body depends on the temperature difference between the surface of the body and the surrounding air. We set this factor equal to zero when it becomes negative, thus ignoring radiation of heat by the body to the air surrounding it.

The stagnation enthalpy is given by

$$h_0 = \frac{V^2}{2} + h_w$$  

(F-6)

where $h_w$ is approximately $2.3 \times 10^4$ J/kg for all altitudes of interest here. The wall enthalpy $h_w$ is taken to be the enthalpy of air evaluated at the wall temperature, and is given approximately by

$$h_w = 1000T_w$$  

(F-7)

(in J/kg) where $T_w$ is taken as the ablation temperature (in Kelvin).

Initially, when a reentering body is at high altitudes, the boundary layer flow of air around the body will be laminar. At lower altitudes, the flow will eventually become turbulent. One commonly hears that the transition from laminar to turbulent boundary layer flow occurs at a Reynolds number of about $5 \times 10^4$. This number is for incompressible flow over a flat plate and assumes lower speeds than those considered here early in the reentry phase. Hypersonic speeds tend to stabilize the flow and the transition can occur at Reynolds numbers several orders of magnitude higher. On the other hand, nose bluntness, surface roughness, and material injected into the boundary layer can lower the transition number. The altitude at which this transition occurs is important since considerably more heat is transferred through a turbulent layer than a laminar layer. For modern reentry vehicles, a typical value for the transition altitude appears to be 20–30 kilometers. We discuss below what assumptions we make about this transition in calculating the heating of the two bomblets.

**Heatshield Calculations**

Once the heating rate on reentry is known and a heatshield material has been chosen, we can estimate two quantities: the thickness of material ablated from the surface of the heatshield and the amount of insulation required to keep the temperature of the inside surface of the heatshield (the “backface temperature”) below some specified level.

The physical processes that take place at the surface of an ablated heatshield are complex and analyzing them in detail is beyond the scope of this study. Therefore, to estimate the amount of heatshield material ablated, we use an approximate technique that involves an “effective heat of ablation,” $q^*$ (in J/kg), which has been empirically determined for a number of materials, and describes the heatshield material’s ability to block heat from entering the body.

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6 Martin, Atmospheric Reentry, p. 114; Katiskas, Ablation Handbook, p. 84.

7 Anderson, Hypersonic and High Temperature Gas Dynamics, p. 274.

8 Gronlund and Wright, “Depressed Trajectory SLBMs,” p. 148.


approximate method is reported typically to estimate the amount of material ablated to within 10 percent or less of the results of a more rigorous analysis that explicitly includes charring of the heatshield material and other physical processes.

For our calculations, we use the thermochemical heat of ablation, defined as

\[ q^* = \frac{dq/dt - (dq/dt)_{rad}}{\rho_{hs}(ds/dt)} \]  \hspace{1cm} (F-8)

where \( dq/dt \) is given by equations (F-2) through (F-5), \( (dq/dt)_{rad} \) is the heat flux radiated from the body, \( \rho_{hs} \) is the density of the heatshield material, and \( ds/dt \) is the recession rate of the heatshield surface. The denominator is simply the rate of ablation of mass per area from the heatshield. This quantity can also be defined as

\[ q^* = C_p(T_w - T_0) + H_v + \eta(h_0 - h_w) \] \hspace{1cm} (F-9)

where the first term is the heat absorbed by the heatshield in raising the temperature to the ablation temperature, the second term is the heat of vaporization of the heatshield material, and the final term is the heat that is blocked from being absorbed by the body. Here \( C_p \) is the specific heat of the heatshield, \( T_0 \) and \( T_w \) are the initial and ablation temperature of the heatshield, and \( \eta \) is called the blowing or transpiration coefficient. This last contribution arises because the emission of ablation products into the boundary layer is found to reduce the amount of heat absorbed by the body compared with what one would expect in the absence of these emissions.\(^{12}\)

Using equation (F-8), the effective heat of ablation can then be used to calculate the rate of mass loss per area due to a heat flux of \( dq/dt \) at the surface:

\[ \frac{dm}{dt} = \frac{dq/dt}{q^*} \]  \hspace{1cm} (F-10)

where \( dm/dt \) is given in units of kg/m². Here we have neglected the radiation term in equation (F-8), which is only a few percent of the other heating term for the conditions we are considering. Neglecting reradiated heat will overestimate the amount of material that is ablated.

The total mass \( \delta m \) ablated per area is then found by integrating \( dm/dt \) over the trajectory, and the total thickness of material ablated is found by dividing \( \delta m \) by the density of the heatshield material.

The insulation requirements are determined using a heat-conduction program that was written for this purpose.\(^{13}\) This program takes as input the surface heating rate as a function of time during reentry, which is calculated from equations (F-2) to (F-5) on the trajectory of the bomblet. The program then calculates the temperature increase of the heatshield surface and the conduction of heat into the heatshield by numerically integrating standard heat conduction equations. (The program assumes spherical symmetry, so that the problem reduces to one dimension.) When the surface temperature becomes sufficiently high, the program calculates the ablation of material at the surface using an effective heat of ablation.\(^{14}\) Since we are only interested in approximate results, the program uses average values of the heatshield material properties over the temperature range of interest, although it could be modified to use material properties as a function of temperature if desired.

**Analysis for the Spherical Bomblet.** The spherical bomblet is taken to be a sphere with a radius of 10 centimeters and a mass of 10 kilograms. Using experimental data for the drag coefficients of spheres,\(^{15}\) one finds that the drag coefficient \( C_D \) has a value of roughly 0.5 for speeds less than Mach 1 (0.3 km/sec) and roughly 0.9 for speeds greater than Mach 1.

Thus, the bomblets will have \( \beta = 3400 \text{ N/m}^2 \) (70 lb/ft²) for high speeds (as they reenter the


\(^{13}\) This program was written by Dr. Jeremy Broughton in June 1999.

\(^{14}\) The program calculates the mass of ablated material slightly differently than the method described above since it only considers ablation to occur when the surface temperature is above an effective ablation temperature, whereas in the method described above the mass ablation rate is calculated over the entire trajectory. Since in this part of the calculation we are only interested in calculating the heat conduction in the body and not the ablated mass, we therefore choose the value of the effective heat of ablation used in the program to give the same amount of ablated material as the method above.

atmosphere) and $\beta = 6200 \text{ N/m}^2$ (130 lb/ft$^2$) for speeds less than Mach 1. For the bomblet and trajectory considered here, this transition occurs when the bomblet reaches an altitude of roughly 12 kilometers (see Figure F-1).

We assume that the bomblet is spinning on reentry, which causes the heat transferred to the body to be averaged over the bomblet’s surface. This averaging reduces the heat loading on any particular part of the heatshield, and simplifies the calculation by making the problem spherically symmetric. One could cause the bomblet to spin by putting sets of asymmetric ridges on the surface of the heatshield to create a torque in the upper atmosphere. These ridges would burn off at lower altitudes, but not until they had already done their job.

With this assumption, we calculate an average local heating rate at a given time by integrating the heating rate over the surface of the sphere and dividing by the surface area. We use the heating equations (F-2) through (F-5) to calculate the heating along the trajectory in two parts: (1) we assume the heating rate is given by the stagnation point formula over an area on the front of the sphere that reaches from $\theta$ equals zero to 20 degrees from the velocity vector, and (2) we apply the flat-plate equations to rings of width $r \times d\theta$ (where $r$ is the sphere radius) and having constant angle with respect to the bomblet velocity, and then integrate the heating over such rings from $\theta$ equals 20 to 90 degrees (see Figure F-2). We assume the boundary-layer flow separates from the body and the heating is zero over the rear hemisphere (for $\theta$ between 90 and 180 degrees).

To be conservative in our analysis, we calculate the heating of the bomblet using the turbulent boundary layer equations for all altitudes. This assumption leads to an overestimate of the heat transfer to the bomblet.

Figure F-3 shows the local heating rates at the stagnation point and at a point 45 degrees around the sphere from the stagnation point, as well as the heating rate averaged over the surface; the latter is used as input to the heat conduction program. (These curves, and those in Figures F-5 and F-6 below, are calculated assuming a wall temperature of 2,700 K, which is appropriate for silica phenolic, as discussed below.)

We calculate that the total heat transferred to the bomblet is roughly $8 \times 10^6 \text{ J}$. This figure is about
3 percent of the total kinetic energy of the bomblet at the start of reentry. The low ballistic coefficient of this bomblet causes it to slow rapidly during reentry. The peak deceleration takes place at 25 kilometers altitude, and the bomblet has a speed of 75 meters per second at ground level (see Figures F-1 and F-4). The time to reach the ground from 150 kilometers altitude is 185 seconds.

By comparison, the Mark 21 reentry vehicle developed by the United States for the Peacekeeper (MX) missile has a ballistic coefficient of about 144,000 N/m² (3000 lb/ft²). On a trajectory having the same reentry speed and angle at 150 kilometers altitude as the bomblet considered here, it would reach the ground in 54 seconds and would impact at 3.4 km/s. Peak deceleration would occur at 6 kilometers altitude—less than 4 seconds before impact. Thus while Figures F-5 and F-6 show that the peak heating rate of the spherical bomblet is 5 to 10 times lower than that of the Mark 21 reentry vehicle, the longer flight time for the bomblet means that the absorbed heat has a much longer time to diffuse toward the interior of the body.

These figures also show the heating rate of a reentry vehicle with a ballistic coefficient of 72,000 N/m² (1500 lb/ft²). This reentry vehicle is assumed to have a nose radius of 5 centimeters and a cone half-angle of 15 degrees, and the transition to turbulent boundary layer flow is assumed to occur at 50 kilometers altitude. This reentry vehicle would reach the ground from 150 km altitude in 58 seconds. Peak deceleration occurs at 10 kilometers altitude, 8 to 9 seconds before impact.

Now that we have calculated the heating rate as a function of time for the trajectory of interest, we can analyze the performance of a heatshield made of a particular material. The first we consider is silica and


16 The Mark 21 heating curves assume the transition to turbulent boundary-layer flow occurs at 30 kilometers altitude.

17 There is considerable information on material properties available in the open literature. See, for example, S.D. Williams and Donald M. Curry, “Thermal Protection
phenolic, also called refrasil phenolic, which is a composite that uses high-purity silica fibers in a cloth that is impregnated with phenolic resin. This material was developed during the 1960s. The values for the material properties we use in our analysis for refrasil phenolic are a density of 1,632 kg/m$^3$, a specific heat of 1,174 J/kg-K, and a thermal conductivity of 0.5 J/m-s-K. We use an ablation temperature of 2,700 K and a wall enthalpy of $3.3 \times 10^6$ J/kg.$^{19}$

We estimate the mass ablated from the surface of the bomblet using the effective heat of ablation, which is shown in Figure F-7 for this material. For the heating rate calculated on the trajectory discussed above, we calculate a total ablated mass of 0.6 kg, or an average depth of ablation of less than 3 millimeters over the surface of the sphere.

We also calculate temperature profiles within the heatshield using the program described above, which numerically integrates the heat conduction equations for the bomblet. The bomblet is assumed to have an initial, uniform temperature of 300 K.$^{20}$ The reentry heating of the bomblet’s surface raises the surface temperature and the heat begins to diffuse inward. When the temperature of the outer surface reaches the ablation temperature, the temperature stops increasing and the heatshield begins to ablate. The program calculates temperature profiles within the heatshield at five times during reentry, with the final one being the time at which the bomblet hits the ground.

Figure F-8 shows the temperature profile at 37 seconds ($t = 0$ is taken to be when the bomblet is at 150 kilometers altitude). The surface is at the ablation temperature and the ablation has caused the outer radius of the bomblet to recede slightly from the original surface at a radius of 10 centimeters. Subsequent profiles are shown in Figure F-9. By 74 seconds, ablation has stopped after the surface has receded about 3 millimeters. The temperature profile at 185 seconds, when the bomblet hits the ground, shows that the temperature would rise only to 350 K at the inner surface of a 2-centimeter-thick heatshield, or to less than 320 K at the inner surface of a 2.5-centimeter heatshield.$^{21}$

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$^{19}$ Katiskas, *Ablation Handbook*, p. 251. Some sources give an ablation temperature of 2,500 K. Using this temperature in the calculations increases the mass ablated by less than 3 percent.

$^{20}$ The temperature of the bomblet could increase or decrease during flight, depending on a number of factors (see Appendix A). Since the heat required to bring the outer layer of the heatshield to the ablation temperature is small compared to the heat dissipated by ablation, this assumption will not affect our results.

$^{21}$ The curves in Figure F-9 were calculated for a heatshield thickness of 4 centimeters. The temperatures noted for the 2- and 2.5-centimeter-thick heatshield ignore the small amount of heat that flows into the interior parts of the heatshield. This approximation only changes the calculated
A shell of this material with an outer radius of 10 centimeters and a thickness of 2 to 2.5 centimeters would have a mass of 3.3 to 4.0 kilograms. In practice, to optimize the design and reduce mass, a thinner shell of ablating material would be used, backed with an insulating layer made of low-density material with low thermal conductivity. In addition, if the bomblet had a metallic shell for structure inside the heatshield, the metal would act as a heat sink and could reduce the amount of heatshield required.

There are also other standard heatshield materials that can be considered. For example, to reduce the mass of the shell, low-density nylon phenolic might be used, since a 2-centimeter-thick shell of this material with an outer radius of 10 centimeters would have a mass of only 1.2 kilograms. Using this material would result in a greater volume of material being ablated than for silica phenolic (the surface recession is about 9 millimeters). However, temperature profiles calculated for nylon phenolic show that at 185 seconds, the inner surface of a heatshield with an original thickness of 2 centimeters would only increase by about 20 K.

**Analysis for the Conical Bomblet.** Another option that we consider is the conical bomblet. The conical shape will result in a higher ballistic coefficient, which will reduce the reentry time and thus the time for heat to diffuse into the bomblet. In addition, this shape can be designed to fall nose first. A contact fuse can then set off an explosive charge that disperses the agent upward out the back of the bomblet.

To calculate the heating and the behavior of the heatshield, we consider the model shown in Figure 7-2 of Chapter 7 and again assume a total mass of 10 kilograms. There is nothing special about this specific configuration—the size and shape could be varied if desired, e.g., to ensure aerodynamic stability during reentry. What we show here is that the heatshield requirements for this type of body can easily be met with simple materials and with a size and mass consistent with the assumed size and mass of the bomblet.

From its shape, one can estimate a ballistic coefficient for this bomblet of about 12,000 N/m² (250 lb/ft²), which we assume is constant throughout reentry.

For this case we calculate the heating rate at two points on the body: at the nosetip (using the stagnation point heating equation (F-2) above), and at a point on the wall 10 centimeters back from the nose. Again, to be conservative in our heating estimates, we use the turbulent heating equations at all altitudes, since these give more severe heating than the laminar equation.

The peak deceleration of this bomblet occurs at 20 kilometers altitude, and the bomblet would have a speed of 150 meters per second at ground level (see Figures F-10 and F-11). The time to reach the ground from 150 kilometers altitude is 115 seconds.

Figure F-12 shows the heating rates during reentry at the two points we consider. We again consider a heat-

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shield made of refrasil phenolic. We find that the heat-shield is ablated a distance of 1.1 centimeters at the nosetip and about 3 millimeters at the location being considered on the wall.

Figure F-13 shows the temperature profile in the heatshield at the nosetip of the cone when it reaches the ground. Recall that the nose is assumed to have a 5-centimeter radius of curvature. These figures show that an original thickness of 2.5 centimeters of heatshield at the nose would keep the backface temperature at less than 330 K at impact.

Figure F-14 shows the temperature profile in the heatshield at impact at the point on the wall 10 cm back from the nose. These figures show that at impact a 2-centimeter-thick heatshield would give only about a 10 K temperature rise at the inner surface of the heatshield, and a 1.5-centimeter-thick heatshield would give about a 60 K rise.

If one considers a 5-centimeter-radius hemisphere of heatshield material at the nose, and 2 centimeters of material on the side walls, the total mass of this heatshield would be about 2 kilograms.

**Figure F-11. Altitude as a function of time for the conical bomblet.**

**Figure F-12. Local heating rates for the conical bomblet at the stagnation point and at a point on the wall 10 cm behind the nose.** (See also Figures F-5 and F-6, which compare these curves to those of two reentry vehicles with high ballistic coefficients).

**Figure F-13. Temperature profile in the heatshield at the nose of the conical bomblet when the bomblet hits the ground.**

*The original surface of the heatshield is at the right side of the graph and the x-axis gives the distance into the heatshield. The figure shows that 1.1 cm of heatshield have ablated from the nose at the stagnation point. The surface temperature is about 600 K but drops to less than 305 K at a distance 3 cm in from the original heatshield surface. The heatshield material is refrasil phenolic.*

**Figure F-14. Temperature profile in the heatshield at a point on the wall of the conical bomblet 10 cm behind the nose, at the time the bomblet would hit the ground.**

*The original surface of the heatshield is at the right side of the graph and the x-axis gives the distance into the heatshield. The figure shows that less than 3 mm of heatshield have ablated and the temperature has dropped to about 310 K at a distance 2 cm in from the original surface. The heatshield material is refrasil phenolic.*
Cooling of the Spherical Bomblet During Midcourse. We show here that the temperature of a bomblet would decrease only slightly during the roughly 30 minutes between its release from the missile and the beginning of atmospheric reentry if it were in darkness during that time. Appendix A notes that a spherical shell in low-earth orbit in the earth’s shadow would have an equilibrium temperature of about 180 K. However, the heat capacity of the bomblets we consider here is large enough that they would remain far from thermal equilibrium after 30 minutes.

We consider the 10-cm-radius spherical bomblet with a 2-cm-thick heatshield of refrasil phenolic and an initial temperature of 300 K. Assuming it radiates like a blackbody, it will radiate 460 W/m² from its full surface area. At the same time, it will absorb infrared radiation from the earth of 240 W/m², which will be absorbed over an effective area of the cross-sectional area of the bomblet. The bomblet assumed here will therefore radiate a net power of about 50 W, or about 9×10⁴ J over 30 minutes.

If we assume the thermal conductivity of the heatshield is very high, so that the entire heatshield is at the same temperature, then the average change in temperature $\Delta T$ of the heatshield due to this loss of heat can be calculated from its heat capacity by

$$\Delta T = \frac{9\times10^4}{c \rho V} = 19 K \quad (F-11)$$

where $c = 1174$ J/kg-K is the specific heat of refrasil phenolic, $\rho = 1632$ kg/m³ is the density, and $V = 0.0025$ m³ is the volume of the heatshield.

Since the actual thermal conductivity of the heatshield is low, there will in reality be a temperature gradient across the heatshield, with the outside surface being cooler than the average and the inside surface being warmer. As a result, the temperature change at the inside surface of the heatshield will be less than the value calculated above.
Appendix G

NASA Air Density Explorer Series
Inflatable Balloon Satellites

In the 1960s the United States launched seven small balloon satellites, the last four of which were successfully put in orbit. These satellites, in the Explorer series, were developed during 1956 and 1957 at NASA’s Langley Research Center. They were used to make measurements of the Earth’s atmospheric density by measuring the effect of atmospheric drag on the balloon’s orbit. The first three attempts to put such balloons into orbit, beginning in October 1958, failed due to malfunctions of their rocket boosters. However, the subsequent four balloon launches were all successful. The first balloon successfully put into orbit was Explorer 9, which was launched on 16 February 1961, and the last was Explorer 39, launched in August 1968.

The balloons had a diameter of 3.7 meters. They were constructed of a laminate made from commercially obtained mylar plastic and aluminum foil. Two different laminate compositions were used. The first three balloons used a three-layer laminate, consisting of two 0.00045-inch-thick aluminum foil sheets bonded to a center layer of 0.001-inch-thick mylar. The last four balloons used a four-layer laminate consisting of alternating layers of 0.0005-inch-thick aluminum and mylar, with the additional inner layer of mylar added to control the temperature of a radio beacon placed inside the balloon.

The materials used to make the laminate were bought commercially in rolls. The laminate was cut into 40 flat gores (a gore is a piece of material that is wider in the middle than at the ends). These pieces were then fabricated into a beach-ball type structure by assembling them over a 3.7-meter-diameter hemisphere. The gores were bonded together using a 3/8-inch overlap between the gores. Two different commercially obtained adhesives were used in bonding the gores together (Goodyear Pliobond and GT-301, a thermosetting plastic made by the G.T. Schjeldahl company).

Starting with the fourth balloon (Explorer 9), a center strip composed of a single layer of mylar was used to divide the balloon into two electrically isolated hemispheres, so the balloon could be used as the antenna for a tracking beacon to be carried inside the balloon (in the case of Explorer 9, this tracking beacon failed shortly after deployment).

Each of the balloons weighed about 10 pounds (4.5 kg), although the deployed weight was greater because of the weight of the radio tracking beacon and its associated batteries and solar cell panels. The deployed weight of Explorer 9 was 6.7 kg.

The radio beacon, which was inside the balloon, had to be kept below a temperature of 333 K (60°C). To reduce the average temperature of the satellite when in sunlight, 3,600 white circles, typically with a diameter of 5.1 cm, were painted on its outer aluminum surface (covering about 17 percent of the surface on Explorer 9 and 25 percent on Explorer 19). A fourth

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5 On Explorer 9, near the location of the beacon, the size and spacing of the white dots was reduced to obtain a more uniform temperature distribution. In addition, an area of about 160 cm² directly over the location of the beacon was painted solid white. See Charles V. Woerner and Gerald M.
inner layer of mylar was also added starting with the fourth balloon. The higher emissivity of mylar relative to aluminum resulted in greater heat transfer due to radiation on the inside of the balloon and thus moderated hot and cold spots on the balloon. These two steps reduced the predicted temperature of the hottest spot on the balloon from about 423 K to 330 K and the temperature difference between the hottest and coldest spots on the balloon from about 116 K to about 53 K.6

In order for the balloon to be useful in making drag measurements, it was essential that it retain its spherical shape. Ground tests showed that in order to obtain the strongest sphere it was necessary during inflation to stress the aluminum foil beyond its yield point, which required an inflation pressure of about 0.1 pounds per square inch. With this initial inflation pressure (the gas was allowed to vent out after inflation), ground tests indicated that the balloon should retain its spherical shape down to an altitude of about 75 miles (121 km).7

The balloon was folded into a cylindrical package with a diameter of 21.6 cm and a length of 28.0 cm. The greatest difficulty in doing this was removing the air from the folds during the folding process. When deployed in space, it was inflated to a pressure of 0.1 pounds per square inch using a small bottle of nitrogen gas. The deployed balloons were spinning at a rate variously reported to be either 30 or 220 revolutions per minute. Prior to deployment, many tests were performed in a vacuum chamber to ensure the balloon deployment and inflation would work reliably.

The balloons were not designed to hold gas for any period of time, and it was calculated that the gas pressure would fall to 10^{-4} mm of mercury at about 18 hours after deployment (at this pressure it was estimated that the heat transfer via the gas would become negligible).8

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In this appendix we discuss how the presence of a warhead inside a balloon would affect the thermal behavior of the balloon.

If the warhead and balloon are at different temperatures, the warhead will transfer heat to (or from) the balloon in several ways:

- radiation
- conduction through any spacers used to position the warhead within the balloon
- conduction through the gas in the balloon
- motion-driven convection of the gas

Below we discuss the different means of heat transfer, then calculate the effect of the warhead on the balloon’s thermal behavior.

**Radiation**

Since we are only making rough estimates, we will model the warhead as a sphere with a surface area of 4 square meters and a diameter of 1.12 meters. We assume the warhead is concentric with the spherical balloon, which has a diameter of 3 meters.

The power transferred by radiation, $P_R$, to (or from) the balloon by the warhead is given by

\[
P_R = \frac{A_W \sigma (T_W^4 - T_B^4)}{1 + \frac{A_W}{A_B} \left[ \frac{1}{\epsilon_W} - 1 \right]} \quad (H-1)
\]

where $A_W$ and $A_B$ are the surface areas of the warhead and balloon, $\epsilon_W$ and $\epsilon_B$ are the infrared emissivities of the warhead’s surface and the inside surface of the balloon, $T_W$ and $T_B$ are the temperatures of the warhead and balloon, and $\sigma$ is the Stefan-Boltzmann constant ($5.67 \times 10^{-12}$ W/cm$^2$K$^4$).

If we assume that the inside surface of the balloon is a blackbody ($\epsilon_B = 1$),\(^2\) this equation simplifies to

\[
P_R = A_W \epsilon_W \sigma (T_W^4 - T_B^4) \quad (H-2)
\]

We assume that the outer surface of the warhead has been given a low emissivity finish (or covered with a thin layer of superinsulation) to reduce the heat transfer. (See Table A-1 in Appendix A for a list of emissivities for different materials.) If we take $\epsilon_W = 0.036$ (corresponding to shiny aluminum) and consider a case in which there is a temperature difference of 10 K between the warhead and balloon ($T_B = 300$ K and $T_W = 310$ K), then using equation (H-2) we find that

\[
P_R = 9.3 \text{ watts} \quad (H-3)
\]

\(^1\) This assumes that the surfaces of the warhead and balloon are diffuse scatterers. If they were specular reflectors instead, the heat transfer would be somewhat less—the denominator in this equation would be replaced by $(1/\epsilon_W + 1/\epsilon_B - 1)$. See Robert Siegel and John R. Howell, *Thermal Radiation Heat Transfer*, 2nd ed. (Washington, D.C.: McGraw Hill, 1981), chapters 8 and 9.

\(^2\) By making this assumption, we will overestimate the power transferred by radiation. However, since the surface area of the balloon is much larger than that of the warhead, the effect of this assumption is small, unless the emissivity of the inside of the balloon is small. For example, if the emissivity of mylar ($\epsilon = 0.5$, see Table A-1 in Appendix A) was used instead of assuming the inside of the balloon was a blackbody, the power transferred would only be decreased by about 12 percent.
Conduction Through Spacers

The power transfer due to spacers, $P_S$, can be made negligible by using spacers with low thermal conductivity. The power transfer through a spacer of cross-sectional area $A$ and length $L$ and with thermal conductivity $\kappa$ is given by

$$P_S = \frac{\kappa A(T_W - T_B)}{L} \quad (H-4)$$

The warhead might be positioned within the balloon using a set of strings that would have very low thermal conductivities. However, to demonstrate that the thermal conductivity can easily be made negligible, we will consider here a set of relatively short and thick spacers: 10 spacers each with a cross-sectional area of 1 cm$^2$ and a length of 10 cm. Assuming these spacers are made of commonly available low-conductivity materials such as phenolic or polystyrene, they would have a thermal conductivity of about $\kappa = 0.03$ W/m-K. With the same 10 K temperature difference between the warhead and balloon used above, we find from equation (H-4) that

$$P_S = 0.03 \text{ watts} \quad (H-5)$$

which is a factor of 300 less than that due to radiation. So the heat transfer through spacers can be neglected.

Conduction Through the Gas Used to Inflate the Balloon

What about heat transfer due to conduction through the gas used to inflate the balloon? We assume that the gas used is nitrogen and that a pressure of 69 Pa (0.01 pounds per square inch, or $7 \times 10^{-4}$ atmospheres) is used to inflate the balloon. As above, we model the warhead as a sphere with a surface area of 4 square meters (and diameter of 1.12 meters), and assume it is concentric with the balloon with a diameter of 3 meters.

The heat transfer between two concentric spheres is given by

$$P_G = \frac{4\pi\kappa(T_2 - T_1)}{\left(\frac{1}{R_1} - \frac{1}{R_2}\right)} \quad (H-6)$$

where $R_1$ and $R_2$ are the radii of the spheres, $T_1$ and $T_2$ are the temperatures of the spheres, and $\kappa$ is the thermal conductivity of the medium between them. The thermal conductivity of nitrogen (at 300 K and one atmosphere) is 0.0258 W/m-K. Since the thermal conductivity of a gas is essentially independent of pressure until the mean free path of the gas molecules becomes equal to the size of the enclosure, this will also be the value of $\kappa$ at a pressure of 69 Pa ($7 \times 10^{-4}$ atmospheres). Assuming the same 10 K temperature difference, equation (H-6) yields

$$P_G = 2.9 \text{ watts} \quad (H-7)$$

Thus, the power conducted through the gas, while smaller than that due to radiation, is not entirely negligible.

However, the attacker could eliminate this means of heat transfer by venting the gas after the balloon was inflated. How rapidly could this be done? The simplest way would be to open a hole in the balloon skin and let the gas vent directly to space. In this situation, a hole with an area of one square centimeter will act as a pump with a speed of 11,700 cm$^3$/sec. The gas pressure in the balloon, $P$, will then be given by

$$P = P_0 e^{-S t/V} \quad (H-8)$$

where $P_0$ is the initial pressure, $S$ is the pump speed, $V$ is the volume of the balloon, and $t$ is the time since the hole was opened. If we use a pair of circular holes 7.6 cm (3 inches) in diameter and assume an initial pressure of 69 Pa, we get

$$P = 69 e^{-0.0754t} \text{ Pa} \quad (H-9)$$

As noted above, the thermal conductivity will not decrease until the mean free path of the gas molecules becomes equal to the size of the enclosure. For our balloon, with a diameter of 3 meters, this transition will occur at a pressure of about 0.0027 Pa. From equation (H-9), we find that reaching this pressure would take about 135 seconds, after which the thermal conductivity of the gas would decrease rapidly. If we assume the conductivity decreases in direct proportion to the pressure after the mean free path exceeds the balloon size, then the conductivity would be reduced by a factor of more than 1,000 four minutes after venting begins.\(^7\)

---


\(^5\) Strong, Procedures in Experimental Physics, p. 97.

\(^6\) A pair of oppositely placed holes is used here to prevent the escaping gas from propelling the balloon.

\(^7\) As shown in Appendix A and also later in this appendix, empty lightweight balloons will generally reach thermal
Convection of the Gas
When a gas heats up, it expands and becomes less dense. In the earth’s gravitational field, the heated, less dense air then rises, resulting in convection. However, since the balloon is in free fall in space, there will be no convection driven by temperature difference. However, it is likely that there will be at least some gas motion. In particular, rotation of either the balloon or the warhead would generate gas motion that could be a significant source of heat transfer.

To minimize the heat transfer by gas convection, the balloon and warhead could be attached to each other (by spacers or sets of strings, for example) to minimize the relative motion between them, and the balloon and warhead could be deployed so that they rotate or tumble only slowly. In such a rotating environment, heated gas would tend to flow inward if the warhead at the center of the balloon is colder than the balloon, because of the equivalence of centrifugal force and gravity. But gas heated by a warm warhead at the center would tend to remain near the warhead, so heat transfer by gas convection would not be significant if the warhead were warmer than the balloon.

Alternatively, as discussed above, the gas could be vented after it was used to inflate the balloon. Thus, the attacker has the option of eliminating the heat transfer via both conduction through the gas and convection of the gas by rapidly venting the gas out of the balloon.

Effect of Warhead on Balloon Temperature
The above discussion indicates that, if the gas is vented out of the balloon, radiation will be the dominant mechanism for transferring heat from the warhead to the balloon. If the gas is retained, thermal conduction through the inflating gas will have an effect, but one smaller than radiation. The effects of convection are more difficult to estimate numerically, but can be minimized by fixing the warhead relative to the balloon, keeping the balloon temperature below that of the warhead, not spinning the balloon, or by venting out the gas.

To illustrate the effect of the warhead on the balloon temperature, we consider two cases. In one case we assume that the total power transferred is equal to that due to radiation, as would be the case if the gas is vented. In the second case, we assume that motion-driven gas convection is the primary heat transfer mechanism: for this case we take the heat transfer to be five times that due to radiation alone. The key point here is that the precise size of the convective heat transfer does not matter: as can be seen from the discussion below, even large variations in the heat transfer from the warhead to the balloon produce results that are qualitatively similar and that are easily hidden from the defense’s sensors.

Figure H-1 compares the thermal behavior following deployment of three lightweight balloons with shiny aluminum foil outer surfaces, masses of 0.5 kg, and initial temperatures of 300 K. The balloons are assumed to be in daylight and tumbling so that all of their surfaces are equally exposed to the sun. One balloon is empty and quickly reaches the expected equilibrium temperature of 454 K. The second balloon contains a warhead with an emissivity of 0.036 and an initial temperature of 300 K, and the heat transfer is assumed to be only due to radiation. As the figure shows, ten minutes after deployment, the temperature of this balloon is reduced by about 11 K, from 454 K to 443 K, relative to the empty balloon. The third balloon also contains a warhead with an emissivity of 0.036; however, here the heat transfer is taken to be five times that due to radiation. After ten minutes, this balloon has a temperature of 410 K.

If the defense knew that the equilibrium temperature of the empty balloon should be 454 K, then these temperature differences could be used to deduce the presence of the warhead. However, the defense will not know the precise surface composition of each of the balloons, so the attacker can easily deny the defense the ability to identify the balloon containing the warhead by using balloons designed to equilibrate over a range of temperatures. As is also shown in Figure H-1, empty aluminum balloons with a small percentage (1–4 percent) of their surface covered with white paint will have equilibrium temperatures varying

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8 In this calculation, the warhead is assumed to have a heat capacity equal to 900 kg of aluminum. We neglect any heating due to the fissile material in the warhead here, but discuss it later in this appendix.
from 390 K to 454 K. Together with the empty balloon with no paint, these temperatures span the temperatures of the balloons containing the warheads. Thus, unless the defense has essentially complete information about the surface composition of the balloons, it cannot determine which balloon contains the warhead based on measurements of the temperature of the balloons.⁹

In fact, while the five empty balloons in Figure H-1 are in thermal equilibrium (subject to the approximation that their surface temperature is uniform), the balloons with warheads are not. This is because the temperatures of the much heavier warheads inside these balloons are still changing, albeit slowly, and therefore the surface temperatures of these balloons are slowly drifting upward. However, this drift is small, about 0.01 K/minute for the balloon with heat transfer equal to five times the radiation heat transfer, and less for the other. As discussed in Chapter 8, in the real world, where the temperature of the balloon will not be uniform over its surface, the complex and changing pattern of temperature variations over the surface of the balloon will easily obscure this small drift.

Moreover, the effect of the warhead on the balloon can be made almost negligible by choosing a balloon equilibrium temperature close to that of the initial warhead temperature. Figure H-2 illustrates such an approach. It shows the temperature variation after deployment for balloons coated with a thin layer of aluminum silicone paint, which gives an equilibrium temperature of 299.3 K.

As the figure shows, adding a warhead (at 300 K and with an emissivity of 0.036), increases the balloon temperature (after ten minutes) by only about 0.01 K. For a balloon containing a warhead where the heat transfer is taken to be five times that due to radiation, the temperature after ten minutes is increased only by about 0.5 K. For both of these balloons, the temperature drift due to the presence of the warhead is negligible: less than 0.00001 K/minute.

As in the previous example, the presence of the warhead can easily be masked by using small amounts of paint. Figure H-2 also shows the thermal behavior of two empty balloons, one with 0.5 percent of its surface covered with black paint and the other with 0.1 percent of its surface covered with white enamel paint. The figure shows, even these very small amounts of paint produce thermal variations that would easily mask the presence of a warhead.

**Heating Due to the Fissile Material in the Warhead**

The discussion up to this point has neglected the heat produced by nuclear reactions taking place in the warhead’s fissile material, which we assume to be plutonium. The thermal power produced by nuclear reactions is 2.5 W/kg for weapon-grade plutonium.¹⁰ The specific heat of plutonium at 300 K is 142 J/kg-K. Thus, if thermally isolated from its environment, weapon-

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⁹ In actual practice, the balloons with the warheads would also be likely to have some amount of paint on their surface, with the amount of paint on the empty balloons adjusted accordingly.

grade plutonium would heat at a rate of 0.018 K/s = 1.1 K/minute = 33 K over a 30-minute ICBM flight. Of course the fissile material in the core of a nuclear weapon is not thermally isolated; there will be some thermal coupling to the rest of the nuclear weapon and to the outside world. The warhead designer must make sure that this coupling is sufficient to keep the core of the weapon from overheating. Note that this issue must be resolved for any nuclear warhead, not just one to be put inside a balloon decoy.

If we assume that a nuclear weapon deployed by an emerging missile state contains 6 kg of weapon-grade plutonium, then the total thermal power produced by nuclear reactions will be 15 W. If we assume the warhead as a whole has a mass of 900 kg with an average specific heat of 0.9 kJ/kg·K, then the total heat capacity of the warhead will be $8.1 \times 10^5$ J/K. If we further assume the warhead heats uniformly (which gives the greatest temperature increase at the surface of the warhead, and hence the greatest effect on the balloon), then for weapon-grade plutonium, the warhead would heat at a rate of $1.9 \times 10^{-5}$ K/s = 0.0011 K/minute = 0.033 K over a 30-minute ballistic missile flight.

How would this nuclear heating affect the above results? To estimate this, we assume that the entire warhead heats at a rate of $1.9 \times 10^{-5}$ K/s.

Recall that an empty aluminum-coated balloon has an equilibrium temperature of 454 K, that adding a warhead resulted in a temperature 10 minutes after deployment of 443 K (assuming the heat transfer is due to radiation), and that this temperature was drifting upward at a rate of less than 0.01 K/minute. Adding in the heating due to nuclear reactions increases the temperature at ten minutes after deployment by about 0.0006 K and leaves the temperature drift rate essentially unchanged.

For balloons coated with aluminum-silicone paint, the empty balloon had an equilibrium temperature of 299.3 K. The balloon with the warhead (assuming the heat transfer is due to radiation) was at a temperature ten minutes after deployment that was 0.01 K higher, with a negligible upward temperature drift. Adding the

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11 The heat generated by reactor-grade plutonium is 5.6 times higher. However, as far as is known, all stockpiled plutonium nuclear weapons (including those of India and Israel) have used plutonium that is weapon-grade or better. The primary problem with reactor-grade plutonium is an increased probability of premature initiation of the detonation because of the increased fraction of Pu-240 in reactor-grade material. This problem can be solved, but it is more difficult to deal with than the greater heating associated with reactor-grade plutonium. A reasonable estimate is that a nuclear weapon built by a country such as North Korea would contain about 6 kg of weapon-grade plutonium. (Marvin Miller, personal communication, December 1999.)

effect of nuclear heating increases the temperature at 10 minutes after deployment by 0.0001 K, with a still negligible upward temperature drift of 0.0002 K/minute.

Thus, it is clear that adding the effects of nuclear reaction heating does not in any way change the conclusion that the thermal presence of the warhead can be easily masked.
Appendix I

Shroud Cooling Requirements

There are several external sources of heat that will cause the shroud to heat up once it has been cooled to a temperature of 77 K, unless cooling continues. Above the atmosphere, the sun will deliver about 1,360 W/m² of radiation at visible and near infrared wavelengths. In addition, about 30 percent of the solar radiation that hits the Earth’s surface, or 410 W/m², will be reflected upward onto the warhead from the Earth below (this is known as the albedo flux). The radiation flux from the Earth is about 240 W/m² in the long wavelength infrared (at a wavelength of about 10 microns). Finally, the nuclear warhead itself radiates infrared radiation, and some of this will penetrate the insulation and heat the shroud.

We now show that maintaining the temperature of the shroud against the various heat inputs will require the evaporation of about 200 grams of liquid nitrogen per minute.

The thermal power absorbed by the outer wall of the shroud, $P_{\text{outer}}$, from sunlight, Earth-reflected sunlight, and infrared Earthshine will be

$$P_{\text{outer}} = \left(\alpha_S (S + S_R) + \varepsilon_{\text{IR}} E\right) A_C$$

(1-1)

where $\alpha_S$ is the solar absorptivity and $\varepsilon_{\text{IR}}$ the infrared emissivity (which is equal to the infrared absorptivity) of the aluminum used to construct the shroud, $S$ is the solar flux (1360 W/m²), $S_R$ is the reflected solar flux from the Earth, $E$ is the Earth infrared flux (240 W/m²), and $A_C$ is the cross-sectional area. For a conical shroud with a height $H$ and a base diameter $D$, the cross-sectional area will depend on the orientation of the shroud with respect to the sun and Earth. It will range in value from $\pi (D/2)^2$ (for a head-on orientation) to $H(D/2)$ (for a side-on orientation). Since we are interested in calculating the amount of coolant required to maintain the shroud temperature, we will use the higher value of $H(D/2)$. Equation (1-1) then becomes

$$P_{\text{outer}} = \left(\alpha_S (S + S_R) + \varepsilon_{\text{IR}} E\right) H(D/2)$$

(1-2)

For aluminum that has been polished, the values of $\alpha_S$ and $\varepsilon_{\text{IR}}$ are approximately 0.20 and 0.031, respectively.¹ We assume a shroud with a height of 3 meters and a base diameter of 1 meter. With these values, we obtain a value for $P_{\text{outer}}$ of 540 watts.

Now we consider the thermal energy transferred to the inner surface of the cooled shroud by infrared radiation from the warhead:

$$P_{\text{inner}} = \varepsilon_{\text{si}} \sigma (T_w^4 - T_s^4) A_s$$

(1-3)

where $P_{\text{inner}}$ is the power absorbed by the inner wall of the shroud, $\varepsilon_{\text{si}}$ is the effective infrared emissivity of the superinsulation between the warhead and shroud, $\sigma$ is the Stefan-Boltzmann constant ($5.67 \times 10^{-8}$ W/m²-K⁴), $T_w$ is the temperature of the warhead, $T_s$ is the temperature of the shroud, and $A_s$ is the surface area of the shroud.² For a conical shroud of height $H$ and base diameter $D$, the surface area is given by $\pi (D/2) [(D/2)^2 + H^2]^{1/2}$. Thus, equation (1-3) becomes

$$P_{\text{inner}} = \varepsilon_{\text{si}} \sigma T_w^4 \pi (D/2) [(D/2)^2 + H^2]^{1/2}$$

(1-4)

² The effective emissivity is defined by $q = \sigma \varepsilon_{\text{eff}} (T_1^4 - T_2^4)$, where $q$ is the energy per unit area and time transferred by radiation between surfaces 1 and 2, $T_1$ and $T_2$ are the temperatures of surfaces 1 and 2, and $\varepsilon_{\text{eff}}$ is the effective emissivity of the insulation between surfaces 1 and 2. Wertz and Larson, Space Mission Analysis, p. 383.
Assuming that the warhead is at room temperature (300 K), and superinsulation has a relatively high effective emissivity of 0.05,\(^3\) the power absorbed by the inner surface of the shroud (with a height of 3 meters and a base diameter of 1 meter) is 110 watts.

Although its effect will be relatively small, we also want to estimate the heat transferred to the inner wall of the shroud from the warhead through the supports that attach the shroud to the warhead. We assume that the shroud inner wall sits on 20 supports resting against the heat shield of the inner warhead, and that each support has a cross section of 1 cm\(^2\) and a length of 2.5 cm. Thus, the power conducted to the inner surface of the shroud through the supports, \(P_{SUPPORTS}\), is given by

\[
P_{SUPPORTS} = \left(\frac{\kappa}{\Delta X}\right) \left[ T_W - T_S \right] A_S \tag{I-5}
\]

where \(\kappa\) is the thermal conductivity of the supports, \(\Delta X\) is the length of each support (in this case, \(\Delta X = 2.5\) cm = 0.025 m), \(T_W\) and \(T_S\) are the temperatures of the warhead and shroud, and \(A_S\) is the total cross sectional area of the supports (in this case \(A_S = 20\) cm\(^2\) = 0.002 m\(^2\)). Although a material with far lower conductivity could be chosen,\(^4\) we will assume the supports are made of Teflon, which we will take to have an average thermal conductivity of \(\kappa = 1.0\) W/m-K.\(^5\) Under these assumptions, for a warhead at room temperature (300 K) and a shroud at liquid nitrogen temperature (77 K), the power transferred to the shroud through the supports will be roughly 18 watts.

The total power absorbed by the shroud, \(P_{SHROUD}\), will thus be

\[
P_{SHROUD} = P_{INNER} + P_{OUTER} + P_{SUPPORTS}
\]

\[
= \left[ 540 + 110 + 18 \right] \text{watts} = 670 \text{ watts}
\]

The heat of vaporization of liquid nitrogen is approximately \(2 \times 10^5\) J/kg = 200 J/g, requiring the evaporation of about 3.4 grams of liquid nitrogen per second or about 200 grams per minute. For 30 minutes in space, the amount of liquid nitrogen coolant needed is 6 kilograms, or about 7.5 liters. If the missile was fired on a trajectory so that it was always in the Earth’s shadow, the amount of nitrogen need to maintain the shroud at 77 K would be reduced by about a factor of five.
Appendix J

Exoatmospheric Hit-to-Kill Intercept Tests

In this appendix we list the 20 intercept tests that the United States has conducted (through January 2000) with exoatmospheric hit-to-kill interceptors, which are the tests that are relevant to the development of the current NMD system. Of these tests, 5 intercept attempts hit the target and 15 missed.

Homing Overlay Experiment (HOE)

The Homing Overlay tests used a large, infrared homing interceptor, which unfurled a fifteen foot diameter set of spokes just prior to intercept.

<table>
<thead>
<tr>
<th>HOE Intercept Attempt</th>
<th>Date</th>
<th>Result</th>
<th>Notes</th>
</tr>
</thead>
<tbody>
<tr>
<td>1</td>
<td>7 February 1983</td>
<td>MISS</td>
<td>The first intercept attempt missed by a large distance. The miss was attributed to problems in the sensor cooling system, which prevented target tracking.</td>
</tr>
<tr>
<td>2</td>
<td>28 May 1983</td>
<td>MISS</td>
<td>The second test was very similar to the first, with the interceptor missing its target by a great distance. While it was able to begin homing, the guidance electronics failed.</td>
</tr>
<tr>
<td>3</td>
<td>16 December 1983</td>
<td>MISS</td>
<td>A software error in the on-board computer was the cause of the third miss. The error prevented the conversion of optical homing data into steering commands.</td>
</tr>
<tr>
<td>4</td>
<td>10 June 1984</td>
<td>HIT</td>
<td>The fourth attempt resulted in a hit for the interceptor, but the target was heated to 100 degrees F to increase its visibility to the interceptor’s infrared sensors. According to reports, the target was acquired at a range of “hundreds of miles” and the closing speed was greater than 20,000 feet per second (6 km/s).</td>
</tr>
</tbody>
</table>

Exoatmospheric Reentry Vehicle Intercept System (ERIS)

ERIS was part of the Strategic Defense Initiative’s ground-based interceptor program and built on technology developed as part of the Homing Overlay Experiment.

<table>
<thead>
<tr>
<th>ERIS Intercept Attempt</th>
<th>Date</th>
<th>Result</th>
<th>Notes</th>
</tr>
</thead>
<tbody>
<tr>
<td>1</td>
<td>28 January 1991</td>
<td>HIT</td>
<td>The target was accompanied on each side by a decoy balloon. The ERIS seeker was programmed to track and hit the center of the three targets. It did not otherwise differentiate between the warhead and decoys. The intercept reportedly occurred at an altitude of 145 nautical miles (270 km) and a closing speed of over 30,000 miles per hour (13.4 km/s).</td>
</tr>
<tr>
<td>2</td>
<td>13 March 1992</td>
<td>MISS</td>
<td>The kill vehicle missed its target by “several meters,” in part because it failed to select the target warhead with enough time remaining to maneuver for a hit. The kill vehicle flew between the target and a single balloon decoy separated from the target by about 20 meters.</td>
</tr>
</tbody>
</table>
Lightweight Exoatmospheric Projectile (LEAP)

In 1992 the Strategic Defense Initiative Organization conducted two test of the LEAP kill vehicle then under development. The LEAP was eventually adopted by the US Navy as the kill vehicle for the Navy Theater-Wide theater missile defense system, and has so far undergone two additional intercept tests in this role.

**LEAP Intercept Attempt 1 (Flight Test 2) MISS** 19 June 1992
This attempt failed when the LEAP missile did not receive data from the ground control system as planned about the target’s speed and position.

**LEAP Intercept Attempt 2 (Flight Test 3) MISS** 22 June 1993
The interceptor apparently missed its target by about 7 meters, but very little information is available about this test.

**LEAP Intercept Attempt 3 (Terrier/LEAP, FTV-3) MISS** 4 March 1995
The first intercept test for the Navy Theater Wide Ballistic Missile Defense System. A software error during the second stage of flight caused the third stage of the missile to fly too high and miss its target.

**LEAP Intercept Attempt 4 (Terrier/LEAP, FTV-4) MISS** 28 March 1995
The kill vehicle did not switch to internal power, reportedly because a battery failed, and it passed 170 meters from the target. The Navy was very optimistic despite the intercept failure and termed the test a “clear success,” with 42 of 43 test objectives met.

Theater High Altitude Area Defense (THAAD)

The Theater High Altitude Area Defense system is the US Army’s ground-based, exo- and high-endoatmospheric interceptor. The system uses a single-stage, solid-propellant missile and a kinetic kill vehicle with infrared guidance in the terminal phase.

**THAAD Intercept Attempt 1 (Flight Test 4) MISS** 13 December 1995
This was the first THAAD test in which the primary objective was a hit-to-kill intercept. A software error caused the Divert and Attitude Control System to misfire and made the kill vehicle veer off course. The trajectory was corrected, but insufficient fuel remained for intercept. A program official said, “the indications are that we should have hit.”

**THAAD Intercept Attempt 2 (Flight Test 5) MISS** 22 March 1996
Twenty seconds into the flight a lanyard connecting the kill vehicle to its supporting electronics module disconnected, effectively shutting down the interceptor before booster separation. The interceptor stopped responding to ground controls, flew past its target, and was subsequently detonated.

**THAAD Intercept Attempt 3 (Flight Test 6) MISS** 15 July 1996
The interceptor’s seeker electronics malfunctioned, overloading the signal processor and preventing target acquisition. An independent review panel found no major problems in the program but recommended that testing be slowed and “more emphasis placed on intercepting a target instead of meeting an aggressive test schedule.” One official said that he “believe[d] they will dramatically restructure the program” if the missile failed to intercept during its next test.

**THAAD Intercept Attempt 4 (Flight Test 7) MISS** 6 March 1997
The Divert and Attitude Control System (DACS) failed and the missile flew out control. One official complained, the DACS “did not work. It never worked. What we don’t know is why it didn’t work.”
Theater High Altitude Area Defense (continued)

THAAD Intercept Attempt 5 (Flight Test 8)   MISS
12 May 1998
Originally scheduled for December 1997, the test was postponed due to problems in the missile’s inertial measurement unit and concerns over system readiness. The test ended early as a short-circuit in the thrust vector control assembly sent the interceptor out of control.

THAAD Intercept Attempt 6 (Flight Test 9)   MISS
29 March 1999
The missile lost track of the target, missed it by about 30 yards and self-destructed. Sources said that the missile “did not take over and make the final adjustments... to intercept the target.” The miss was eventually attributed to a failure in one of the thrusters used to steer the interceptor. The Pentagon initially reported success in 16 of its 17 test goals, but later stated that 2 out of 4 was more accurate.

THAAD Intercept Attempt 7 (Flight Test 10)   HIT
10 June 1999
Following several test delays, the THAAD hit—the first time in seven attempts. Brig. Gen. Richard Davis, USAF, was very optimistic: “The technology can work. We’ve shown it to be able to work. No longer will we say that the design is flawed.” This test used a Hera target missile flown on a highly lofted trajectory. The target flew at about 2 kilometers per second at intercept and the intercept occurred at 60–100 kilometers altitude. While the THAAD missile hit its target, no countermeasures were employed, making this a less difficult scenario than THAAD could face from a real-world threat.

THAAD Intercept Attempt 8 (Flight Test 11)   HIT
2 August 1999
The test used a Hera target missile flown on a highly lofted trajectory. The 4-meter-long reentry vehicle separated from the missile booster. Intercept occurred at above 80 kilometers altitude and probably at well above 100 kilometers. The target was traveling at about 2 kilometers per second at intercept. Again, no countermeasures were employed.

National Missile Defense (NMD)
The NMD ground-based interceptor consists of an exoatmospheric kill vehicle (EKV) on top of a booster. The booster will be a three-stage missile based in an underground silo. The NMD EKV has its own seeker, propulsion, communications, guidance, and computers to support targeting decisions and maneuvers.

NMD Intercept Attempt 1 (IFT-3)   HIT
2 October 1999
The test was originally scheduled for June 1999, but was postponed several times, reportedly due to a series of minor problems with the kill vehicle. A surrogate booster carried the prototype exoatmospheric kill vehicle from the Kwajalein Missile Range to intercept a target launched from Vandenburg AFB, California. The intercept reportedly occurred at about 140 miles (225 km) altitude at a closing speed of 15,000 miles per hour (6.7 km/s). The NMD ground-based radar observed the test but was not used to guide the kill vehicle—instead a global positioning system transmitter on the mock warhead (along with a backup C-band radar beacon) told the interceptor missile where to release the kill vehicle. In January 2000, the Pentagon acknowledged a series of anomalies in the test that led to the kill vehicle initially being unable to find the mock warhead. Eventually, the kill vehicle started to home instead on the bright balloon decoy that was included in the test. Fortuitously, the balloon and warhead were close enough together that the warhead then appeared in the field of view of the kill vehicle, which was then able to home on and intercept the warhead. According to the 1999 annual report by the Pentagon’s Director of Operational Testing and Evaluation, there is no basis to classify the test as either a success or failure since it is unclear whether the intercept would have occurred if the brighter balloon had not been present.

NMD Intercept Attempt 2 (IFT-4)   MISS
18 January 2000
This test differed from IFT-3 in that it incorporated other components of the system, including the Defense Support Program early warning satellites, the prototype ground-based radar on Kwajalein, and the battle-management system in Colorado. A failure of the two infrared sensors on the kill vehicle caused it to miss the mock warhead, reportedly by a distance of 100 feet.