FINAL REPORT

Purchase Order No. NA-P7-1903

Project No. 430-002-02X

PREDICTION OF AIRCRAFT DAMAGE TIME IN POST-CRASH FIRES

JUNE 1968

Prepared for

DEPARTMENT OF TRANSPORTATION
FEDERAL AVIATION ADMINISTRATION
NATIONAL AVIATION FACILITIES EXPERIMENTAL CENTER
Atlantic City, New Jersey 08405

by

University Engineers, Inc.

Norman, Oklahoma

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Prepared by
J. Reed Welker

This report has been prepared by University Engineers, Inc. for the Department of Transportation, Federal Aviation Administration, National Aviation Facilities Experimental Center, Atlantic City, New Jersey under Purchase Order No. NA-P7-1903. The contents of this report reflect the views of the contractor, who is responsible for the facts and the accuracy of the data presented herein, and do not necessarily reflect the official views of policy of the FAA. This report does not constitute a standard, specification or regulation.

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ABSTRACT

A mathematical model was formulated which permits calculation of the time required for damage of the aluminum skin covering an aircraft fuselage when exposed to fire. The damage time was defined as the time required for melting of the aluminum skin.

The model was developed through consideration of the heat transfer rates by convection and radiation. The resulting differential equation was solved using a numerical technique. The results indicate that the minimum time for skin damage for the largest commercial aircraft now in service is less than 40 seconds. The predictions made through the use of the model correspond closely to measurements made by FAA on full-size aircraft models.

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INTRODUCTION

The occurrence of fires following crashes of private, commercial, and military aircraft leads to loss of life which could be prevented if the post-crash fire could be prevented. However, the prevention of post-crash fires is difficult and usually impossible. Systems are therefore required which will permit suppression of the fire to such an extent that evacuation of personnel from the aircraft is practicable.

The greatest loss of life occurs when a commercial aircraft crashes. The large number of passengers aboard the aircraft cannot be evacuated through the fire, and the aircraft fuselage does not offer a period of protection long enough to permit suppression of the fire. The passengers are therefore trapped inside the aircraft cabin by the fire.

As long as the aircraft fuselage remains intact following a crash, the passengers are afforded some protection against the high temperatures, lack of oxygen, and noxious gases produced by the fire. However, commercial airliners are constructed primarily of aluminum alloys in order to reduce weight. Aluminum alloys of the types used in aircraft construction melt at temperatures significantly lower than those of flames from burning hydrocarbons. The aluminum skin, which is kept as thin as possible consistent with structural requirements, is rapidly melted. The passengers are therefore exposed to the effects of the fire relatively soon after the crash, and there is insufficient time for fire suppression and rescue operations to be successful.

Up to now, there has been no way available to predict the time available for fire suppression and rescue operations. The purpose of the work covered by this report is to formulate a mathematical model which will permit estimation of the time required for the fire to melt the aluminum skin of an aircraft. The time to melting is taken as the maximum time available for fire suppression in order to permit rescue of the passengers.

DEVELOPMENT OF THE MODEL

The development of the mathematical model was based on heat transfer to and from the aircraft during the fire exposure. Although brief consideration was given to instances where the fire did not surround the aircraft, they are not covered in this report. Rather, this report covers the fire situation of maximum danger to the passengers: the case where the fire surrounds or directly contacts the aircraft.

Figure 1 shows a simplified model of the aircraft skin backed by a layer of thermal insulation.

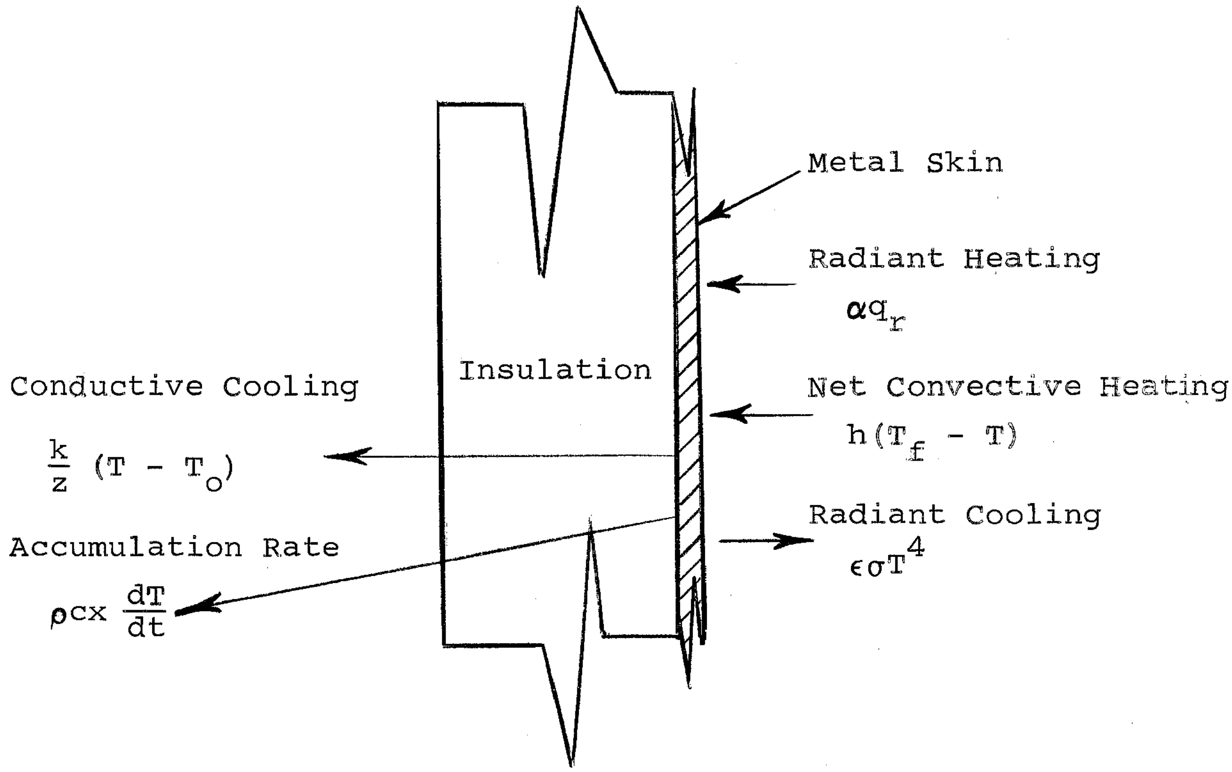


Figure 1. Simplified Model of Aircraft Heating.

In the model, heat gain to the aircraft skin is assumed to be by radiation and convection from the fire. Heat loss from the aircraft skin is due to radiation, convection, and conduction. The difference between the heat gain and heat loss is accumulated by the skin and raises its temperature. The following terms are therefore included in the heat balance:

Radiation heating	$= \alpha q_r$	(1)
Radiation cooling	$= \epsilon\sigma T^4$	(2)
Net convective heating	$= h(T_f - T)$	(3)
Conductive cooling	$= \frac{k}{z}(T - T_o)$	(4)
Accumulation rate	$= \rho c x \frac{dT}{dt}$	(5)

The terms in Equations 1 through 5 are defined as follows:

T = aircraft skin temperature
T_o = temperature inside insulation layer
T_f = flame temperature
α = total absorptance of aircraft skin
q_r = radiant heat output of fire
ϵ = total emittance of aircraft skin
σ = Stephan-Boltzman constant
h = convective heat transfer coefficient
k = thermal conductivity of insulation
z = thickness of insulation
ρ = density of aircraft skin
c = heat capacity of aircraft skin
x = thickness of aircraft skin
t = time

Since

$$\text{Input} - \text{Output} = \text{Accumulation} \tag{6}$$

Equations 1 through 5 can be combined to obtain

$$\rho c x \frac{dT}{dt} = \alpha q_r + h(T_f - T) - \epsilon \sigma T^4 - \frac{k}{z}(T - T_o) \tag{7}$$

Equation 7 relates the rate of temperature buildup to the net heat gained by the aircraft skin. In deriving Equation 7, several assumptions have been made in order to simplify the model. The temperature throughout the aircraft skin was assumed to be uniform because the skin is thin and its thermal conductivity is high. The properties of the metal were assumed to be known and constant over the temperature range in question. The radiant heat transfer from the flame to the aircraft was assumed to be constant and the convective heat transfer coefficient was assumed to be constant.

Equation 7 does not account for the amount of energy required to melt the aluminum skin of the aircraft. Since the aluminum is an alloy, it melts over a temperature range rather than at a particular temperature. If it is assumed that the fraction of aluminum melted over a given melting temperature range is proportional to the fraction of the melting temperature range traversed, the heating rate necessary for melting can be given by

$$q_m = \frac{\rho x \ \Delta H_f}{(T_E - T_B)} \frac{dT}{dt} \tag{8}$$

In Equation 8

q_m = heating rate for melting

ΔH_f = heat of fusion

T_B = temperature at beginning of melting

T_E = temperature at end of melting

If the energy required for melting is included in the heat transfer equation, it becomes

$$\left[\rho c x + \frac{\rho x \ \Delta H_f}{(T_E - T_B)}\right] \frac{dT}{dt} = \alpha q_r + h(T_f - T) - \epsilon \sigma T^4 - \frac{k}{z}(T - T_o) \qquad (9)$$

Equation 9 can only be used after the initial melting temperature is reached. At temperatures below the initial melting temperature, Equation 7 must be used.

If the skin material does not melt on exposure to fire (for example, a stainless steel skin), Equation 7 can be used throughout the heating cycle and can be used to calculate the maximum temperature reached during fire exposure. The maximum temperature is calculated by setting the accumulation term in Equation 7 equal to zero. Thus,

$$\alpha q_r + h(T_f - T_{max}) - \epsilon \sigma T_{max}^4 - \frac{k}{z}(T_{max} - T_o) = 0 \qquad (10)$$

where T_{max} is the highest temperature reached. Equation 10 can be solved by trial and error to obtain the maximum temperature.

SOLUTION OF THE MODEL

Both Equation 7 and Equation 9 must be used for calculation of the failure time for aluminum aircraft skin. Equation 7 applies until the temperature at which melting begins is reached, and Equation 9 applies from the start of melting until melting is complete. Both equations are non-linear first order differential equations, and neither can be solved analytically. Each requires an initial condition for its solution.

In order to simplify the numerical solution of Equations 7 and 9, they were written in the form

$$\frac{dT}{dt} = A_1 + B_1 \ T + C_1 \ T^4 \qquad (11)$$

and

$$\frac{dT}{dt} = A_2 + B_2 T + C_2 T^4 \tag{12}$$

Equation 11 corresponds to Equation 7 and Equation 12 corresponds to Equation 9. The constants are given by

$$A_1 = \frac{\alpha q_r + hT_s + \frac{k}{z} T_o}{\rho c x} \tag{13}$$

$$B_1 = - \frac{(h + k/z)}{\rho c x} \tag{14}$$

$$C_1 = - \frac{\epsilon \sigma}{\rho c x} \tag{15}$$

$$A_2 = \frac{\alpha q_r + hT_s + \frac{k}{z} T_o}{\rho c x + \frac{\rho x \Delta H_f}{(T_E - T_B)}} \tag{16}$$

$$B_2 = - \frac{(h + k/z)}{\rho c x + \frac{\rho x \Delta H_f}{(T_E - T_B)}} \tag{17}$$

and

$$C_2 = - \frac{\epsilon \sigma}{\rho c x + \frac{\rho x \Delta H_f}{(T_E - T_B)}} \tag{18}$$

The initial condition applied to Equation 11 is

$$T = T_o \text{ @ } t = 0 \tag{19}$$

since the aluminum is initially at the temperature of the surroundings. The initial temperature for Equation 12 is taken as the initial melting temperature at the time, t, at which the initial melting temperature is reached according to the calculations of Equation 11. Since Equation 12 only applies during the melting period, calculations are stopped when the end of the melting range is reached.

If the aircraft skin is a non-melting material such as stainless steel, only Equation 11 is used, and the calculations are continued until the steady state solution is approached.

The solutions to Equation 11 and 12 were obtained using the Runge-Kutta technique, which is explained in standard books, for example, Mickley, Sherwood, and Reed (6).

Calculations were made for stainless steel and aluminum aircraft skins using the data in Table 1. These materials were chosen to correspond with tests made by the Federal Aviation Administration at the National Aviation Facilities Experimental Center (NAFEC) at Atlantic City. The results of the NAFEC tests are discussed briefly later in this report.

Some of the parameters in Table 1 are not well known, and must be estimated in order for the equations to be solved. The primary mechanisms for heat transfer within the flame are radiation and convection. Heat transfer by radiation depends not only on the intensity of the source but also on the absorptance of the receiver. The radiant output of the fire, q_r, was assumed to be equal to 31,000 Btu/hr-ft^2, a value obtained by Copley in fire tests using JP-4 as the fuel (2). Since soot deposits rapidly darken the aircraft skin, the absorptance, α, was assumed to be unity. Likewise, the emittance, ϵ, for the surface was assumed to be unity. The convective heat transfer coefficient, h, was estimated to be 5 Btu/hr-ft^2. The estimate was based on forced convection at gas velocities of about 20 ft/sec, and corresponds quite closely to recent data obtained by Neill in direct flame contact heat transfer measurements (7). The flame temperature, T_f, was taken to be about 2000 °F, a value based on optical pyrometer readings on hydrocarbon flames.

It should be pointed out that any parameter dependent on flame properties is not constant. Fluctuations occur which have periods ranging from a fraction of a second to at least several seconds, depending on the turbulence of the flame and the gross movement of the flame due to the effects of external factors such as the wind. However, when the thermal sink is large enough, the small-scale fluctuations, such as those due to turbulence, are damped out.

TABLE 1

NUMERICAL VALUES OF FIRE DAMAGE MODEL CALCULATIONS

Parameters	ALUMINUM		STAINLESS STEEL	
	Value	Ref	Value	Ref
q_r	31,000 $\frac{\text{Btu}}{\text{hr-ft}^2}$	2	31,000 $\frac{\text{Btu}}{\text{hr-ft}^2}$	2
k	0.7 $\frac{\text{Btu}}{\text{hr-ft}^2}$ -°F-in	3	0.7 $\frac{\text{Btu}}{\text{hr-ft}^2}$ -°F/in	3
z	0.5 inches	3	0.5 inches	3
ρ	175 lb/ft^3	3	508 lb/ft^3	3
c	0.23 Btu/lb-°F	8	0.12 Btu/lb-°F	3
ΔH_f	170 Btu/lb	10	NA	
T_B	900°F	3	NA	
T_E	1200°F	3	NA	
h	5 $\frac{\text{Btu}}{\text{hr-ft}^2}$	(a)	5 $\frac{\text{Btu}}{\text{hr-ft}^2}$	(a)
T_o	80°F	(a)	80°F	(a)
T_f	2000°F	(a)	2000°F	(a)
α	1.0	(a)	1.0	(a)
ϵ	1.0	(a)	1.0	(a)

(a) See discussion in text

FAA FIRE TESTS

Before presenting the results of the calculations on the fire damage model, a brief discussion of the FAA fire tests is in order. Eight tests were run by the FAA at NAFEC. The tests were made using a full-scale section of a Boeing 707 fuselage with Kaowool insulation between the regular aircraft skin and an outer aircraft skin which was being tested. The original test configuration was described by Conley (1). As originally planned, tests were to be run with fires to be burned upwind of the fuselage as well as in direct contact with the fuselage.

The first series of tests was composed of four runs using stainless steel as the outer skin (or test skin) on the aircraft test section (4). A sketch of the test setup appears in Figure 2.

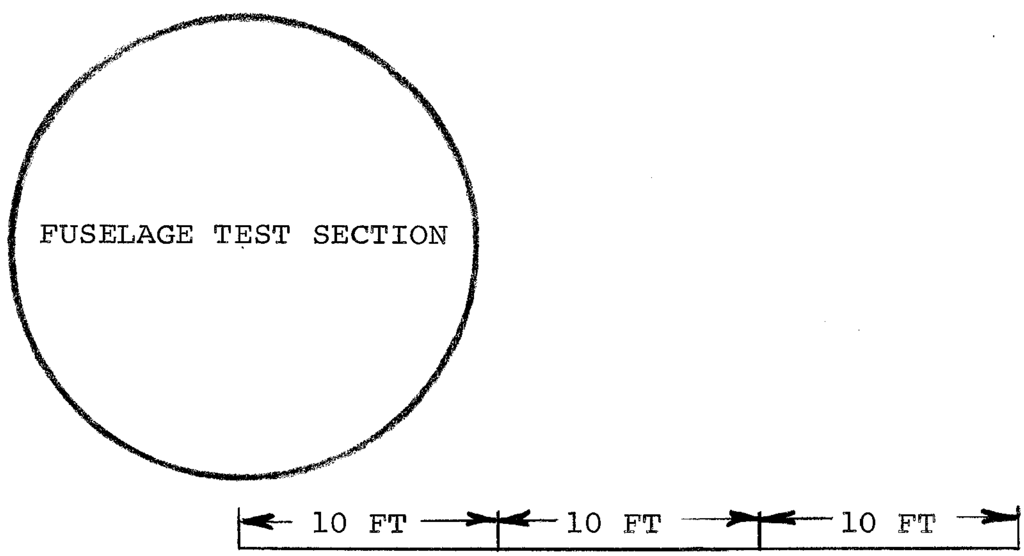


Figure 2. Schematic Diagram of FAA Test Setup for Stainless Steel Tests (Not to Scale).

Three fire pits were dug, each 10 feet wide and 30 feet long. The first pit was directly beneath the aircraft test section, and the second and third were directly upwind of the first. The four tests were run as follows: Test No. 1 - Pit No. 3, Test No. 2 - Pit No. 2, Test No. 3 - Pit No. 1, and Test No. 4 - both Pits No. 1 and No. 2. These fires were designed to burn under relatively mild conditions in order to determine the integrity of the instrumentation and check the test setup. It was found that the ambient wind was strong enough to cause incomplete fire coverage in some of the tests, which resulted in low heating to the test section.

The second series of tests utilized aluminum test sections attached in spaces which were cut out of the stainless steel skin (5). The fires for the aluminum tests burned above a pit 50 feet square located as shown in the sketch in Figure 3. This larger pit was designed to provide relatively complete fire coverage and maximum fire exposure.

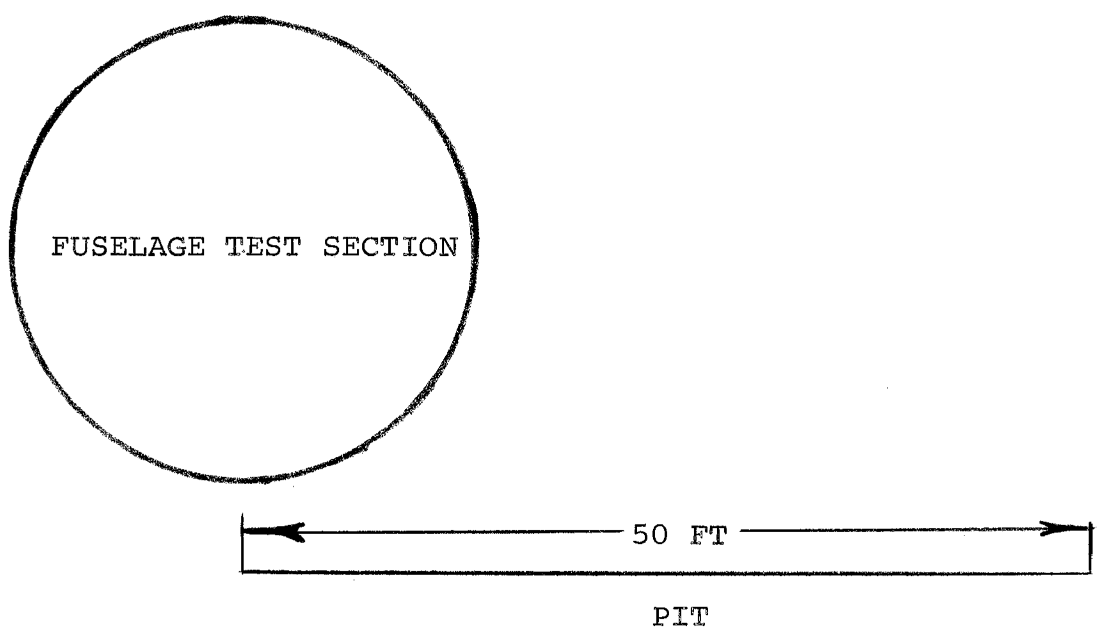


Figure 3. Schematic Diagram of FAA Test Setup for Aluminum Tests (Not to Scale).

The backing for the aluminum tests differed from that of the stainless steel tests. The insulation used was a standard two-inch-thick layer of Type AA glass fiber with a backing sheet of polyvinylchloride plastic. This test setup was used in order to simulate more nearly actual aircraft construction. The aluminum tests were run using four thicknesses of skin material in order to measure the failure response over a wider range of conditions.

STAINLESS STEEL RESULTS

The results of the calculations made for a stainless steel skin 0.031 inches thick are shown in Figure 4. The line is the result of calculations made until the skin temperature reached about 1500°F. The parameters shown in Table 1 were used in Equation 7 to obtain the calculated results. The open points on Figure 4 are the results of the FAA fire tests (4). An examination of the original FAA data showed a delay of approximately 13 seconds from the point of ignition until the fire built up sufficiently to cover the test area on the simulated aircraft fuselage. This delay time due to fire buildup has been used to adjust the data points, as shown by the solid points in Figure 4. Each solid point represents the same reading as the open point at the same temperature, but it has been shifted to the left side of the graph by 13 seconds. The adjusted data are seen to correspond quite well to the predicted result.

The reason for shifting the data points instead of the calculated curve is based on an analysis of the situation prevailing following a crash and on the goal of the mathematical model. Following a crash, it might be expected that fuel would be splashed over the fuselage as well as on the ground around it. Total maximum involvement of the aircraft in the fire would be expected to occur with little or no delay. Since the goal of the mathematical model is to predict the maximum fire hazard, the data were shifted to correspond to immediate involvement of the fuselage.

It should be pointed out that the FAA data shown in Figure 4 were taken from Test No. 2. Data from the other three stainless steel tests showed slower temperature rise in the aircraft skin. The slower rise was due to poor fire coverage, which was caused by wind conditions at the time of the test.

Figure 5 shows the calculated damage time for stainless steel as a function of the thickness of the stainless steel skin. Lines are shown for skin temperatures of 900°F and 1500°F. The two data points shown are adjusted data from Figure 4. They

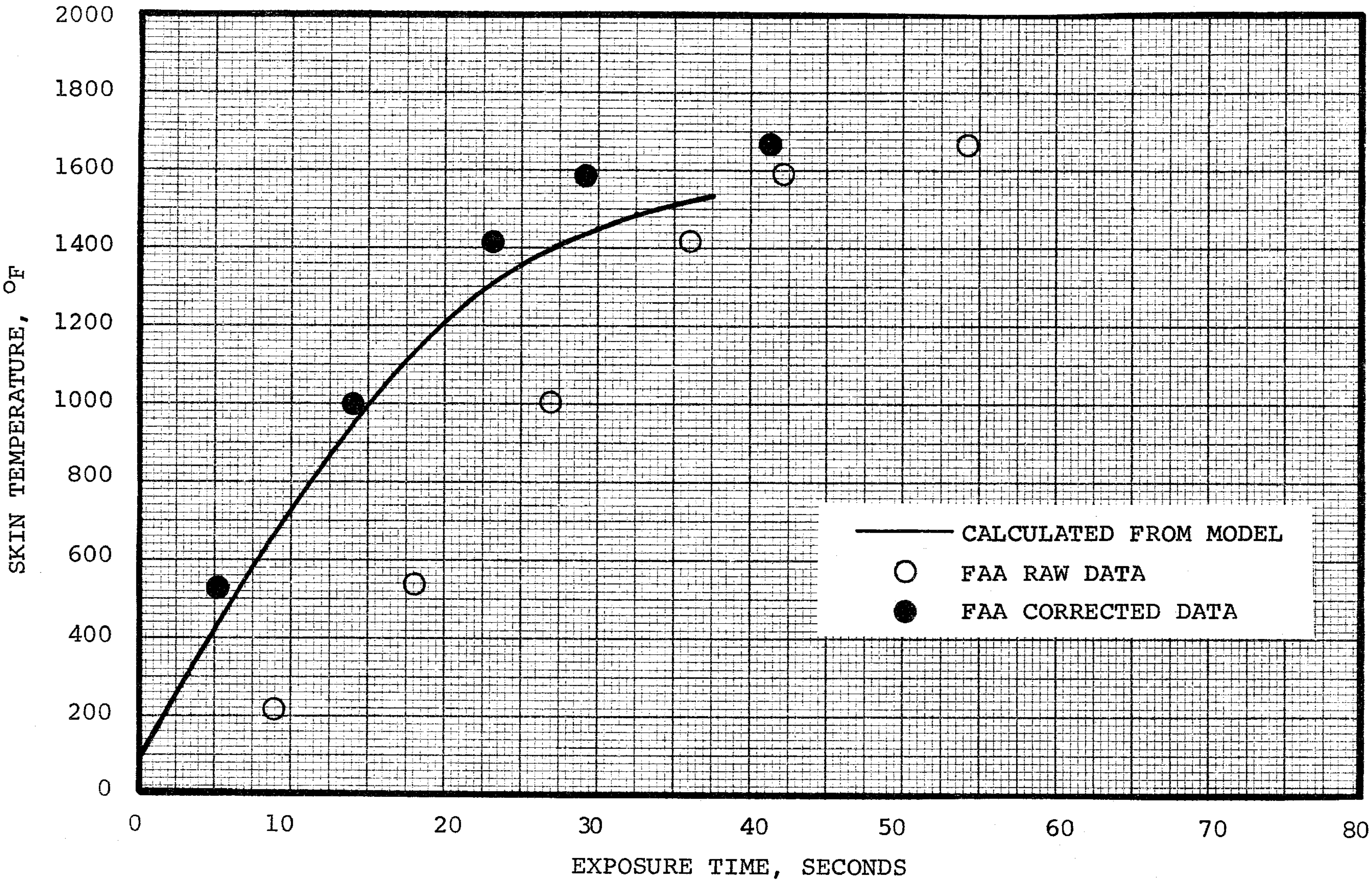


Figure 4. Skin Temperatures for 0.031-Inch Stainless Steel

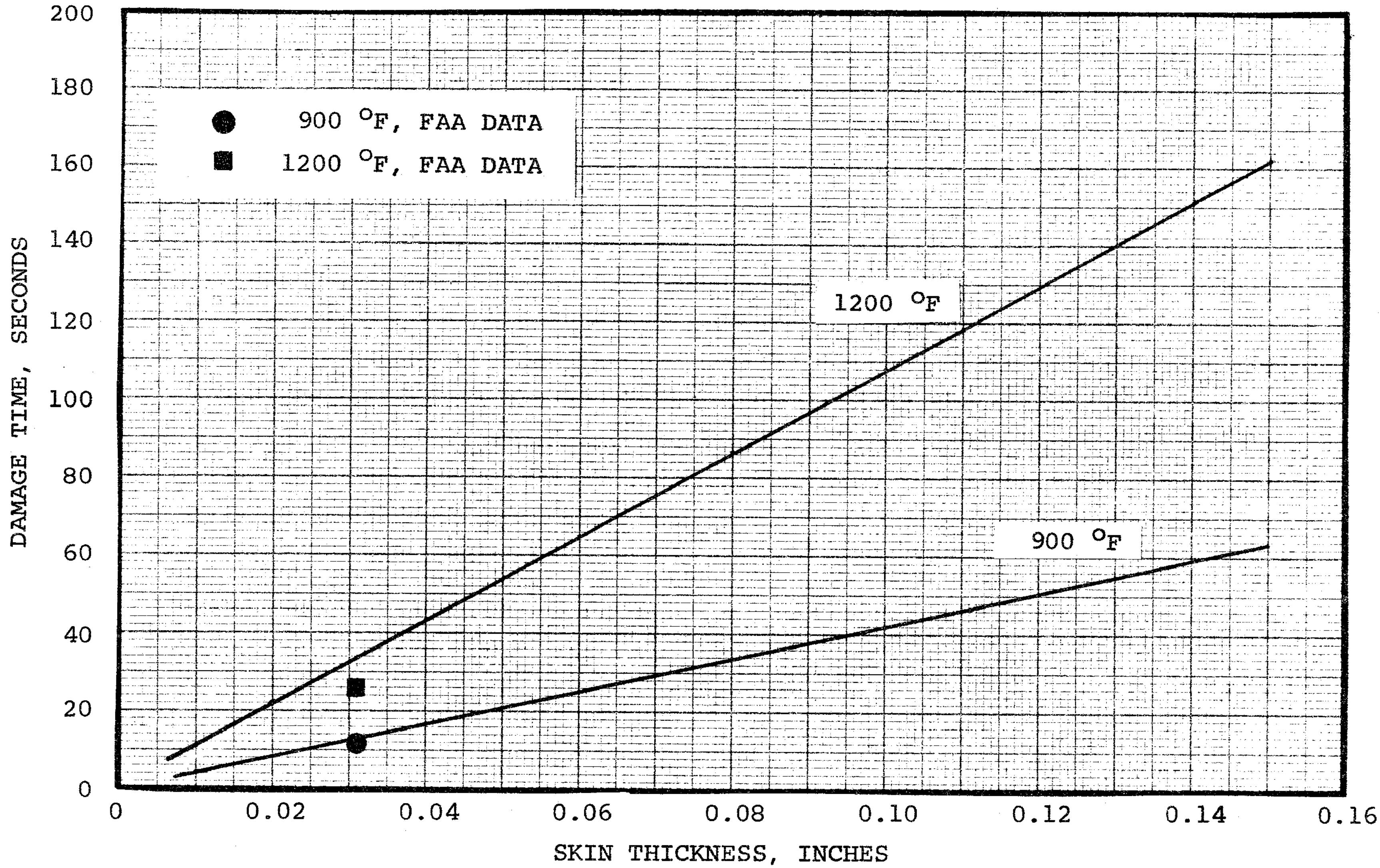


Figure 5. Damage Times for Stainless Steel Aircraft Skin

fit the calculated curve quite well. Data points from other tests are not included because the aircraft skin was not fully involved in the fire.

ALUMINUM RESULTS

The results of the calculations made for aluminum aircraft skins are shown in Figure 6 for a thickness of 0.020 inches and in Figure 7 for a thickness of 0.090 inches. The data from the FAA fire tests (5) are shown on each of the figures. The open points are the direct temperature measurements and the solid points have been adjusted as described for the stainless steel tests. The calculated curves give a reasonable fit to the experimental data, although there are some deviations. The 0.090-inch test shows a rather wide deviation from the calculated curve as the melting point was reached. The reason for the deviation is not obvious, particularly in view of the close agreement up to about 900°F. It is possible that as the aluminum sheet began to soften, it became partially dislodged and was heated on both sides.

Figure 8 indicates more strongly the non-typical results of the 0.090-inch tests. In Figure 8, the damage time for aluminum at two levels is plotted as a function of the aluminum thickness, assuming maximum fire exposure. The two levels chosen for Figure 8 were the temperature at the start of melting and the temperature at which melting was complete. For the aluminum alloys in these tests (2024-T3 and 7075-T6) the beginning temperature could be approximated by 900°F and the ending temperature could be approximated by 1200°F. The calculated and experimental points show reasonable agreement except for the anomalous point for the 0.090-inch test.

DISCUSSION OF RESULTS

The results of comparisons of stainless steel and aluminum calculations and experimental test results indicate that the calculations are adequate for use as a method of estimating the approximate time required for damage to an aircraft in a post-crash fire. The elapsed time at which the fire has burned through the aircraft skin can be calculated if the aircraft skin thickness is known. However, for most aircraft, the skin thickness varies at different locations on the fuselage.

Figure 9 shows the minimum aircraft skin thickness as a function of the gross weight of the aircraft, according to data

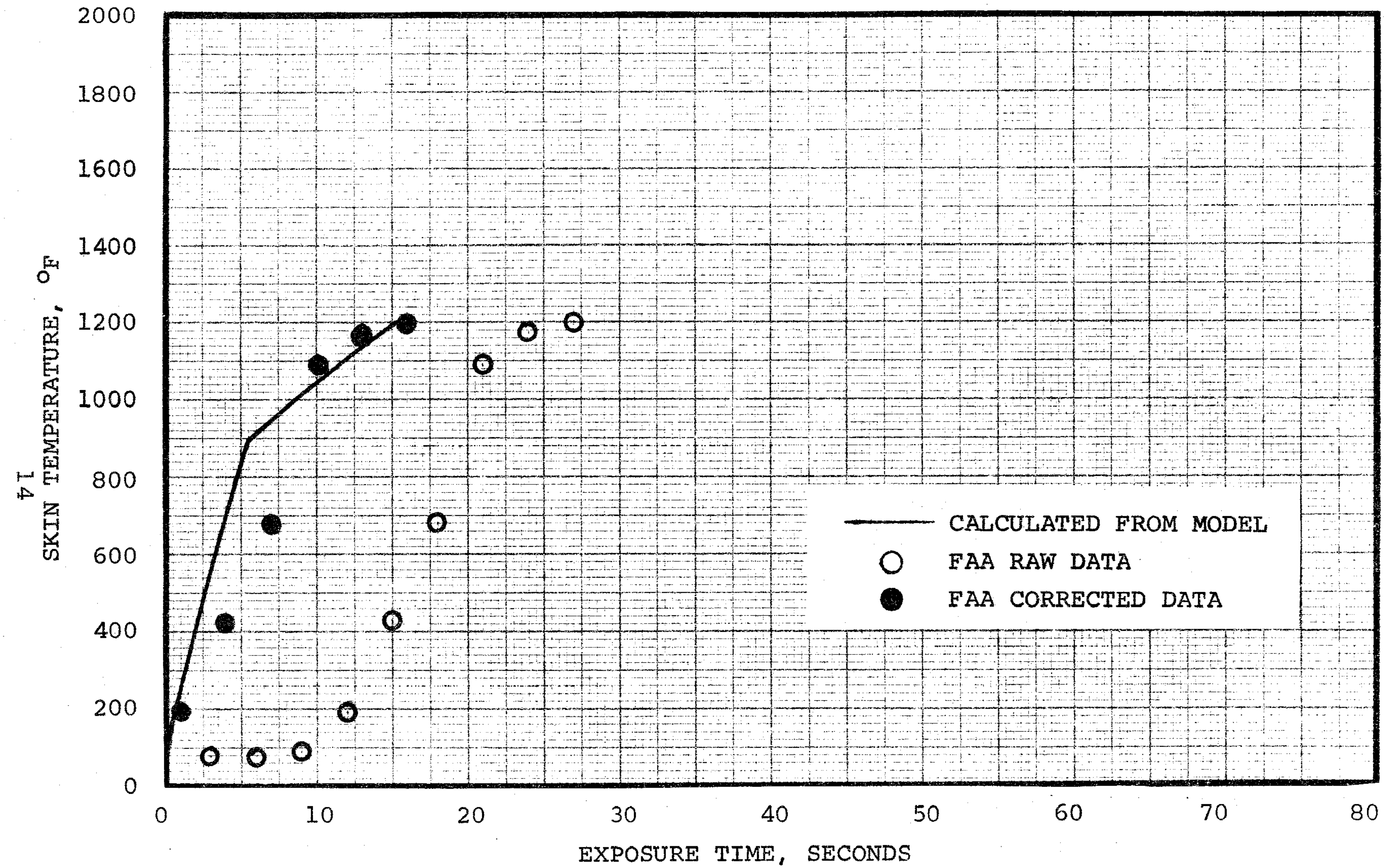


Figure 6. Skin Temperatures for 0.020-Inch Aluminum

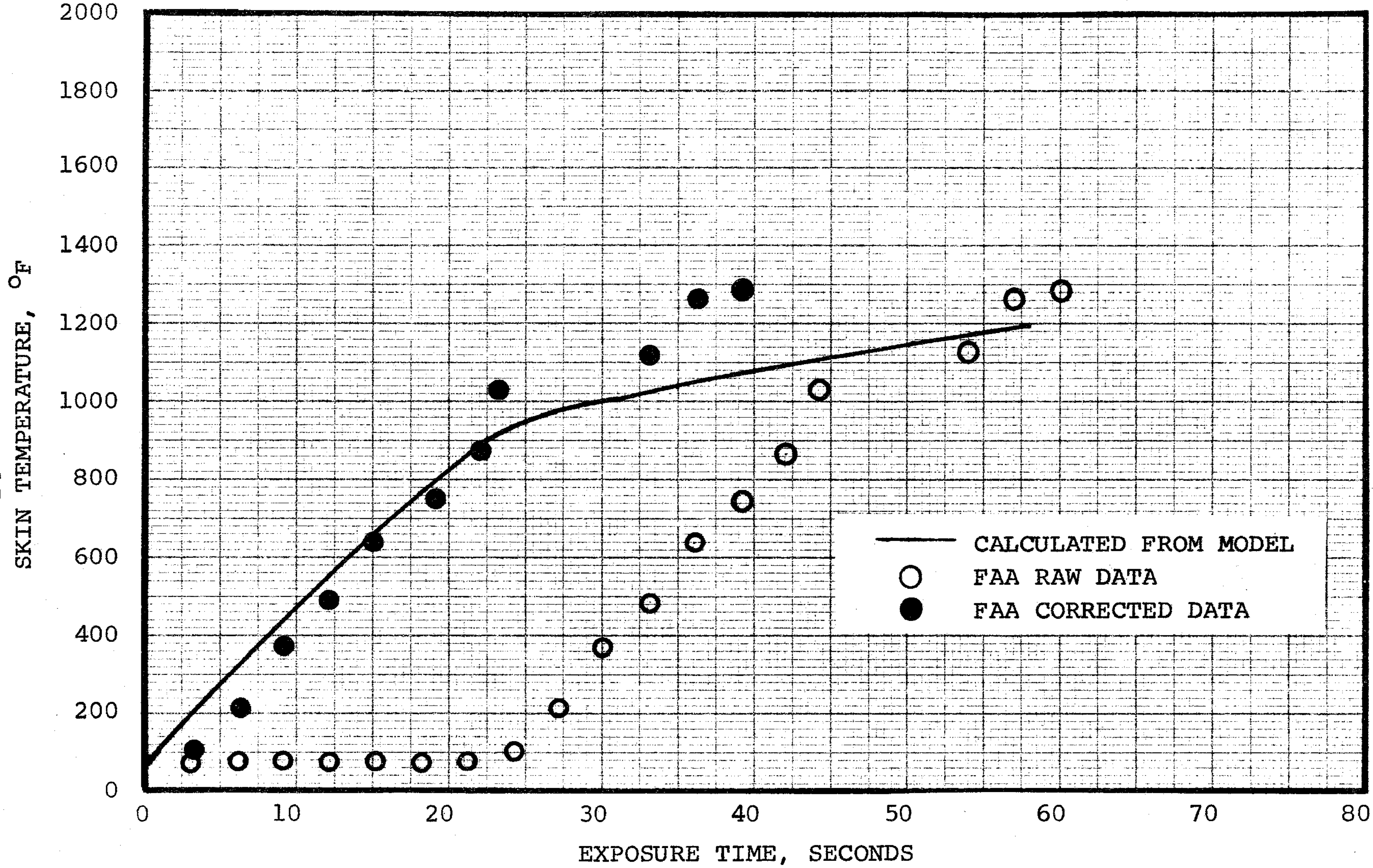


Figure 7. Skin Temperatures for 0.090-Inch Aluminum

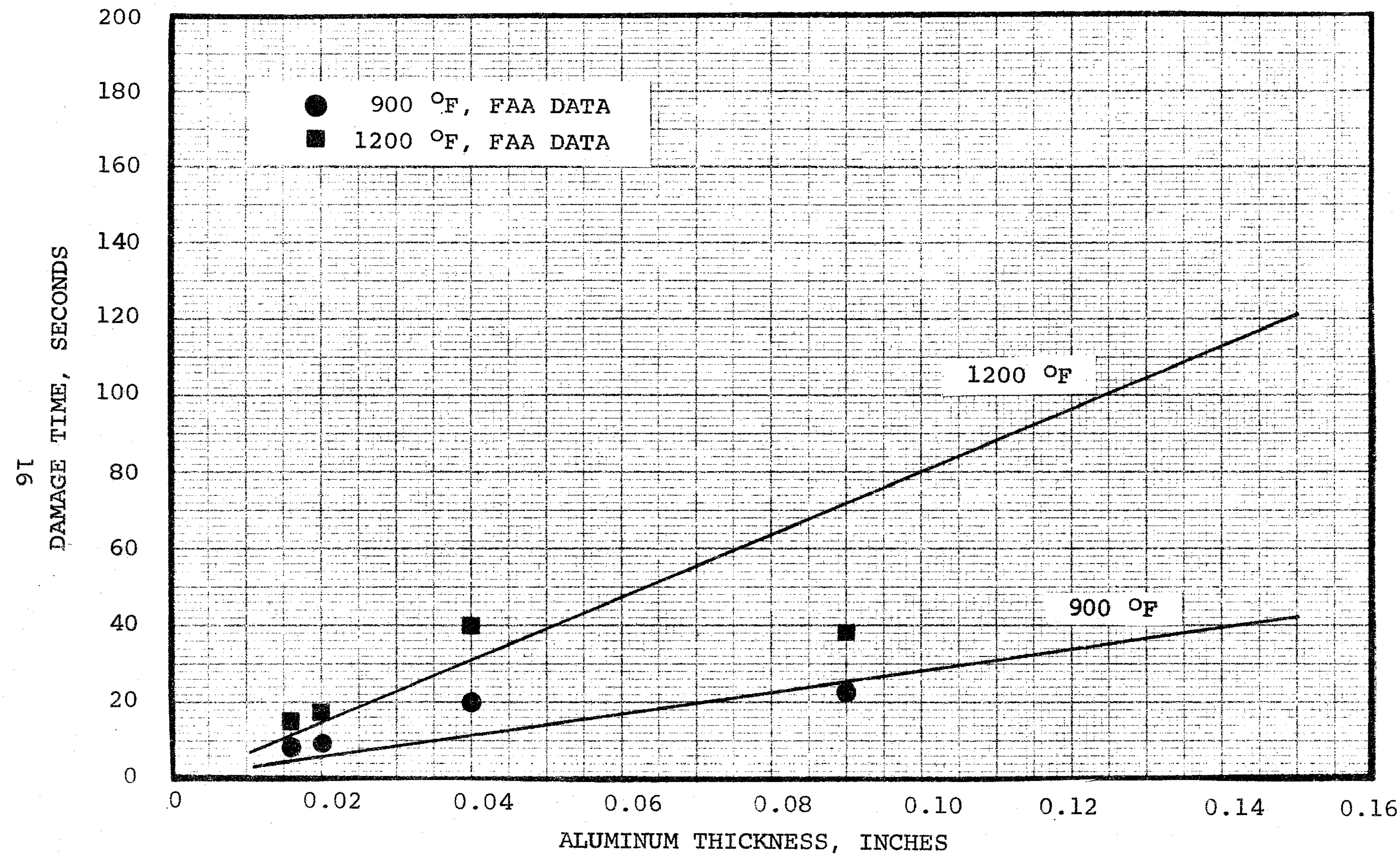


Figure 8. Damage Times for Aluminum Aircraft Skins

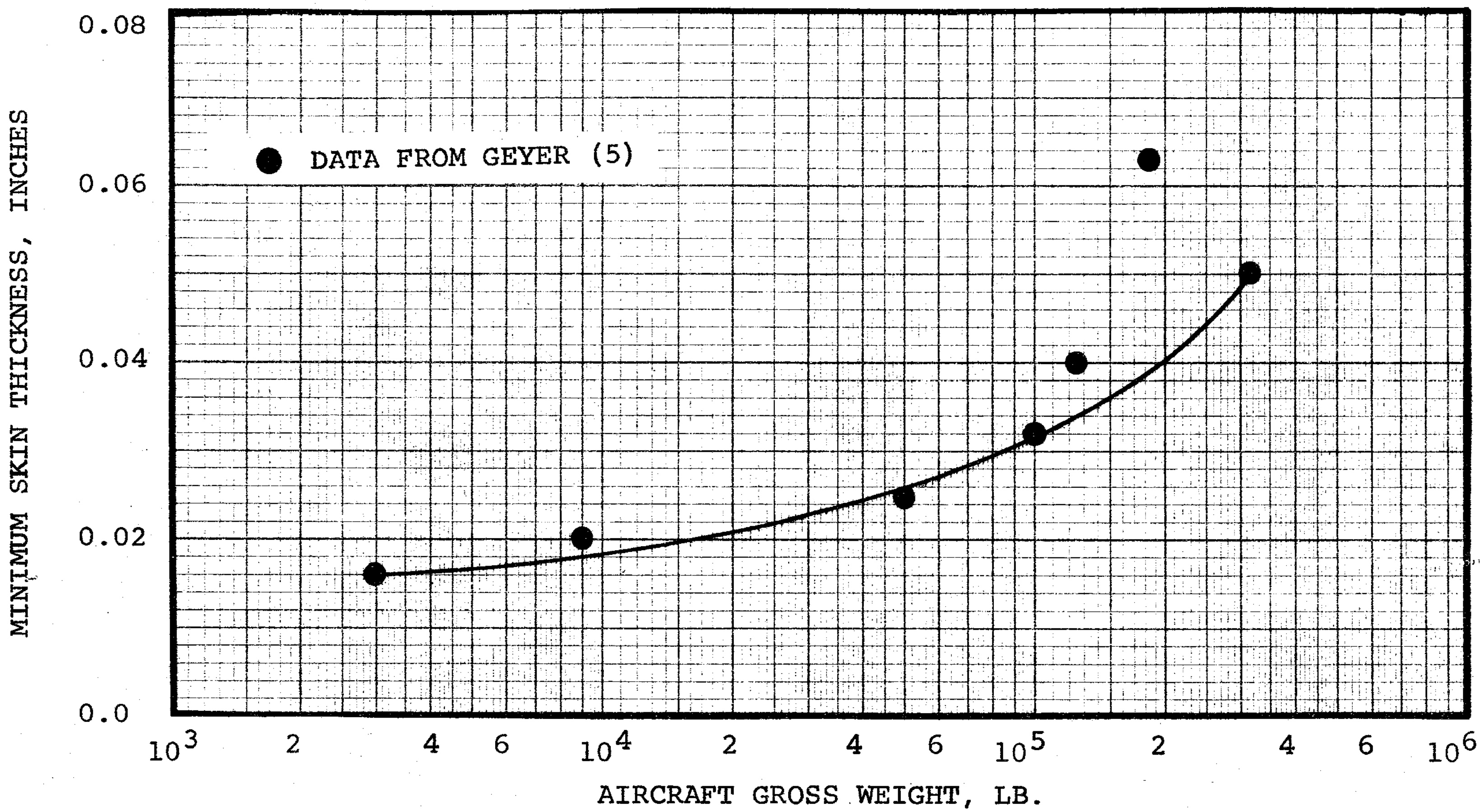


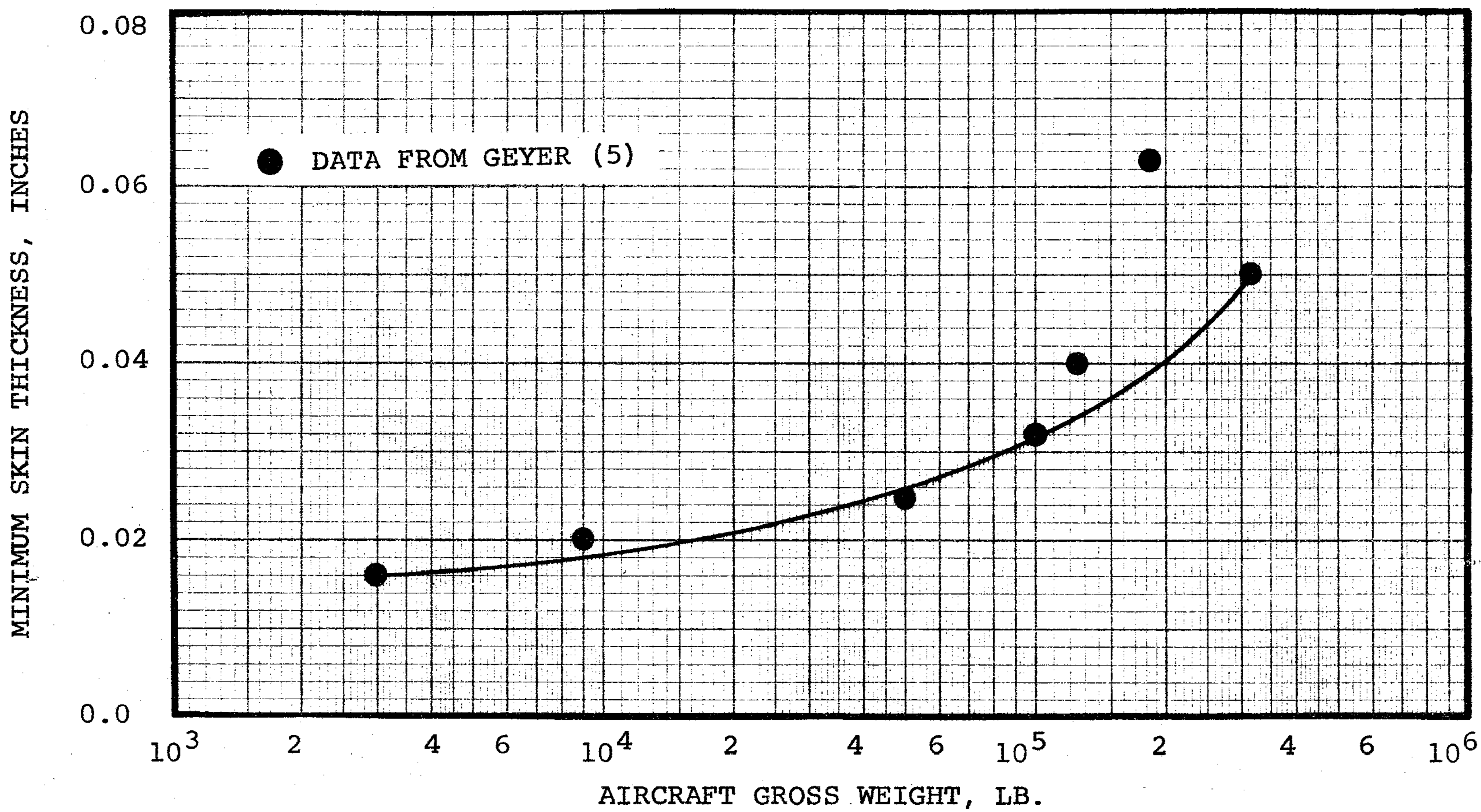
Figure 9. Minimum Skin Thickness of Current Commercial Aircraft

collected by Geyer (5). The curve shown gives a reasonable fit to the data points, which are for aircraft of several manufacturers and range from small single-engine aircraft to inter-continental jet aircraft. It must be emphasized that Figure 9 gives the minimum skin thickness for a given aircraft gross weight; the maximum skin thickness on the same aircraft may be several times the minimum.

A crossplot can now be made using Figures 8 and 9. From Figure 9, the minimum skin thickness of the aircraft is found. Then the time required for burning through the skin, taken as the time required to reach 1200°F, is found from Figure 8. The result is then plotted, obtaining a curve showing the minimum skin melting time as a function of the aircraft gross weight, as shown in Figure 10. The curve in Figure 10 shows that the aircraft skin melting time varies from as little as about 10 seconds for small aircraft to nearly 40 seconds for the large aircraft. These melting times are based on immediate fire involvement and a large fire, so they represent the minimum time available for fire suppression before the fire penetrates the cabin. Should ignition not occur immediately, or if a short time was required for the fire to build up, more time would be available for suppression. However, neither of these can be counted on in a post-crash situation. Therefore, fire suppression techniques and equipment should be designed for use within the minimum time or changes in aircraft design should be made to extend the minimum skin melting time if protection of passengers and crew is to be obtained.

The problem of extinguishment is a rather difficult one to solve. For example, the fire ensuing the crash of a large jet aircraft will cover an area of approximately 9000 ft^2 (5). Recent data (9) indicate that even if fire fighting equipment was available on the spot, more than a minute would be required for fire control. Even though partial extinguishment would provide some aid to passengers, it is unlikely that the fire fighting equipment could reach the scene in time to provide significant aid. It therefore appears that design of on-board protection systems or methods of extending the protection time should be investigated.

There are several possibilities for improving the skin damage times for aircraft. For example, non-melting skins and heat resistant cabin insulations might be used. A stainless steel clad skin might provide some added protection but would add to the aircraft weight. An intumescent paint might be used for coating the aircraft. In a fire environment, the paint would foam up and help to insulate the skin from the heat of the fire.



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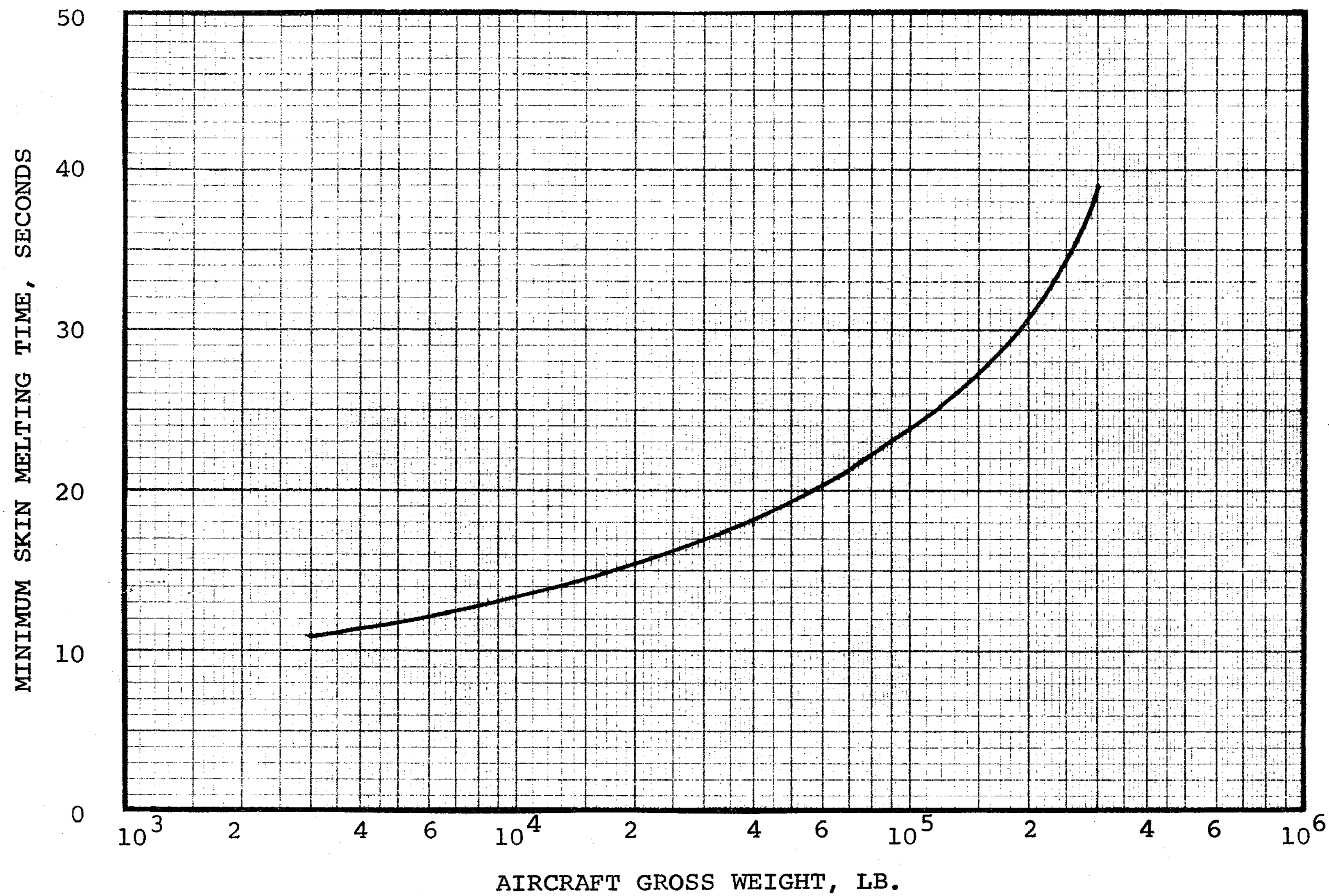

Figure 10. Minimum Skin Melting Time for Aircraft Involved in Maximum Intensity Post Crash Fires

One other factor should be mentioned. The windows on aircraft are usually made of an acrylic plastic. The plastic is flammable and would surely ignite during fire exposure. The burning time of the windows was not considered in this study. It is not known whether the windows would burn through before the aircraft skin melted or whether they would provide better protection.

CONCLUSIONS

The following conclusions are drawn based on the results of this study:

1. The mathematical model developed in this report is adequate to predict the damage time (based on skin melting) of aircraft.

2. The damage time, which is less than 40 seconds for current commercial aircraft, is too short to permit adequate rescue operations using current equipment on potential fires involving current aircraft.

RECOMMENDATIONS

The following recommendations are made based on the results of this study.

1. Studies should be made to determine if the acrylic windows of aircraft would offer protection from fires at least equal to the skin melting time.

2. Studies should be made on aircraft skin design to find a method such as intumescent paint or improved skin material which would extend the damage times of current aircraft skins.

3. On-board protection and escape equipment should be studied.

Both items 1 and 2 could be carried out in the laboratory on small-scale samples, and item 3 could be studied theoretically, all at relatively small cost compared to large outdoor tests, particularly in view of the possible improvement in aircraft safety.

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