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TECHNICAL MEMORANDUM

X-711

STRUCTURAL HEATING EXPERIENCES ON THE X-15 AIRPLANE

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TECHNICAL MEMORANDUM X-711

STRUCTURAL HEATING EXPERIENCES ON THE X-15 AIRPLANE* **

By Eldon E. Kordes, Robert D. Reed, and Alpha L. Dawdy

SUMMARY

A survey of maximum structural temperatures measured on the X-15 airplane during speed flights up to a Mach number of 6 is presented. Structural problems caused by local hot spots and discontinuities encountered during the flight are also discussed.

In general, the hot-structure concept used for the primary structure of the X-15 airplane has proven to be satisfactory. Problems have arisen only from local hot spots and discontinuities. These problems include windshield-glass failures, airflow through openings in the external structure, and structural discontinuities.

Comparison of calculated and measured internal temperatures has shown that satisfactory thermal gradients through the structure can be calculated from known heat input to the exposed surfaces.

INTRODUCTION

The X-15 airplane was designed primarily for four missions: two speed missions to 6,600 feet per second and two altitude missions to 250,000 feet. Structural temperatures calculated from extrapolated wind-tunnel data for these missions resulted in the hot-structure design of the X-15. This design required the use of unconventional materials and design techniques. During the flight program considerable experience has been gained on the behavior of this structure in a hypersonic environment.

This paper shows the magnitude of structural temperatures measured during the flight program and describes structural problems that have developed as a result of structural heating.

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AIRPLANE STRUCTURE

The X-15 fuselage structure is basically monocoque or semimonocoque as indicated in figures 1 and 2. The external surface is Inconel X, and titanium is used extensively for the internal structure. The forward-fuselage section contains the double-walled pressure compartments for the pilot and instruments. The center-fuselage section is formed by the oxidizer tank ahead of the wing and the fuel tank with frames for supporting the wing. The rearward fuselage structure supports the empennage, main landing gear, and engine. The wing is of multispar construction of taper-milled Inconel X skin with titanium substructure. The horizontal and vertical tails are two-cell box structures with stabilizing ribs. The wing, horizontal tail, and vertical tail have segmented leading-edge heat sinks of Inconel X. A tunnel along each side of the fuselage for housing control cables, hydraulic lines, and instrument wiring is formed by removable panels. Throughout the structure, extensive use has been made of corrugations and beading to minimize thermal stress.

INSTRUMENTATION

The thermocouple instrumentation on the X-15 is shown in reference 1. Many areas of the X-15 airplane do not contain thermocouples, and in the areas where thermocouples are available, the spacing and location do not permit detection of local hot spots and severe gradients that may develop. A method that has been successful in obtaining qualitative measurements of maximum temperature distribution in conjunction with thermocouple measurements is the use of temperature-sensitive paints.

DISCUSSION OF RESULTS

Flight Profiles

The X-15 flight program has proceeded toward the design speed and altitude mission in small increments. For example, the speed increments have been accomplished by flying along a trajectory similar to the mission trajectory and programing engine shutdown to give the desired Mach number. For all speed flights the peak Mach number occurs at engine shutdown. Recovery portions of each flight have been used to obtain stability data and evaluate handling qualities; hence, the recovery was different for each flight. The differences in the flight profiles make it almost impossible to obtain systematic research data on structural heating; therefore, this paper presents only examples of

temperature levels and distribution experienced during this program. From consideration of structural safety, the present program has pointed out structural problems while they are still minor.

Maximum Measured Temperatures

Distributions.- The maximum skin-temperature distribution measured on the flight to maximum speed is presented in figure 3. These temperatures occurred during the recovery portion of the flight several minutes after engine shutdown and, because of the maneuvers performed, cannot be attributed entirely to peak Mach number. Thermocouple locations for the forward fuselage and wing lower skin are shown by the solid symbols in the sketches and measured temperatures by the open symbols in the plots. The solid curves, included for reference, are for the calculated maximum temperature distribution for a design speed mission. The dip in the curves near the 5-percent fuselage station is attributed to the thicker skin in this region, and the rise is caused by the insulation in the cockpit area, which blocks internal radiation. The low temperatures near the 40-percent station are caused by the liquid-oxygen tank. On the wing, the higher temperatures near the trailing edge are a result of the thinner skin. Effects of the heat sink of the internal structure are not shown, since internal temperature measurements are not adequate. These data are the temperatures measured on a speed flight to a Mach number of 6.04 and are representative of the levels reached during the X-15 program. It should be mentioned that on all speed flights above a Mach number of 4, the maximum temperatures followed the same trends.

The large gap between measured temperatures and the temperatures predicted for design can be attributed to values of heat transfer used in the design calculations and differences in the flight profiles. With a flight profile chosen to minimize heating effects, the high-speed flights have been accomplished without extreme structural temperatures.

Locations.- The maximum temperatures measured at various points on the X-15 during the flight program are summarized in figure 4. Maximum temperatures are shown for the canopy frame, ball nose, lower fuselage, side fairings, lower wing skin, wing leading edge, lower ventral, lower speed brake, and horizontal tail. These maximum temperatures did not all occur on the same flight; however, they serve to illustrate the highest temperature levels that the structure has experienced up to this time, with the exception of local hot spots which are discussed subsequently.

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Wing Spar Temperatures

Temperature time history.- The temperatures presented thus far do not show variation during the flight or the gradient through the structure. In order to illustrate these quantities, typical temperature histories for the front spar of the wing at the midsemispan are presented in figure 5. These temperatures were measured on a flight to a maximum Mach number of 5.28.

The sketch shows the thermocouple location on the lower skin, the lower spar cap, the web, and the upper skin. The number by each thermocouple is used to identify the curve. Time is measured from launch from the B-52 after a "cold soak" at an altitude of 45,000 feet, and the time of peak Mach number is given for reference. On this flight the lower skin temperature shown by the solid curve increased at 6.25° per second during powered flight, and the maximum temperature occurred about 150 seconds after peak Mach number. This time corresponds to the time of maximum temperature difference between the lower skin and the spar web of about 570° F.

Temperature gradient.- From the standpoint of thermal stresses in the structure, the temperature gradient, together with the temperature level, defines the most severe condition on each flight. The measured data give the temperature levels but, with the limited number of thermocouples on the spar, the complete thermal gradient cannot be determined from flight measurements and must be obtained from analysis. Calculated gradients are compared with flight data for the time of maximum gradient in figure 6. The temperature is shown as a function of the wing thickness measured from the lower surface. The solid curve was obtained from calculations, and the points are from the flight measurements at the numbered points shown on the sketch. For these calculations, the heat transfer to the external skin was determined first on the basis of the time history for the skin temperatures by the method described in reference 1. This heat input was used to compute the thermal gradient through the spar and included 4 inches of each cover sheet. The spar cross section was divided into 30 elements for the calculations. The calculated and measured temperatures agree well at the four thermocouple locations shown in the sketch.

Calculated stress.- The thermal gradient in figure 6 has been used to calculate the thermal stresses in this isolated spar element. The stress calculations have neglected the interaction with the adjacent structure and are used primarily to monitor the changes in the thermal-stress level for various flight conditions. Calculated thermal stresses for the gradient shown are presented in figure 7. The curves on the right indicate the variation of normal stress through the spar, and the curves on the lower left show the variation of the normal stress in the lower skin. Included for reference are the thermal stresses calculated

for a design speed mission. The stress levels for the Mach 5.28 flight
at 225 seconds are well below the stresses predicted for this design
condition, except in the lower skin at the spar cap where the compressive
stress is higher for the Mach 5.28 flight.

Structural Problem Areas

Some of the areas of the X-15 where structural problems have
developed on the fuselage and wing as a result of heating or thermal
stresses are shown in figure 8. They include the side fairings, nose-
gear compartment, canopy seal, canopy glass, and wing leading edge.

Skin buckling.- The first temperature problem occurred on the side-
fairing panels along the liquid-oxygen (lox) tank before the X-15 was
flown. Pronounced elastic buckles appeared in the panels as a result
of tank contraction when the tank was filled for the first time. The
buckling was relieved by adding a 1/8-inch expansion joint to the tunnel
fairing near the wing leading edge.

After the flight on March 7, 1961, in which a Mach number of 4.43
was reached, several permanent buckles were formed in the outer sheet
of the fairing between the corrugations near the edge of a panel. Since
these fairing panels are required to carry local air loads only, these
buckles did not seriously affect the structural integrity. The maximum
temperatures measured on the side fairing during this flight are shown
in figure 9 for two fuselage stations in the area of the liquid-oxygen
tank. The insert is a photograph of a typical buckle in the fairing
panel. This buckle occurred near the wing leading edge on the left side
of the airplane. The scales help show the extent of the buckle; the
depth of the buckle is about 1/4 inch. At fuselage station 202, just
forward of the tank, the temperature was 590° F on the lower fairing and
at fuselage station 335, just behind the tank, the temperature was 480° F.
No measurements were available on the tank. The temperatures shown
occurred after engine shutdown, which, on this flight, left about
20 percent of the fuel still in the tanks. The cold tank, about -260° F
for liquid oxygen, together with the high skin temperatures on the
fairings resulted in large gradients, and hence the buckles. The
important results found were that these thermal gradients between the
liquid-oxygen tank and the fairings were actually higher than calculated
for the original design. The design was based on complete fuel burnout
before the maximum skin temperatures were encountered. As a result of
this experience, four expansion joints were installed in the fairing
forward of the wing to give a total expansion capacity of slightly
over 1 inch. To date, this modification has prevented additional
permanent buckles in the fairing for similar flights to higher temper-
atures.

The surface irregularities produced by the buckles were expected to cause local hot spots on high-speed flights. As a check on this effect, the buckle areas were painted with temperature-sensitive paint for the flight to a Mach number of 4.6. The results showed that the maximum temperature in the buckle area was essentially the same as in other areas on the panel, and there was no evidence of local hot spots.

Damage from airflow into the structure.- Another heating problem that has developed on the X-15 is caused by airflow into the interior of the structure. This flow has resulted in unexpected high temperatures around the speed-brake actuators, and loss of instrumentation wires at the roots of the wing and tail surfaces. On the forward fuselage, the seal of the canopy has been damaged because of a slight raising of the front edge by cabin pressure, which allowed hot air to flow against the seal. This problem has been solved by attaching a shingle-type strip to the fuselage just ahead of the canopy joint to prevent airflow under the edge of the canopy. A similar problem has developed in the nose-gear compartment. The small gap at the rear end of the nose-gear door was sufficiently large to allow the airstream to enter the compartment and strike the bulkhead between the nose-gear compartment and the cockpit. This stream caused a local hot spot. Aluminum tubing, for the pressure-measuring system, is attached to this bulkhead in the nose-gear compartment, and during a flight to a Mach number of 5.2, portions of the tubing melted away. A photograph of this damage is shown in figure 10. Pictured are the aluminum tubing with the damaged area and the titanium bulkhead between the nose gear and pilot compartment. The bulkhead was heated to about 530° F, which was sufficiently high to scorch the paint on the bulkhead in the pilot's compartment and cause smoke in the cockpit. Since this flight, an Inconel compression seal has been added to the rear end of the nose-gear door and additional protection has been provided by placing a baffle plate across the compartment just behind the door opening.

Glass failure.- The windshield glass originally installed on the X-15 was soda-lime tempered plate glass. This choice was based on a predicted maximum temperature of 740° F. Data obtained on early flights indicated that outer-face temperatures near 1,000° F could be expected with a differential temperature between faces of 750° F. It was apparent that soda-lime glass would not withstand these temperatures and that alumino-silicate glass should be a satisfactory replacement. The alumino-silicate glass has higher strength and better thermal properties which reduce the expected temperature and gradients to about 70 percent of those predicted for soda-lime glass. The alumino-silicate glass withstood thermal tests to temperatures which were about 1.5 times greater than the expected flight values. Subsequently, the alumino-silicate glass was installed in all three X-15 airplanes; however, one of the soda-lime windshields was inadvertently installed at a later date, and it fractured during recovery from an altitude flight to

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217,000 feet. On a speed flight to a Mach number of 6.04, one of the alumino-silicate glass panels also fractured. In both cases the glass fragments remained in place during the remainder of the flight. Photographs of the fractured panels are shown in figures 11 and 12. Figure 11 shows the soda-lime glass and figure 12 the alumino-silicate glass. The fracture pattern in figure 11 is not typical of tempered glass, but the pattern in figure 12 is typical. In both cases, the retainer frame buckled near the center of the upper edge of the glass and created a local hot spot at this point. Failure of both glass panels started adjacent to this buckle. Subsequent to the last failure, the retainer was changed from 0.050-inch-thick Inconel X to titanium that is 0.10 inch thick in order to eliminate buckling and, hence, the local hot spot.

Damage to wing leading edge.- Structural problems have developed on the wing leading edge because of thermal gradients and local hot spots not detected by the thermocouples. In order to study the overall temperature levels on the wing structure, the temperature-sensitive paint discussed previously has been used. Paint is applied to the surface of the wing and tail before a flight. After the flight, the color changes and patterns are examined to determine gross skin temperature. Figure 13 is a photograph showing the paint patterns on the bottom wing surface after one of the early flights. This figure shows the wing lower surface, the fuselage with frost on the liquid-oxygen tank, the wing leading edge, and the wing tip. The "fence" on top of the wing is actually the tip of the vertical stabilizer. The light areas on the wing surface reached maximum temperatures between 250° F and 400° F. The dark areas represent temperatures above 400° F. The heat sink of the internal structure is clearly seen. Note the wedge-shaped dark areas of high temperature that start at four points on the leading edge and extend back over the wing. These areas begin at the expansion joints in the leading-edge heat sink. On the first flight above a Mach number of 5, these areas of local heating were much more pronounced.

The temperature distributions in the vicinity of these slots, on a flight to a Mach number of 5.3, are shown in figure 14. These data were obtained from the paint pattern since no thermocouples were located in this region; however, the paint colors obtained were correlated with thermocouple data at other points on the wing. This figure shows a segment of the wing leading edge, the expansion joint, and a section of the lower skin. The expansion joints are slots about 0.080-inch-wide cut in the heat sink. The average leading-edge temperature was 830° F and just outboard of the slot on the leading edge is a small area with temperatures above 1,000° F. An area between 970° F and 1,000° F extends rearward on the skin about 8 inches, and the average skin temperature away from the slot is below 800° F.

On this flight, permanent interrivet buckles were formed directly behind the three outboard slots of the leading edge. The type of buckle

and the location are illustrated by the upper sketch in figure 15. This sketch shows a portion of the leading-edge heat sink, the expansion slot, the external skin with the buckle, and the fastener location. Note that the fastener spacing directly behind the slot is wider than the spacing along the solid portion of the leading edge. Subsequent analysis of the leading-edge structure has indicated that several factors contributed to the permanent buckling of the skin. One factor is the thermal stresses in the skin caused by the high gradients around the local hot spot. Another factor is the wide fastener spacing through the leading edge at the slot. A third reason for the buckles is the fact that the original segmentation of the leading-edge heat sink did not adequately relieve the thermal-induced compression loads. The skins at the slots acted as a splice plate for the heat-sink bar and thus were buckled in compression.

In order to minimize this buckling problem, three design changes have been made. Two of the changes are shown in the lower sketch. An 0.008-inch-thick Inconel tab welded along one edge was installed over each slot to prevent tripping the boundary layer, and thus to minimize the local hot spots, as explained in reference 1. A fastener was added at the slot to decrease the fastener spacing and to increase the allowable skin buckling stress. In order to reduce the load that the skin splice must carry at each slot, the third change was to add expansion slots with cover tabs in three of the outboard segments of the leading edge.

No additional damage has occurred at the original slots; however, the original slots had a shear tie to prevent relative displacement of the leading-edge segments, whereas shear ties at the new slots could not be provided without costly rework at the structure. A structural analysis showed that sufficient shear stiffness was present in the leading edge to meet the design requirements without shear ties, but relative displacement of the leading-edge segments was expected at the new slots. The extent of this relative displacement during the last speed flight is shown in figure 16. This photograph shows the leading-edge segments and the cover tab at one slot. Examination of the deformed cover tab and the wing skins indicates the magnitude of this displacement, which was over 1/8 inch at this slot. Modifications to the leading-edge structure which are under consideration include the addition of shear ties at the new slots.

Wing Isotherms

Overall temperature on the wing upper surface obtained during the speed flight to a Mach number of 5.30 is shown by the isotherms in figure 17. These isotherms were obtained from the color patterns of the temperature-sensitive paint. The term "isotherm" is used in a broad sense, since all color changes do not occur at exactly the same time.

It can be seen that the region near the wing root is less than 380° F
and along the leading edge several regions are above 750° F. Small
areas of high temperature directly behind the expansion joints are still
present, even with the cover tabs, but the affected area is much smaller
and does not extend forward to the leading edge. Similar applications
of the temperature-sensitive paint have been used on other areas of the
X-15 to obtain qualitative measurements of the temperatures. These
areas include the cockpit canopy, horizontal and vertical tails, speed
brakes, fuselage nose area, and protuberances such as probes, antennas,
and vent lines.

CONCLUDING REMARKS

Maximum temperatures measured on the X-15 airplane show that speeds
in excess of a Mach number of 6 have been accomplished without extreme
structural temperatures. Comparison of calculated and measured internal
temperatures has shown that satisfactory thermal gradients through the
structure can be calculated from known heat input to the exposed
surfaces. In general, the hot-structure concept used for the primary
structure of the X-15 airplane has proven quite satisfactory. However,
structural problems have developed during the flight program as a result
of local hot spots and discontinuities in the structural elements.
Many of these problems pertain to the X-15 only; however, thermal
problems with windshield glass, airflow through openings in the
external structure, and structural discontinuities are expected to
appear on all hypersonic vehicles until adequate design information is
available in these problem areas.

Flight Research Center
    National Aeronautics and Space Administration
        Edwards, Calif., November 20, 1961

REFERENCE

1. Banner, Richard D., Kuhl, Albert E., and Quinn, Robert D.: Prelim-
        inary Results of Aerodynamic Heating Studies on the X-15 Airplane.
        NASA TM X-638, 1962.

X—15 STRUCTURE
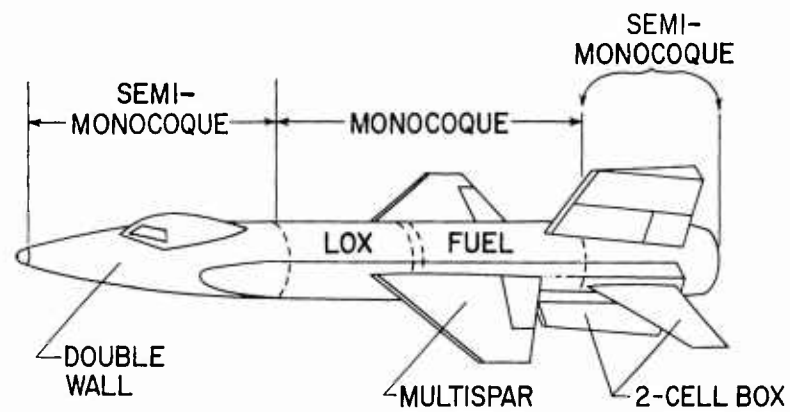
Figure 1

STRUCTURAL DETAILS
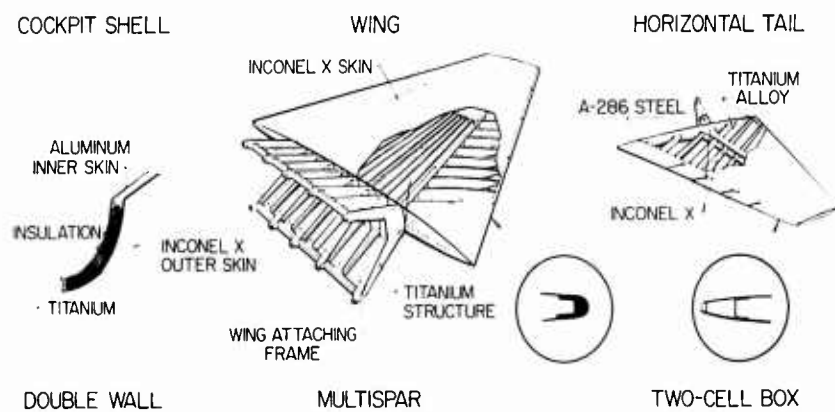
Figure 2

MAXIMUM SKIN TEMPERATURES
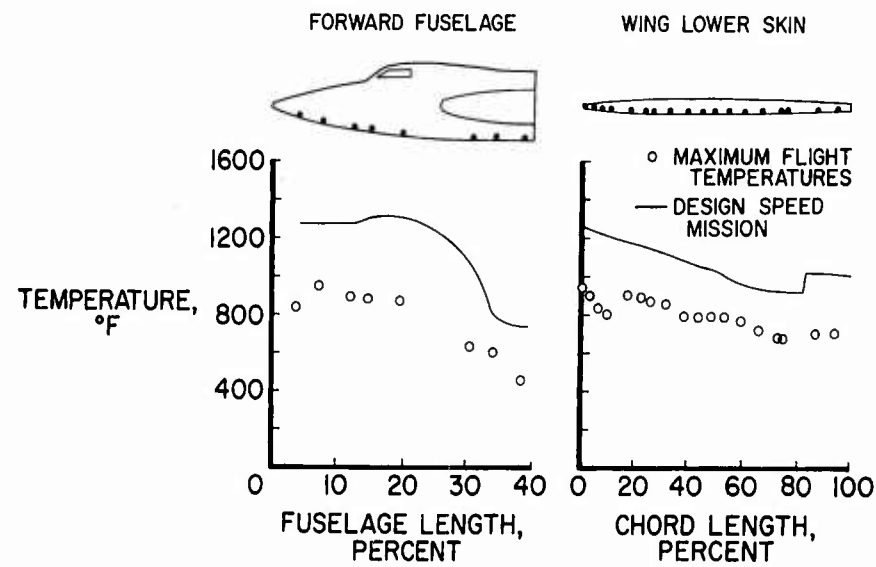
Figure 3

SUMMARY OF MAXIMUM TEMPERATURES
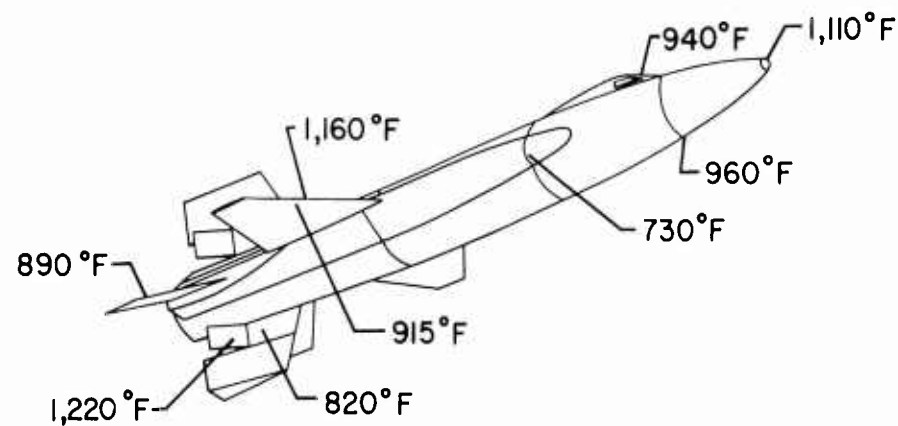
Figure 4

WING–SPAR TEMPERATURES DURING
FLIGHT TO MACH NUMBER OF 5.28
FRONT SPAR MIDSEMISPAN
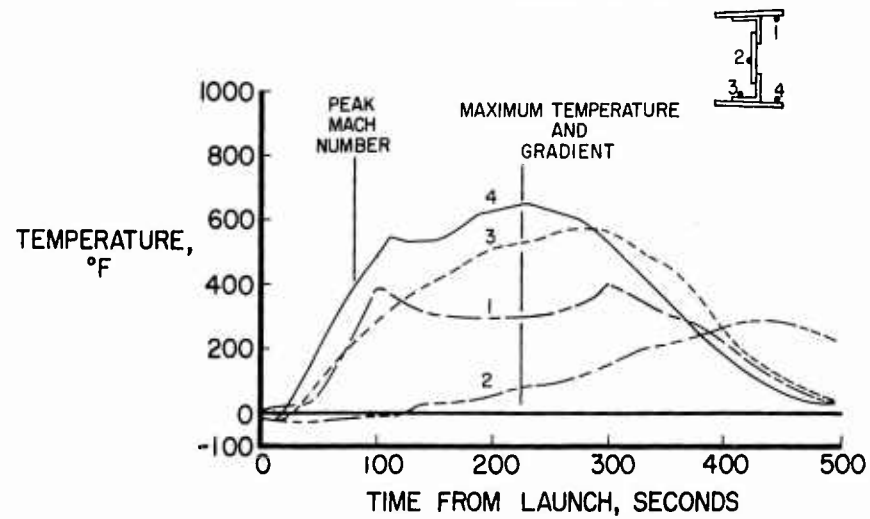
Figure 5

CALCULATED SPAR TEMPERATURES AT
TIME = 225 SECONDS

FLIGHT TO MACH NUMBER OF 5.28

Figure 6

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MAXIMUM THERMAL STRESS
## FRONT SPAR MIDSEMISPAN

STRESS, LB/SQ IN.
SPAR

——— FLIGHT TO MACH NUMBER OF 5.28

- - - - DESIGN SPEED MISSION

STRESS, LB/SQ IN.

LOWER SKIN
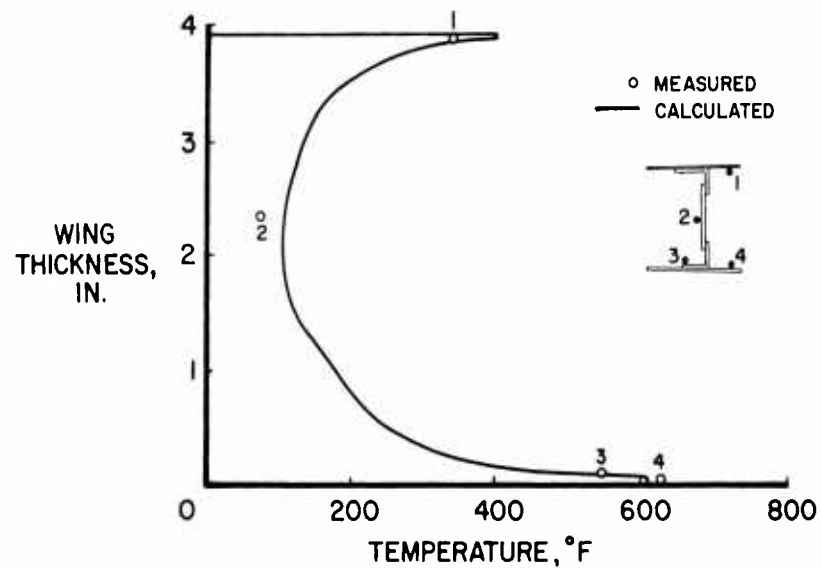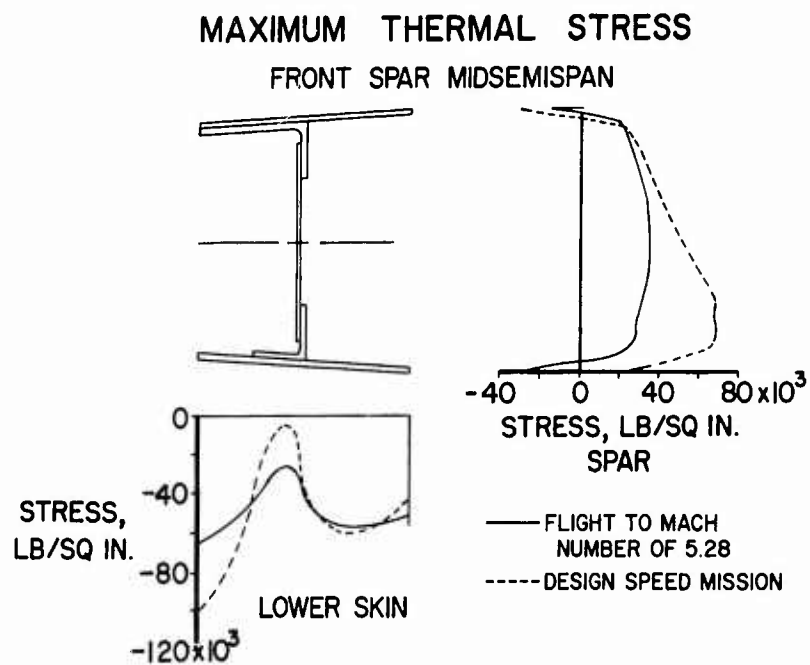
Figure 7

THERMAL PROBLEM AREAS
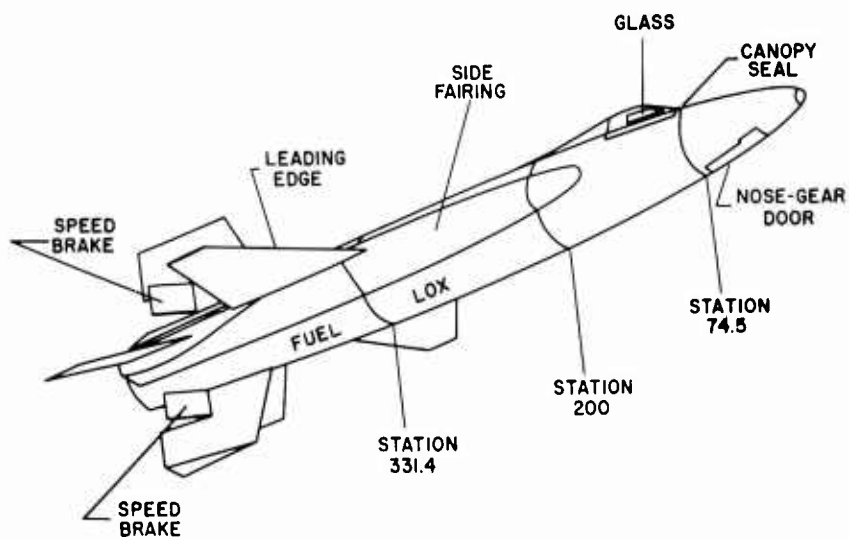
Figure 8

MAXIMUM SIDE-FAIRING TEMPERATURES DURING FLIGHT TO MACH NUMBER OF 4.43
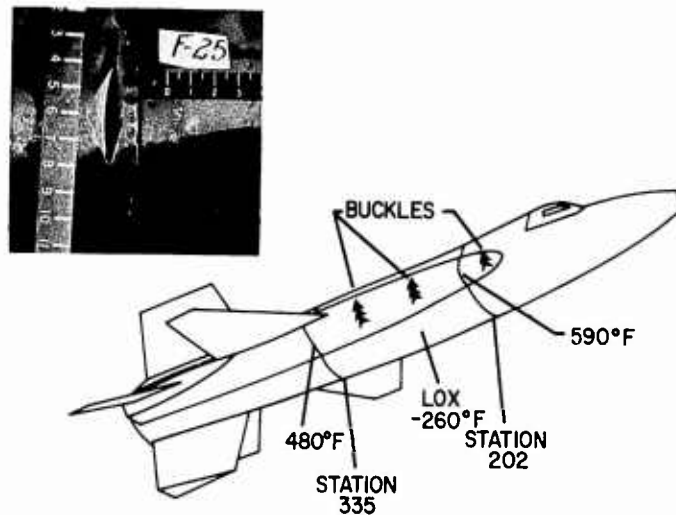
Figure 9

HEAT DAMAGE TO ALUMINUM TUBING

Figure 10

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DAMAGED WINDSHIELD GLASS FOLLOWING FLIGHT TO 217,000 FEET ALTITUDE

Figure 11

DAMAGED WINDSHIELD GLASS FOLLOWING FLIGHT TO MACH NUMBER OF 6.04
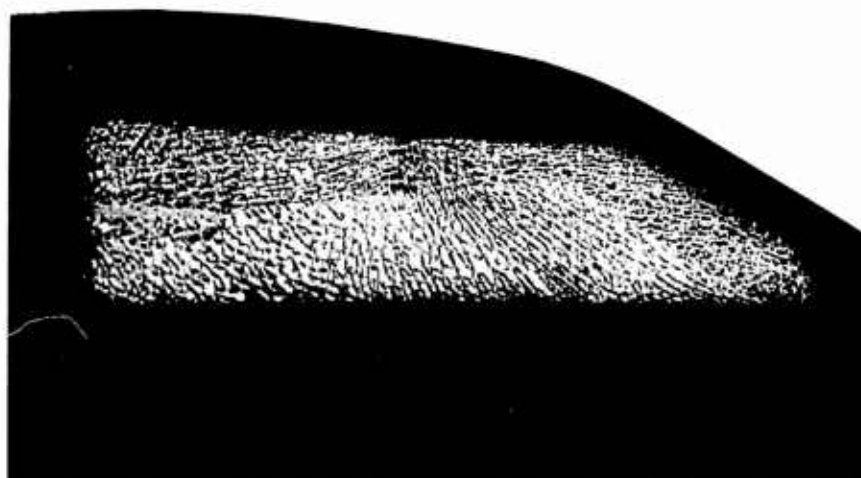
Figure 12

TEMPERATURE-SENSITIVE-PAINT PATTERNS
ON X-15 WING
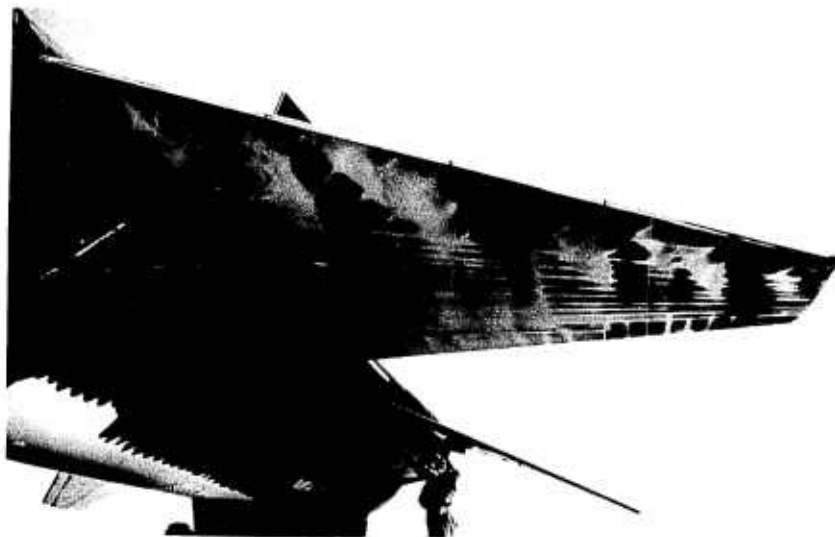
Figure 13

TEMPERATURE DISTRIBUTION AFT OF
LEADING-EDGE EXPANSION SLOT
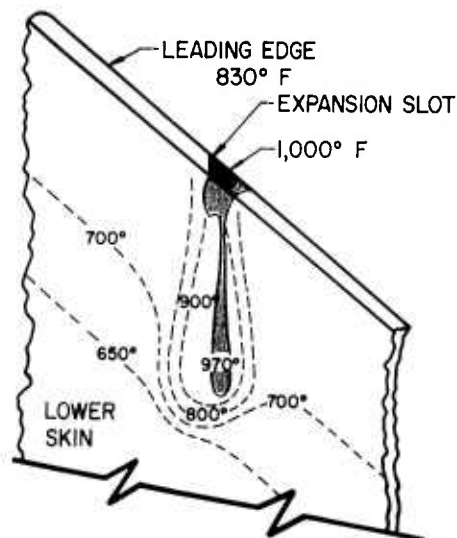
Figure 14

WING SKIN BUCKLE FOLLOWING FLIGHT FLIGHT TO MACH NUMBER OF 5.28
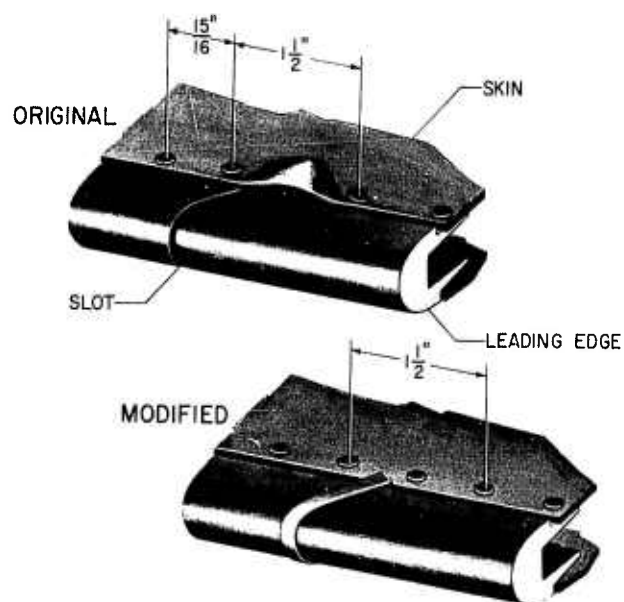
Figure 15

LEADING-EDGE DISPLACEMENT AT NEW EXPANSION SLOTS DURING FLIGHT TO MACH NUMBER OF 6.04

Figure 16

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WING ISOTHERMS FOLLOWING FLIGHT
TO MACH NUMBER OF 5.30

TEMPERATURE, °F
------- 750°
———— 570°
—·—·— 380°

Figure 17

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affected areas to eliminate the problems are discussed.

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