LOW-DENSITY WIND TUNNEL

AERODYNAMIC HEATING - INTRODUCTION

Jackson R. Stalder or Morris Rubesin

The NACA has undertaken a long-range research program to investigate the general field of aerodynamic heating and heat transfer at supersonic speeds.

In general, the high-speed heat-transfer problem has two major practical aspects: First, the effect of heat transfer upon the aerodynamic characteristics of wings and bodies and, secondly, the effect of heat transfer (or aerodynamic heating) upon structural requirements for high-speed aircraft. Both of these practical problems require extensive research in order to provide information for the design of successful high-speed aircraft and missiles.

The problem of aerodynamic heating and heat transfer is a subphase of the more general problem of high-speed gas dynamics. It can be shown that the general field of gas dynamics may be divided into three major regimes and that the division lines, although not sharply defined at present, are marked by the ratio of the average length of path traveled by a gas molecule between successive collisions and a characteristic body dimension. In terms of conventional parameters, we may show these three major regimes on a Mach number-Reynolds number chart (fig. 11(b)).

The region of conventional gas dynamics is shown in the right-hand side of the chart. The dots represent typical aircraft flying at several conditions. For example, this dot represents an airplane with a 5-foot chord wing flying at a Mach number of 1.3 at 30,000 feet altitude. The straight red line represents the range of Mach and Reynolds numbers covered by the wasserfall missile during a portion of its flight. In this region the air density is high enough that the mean free molecular path is negligible in comparison with body dimensions.

The center portion of the chart delineates the slip-flow regime, so-called because a flowing gas in this regime slips along a solid boundary rather than sticking to the surface as occurs in the region of conventional gas dynamics. The red line represents the range of Mach and Reynolds numbers encountered by the V-2 missile during a portion of its flight. It may be seen that the V-2 traverses a portion of this slip-flow regime.

The region to the extreme left of the chart is characterized by long mean free paths and is known as the free-molecule flow regime. This regime is encountered in flight at altitudes of 100 miles and higher for conventional-sized missiles. The top dot represents a 50-foot missile traveling at 10,000 feet per second at 100 miles altitude while the lower dot represents the same missile traveling at 5,000 feet per second at 75 miles altitude.

Research is presently under way in all three of these gas-dynamic regimes. The area enclosed by the yellow line represents the range of variables which will be covered by a high-speed wind-tunnel, now under construction, which is designed especially for heat-transfer studies. The area enclosed by the pink line represents the testing range of a low-density wind tunnel now undergoing preliminary testing. At this time, I should like to introduce

Mr.__________, who will describe some heat-transfer research that has been done in the region of conventional gas dynamics.

HEAT TRANSFER IN THE LAMINAR BOUNDARY-LAYER REGION OF BODIES OF REVOLUTION IN STEADY SUPERSONIC FLIGHT

Forrest Gowen

Aerodynamic heating has been recognized as a serious problem for nearly 20 years, and the flight of aircraft at transonic and supersonic speeds in recent months has served to emphasize the problem. The Germans during World War II, conducted numerous investigations on the subject of aerodynamic heating and on methods for the alleviation of its effects.

They considered the heat transfer to plane surfaces and conical bodies, but for long range supersonic aircraft the effect of longitudinal body curvature on the local heat-transfer coefficients, must be known in order to design the lightest possible surface cooling system. Therefore, it was necessary to develop a method for the calculation of the local rate of heat transfer to any body of revolution in a supersonic, or compressible air flow. Such a method has been developed for the laminar boundary-layer region of thin bodies of revolution and has been published as an NACA Technical Note.

A partial check of the validity of the method is demonstrated by the fact that at subsonic Mach numbers it gives values of heat-transfer coefficients which are in good agreement with those given by the experimentally proven method for subsonic flow developed by Look and Allen in 1943.

In order to further check the compressible flow method a simple electrically heated 20° cone was tested in the Ames 1- by 3-foot supersonic wind tunnel No. 1 at a Mach number of 1.53 and a Reynolds number of one million. These test conditions provided a completely laminar boundary layer on the cone surface. Local heat-transfer rates were measured along the surface of the cone and the local heat-transfer coefficients were calculated (fig. 11(c)). The values of heat transfer coefficient given by the theory are shown by the yellow line, and, in the area of the cone where the measurements were made the agreement between theory and experiment is considered to be satisfactory. The heat-transfer coefficients near the tip of the cone are very high because the boundary-layer thickness at the tip is nearly "0" and the heat-transfer coefficients decrease along the cone as the boundary-layer thickness increases.

Since the experiment and theory agree satisfactorily in the region in which measurements were made, the theory was applied to the calculation of local cooling rate required to maintain a desired surface temperature on an example aircraft with a laminar boundary layer. The longitudinal distribution of local cooling rate required to maintain a desired surface temperature on the example aircraft is shown by this curve (fig. 11(d)). For an altitude of 40,000 to 100,000 feet and a flight Mach number of 1.2 to 3.0. The shape of the distribution curve for a given aircraft changes very little with increasing Mach number, but the values of local cooling rate increase rapidly.

The total cooling rate, or the capacity of the cooling system required to maintain a given surface temperature, is shown in the next chart (fig.11(e)) as a function of Mach number for the example aircraft. The fuselage of the aircraft is 44 feet long and is considered to be flying at an altitude of 100,000 feet. For these flight conditions the boundary layer could be expected to be almost completely laminar if the surface is sufficiently smooth. Because of the very low-ambient air temperature at the indicated altitude no cooling is required to maintain a reasonable surface temperature at the low supersonic Mach numbers, but as the flight Mach number is increased the cooling rate required to maintain a particular surface temperature increases very rapidly.

Even at high altitudes where low-ambient air temperatures prevail, the practical operation of long-range supersonic aircraft at high Mach numbers will be dependent on the provision of adequate insulation and cooling for the aircraft structure, equipment, payload, and occupants.

Mr. __________ will now discuss one of the structural problems which results from aerodynamic heating.

TEMPERATURE GRADIENTS IN THE WING STRUCTURE OF AN AIRPLANE AS A RESULT OF TRANSIENT TEMPERATURE CONDITIONS AS ENCOUNTERED DURING DIVES

Thorval Tenderland or George H. Holdaway

I will discuss temperature gradients in the wing structure of a high-speed airplane as a result of rapidly changing surface temperatures. Due to a differential expansion associated with temperature gradients, thermal stresses are produced in the structure.

The flight conditions which may cause temperature gradients are: (1) Rapid rates of change of the ambient-air temperature such as occur in a steep dive or climb and (2) sudden changes in speed or degree of friction heating.

The experimental technique used to obtain an indication of the magnitude of these temperature gradients was to instrument a P-80 pursuit-type airplane with thermocouples throughout the wing structure. The airplane was then dived in a series of dives from 35,000 feet to 5,000 feet.

The results of the dive tests indicated that temperature differences in the wing structure resulted primarily from (1) Differences in mass of the components of the structure and (2) the rate of change of the boundary-layer temperature. The effects of differences in mass are indicated in the following chart (fig. 11(f) by the chordwise temperature distribution as found at the termination of a high-speed dive. These surface temperatures were measured at station 76 of the right wing. Due to the relatively large mass of the spar caps, their temperatures lagged that of the adjacent skin from 35°F to 50°F.

The time at which the maximum temperature differences occurred was at the pull-out from the dive or shortly thereafter or when both the thermal stresses and the stresses from aerodynamic loads would be large. Calculated spanwise thermal stresses which would result from the temperature differences in the upper wing surface are shown by the following curve. As may be noted from the temperature distribution, the spar caps were relatively cold as compared to the adjacent skin. Therefore, the skin would be restrained from expansion in a spanwise direction by the cooler spar caps. As a result the skin would be in compression while the spar caps would be in tension. The maximum stress, a compression stress of about 6,000 pounds per square inch, would occur in the skin near the point of attachment to the spar caps. This stress is of significant magnitude, however, the importance of the thermal stresses for this airplane depends upon the stresses from aerodynamic loads at the pull-out from the dive and whether the two types of stresses are additive.

To predict the temperature differences for other structural arrangements and at higher rates of vertical descent, calculations were made for the temperatures of the skin and spar cap during one of the dives. This chart (fig. 11(g)) shows a comparison between calculated and measured temperatures of the skin and the spar cap at the 20-percent-chord. To calculate the skin and spar-cap temperatures, the arrangement on the test airplane was assumed to be idealized in shape as shown. The calculated skin and spar-cap temperatures are somewhat higher than the measured skin and spar-cap temperatures. However, the calculated temperature differences are in good agreement with the measured temperature differences and the method provides a means for predicting temperature differences to be anticipated for other structural arrangements and for dives beyond the range of those in the tests.

The effects of increases in the rate of vertical descent and increases in the mass of the structure are shown by this chart (fig. 11(h)). These three curves show calculated temperature differences for three different size skin and spar-cap arrangements at the termination of assumed dives from 35,000 feet to 5,000 feet. One of the three arrangements corresponded to that on the test airplane. The remaining two were similar, but heavier, to correspond to probable sizes required for strength at the higher diving speeds.

As may be noted, a skin and spar-cap arrangement such as on the test airplane would experience a large temperature difference at the higher rates of vertical descent. However, as the skin is increased in thickness a reduction in the temperature differences results. Thus, the increased structural mass required for strength at the higher diving speeds tends to compensate for the increased rate of change of the boundary-layer temperature. The extent of this compensation will depend upon the particular airplane and the conditions of the dive.

At this time I would like to introduce ____________, who will discuss heat transfer at high altitudes.

HEAT TRANSFER AT HIGH ALTITUDES

Glen Goodwin

One of the factors which may limit the speed of missiles or aircraft traveling in the upper atmosphere is the friction heating of the skin and structure. Meteorites which heat and burn when entering the earth's atmosphere at high speeds are ready evidence of this frictional-heating effect. Analytical or experimental information concerning high altitude, or low density, heat transfer is relatively meager. The slip-flow regime has proven intractable to mathematical analyses, and experimental investigations are practically non-existent. The molecular-flow regime has been investigated analytically but has not been investigated experimentally. I should like to discuss here some results of an analysis of the molecular-flow regime which will indicate temperatures likely to be experienced by bodies traveling in the upper atmosphere and to describe an apparatus to be used for the experimental study of heat-transfer problems in high-speed, high-altitude flight.

This figure (fig. 11(1)) indicates the surface temperatures of uncooled bodies for various flight speeds at an altitude of 75 miles with the effect of solar radiation neglected. This corresponds to the case of nocturnal flight. These data were calculated by Stalder and Jukoff for conditions of free-molecular flow using the kinetic theory of gases. The body is a flat plate with the two surfaces thermally insulated from one another and inclined at various angles of attack. Increasing the velocity of the plate causes the temperature of the front surface to increase while the temperature of the rear surface decreases. Both surfaces are at the same temperature for zero flight speed at all angles of attack and for zero angle of attack and all flight speeds. At 24,000 mph which approximately is the velocity of escape from the earth's gravitational pull at this altitude, the front surface of the plate inclined to 90° reaches temperatures around 2000° F while the rear surface which is completely shielded from the oncoming stream of molecules approaches a temperature of absolute zero. This large temperature difference, approximately 2500° F, is indicative of thermal stress problems that may be anticipated. For the same velocity and the plate inclined to 5° the temperature of the front surface drops to 1000° F while the rear surface which now is shielded only partially from the oncoming stream of molecules reaches a temperature of -400° F.

This figure (fig. 11(j)) indicates the stagnation temperature or steady state equilibrium temperature of a flat plate as a function of altitude for a flight speed of 15,000 mph and for night and day time conditions. First consider the effect of altitude on surface temperature. At low altitudes where the density is relatively high, the surface temperature reaches very high values. As the altitude is increased, the air density decreases and thus the surface temperature decreases. Second, consider the effect of solar radiation which is done by considering daytime conditions as compared to night time conditions. Solar radiation becomes important for these conditions only above 80 miles where the frictional heating has decreased until it is comparable to the heat received from the sun.

For all practical purposes frictional heating may be neglected for altitudes above 125 miles and flight velocities of 15,000 miles per hour or less.

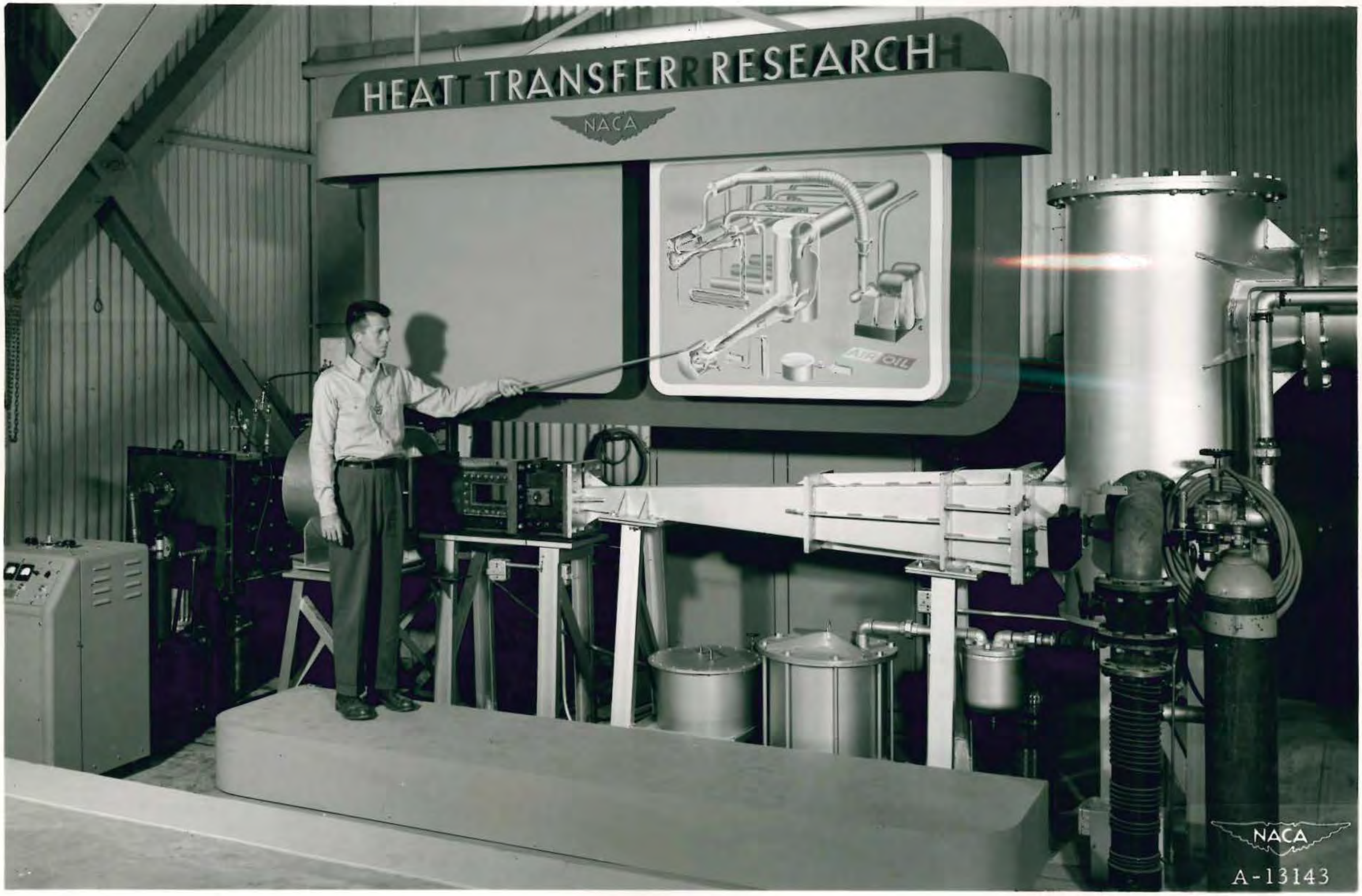
The results presented on these last two charts are results of an analytical analysis in the molecular flow regime. It is the purpose of research now in progress to investigate experimentally, heat transfer in the slip flow and free molecule flow regimes. To do this, conditions in the laboratory must be made to simulate conditions at high altitudes. This is done by our low-density supersonic wind tunnel located here and schematically shown on the next chart (fig.11(k)).

This tunnel has a test section 3 by 3 inches and operates for Mach numbers from 1.5 to 3.5 at test section pressures from 200 to 10 microns (1 micron is 1/1000 of a mm) which corresponds to altitudes from 40 to 60 miles. The Reynolds number range is from 100 to 1000.

The tunnel is a once-through continuous-flow tunnel. The air flows through a drying tank containing silica gel as the drying agent, through a flow rate meter, and then through a needle valve into an upstream settling tank, where the pressure ranges from 1 to 15 mm of Hg. From the tank the air is expanded through a supersonic nozzle into the test section and then through a diffusor into a downstream settling tank. Four booster-type oil diffusion pumps, which operate on the ejector principle, are connected to the downstream tank by a manifold, drawing the air from the tank at a pressure of approximately 100 microns and discharging it at 3 mm of Hg to a large mechanical vacuum pump which discharges it to the atmosphere.

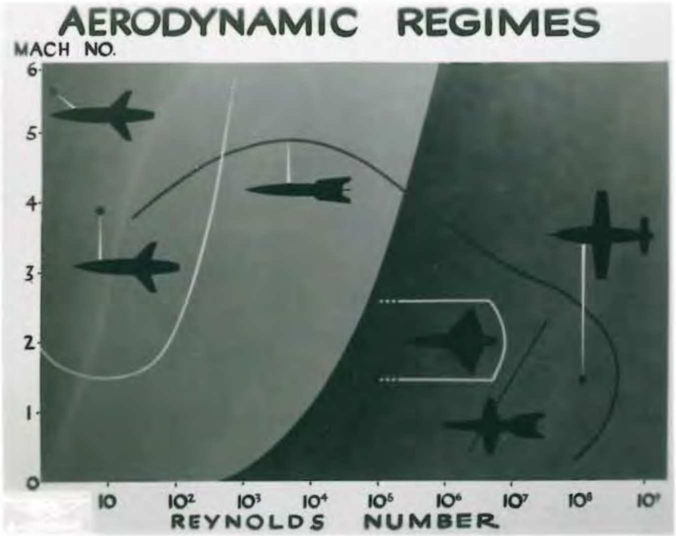
New and varied problems in instrumentation are involved in this system. The test section pressure is too low for successful use of optical apparatus such as Schlieren. Measurements of pressures are difficult but not impossible.

Use of the tunnel will not be confined to heat transfer exclusively, but will also include flow phenomena in the slip and molecular flow regimes.

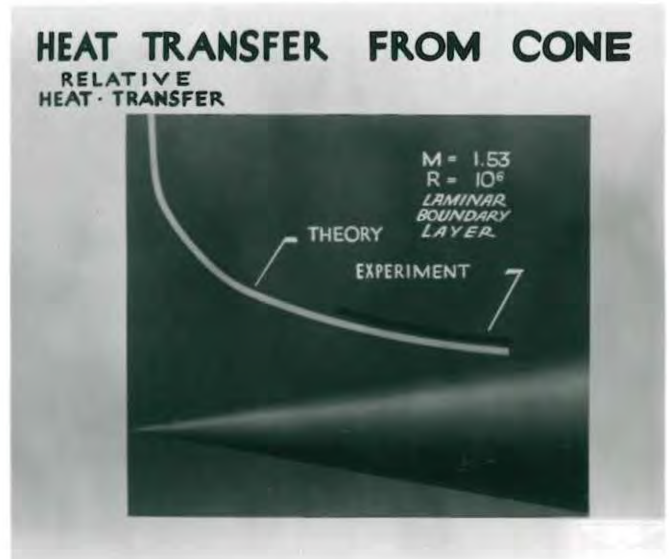


(a) General view.

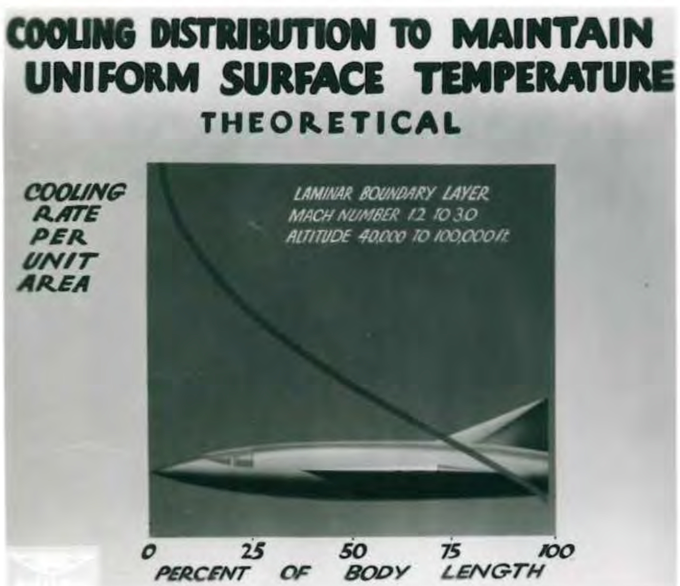
Figure 11.- Low-density wind-tunnel exhibit.



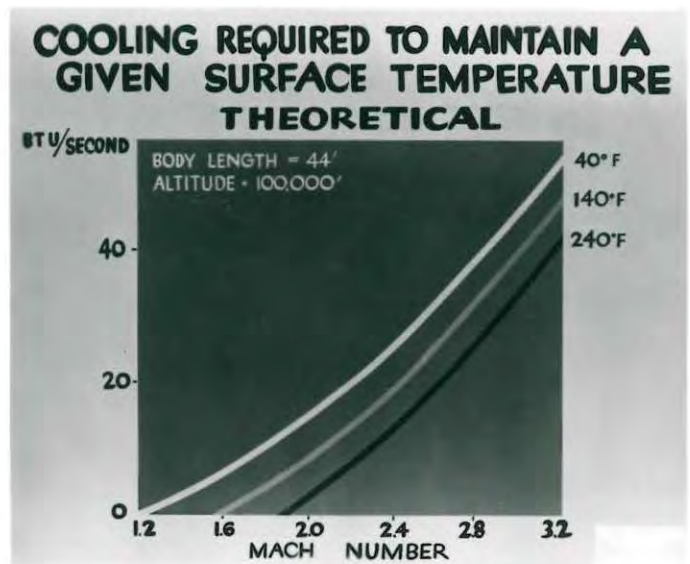
(b) First chart.



(c) Second chart.



(d) Third chart.



(e) Fourth chart.

Figure 11.– Continued

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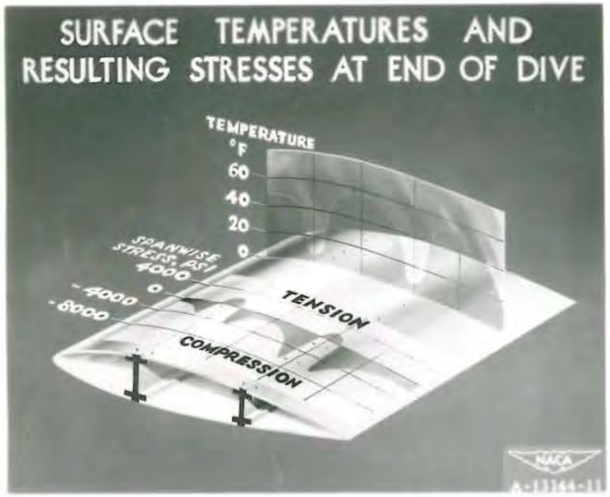
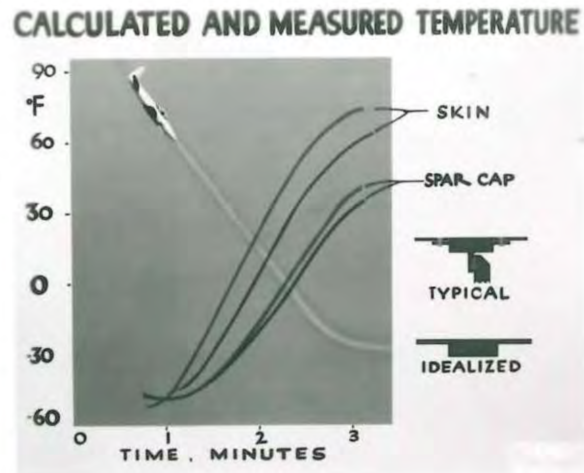
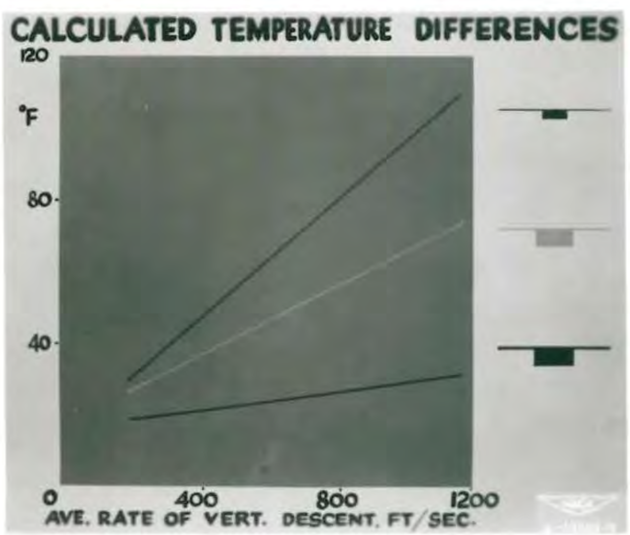


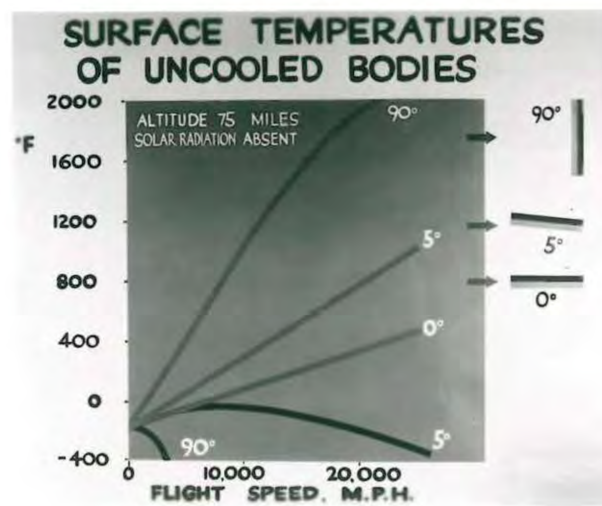
(f) Fifth chart.



(g) Sixth chart.

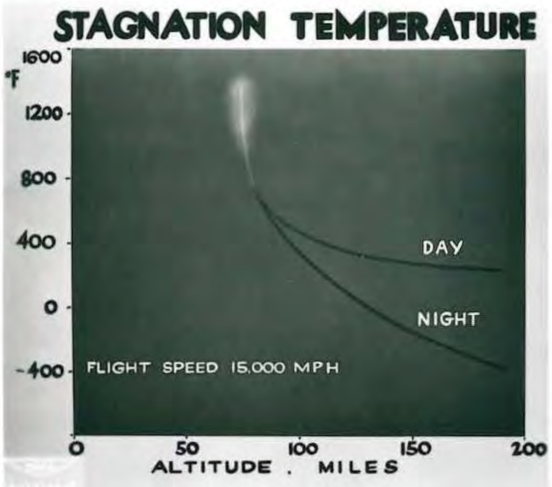


(h) Seventh chart.

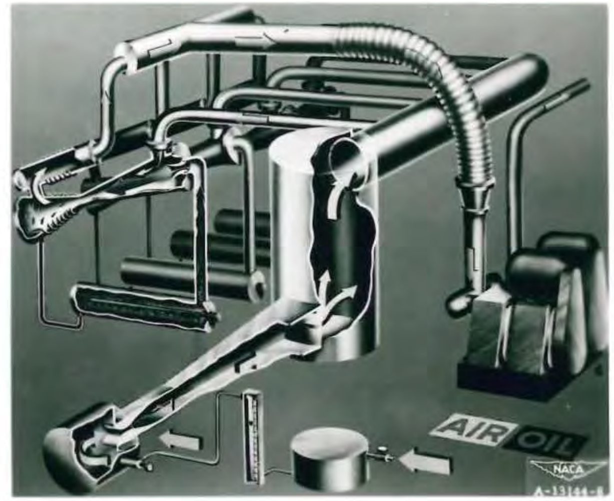


(i) Eighth chart.

Figure 11.– Continued.



(j) Ninth chart.



(k) Tenth chart.

(l) Eleventh chart.

Figure 11.– Concluded.

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