NATIONAL ADVISORY COMMITTEE FOR AERONAUTICS

WARTIME REPORT

ORIGINALLY ISSUED
November 1943 as
Advance Restricted Report 3K10

FLIGHT TESTS OF THERMAL ICE-PREVENTION EQUIPMENT ON A LOCKHEED 12A AIRPLANE

By Richard Scherrer

Ames Aeronautical Laboratory
Moffett Field, California

NACA

WASHINGTON

NATIONAL ADVISORY COMMITTEE FOR AERONAUTICS

ADVANCE RESTRICTED REPORT

FLIGHT TESTS OF THERMAL ICE-PREVENTION EQUIPMENT ON A LOCKHEED 12A AIRPLANE

By Richard Scherrer

SUMMARY

The development of thermal ice-prevention equipment by the National Advisory Committee for Aeronautics has been continued by the application of a heated-air ice-prevention system to a Lockheed 12A airplane. The test data indicate that thermal ice-prevention equipment is practical and that the method of utilizing the exhaust gases as a source of heat is satisfactory.

The design tested was satisfactory in all the icing conditions encountered, but some parts of the design provided only marginal protection in the more severe conditions.

As a result of many flights in icing conditions it is apparent that aerodynamic limitations to extended flight in inclement weather can be eliminated by the use of thermal ice-prevention equipment combined with the removal of all protuberances from the airplane surfaces.

INTRODUCTION

The development of thermal ice-prevention equipment by the NACA has been continued by the application of a heated-air ice-prevention system to a Lockheed 12A airplane.

This project was a continuation of the work reported in references 1, 2, and 3. The results reported in reference 1 demonstrated the practicability of ice prevention on airplanes by the use of heat. The design discussed in reference 1 used a combination of convective and radiant heating from an engine exhaust tube enclosed within the wing leading edge, and an air-heated empennage. The design analysis of reference 2 indicated that an ice-prevention system using heated air for the wing and the windshield, as well as the empennage, would be practical. The present heated-air distribution system in the wing of the Lockheed 12A airplane is similar to that used in the Consolidated B-24D airplane described in reference 2, and the basis for the wing design was discussed in reference 3. A description of the ice-prevention system of a Junkers 88 airplane, which is similar to that used on the Lockheed 12A airplane, is given in reference 4.

The construction and preliminary flight tests of the ice-prevention system for the Lockheed 12A airplane were conducted at the Ames Aeronautical Laboratory, Moffett Field, Calif. The flight tests in icing conditions were conducted at the NACA Ice Research Project, Minneapolis, Minn., during the period from January 10 to March 28, 1943.

EQUIPMENT AND INSTALLATION

The NACA Lockheed 12A airplane on which the ice-prevention research has been conducted, is shown in figure 1. The ice-prevention equipment installed in the airplane consisted of air-heated wing leading edges, horizontal stabilizer, center fin, and windshield. The system also provided air for cabin heating. The heated portion of the wing leading edge extended from wing station 123, which corresponds to the wing panel splice, to the wing tip. The air-heated center vertical fin (added by the NACA), horizontal stabilizer, and windshield installations were the same as shown in figures 3, 4, and 5 of reference 1.

The heated air for the thermal ice-prevention equipment was obtained from cross-flow heat exchangers in the exhaust system of each engine. The heat exchangers are shown in figure 2 and were designed and fabricated by the NACA from Alcoa 122 aluminum alloy. The notch in the external fins shown in figure 2 provided for a baffle separating the incoming and outgoing air streams. The heat exchanger as shown weighed 25 pounds. This heat exchanger, design No. 22 in the series being tested at AAL, was designed to transfer 100,000 Btu per hour with an air-flow rate of 1600 pounds per hour and a flight speed of 150 miles per hour in the Lockheed 12A airplane.

The external fin design of the heat exchanger was based on reference 5 and the internal heat transfer was computed by a method for gas flow in narrow gaps (reference 6).

The exhaust-gas-to-air heat-exchanger assemblies were each incorporated in two units, the cast-aluminum heat exchanger and a double-tube heat exchanger. The cast-aluminum heat exchangers provided heated air for the wings and the empennage; while the double-tube heat exchangers provided for cabin heating and windshield ice prevention. The heat-exchanger assembly is shown in figures 3(a) and 3(b).

The air circulation through the exchanger and the air outlets to the wing and empennage leading edges is shown in figure 3(b). The exchanger assembly is shown mounted in the engine exhaust system in figure 4. The air inlet and the complete installation enclosed in the nacelle and fairing are shown in figure 5.

The wing-leading-edge air-distribution system is shown in various stages of assembly in figure 6. Figure 6(a) shows the leading-edge skin assembled with the chordwise corrugations and ribs. The heated air was ducted from the heat exchanger, through a discharge-valve assembly, into the leading-edge duct which is formed by the assembly shown in figure 6(a) and a baffle, which is shown attached to the wing structure in figure 6(b). The air was distributed spanwise in the D-shape leading-edge duct and then flowed chordwise through the gaps formed by the corrugations. (See section A-A, fig. 7.) The complete wing panel is shown in figure 6(c). Thin aluminum-alloy angles were spot-welded to the corrugations, at the air outlets, between the corrugated inner skin and the outer skin. Adjustments were made to the air distribution by bending the protruding strips over the air-outlet openings. (See section A-A, fig. 7.) Air passed from the corrugations into the interior of the wing and, with the exception of that drawn off at the wing tips, was discharged at the aileron gap. Heated air for the wing tip was drawn from the interior of the wing through the gap between the leading-edge double skins by the suction at the air outlet on the upper surface, as shown in figure 7, section B-B.

The modifications to the wing structure added 26 pounds to the weight of the airplane. The structure of the Lockheed 12A wing consists of a single main spar, a rear shear beam, and a stressed skin and leading edge.

This structure was not changed by the provisions for heating the leading edge. The addition of the inner corrugated skin and baffle plate was considered to more than compensate for the reduction in strength of the material due to the elevated temperatures at which the leading edge was to operate.

A schematic diagram of the duct system installed in the airplane also is shown in figure 7. Seven valves shown in the figure were provided in the duct system to vary the heated-air distribution. Valves were installed at the outlets to the heat exchangers in order to discharge the heated air directly into the air stream when heating is not required. A valve was provided in the empennage duct to adjust the quantity of air flowing to that part. Two valves were provided in the windshield duct; one was operated the the copilot to adjust the air flow to the windshield and the other by the observer to regulate the amount of heated air flowing into the cabin. In severe icing conditions the cabin-heat valve can be shut, thereby providing greater protection for the windshield. The windshield and empennage designs were the same as reported in reference 1 and were supplied with similar heat quantities. The gap between the inner and outer leading-edge skins was reduced from about 3/16 inch to an average of 1/8 inch to improve the internal heat transfer.

The temperature- and pressure-recording instruments installed in the airplane cabin are shown in figure 8. Temperatures were measured at 56 points in the wings and the empennage, and pressures were measured at 20 points. The thermocouples and the pressure tubes were connected to recording instruments by an electric-motor-driven selector-switch-and-valve assembly. The temperatures were recorded on film by two recording galvanometers and the pressures were recorded by an NACA-type two-cell airspeed recorder. The locations of the thermocouples are shown in figures 9 and 10. Both the free-stream total and static pressures were recorded as reference pressures. The rate of air flow through the heat exchangers was evaluated from a calibration of the static pressure drop. The calibrations were made in flight with venturi meters over the normal range of fuel-air ratios and air-flow rates. The rate of flow to the empennage was measured with a venturi meter in the duct.

The air-inlet, air-outlet, and exhaust-gas temperatures were read on a millivoltmeter calibrated in degrees

Fahrenheit and shown on the observer's instrument panel in figure 11. The exhaust-gas thermocouples were unshielded and no corrections were made for radiation effects.

The service airspeed head on the airplane is located on a mast extending down from the fuselage at a point approximately 2 feet aft of the nose. Difficulty was experienced with this installation in icing conditions because ice accretions on the support mast affected the static pressure at the airspeed head. In order to eliminate this difficulty, two static-pressure orifices were installed in the surface of the rear portion of the fuselage to obtain a more consistent and exact static-pressure measurement under all conditions.

TESTS AND RESULTS

The flight tests in natural icing conditions were planned with the cooperation of the U. S. Weather Bureau and the Northwest Airlines dispatch office. After flying into the region of the icing conditions, a preliminary survey of the vertical and horizontal extent of the icing region was made, recording ambient-air temperature, altitude, time, and observations on the severity of the icing conditions. The flight was then continued in the icing condition selected.

During the icing flights, records were taken of the temperatures and the pressures in the ice-prevention system. Photographs and notes of ice formations and meteorological conditions that were encountered were used as a basis for changes to the ice-prevention system during the progress of the tests.

The thermal data are presented in tables I and II and in figures 12 and 13. The conditions of each flight, altitude, speed, air temperature, and type and approximate rate of icing are given in table I. The approximate rate of icing is noted as a basis of comparison of the severity of the icing conditions encountered. The icing rates noted are not believed to have any meteorological significance since the icing rate varies with airspeed and the contour of the surface on which the ice forms. Table II lists the wing and empennage temperatures. The heat-exchanger performance curves derived from data which were taken during the icing and preliminary flights are given in figure 12. The wing-surface temperature rise at stations 127, 200, and 272 for various conditions is shown in figures 13(a) to 13(i). The photographs of ice on the airplane (figs. 14 to 21) indicate the types and the extent of the ice accretions.

The average heat input to the right panel for all the flight tests was 89,500 Btu per hour; while the average input to the empennage was 22,000 Btu per hour. The average heat loss through the wing leading-edge skin was 36,000 Btu per hour in clear air and 43,500 Btu per hour in icing conditions. The average drop in the temperature of the air as it passed through the corrugations was 125° F in clear air and 152° F in icing conditions. The air temperatures at the corrugation outlets indicated that the air distribution was satisfactory with air-flow rates of 1400 to 1500 pounds per hour, but became irregular when the air flow was reduced to approximately 900 pounds per hour. The average wing-surface temperature rise was approximately 130° F in clear air and varied from 90° to 130° F in icing conditions. The thermal data given in tables I and II for the empennage system supplement those previously reported in reference 1

Figures 14 and 15 show ice accretions which formed at the ends of the leading edge of the horizontal stabilizer when the air-outlet gap between the inner and outer skins was 1/16 inch. These formations occurred at the same location on both the right and left stabilizer leading edges. The air-outlet gap then was increased to 1/8 inch at the ends so as to increase the air flow through the gaps in the region where the ice had formed. This increased air flow prevented ice formations on the leading edge in succeeding flights.

During an initial flight in icing conditions the ice-prevention system was turned off and ice allowed to form before the heat was turned on again. Figure 16 shows the wing leading edge during the ice-removal process. Approximately 2 minutes were required to remove all ice accretions. The heating of the wing tips was not sufficient to prevent ice at air temperatures below 25° F. A typical ice formation on the wing tip is shown in figure 17.

Freezing of water which had been prevented from forming ice at the stabilizer leading edge occurred under some conditions on the upper surface, as shown in figure 18.

Figures 19 and 20 show the results of flying in a freezing rain with a free-air temperature of 19° F. Ice formed on the pilot's windshield, as shown in figure 19, after which the heated air to the cabin was shut off, directing all the heat to the windshield. With this condition, the ice was removed. The removal process was quite gradual, indicating that a critical condition for the windshield ice-prevention system had been approached. The stabilizer collected ice just aft of the leading edge, as shown in figure 20, during the same flight. The ice formed in small sheets up to 1/4 inch thick and intermittently blew away. Ice did not form on the leading edge of the horizontal stabilizer. The combination of the relatively large supercooled water drops required for rain and the ambient-air temperature (19° F) indicates that the icing condition during this flight was quite severe.

The heat-exchanger performance was in good agreement with the design values, as shown in figure 12. The heat exchangers were inspected several times during the flight operations at the Ice Research Project in order to determine if there had been any deterioration of the aluminum castings. The first inspection indicated that there had been some erosion and melting of the tips of the internal fins by the exhaust gases. Later inspections, after more than 50 hours of service, showed that there had been little or no additional deterioration of the internal fins.

DISCUSSION

The results obtained with the thermal ice-prevention equipment installed on the Lockheed 12A airplane are in agreement with those reported in reference 1 and with tests of more recent installations in other airplanes. No difficulty was experienced with the operation of the thermal ice-prevention equipment and the flight tests were conducted safely in all the conditions encountered.

The heat supplied to the wing panels was approximately 2000 Btu per hour per square foot in dry air and 2600 Btu per hour per square foot in icing conditions. In tests with an XB-24F airplane, approximately 40,000 Btu per hour were transferred through 45 square feet of surface area with an air flow of 1600 pounds per hour in icing conditions; while in the tests with the Lockheed 12A airplane 43,500 Btu per hour were transferred through 16.5 square feet with 1200 pounds per hour of air in similar conditions. Even with the large heat input, the wing did not have a comparably high average surface temperature. The average wing-surface temperature rise, recorded in the XB-24F airplane tests, was 100° F in clear air and 80° to 105° F in icing conditions; while the temperatures for the Lockheed 12A airplane were 130° F and 90° to 130° F, respectively, with almost three times the unit heat input.

The relatively large heat quantity transferred, per unit area, to the wing skin of the Lockheed 12A airplane was due to the large mass flow through the corrugations with consequent high internal heat-transfer coefficients. The wing-surface temperatures did not increase in the same ratio as the unit heat quantities transferred to the skin, probably because the external heat transfer was greater than on other designs.

The boundary-layer characteristics of the airfoil sections govern the rate of external heat transfer. The NACA 230-series sections used on the Lockheed 12A airplane normally would have less chordwise extent of the laminar boundary layer than the Davis sections used on the XB-24F airplane. The early onset of turbulent flow would tend to increase the external heat transfer. The increase in external heat transfer could have been caused also by premature transition due to surface irregularities that were apparent.

The fact that more heat per unit area was necessary to maintain the wing surface at temperatures only slightly above those noted in the XB-24F tests indicate that the use of a fixed quantity of heat per unit area protected may have definite limitations as a design criterion.

A comparison of the heat quantities and skin temperatures of the wing and empennage indicates that a greater portion of the total output from the heat exchangers could have been directed to the empennage and would have improved the performance of these parts.

The importance of having an airplane that is to be flown in inclement weather aerodynamically clean cannot be overstressed. It has been noted during many flights in icing conditions that ice will form on any object extending from the surface of the airplane. Figure 21 shows small ice accretions on the heads of 3/32-inch-diameter brazierhead rivets. Other objects on the Lockheed 12A airplane which normally collected ice were the radio antennas and antenna masts, pitot-static head mast, D-F loop housings, door handles, cabin cold-air scoops, tail wheel, and navigation lights. Ice accretions on these and other protuberances cause large loads and aerodynamic disturbances.

CONCLUSIONS

1. The thermal ice-prevention equipment in the Lockheed 12A airplane was satisfactory in all the icing conditions encountered and confirmed the practicability of heated-air systems using the exhaust gases as a source of heat.

2. Aerodynamic limitations to extended flight in icing conditions can be eliminated by the use of thermal ice-prevention equipment combined with the elimination of all protuberances from the surfaces of the airplane.

Ames Aeronautical Laboratory,
National Advisory Committee for Aeronautics,
Moffett Field, Calif.

REFERENCES

1. Rodert, Lewis A., Clousing, Lawrence A., and McAvoy, William H.: Recent Flight Research on Ice Prevention. NACA A.R.R., Jan. 1942.

2. Jones, Alun R., and Rodert, Lewis A.: Development of Thermal Ice-Prevention Equipment for the B-24D Airplane. NACA A.C.R., Feb. 1943. (Classification changed to "Restricted" Aug. 1943.)

3. Rodert, Lewis A., and Jackson, Richard: Preliminary Investigation and Design of an Air-Heated Wing for Lockheed 12A Airplane. NACA A.R.R., April 1942.

4. Rodert, Lewis A., and Jackson, Richard: A Description of the Ju 88 Airplane Anti-Icing Equipment. NACA R.B., Sept. 1942.

5. Biermann, Arnold E., and Ellerbrock, Herman H., Jr.: The Design of Fins for Air-Cooled Cylinders. Rep. No. 726, NACA, 1941.

6. McAdams, William H.: Heat Transmission. McGraw-Hill Book Co., Inc., 1942.

TABLE I.- RESULTS OF THERMAL ICE-PREVENTION EQUIPMENT TESTS WITH THE LOCKHEED 12-A (HEATER NO. 22)

Flight number	17	18	21	17	20	22	29	31	31
Run number	2	1	1	1	1	1	1	1	2
Pressure altitude, ft	1,500	2,800	5,000	2,500	4,800	3,000	3,400	3,700	3,500
Indicated airspeed, mph	152	157	144	143	130	-------	149	166	152
Free air temperature, deg F	25	24	21	25	26	19	24	12	10
Type of ice	None	None	None	Glaze	Glaze	Freezing rain	Glaze	Rime	Rime
Approx. rate of icing,[a] in./hr	-------	-------	-------	-------	5	2	1	1/2	1
Fuel-air ratio	0.080	0.080	0.080	0.080	0.080	0.080	0.080	F.R.	0.080
Exhaust-gas temperature, deg F									
Left	1,280	1,280	1,200	1,230	1,170	1,320	-------	-------	-------
Right	1,250	1,270	1,220	1,300	1,300	1,330	1,350	1,230	1,350
Air-inlet temperature, deg F	30	45	30	40	40	30	40	20	25
Air temperature rise in heat exchanger, deg F									
Left	278	275	273	280	273	318	287	286	297
Right	272	272	275	303	350	327	313	280	298
Rate of air flow, lb/hr									
Left	1,900	1,870	1,960	1,840	1,450	1,500	1,600	1,790	1,740
Right	1,720	1,720	1,825	1,610	1,050	1,300	1,200	1,870	1,460
Heat quantity, lb/hr									
Left	128,000	125,000	130,000	125,000	101,000	116,000	111,000	124,000	125,000
Right	113,000	113,000	121,000	118,200	93,000	103,000	91,300	127,000	105,000
Air temperature at tail duct venturi, deg F	250	250	240	270	280	280	270	235	250
Air temperature at empennage, deg F	170	175	170	188	193	193	190	159	180
Rate of air flow to empennage, deg F . . .	-------	-------	-------	-------	-------	-------	507	655	685
Heat loss in duct from heater to venturi, Btu/hr	-------	-------	-------	-------	-------	-------	8,620	10,800	11,640
Heat loss in duct from venturi to empennage, Btu/hr	-------	-------	-------	-------	-------	-------	9,840	12,080	11,640
Heat at empennage, Btu/hr	-------	-------	-------	-------	-------	-------	18,500	22,100	25,500
Total heat from heater to empennage, Btu/hr	-------	-------	-------	-------	-------	-------	36,960	44,980	48,780
Heat added to the right-hand wing panel, Btu/hr	[b]86,000	[b]86,000	[b]96,000	[b]95,000	[b]76,000	[b]85,000	72,820	104,510	81,110

[a]Rate of icing observed on inboard wing leading edge. [b]Interpolated values.

TABLE II.- TEMPERATURES IN THE LOCKHEED 12-A THERMAL ICE-PREVENTION EQUIPMENT

Flight number	17	18	21	17	20	22	29	31	31
Run number	2	1	1	1	1	1	1	1	2
Pressure altitude, ft	1500	2800	5000	2500	4800	3000	3400	3700	3500
Indicated airspeed, mph	152	157	144	143	130	--------	149	166	152
Free air temperature, deg F . .	25	24	21	25	26	19	24	12	10
Type of ice	----	----	----	Glaze	Glaze	Freezing rain	Glaze	Rime	Rime
Exchanger air-outlet temp., deg F	292	317	305	343	390	357	353	300	323
Air temperature out of leading-edge corrugations, deg F									
W4, sta.127, top	167	171	160	187	203	190	184	153	173
W5, sta.127, bottom	215	225	169	237	270	253	243	190	225
W36, sta.170, top	188	200	188	188	216	192	223	175	202
W37, sta.170, bottom	193	207	200	193	230	216	230	180	206
W34, sta.200, top	168	175	175	168	192	180	192	159	175
W35, sta.200, bottom	170	180	180	173	200	185	200	163	180
W38, sta.236, top	154	166	161	151	176	163	180	147	161
W39, sta.236, bottom	149	161	151	144	170	159	172	145	156
W12, sta.272, top	176	181	176	171	190	188	188	160	170
W13, sta.272, bottom	186	197	190	181	203	210	208	167	190
Wing tip air temperatures, deg.F									
W40, air inlet	137	147	140	133	151	144	161	130	137
W41, sta.274, air outlet . . .	43	50	47	48	38	38	45	40	38
W42, sta.284, air outlet . . .	38	48	45	48	38	36	43	45	35
Fin leading-edge temp. rise, deg F									
F1, sta. 4	----	----	----	----	----	--------	----	----	----
F2, sta.31	----	6	30	5	----	--------	14	23	17
F3, sta.22.5	25	44	30	19	20	3	17	28	24
F4, sta.14.5	30	46	34	23	22	5	19	33	28
Stabilizer temperatures, deg F									
S1, sta.100 L.E. skin rise . .	----	----	----	----	----	--------	----	----	----
S2, sta.100 duct air	133	135	127	138	134	135	142	122	127
S3, sta.58.5 L.E. skin rise .	48	51	52	20	32	43	38	53	45
S4, sta.59 top, skin rise . .	54	56	59	40	41	63	63	62	50
S5, sta.59 duct air	125	122	122	138	134	137	135	116	122
S6, sta.18 L.E. skin rise . .	35	38	39	28	17	36	31	43	35
S7, sta.18 duct air	170	175	170	188	193	193	190	159	170

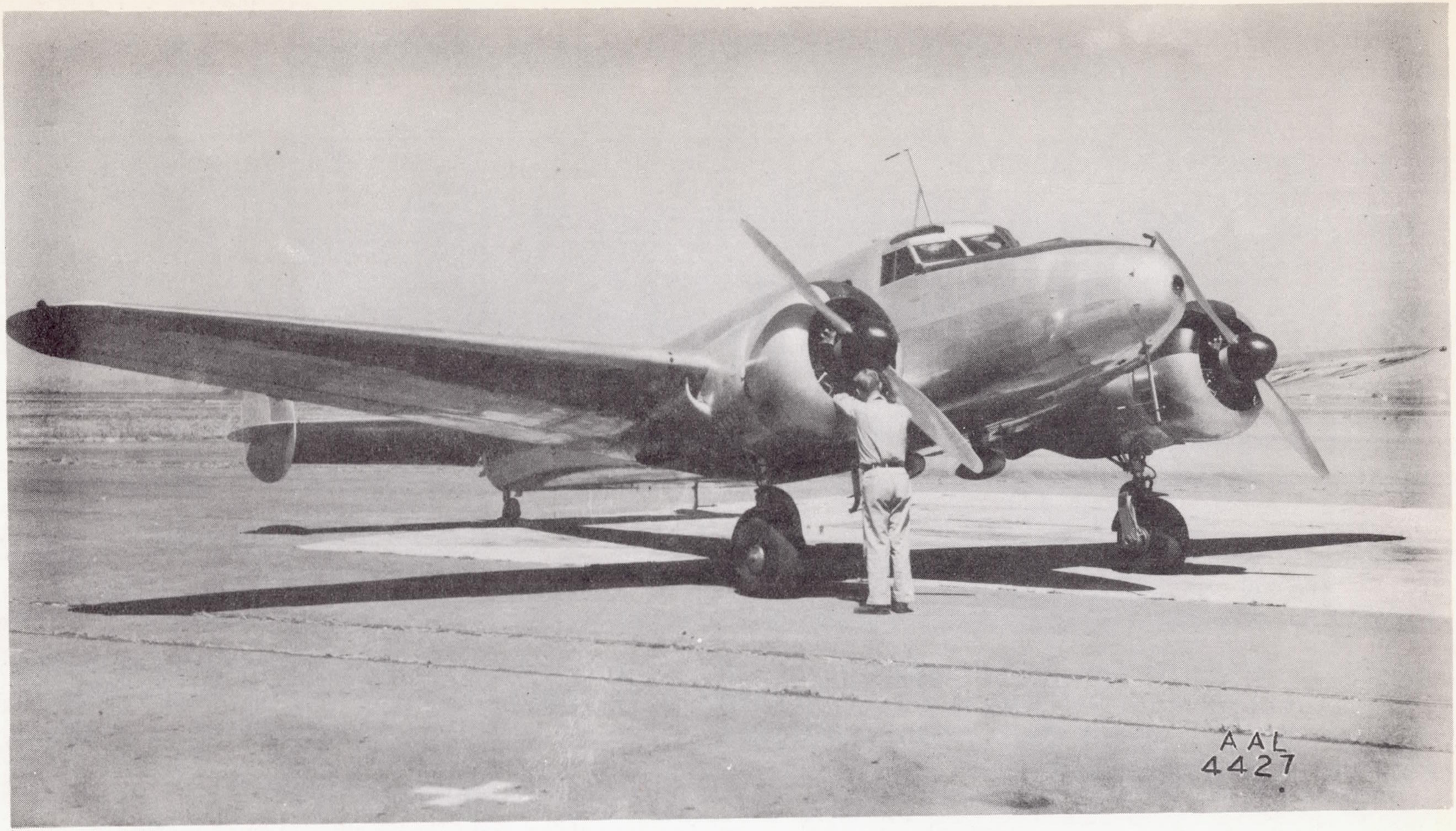
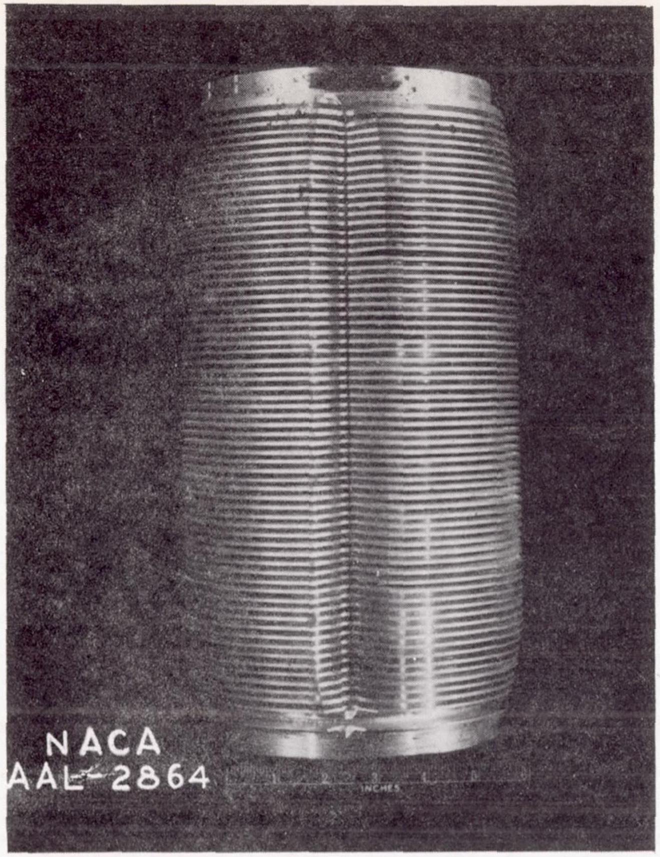
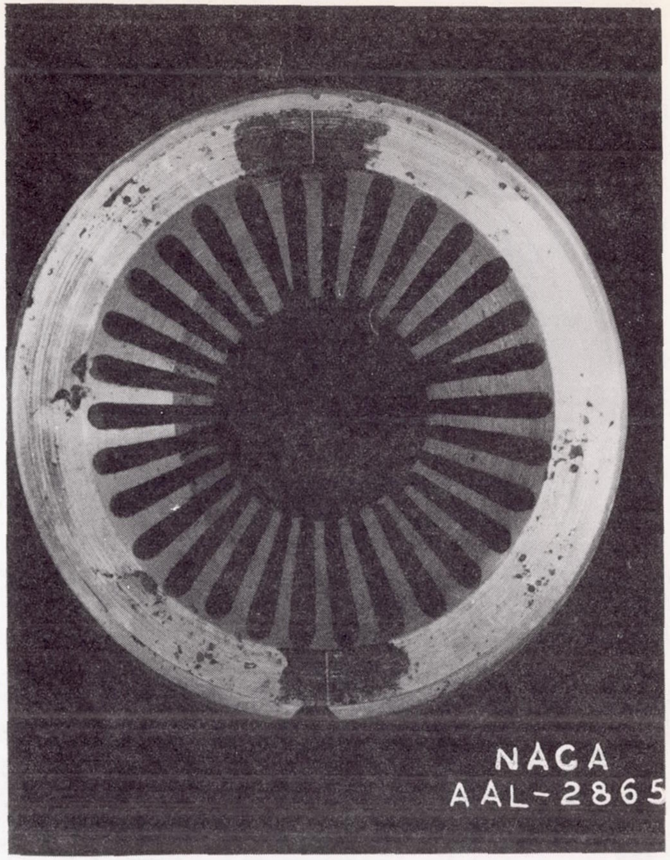


Figure 1.- The NACA Lockheed 12A airplane equipped with heated-air thermal ice-prevention equipment.

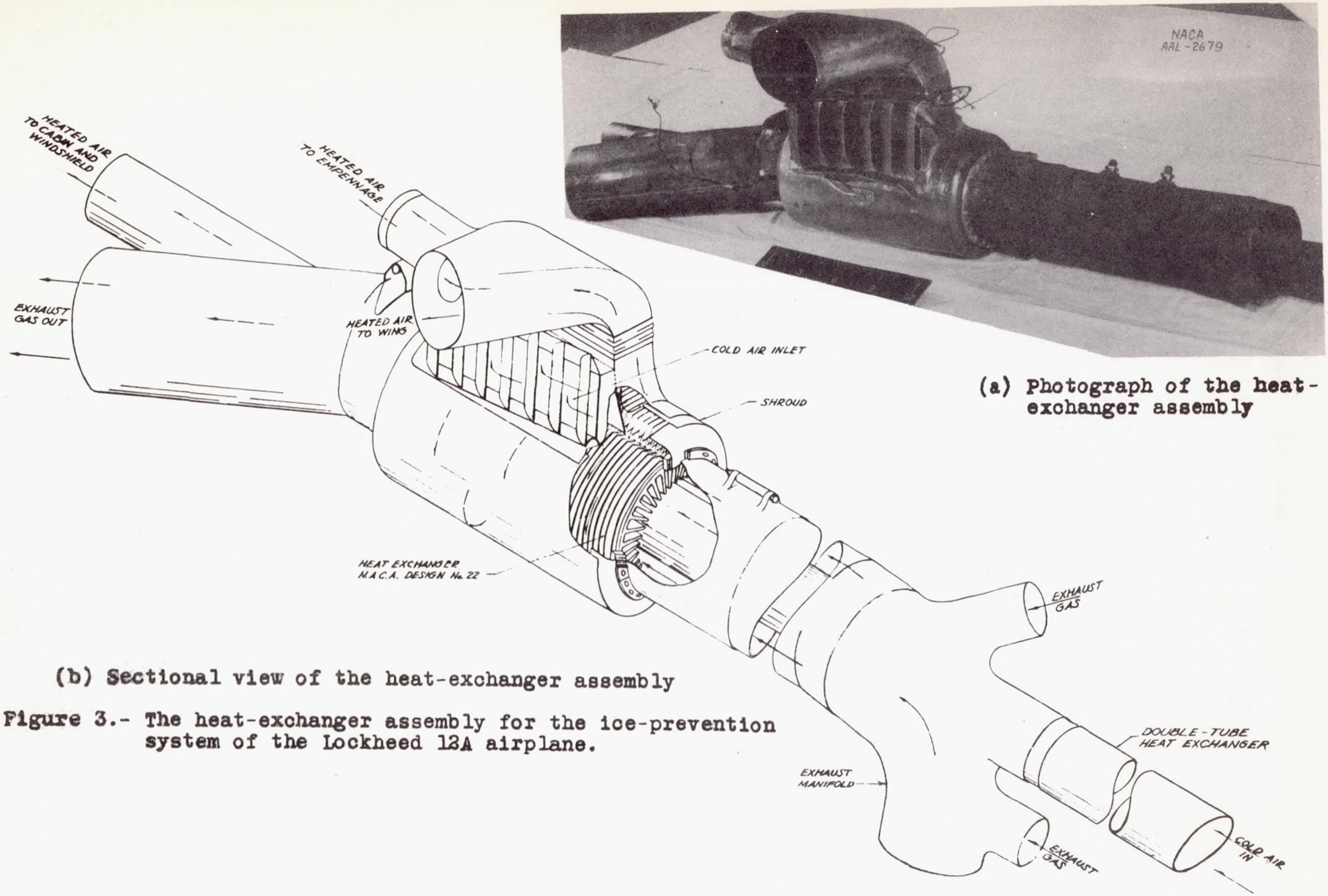


(a) Side view



(b) End view

Figure 2.- The cast-aluminum heat exchanger as used in the ice-prevention system of the Lockheed 12A airplane. NACA heat exchanger Design No. 22.



(a) Photograph of the heat-exchanger assembly

(b) Sectional view of the heat-exchanger assembly

Figure 3.- The heat-exchanger assembly for the ice-prevention system of the Lockheed 12A airplane.

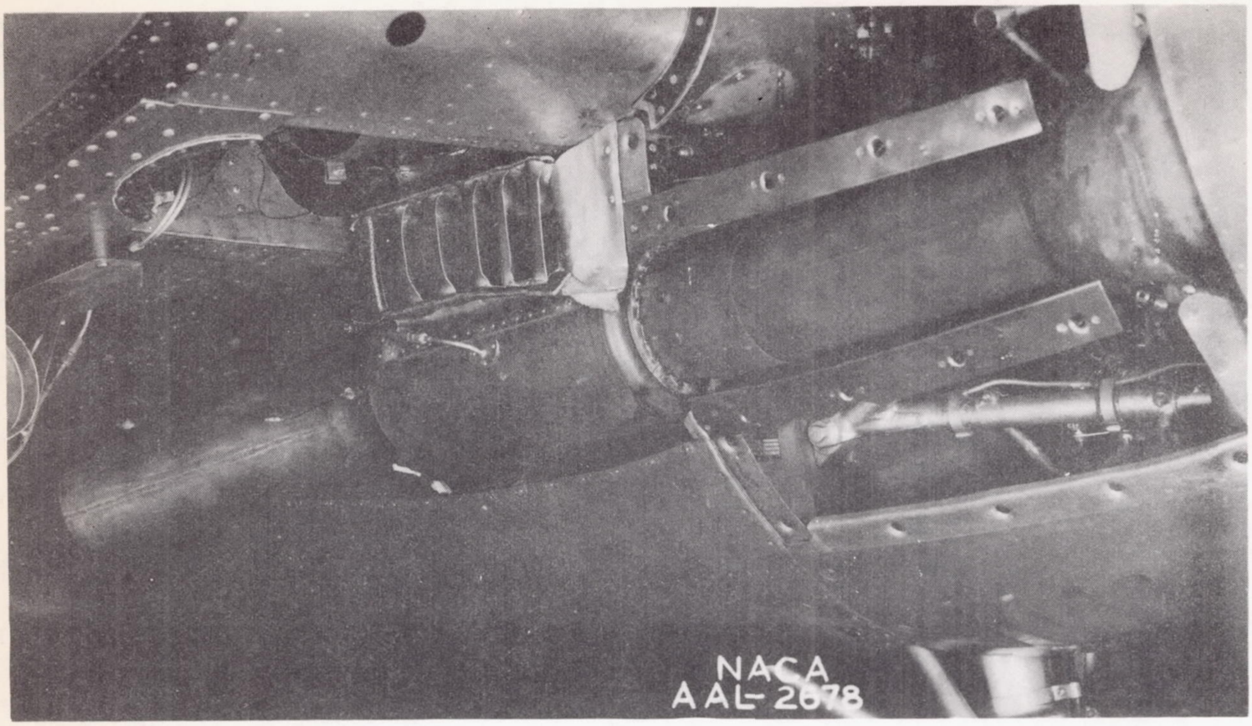


Figure 4.- The heat-exchanger assembly installed in the right nacelle of the Lockheed 12A airplane.

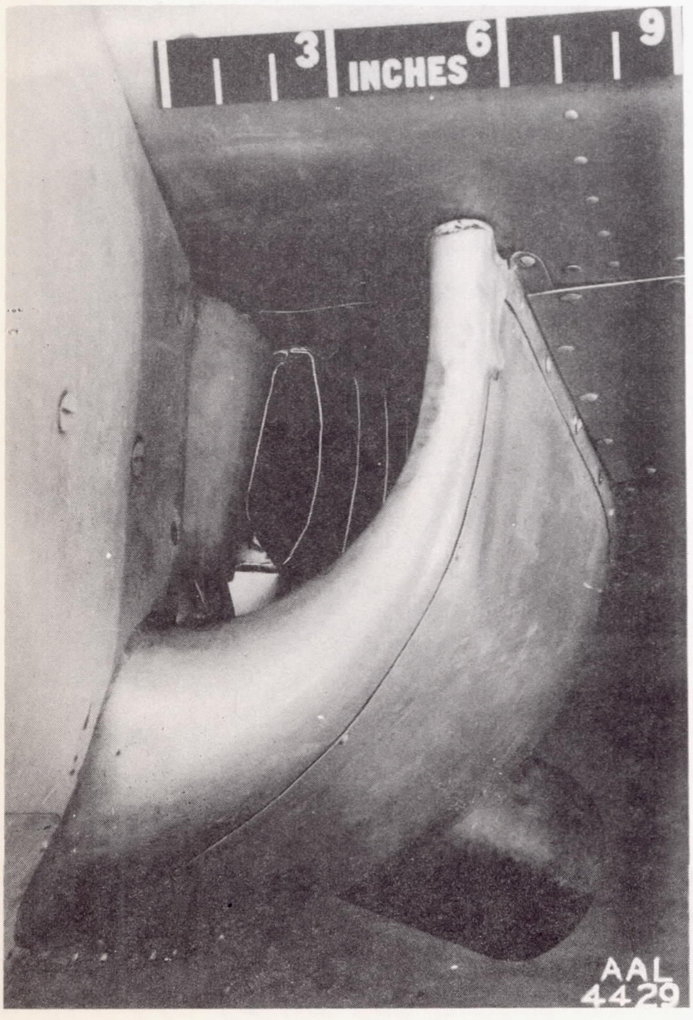
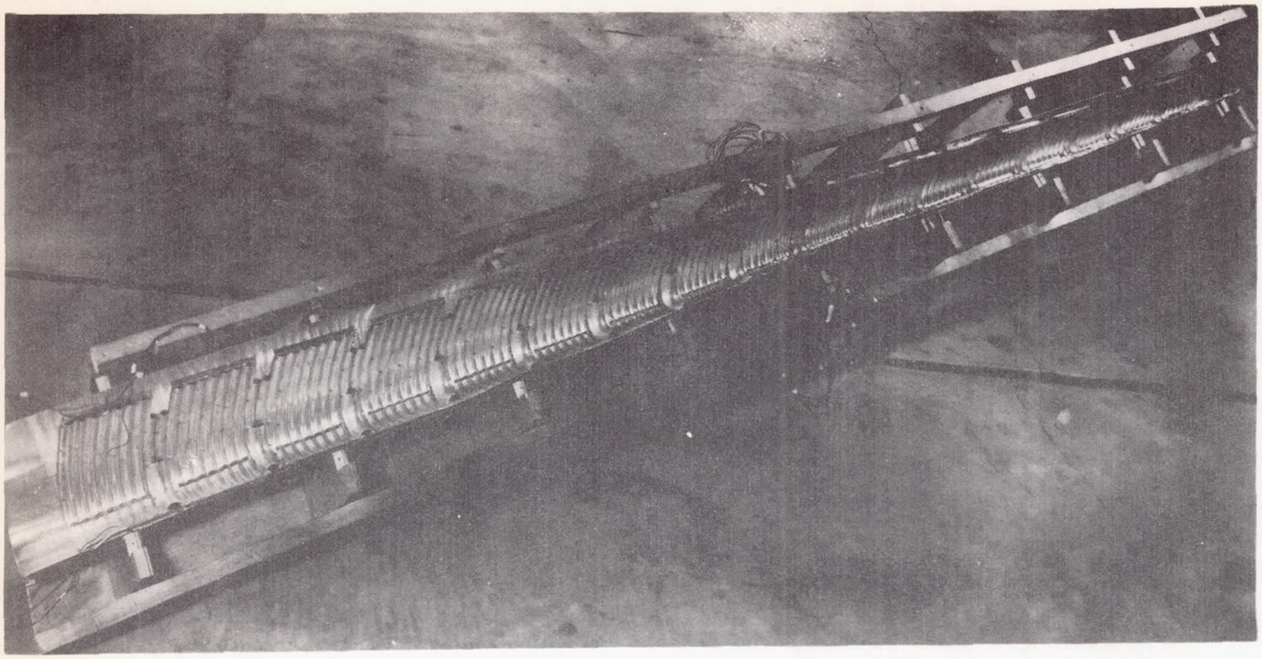
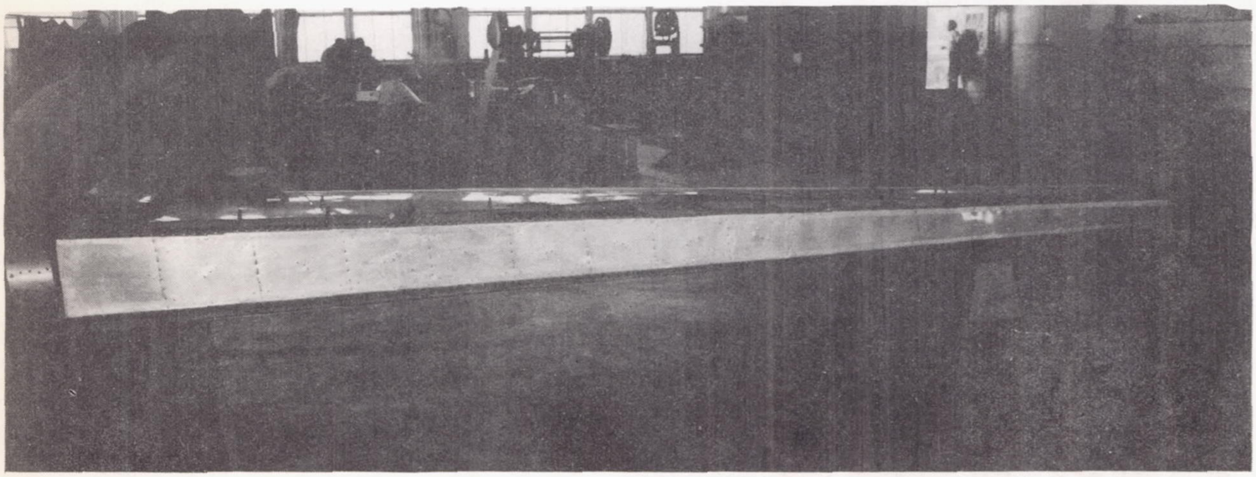


Figure 5.- The heat-exchanger air inlet and fairing on the left nacelle of the Lockheed 12A airplane.

A-49

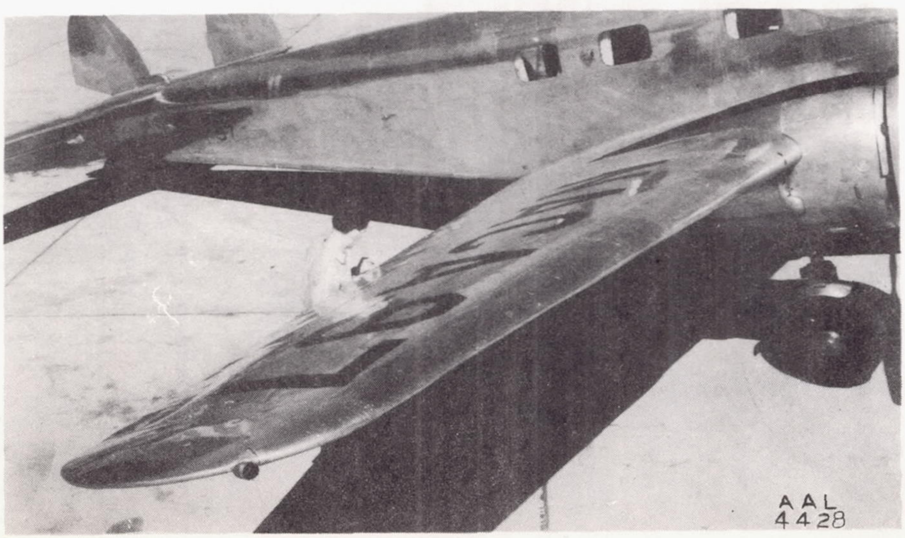


(a) Corrugated inner skin



(b) Leading-edge baffle plate

Figure 6.- The right-hand wing panel for the Lockheed 12A ice-prevention system.



(c) Completed wing panel

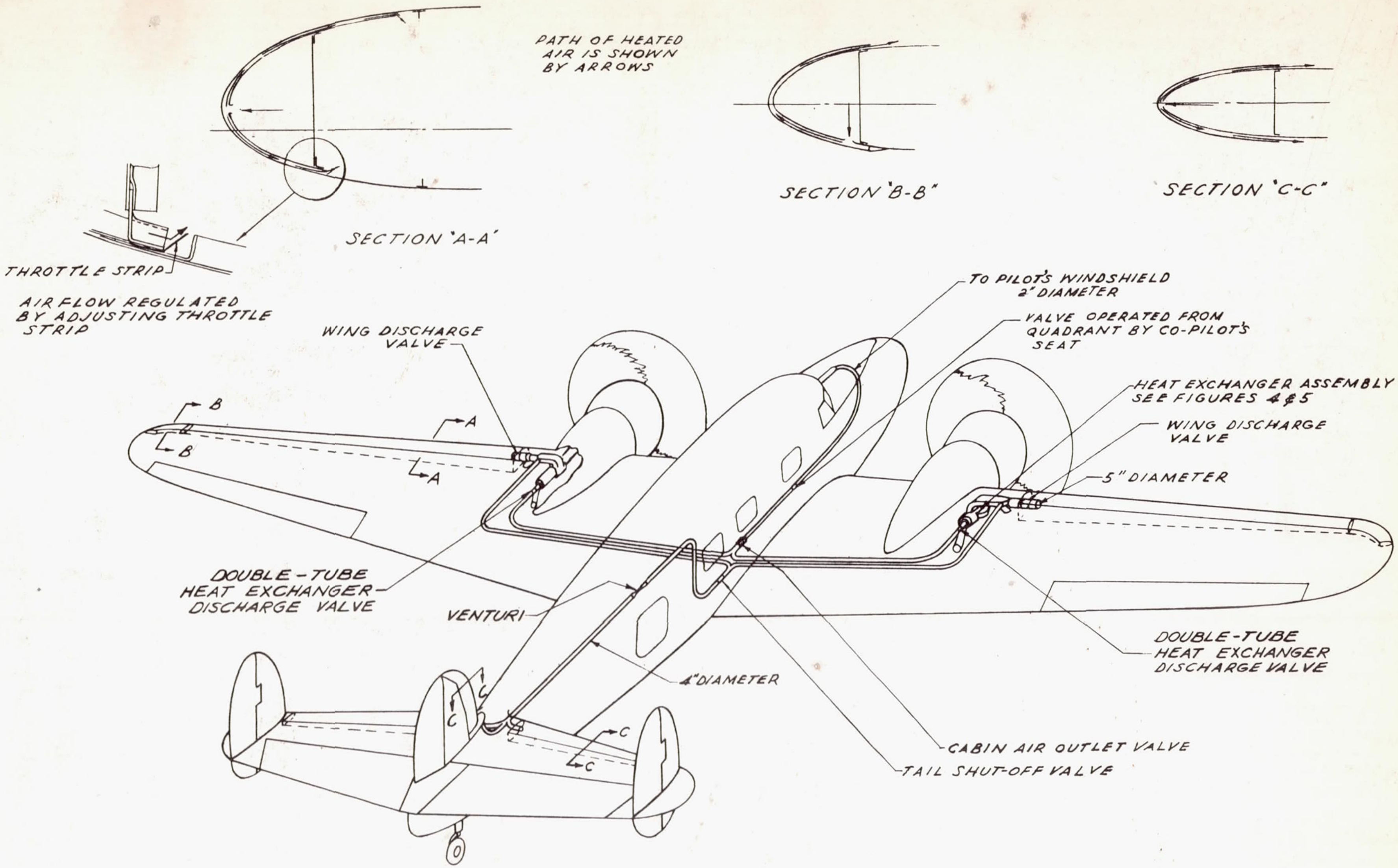


FIGURE 7.— DIAGRAM OF THE ICE PREVENTION EQUIPMENT AND AIR DUCT SYSTEM IN THE LOCKHEED 12-A AIRPLANE

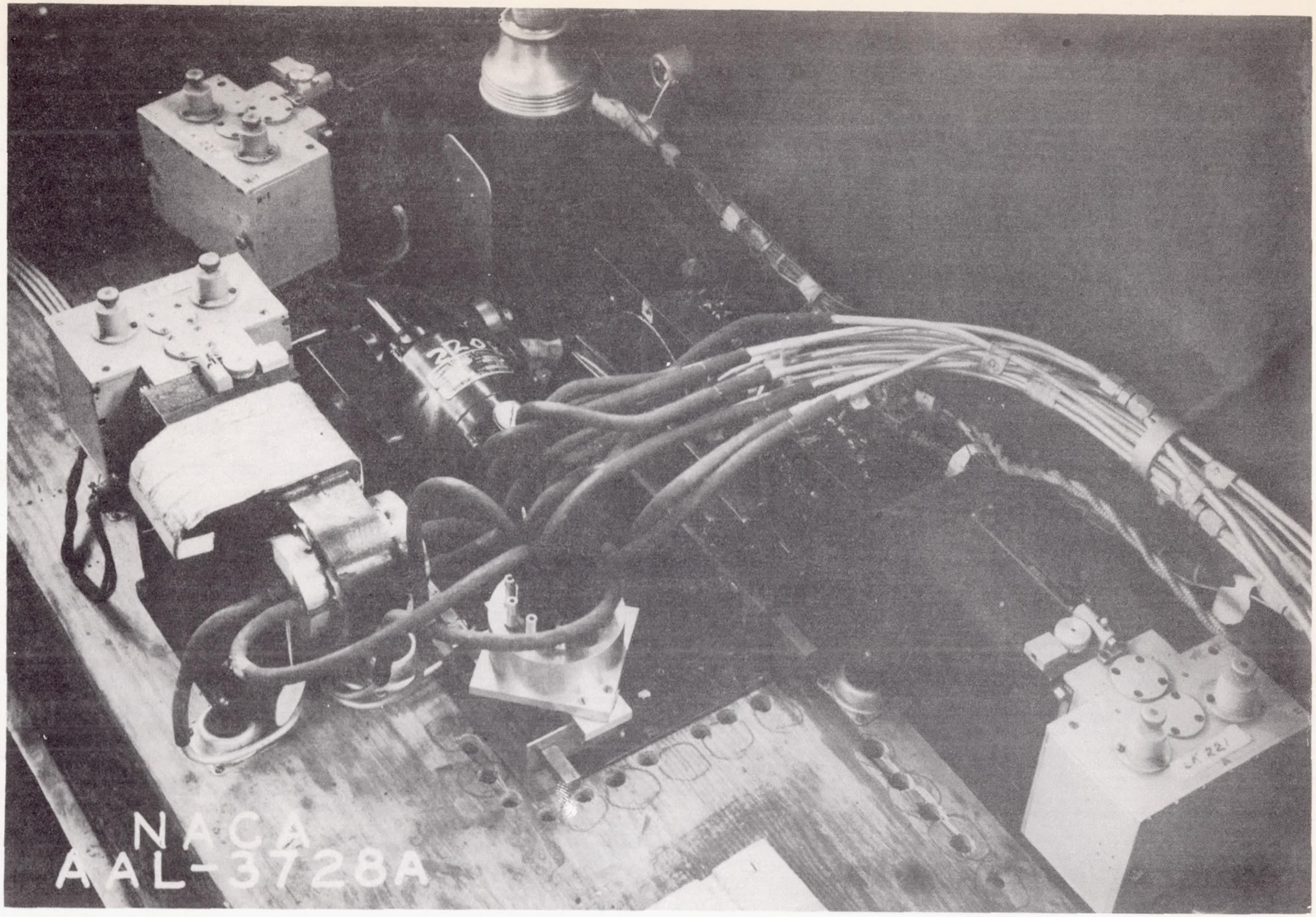


Figure 8.- Temperature-and pressure-recording instruments installed in the Lockheed 12A airplane.

A-49

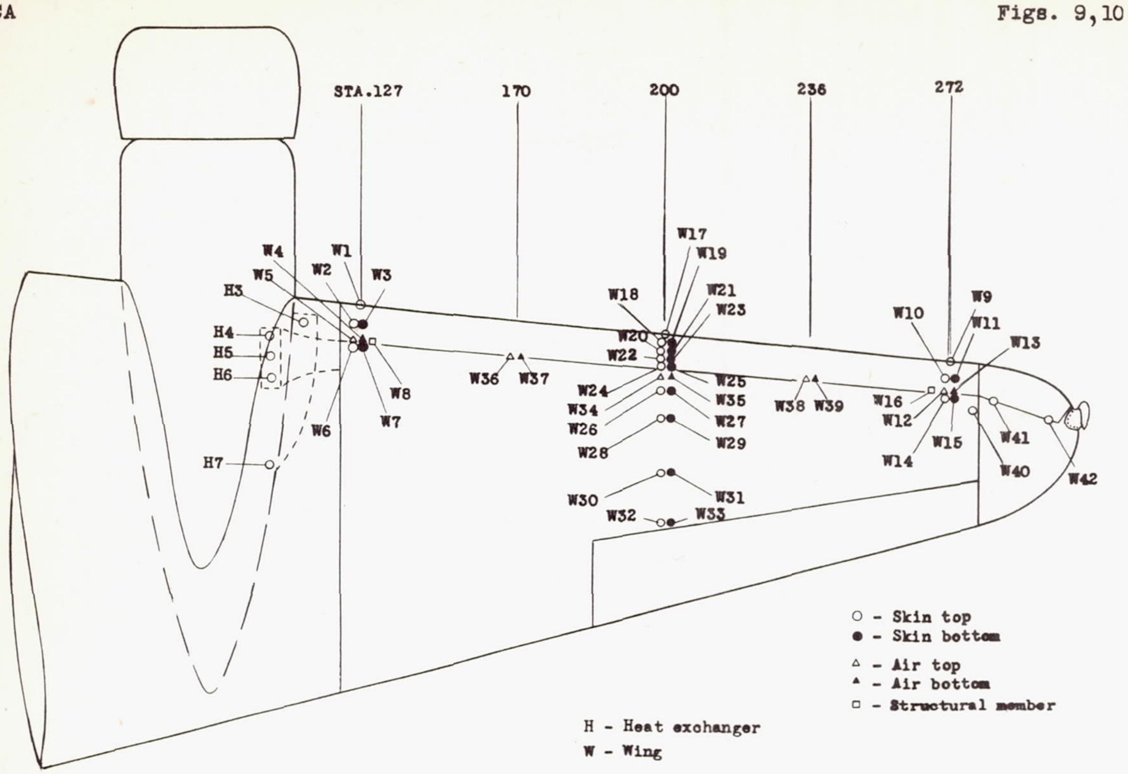


FIGURE 9.— THERMOCOUPLE LOCATIONS IN THE RIGHT WING PANEL OF THE LOCKHEED 12A AIRPLANE

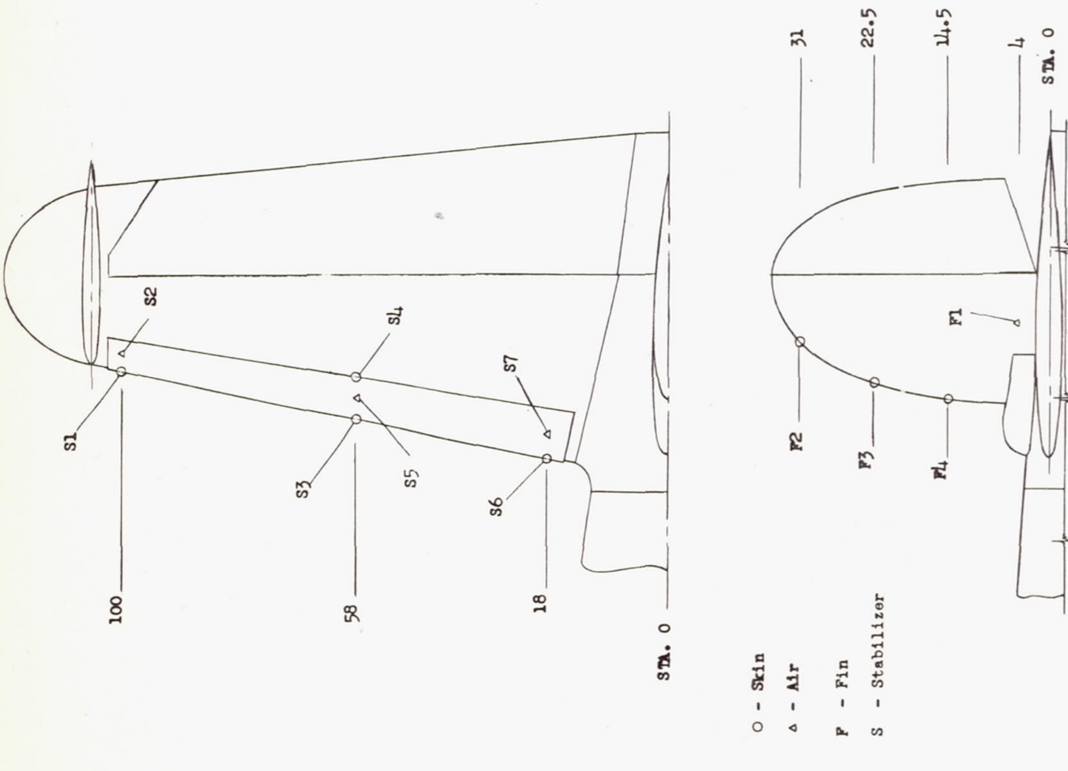


FIGURE 10.— THERMOCOUPLE LOCATIONS IN THE EMPENNAGE OF THE LOCKHEED 12A AIRPLANE

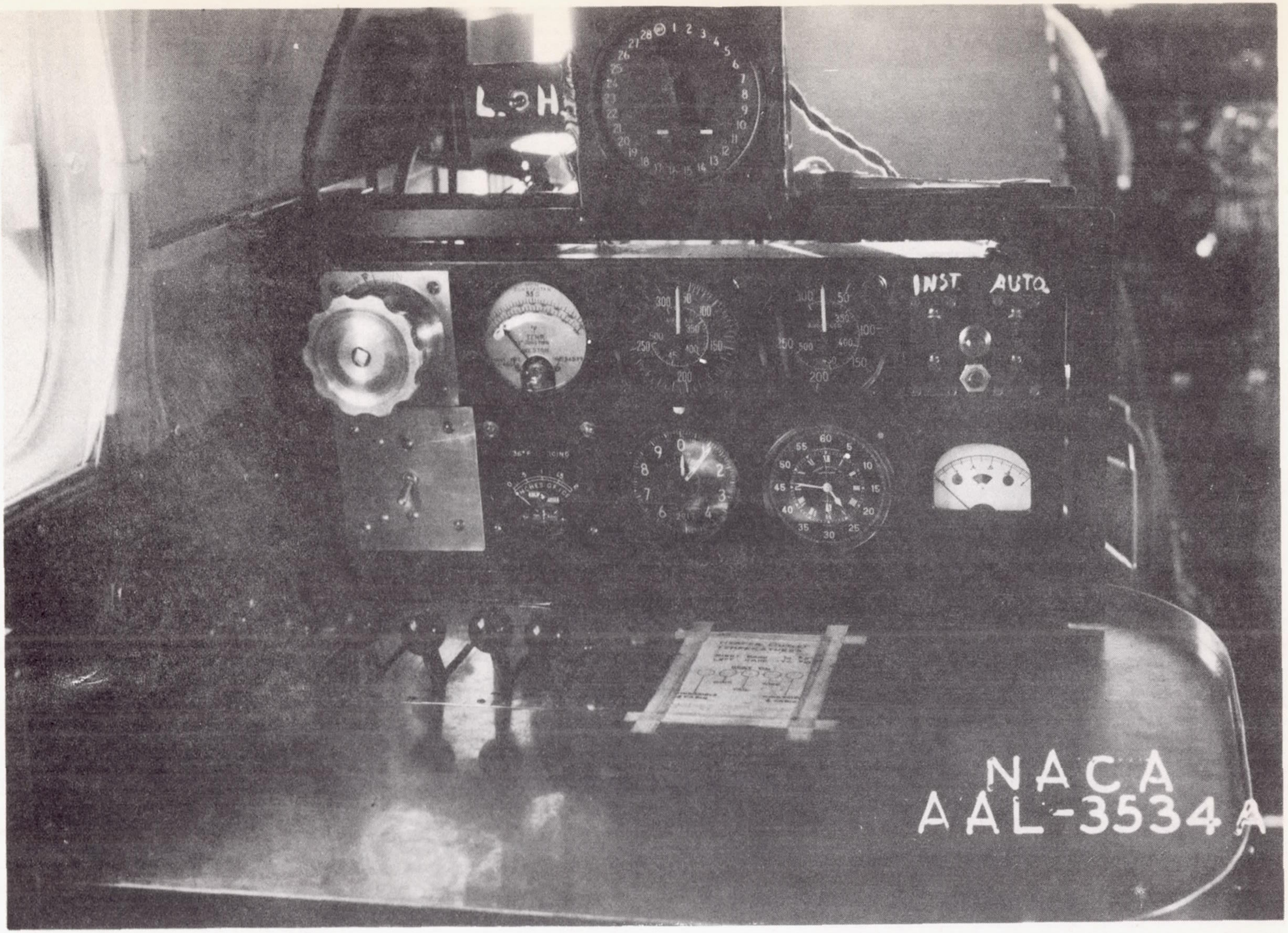


Figure 11.- Observer's instrument panel in the Lockheed 12A airplane.

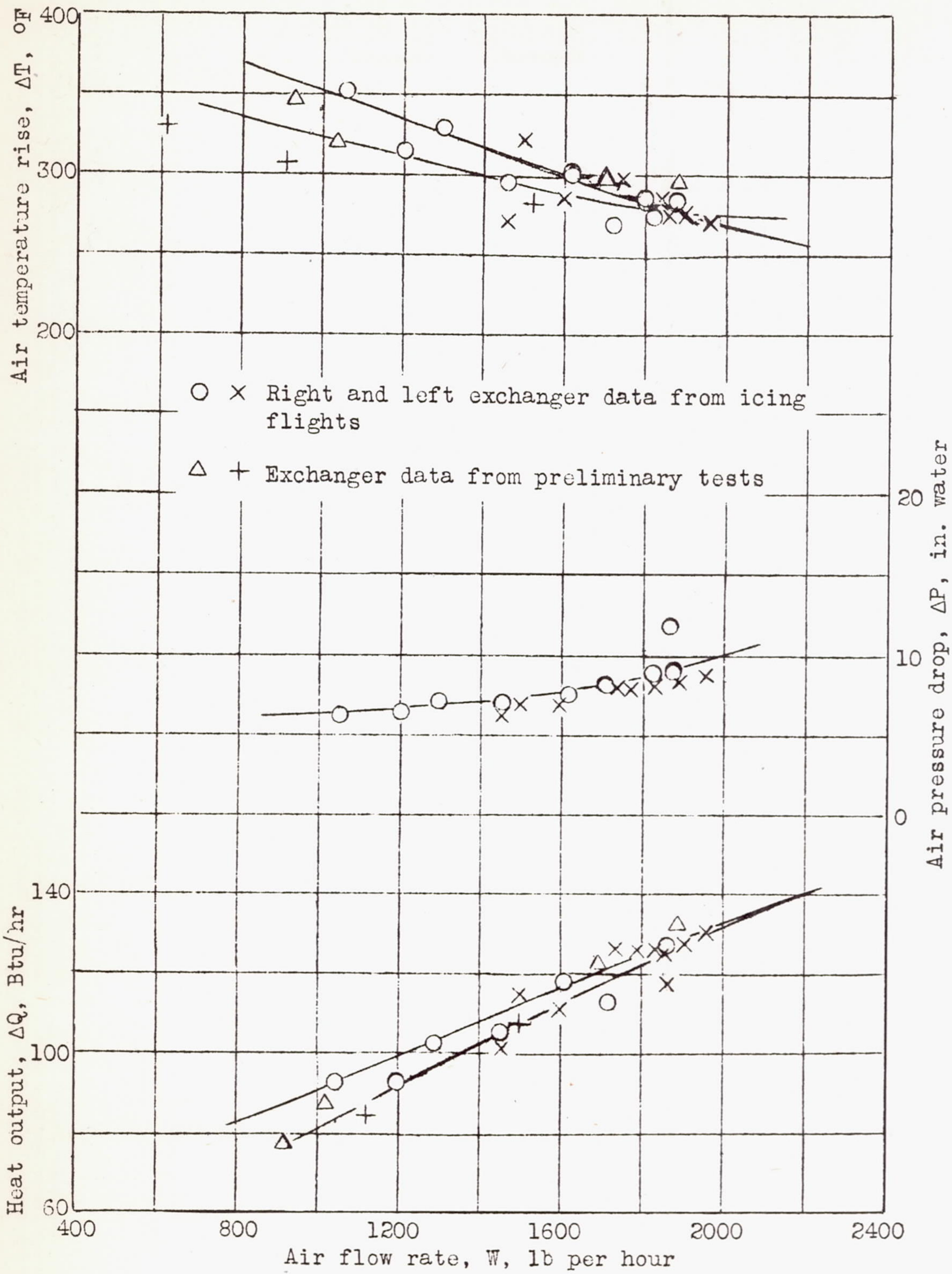
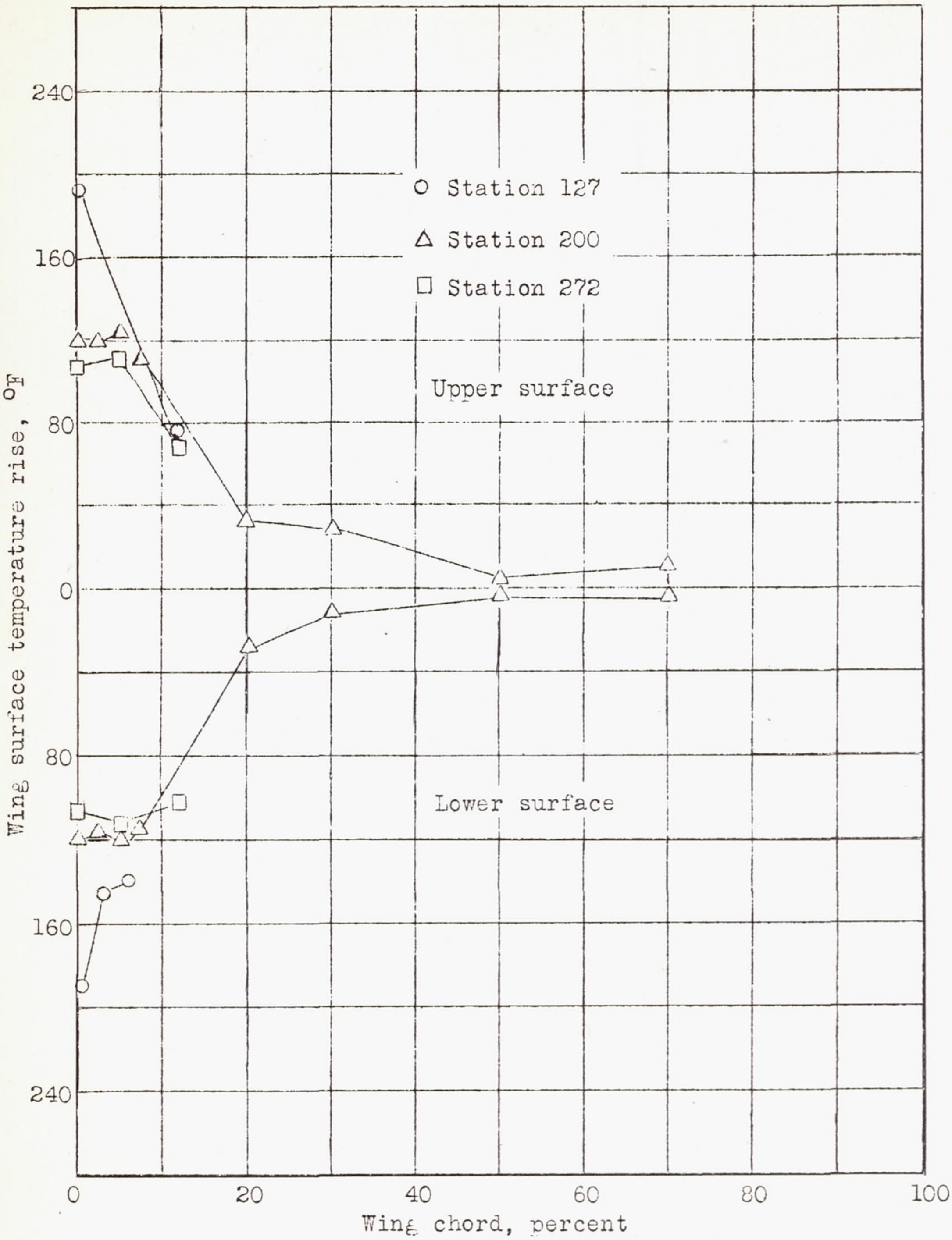


Figure 12.- Performance of heat exchanger, NACA design No. 22, as installed in the Lockheed 12A airplane.

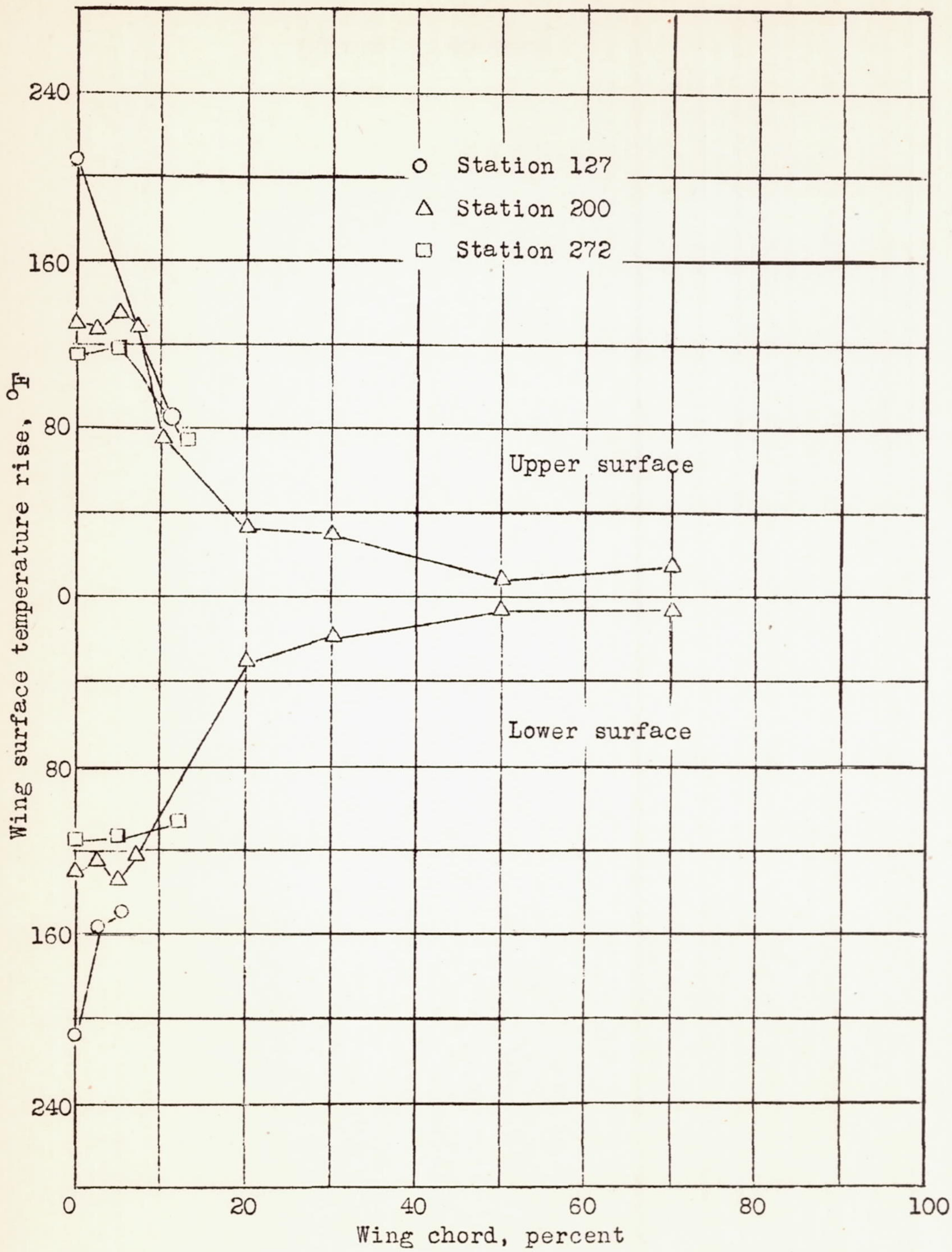
A-49



(a) Flight 17; run 2; clear air; free air temperature 25 °F; fuel-air ratio 0.080; air temperature rise 272 °F.

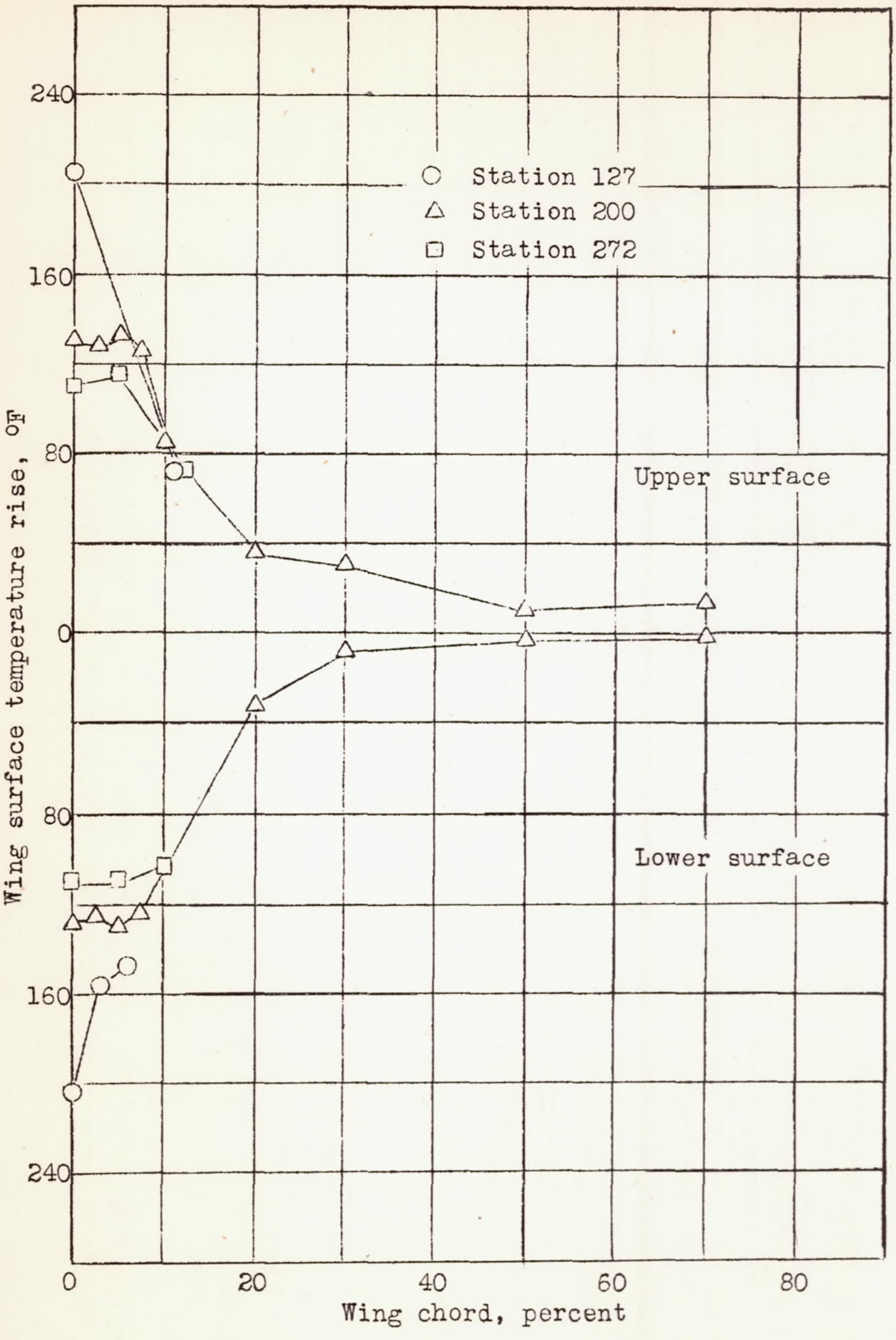
Figure 13.- Wing surface temperature rise at wing stations 127, (a to i) 200 and 272.

A-49



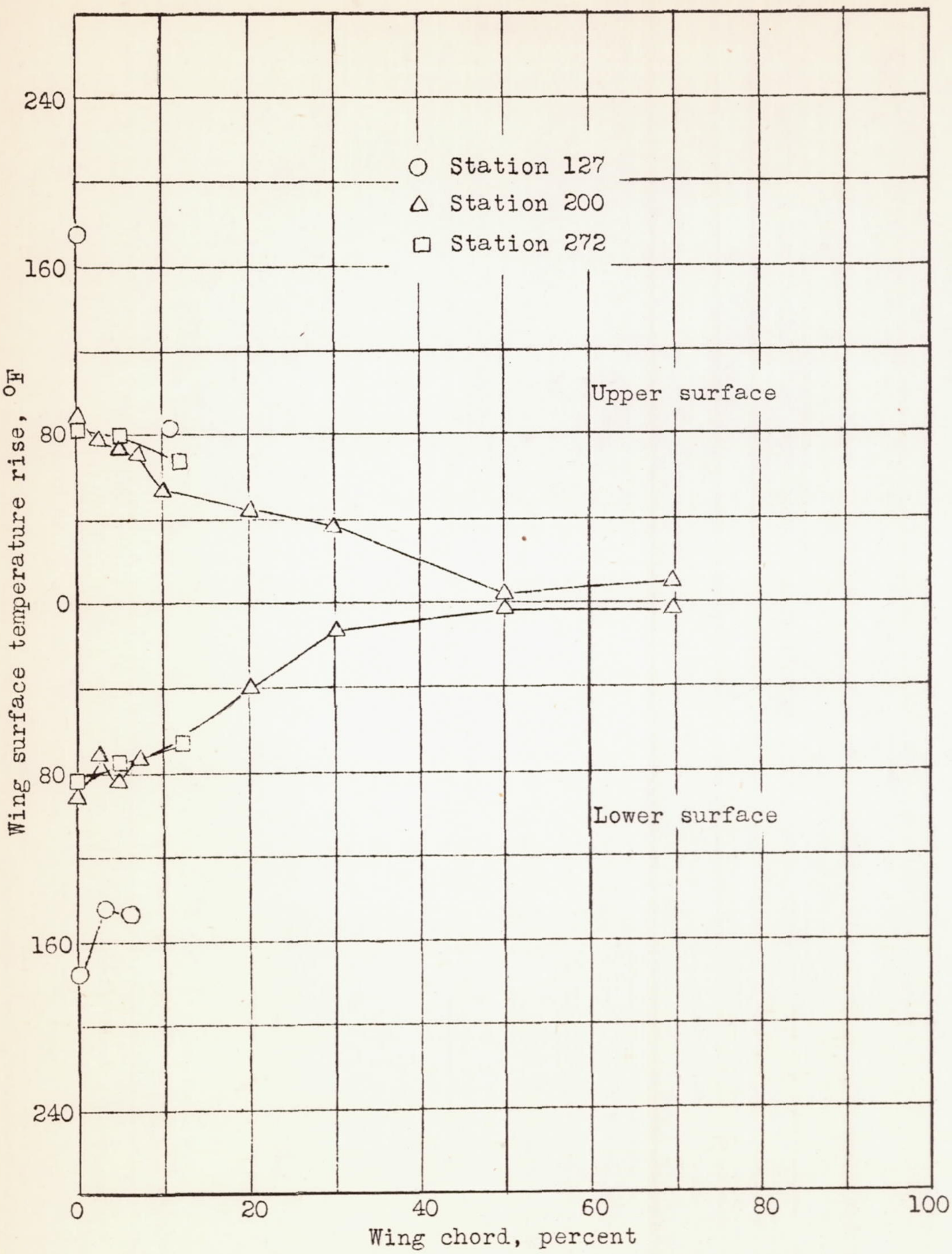
(b) Flight 18; clear air; free air temperature 25 °F; fuel-air ratio 0.080; air temperature rise 272 °F.

Figure 13.- Continued.



(c) Flight 21; clear air; free air temperature 21 °F; fuel-air ratio 0.080; air temperature rise 275 °F.

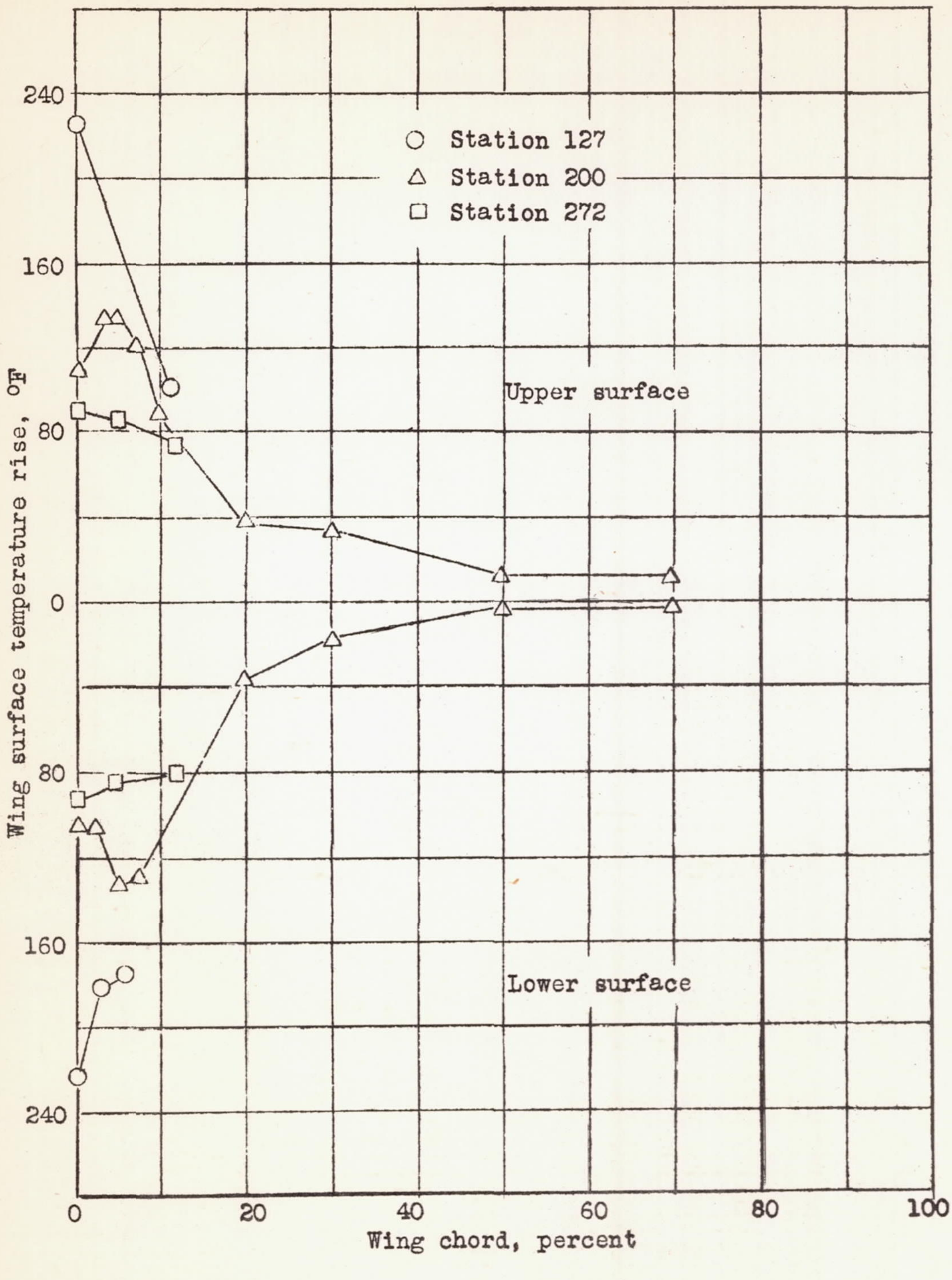
Figure 13.- Continued.



(d) Flight 17; run 1; light glaze ice; free air temperature 25 °F; fuel-air ratio 0.080; air temperature rise 308 °F.

Figure 13.- Continued.

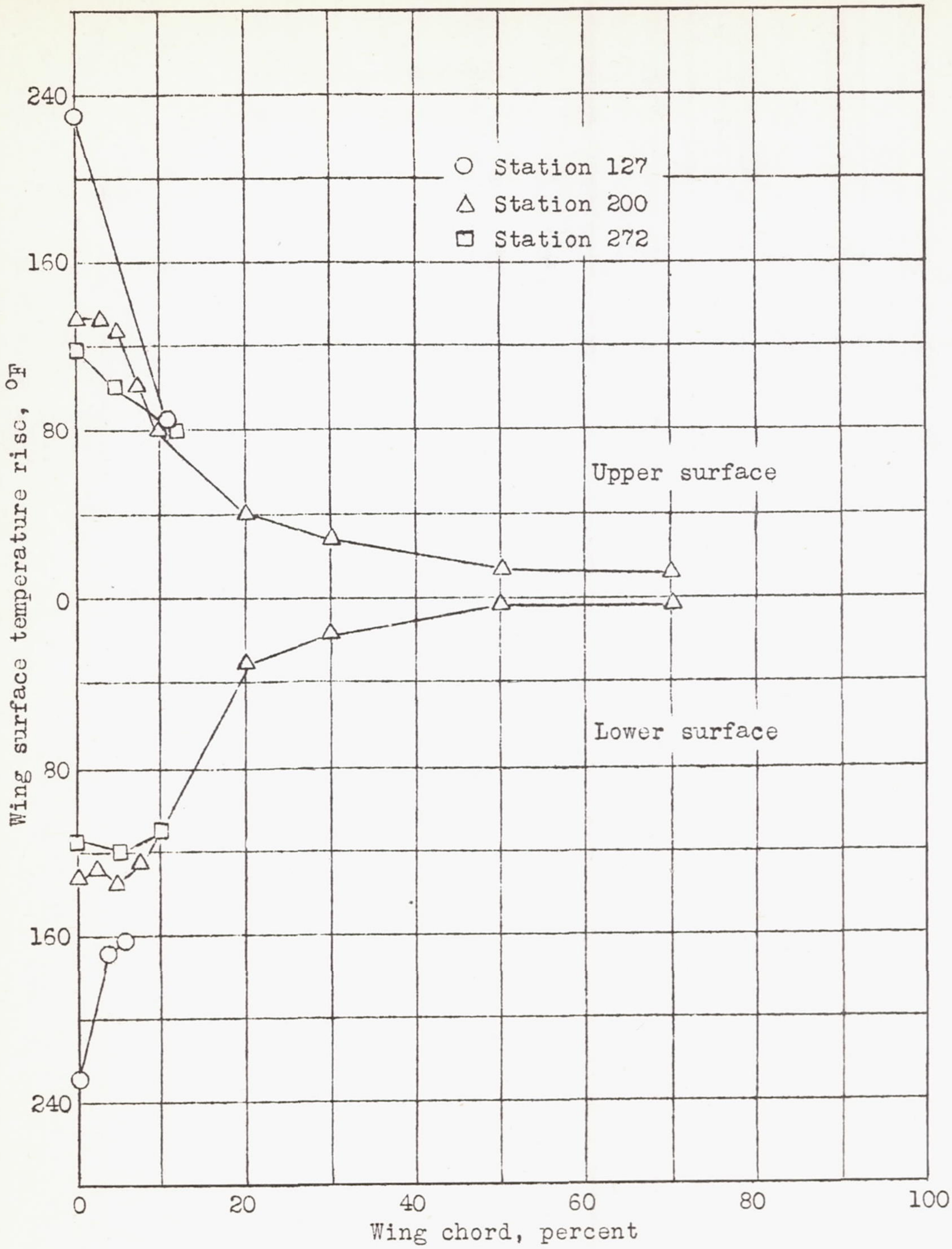
A-49



(e) Flight 20; severe glaze icing (approx. 5 lb per hour); free air temperature 26 °F; fuel-air ratio 0.080; air temperature rise 350 °F.

Figure 13.- Continued.

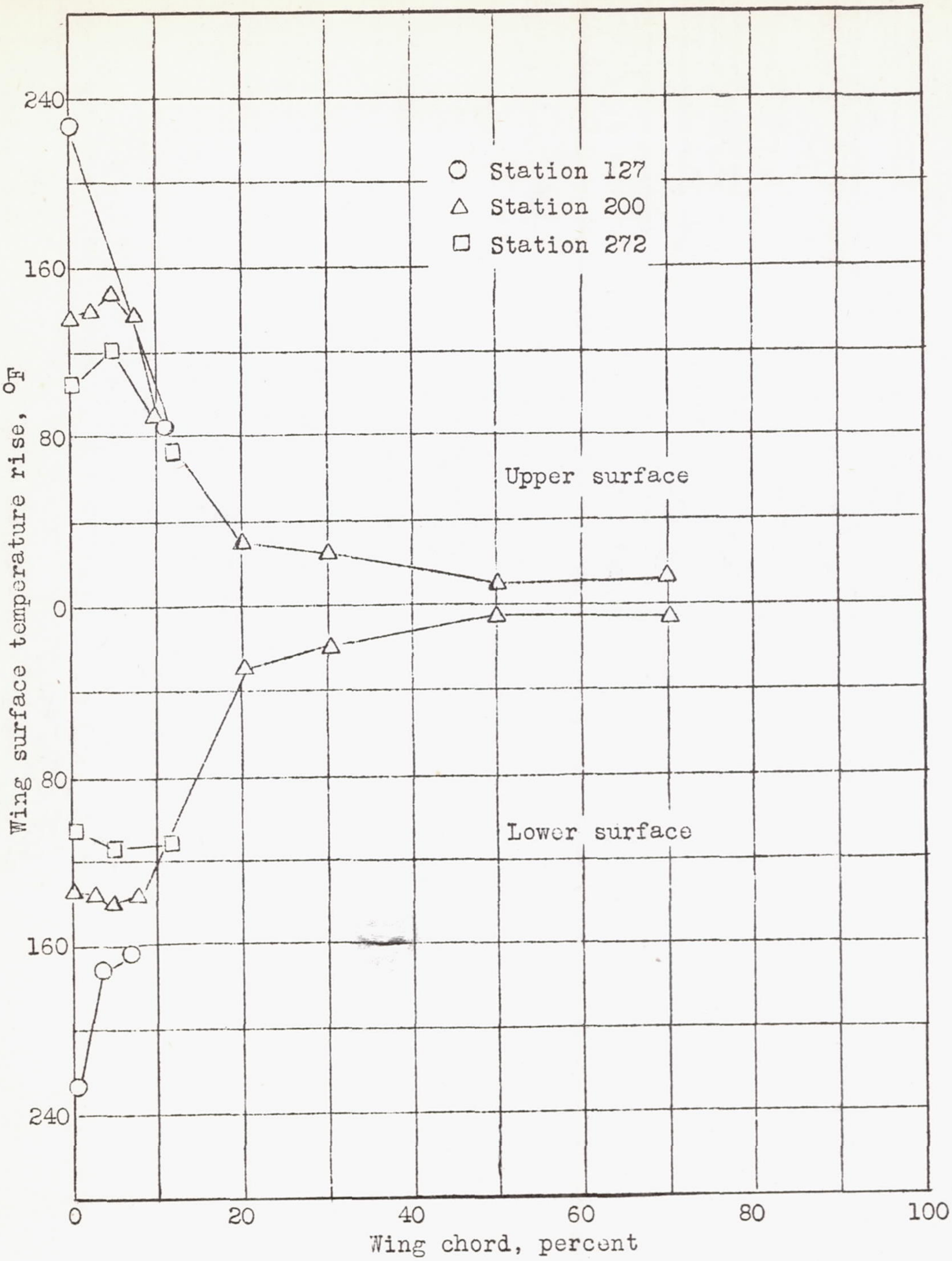
A-49



(f) Flight 22; freezing rain (2 to 3 in. per hour); free air temperature 19 °F; fuel-air ratio 0.075; air temperature rise 327 °F.

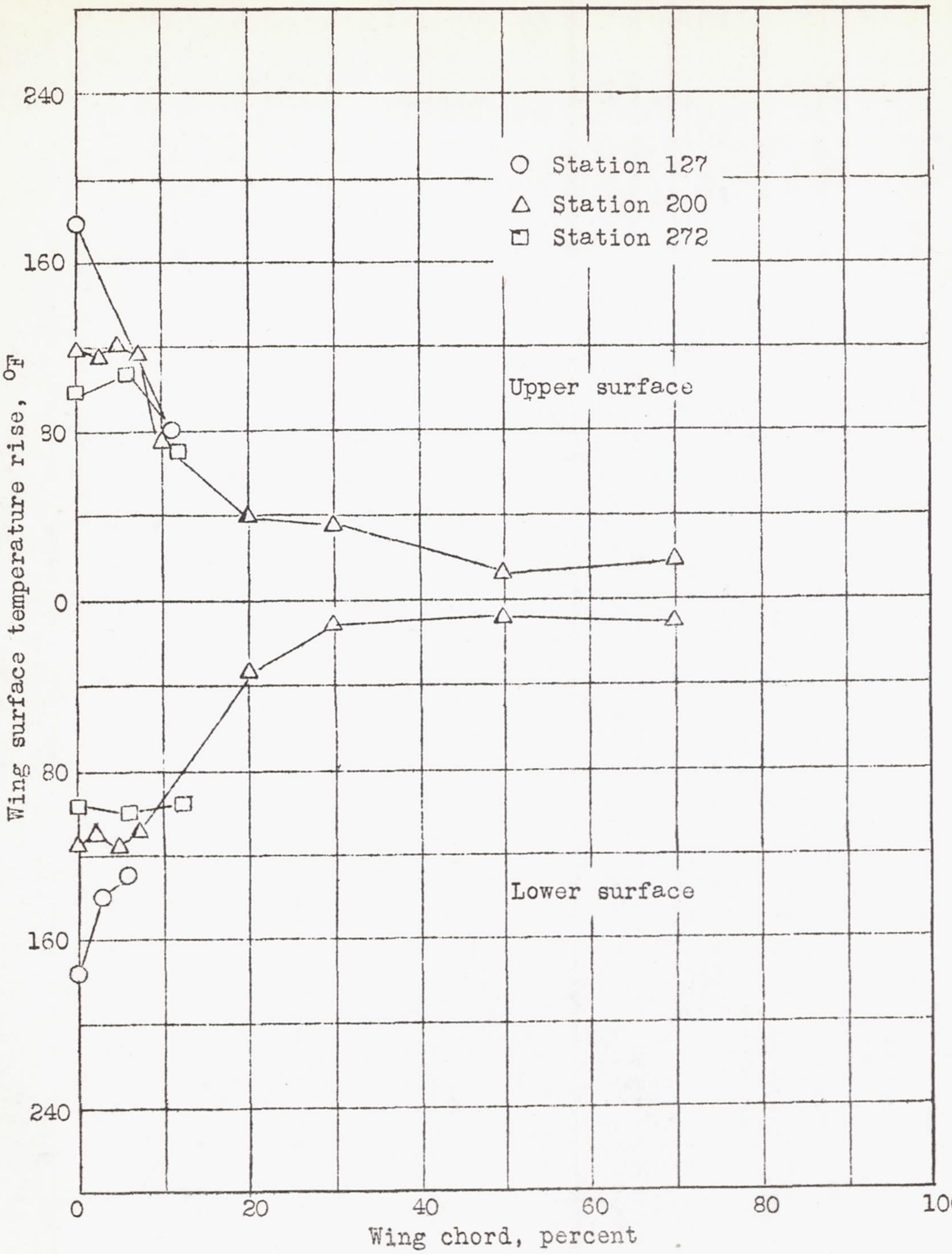
Figure 13.- Continued.

A-49



(g) Flight 29; light rime ice (approx. 1 in. per hour); free air temperature 24 °F; fuel-air ratio 0.080; air temperature rise 313 °F; heat quantity, 72,820 Btu per hour.

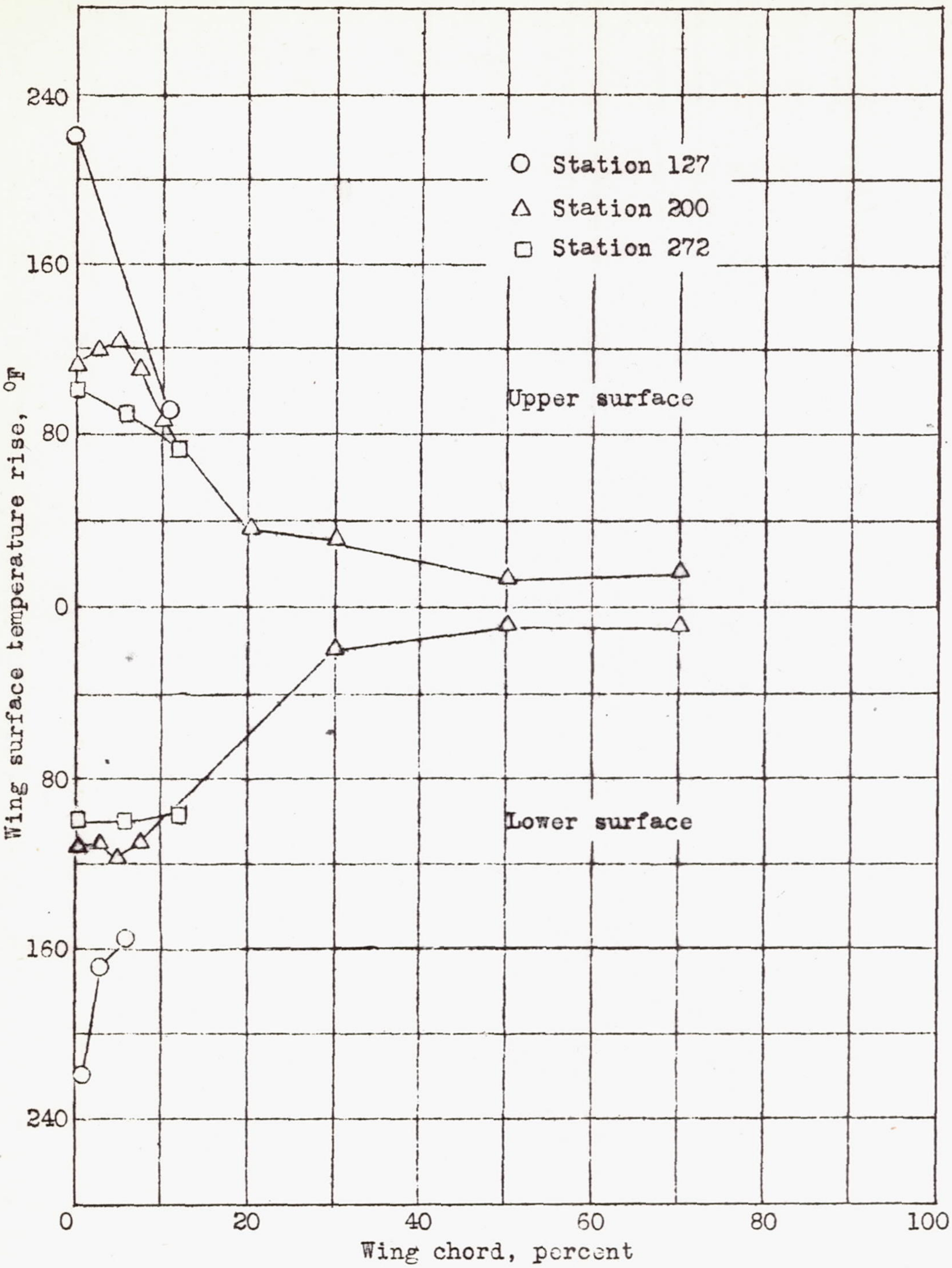
Figure 13.- Continued.



(h) Flight 31; run 1; light rime ice (approx. 1/2 in. per hour); free air temperature 12 °F; fuel-air ratio approx. 0.11; air temperature rise 280 °F; heat quantity to wing 104,000 Btu per hour.

Figure 13.- Continued.

A-49



(i) Flight 31; run 2; rime ice (approx. 1 in. per hour); free air temperature 10 °F; fuel-air ratio 0.080; air temperature rise 298 °F; heat quantity to wing 81,100 Btu per hour.

Figure 13.- Concluded.

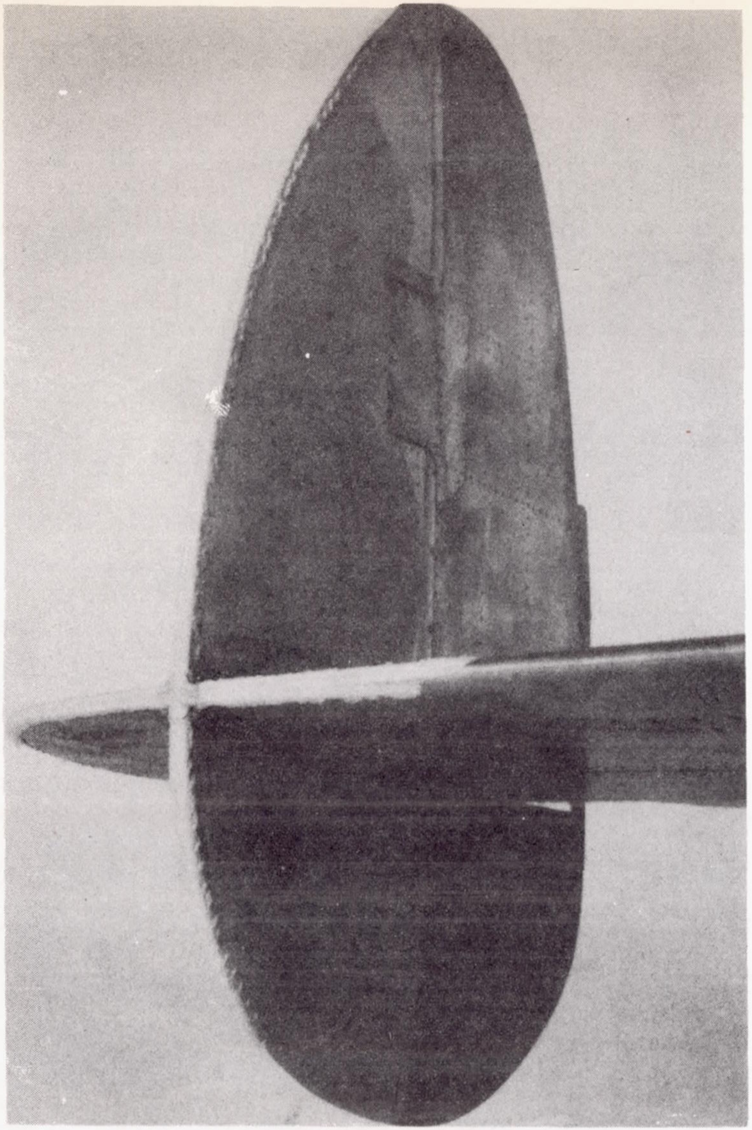


Figure 14.- Ice accretion on the outboard leading edge of the horizontal stabilizer of the Lockheed 12A airplane before adjustment of air-outlet gap.

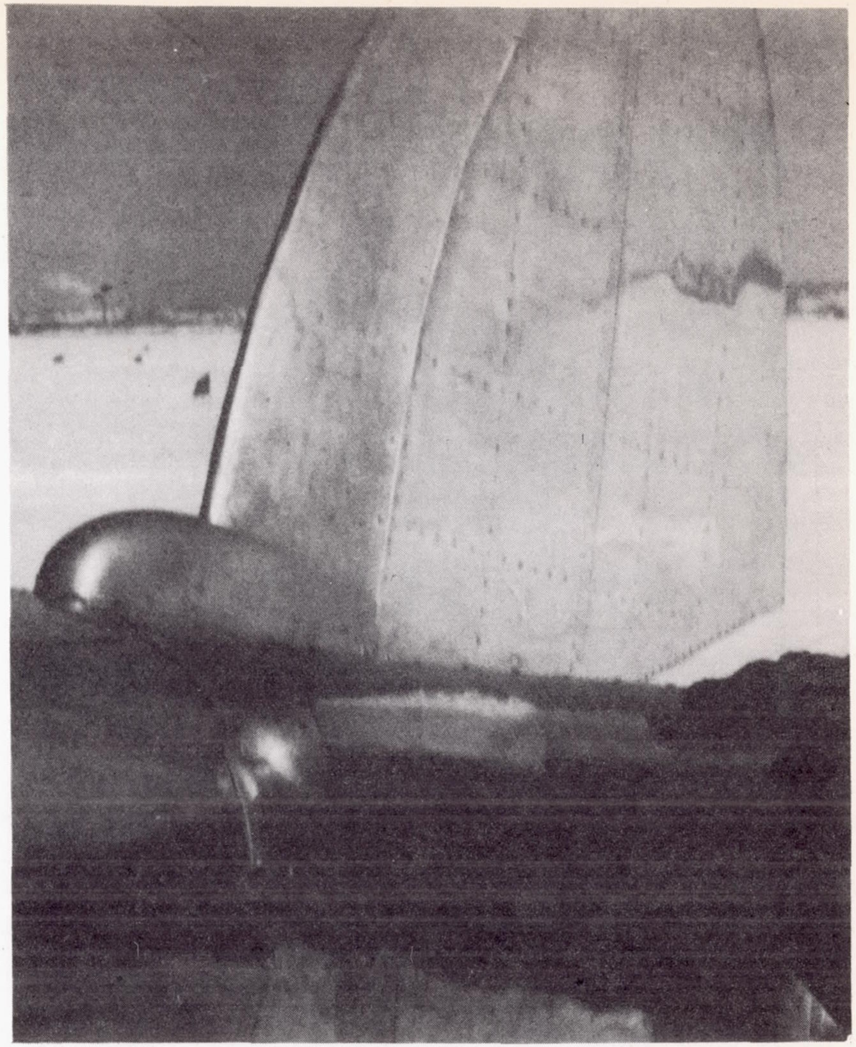


Figure 15.- Ice accretion on the inboard leading edge of the horizontal stabilizer of the Lockheed 12A airplane before adjustment of air-outlet gap.

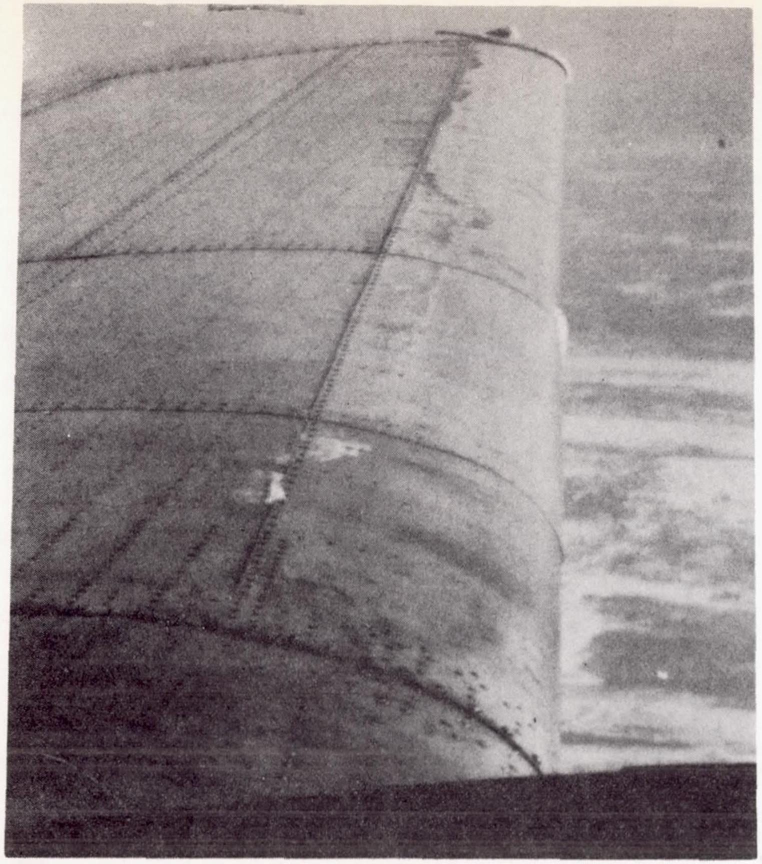
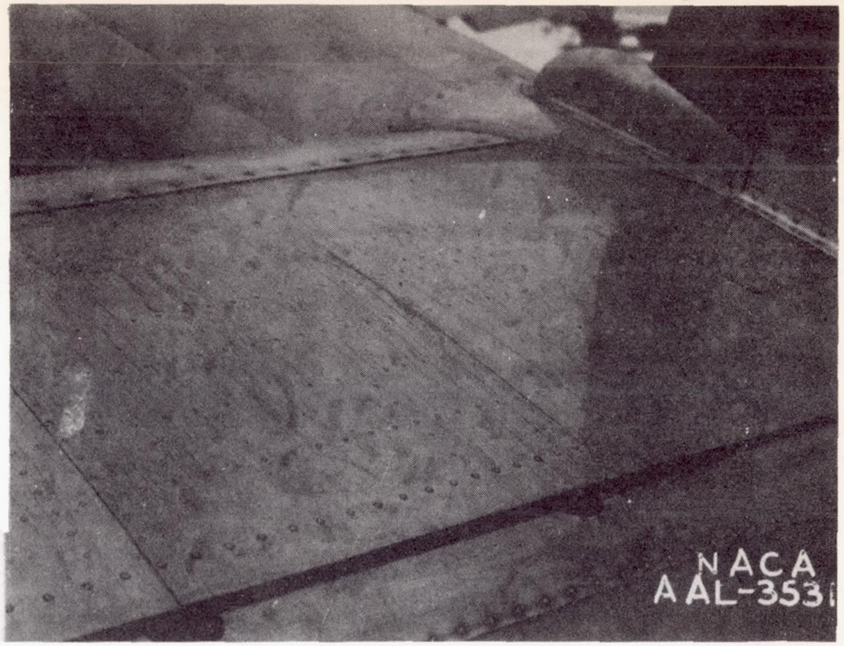


Figure 16.- Ice formations during the process of removal from the wing of the Lockheed 12A airplane. The wing was completely cleared within 2 minutes after the application of heat.

Figure 17.- A typical ice formation on the wing tip of the Lockheed 12A airplane.

Figure 18.- Ice accretions caused by the freezing of water aft of the heated area of the stabilizer of the Lockheed 12A airplane.



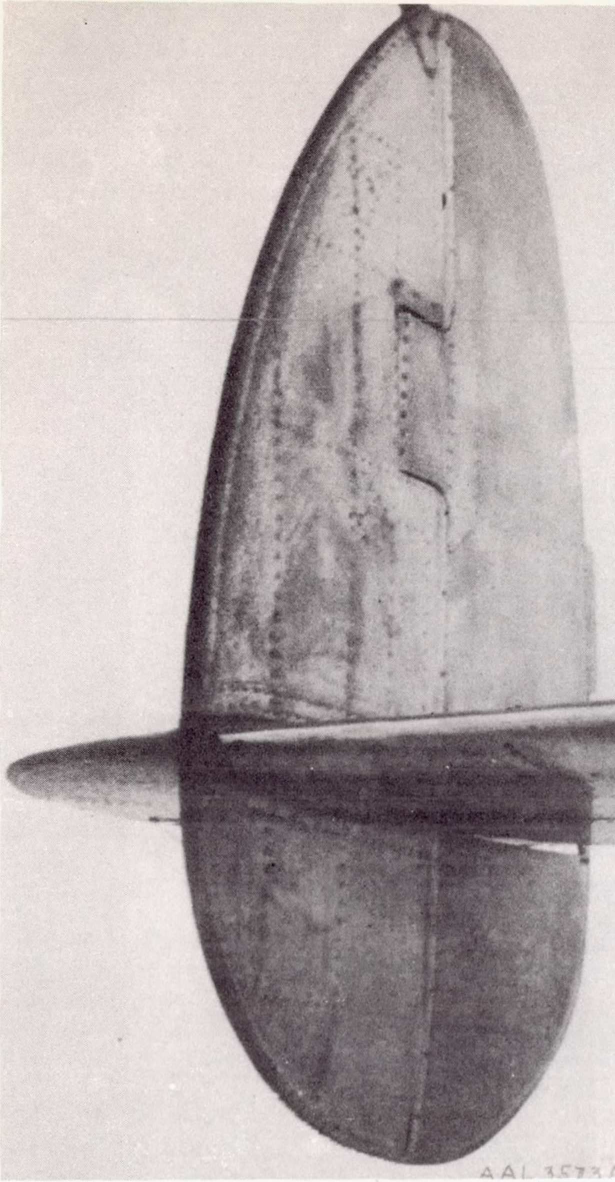
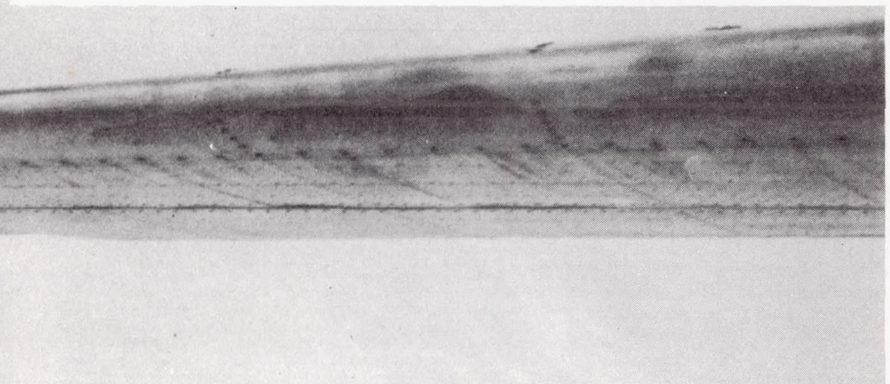



Figure 20.- Ice accretions on the horizontal stabilizer of the Lockheed 12A airplane during a flight in freezing rain.

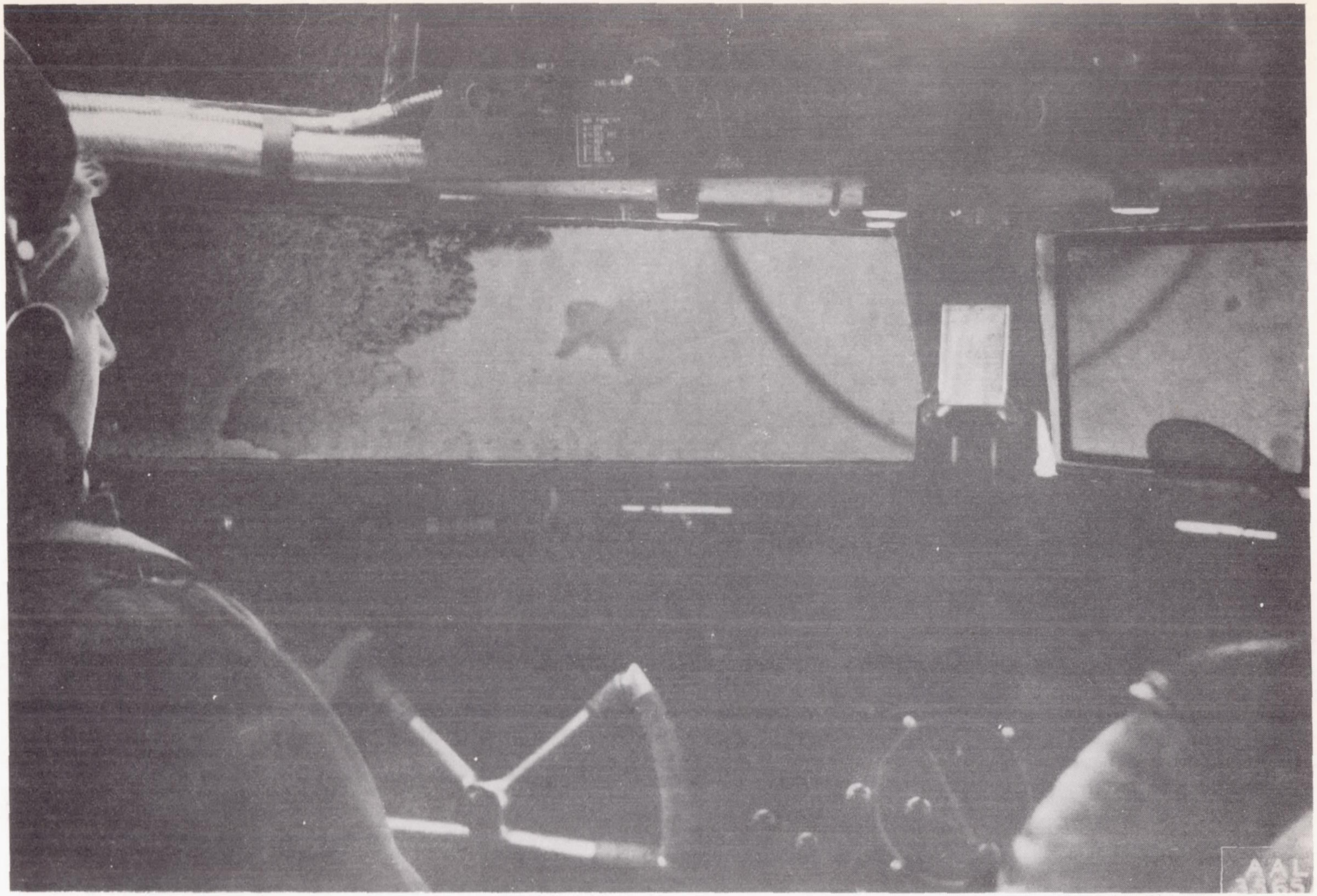


Figure 19. - Ice accretion on the pilot's heated windshield during a flight in freezing rain with the Lockheed 12A airplane.

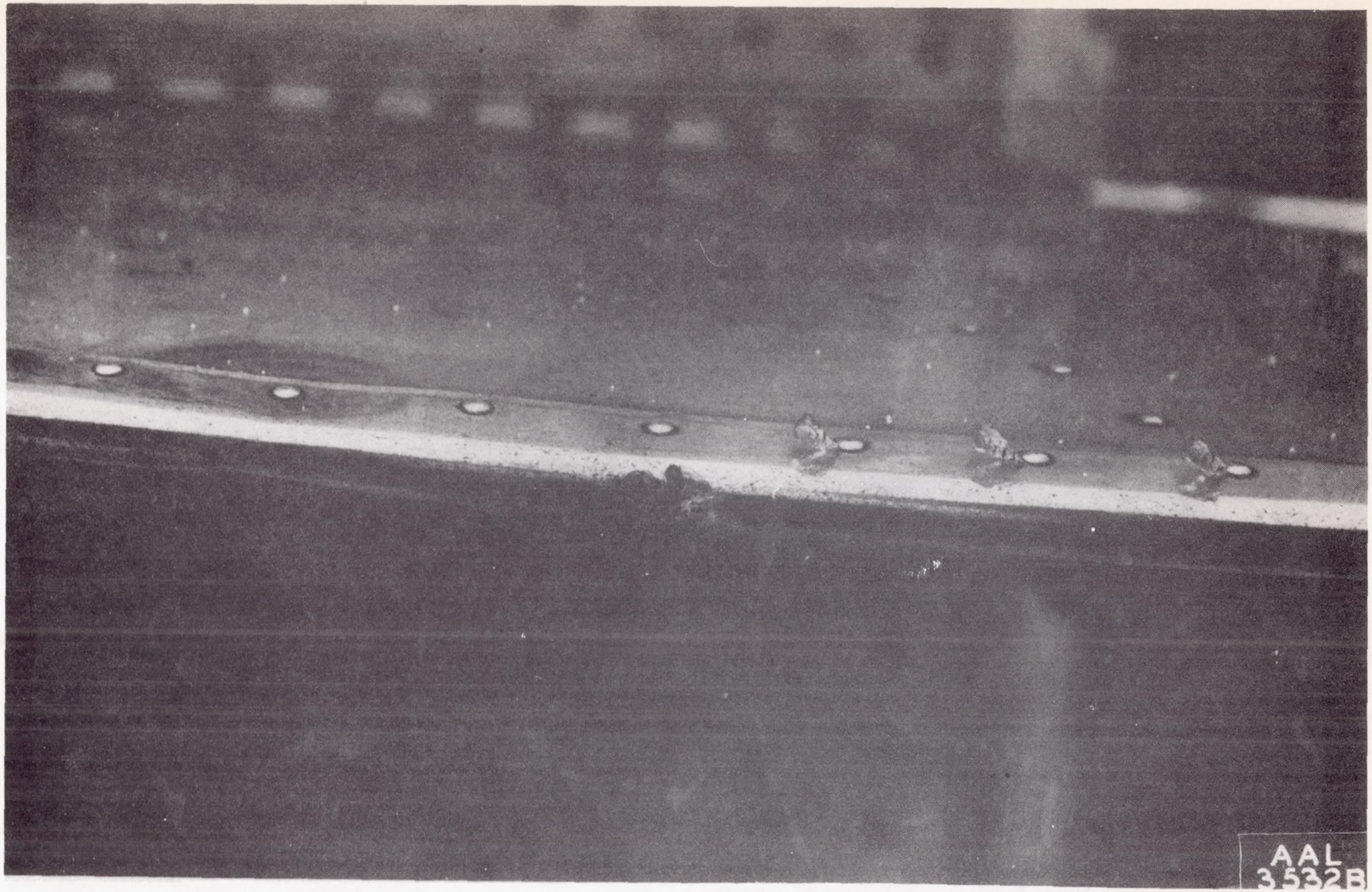


Figure 21. - An example of ice accretions on small protuberances from the surfaces of airplanes.