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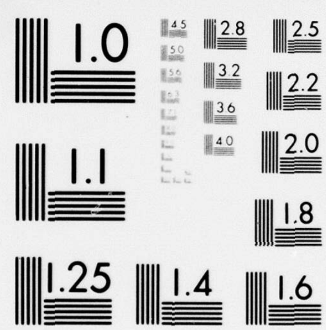

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FEASIBILITY OF NON-CATAPULT EJECTION AND HAZARD OF AN EJECTION SEAT ROCKET PLUME

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Efforts conducted by the Navy focused on feasibility of utilizing a rocket propelled ejection seat without a catapult. It was speculated that elimination of the catapult would result in significant weight reduction. However, between time of rocket ignition and the time the ejecting crewmember leaves the cockpit, there is a burn hazard to the crewmember created by the rocket exhaust plume from his own rocket as it scatters about the cockpit. The crewman may be exposed to temperatures as high as 5000°F during ejection.

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20. Extreme temperature and blast pressure also create severe erosive effects. Methods for protecting the crewmember from the rocket plume which were investigated included venting the plume, containing/shielding the plume, and quenching afterburn of rocket propellant with inert gas. Engineering trade-offs such as weight penalties, complexity of maintenance, cost and anomalies on overall escape system performance were compared between utilization of the catapult and rocket plume protection methods. Based on these engineering trade-offs it was concluded that the catapult is still recommended for use in current aircraft cockpits.

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BACKGROUND

In recent years the Navy has investigated the possibility of developing an ejection seat which could be propelled solely by rocket; this concept implies elimination of the catapult system which is common to all ejection seats and is used in conjunction with a rocket system. The catapult thrusts the crewmember and ejection seat upward for the first several feet before the rocket ignites to continue the propulstion sequence. Elimination of the catapult through incorporation of only rocket propulsion would reduce weight by approximately 10 to 20 pounds (4.54 to 9.08 Kg).

However, there is a thermal hazard for non-catapult ejection created by the rocket plume (rocket exhaust). Due to the dimensional confines of the aircraft cockpit, the rocket plume (which is about three to four times the length of the seat when in free space) reflects off the cockpit floor and walls. The crewmember is engulfed by hot gases and struck by high blast pressures from the rocket exhaust from his own ejection seat during the time from rocket ignition until seat/aircraft separation. (This problem is not to be confused with another common ejection problem where a crewmember in a multi-manned aircraft receives burn injuries from the rocket plume of another seat during sequenced ejections.)

For the preliminary development phase a thrust versus time profile for the ejection seat rocket was selected to give the best escape benefit for various aircraft attitudes. The rocket plume impingement hazard was investigated for an ejection seat rocket which had a thrust versus time profile as shown in figure 1. The seat/man weight for this profile is assumed to be approximately 350 pounds (158.8 Kg). For this thrust versus time profile it takes approximately 0.40 seconds of rocket burn before the seat/man leaves the aircraft. Film coverage of experimental rocket ejection tests reveal that the crewman is still subjected to hot gases from the reflected plume for several feet after the seat leaves the guide rails. It must also be remembered that life support equipment and recovery system components must also be protected from the damaging effects of the plume.

The best data to date pertaining to the thermal hazard of non-catapult ejection was obtained with placement of white laboratory rats on a test dummy during a rocket ejection seat firing[1]. Rats on the dummy of the ejected seat were situated on the face, right shin, left hip and right shoulder. The rats on the face and hip were badly burned; the rats on the shin and shoulder received second-degree burns. The results of this test verify the damaging effects of the rocket exhaust in the cockpit as theorized in the following text.

PROBLEM DEFINITION

Before attempting to design a system for protecting the crewmember from the rocket plume, it is desirable to define the problem, that is, to determine

[1]*Stoll, A.M.; Assessment of Thermal Hazard from "MEW" Type Ejection Seat Firing; letter report concerning Report No. NADC-MR-7024, Naval Air Development Center, Warminster, Pa., 18 Feb 1971.*

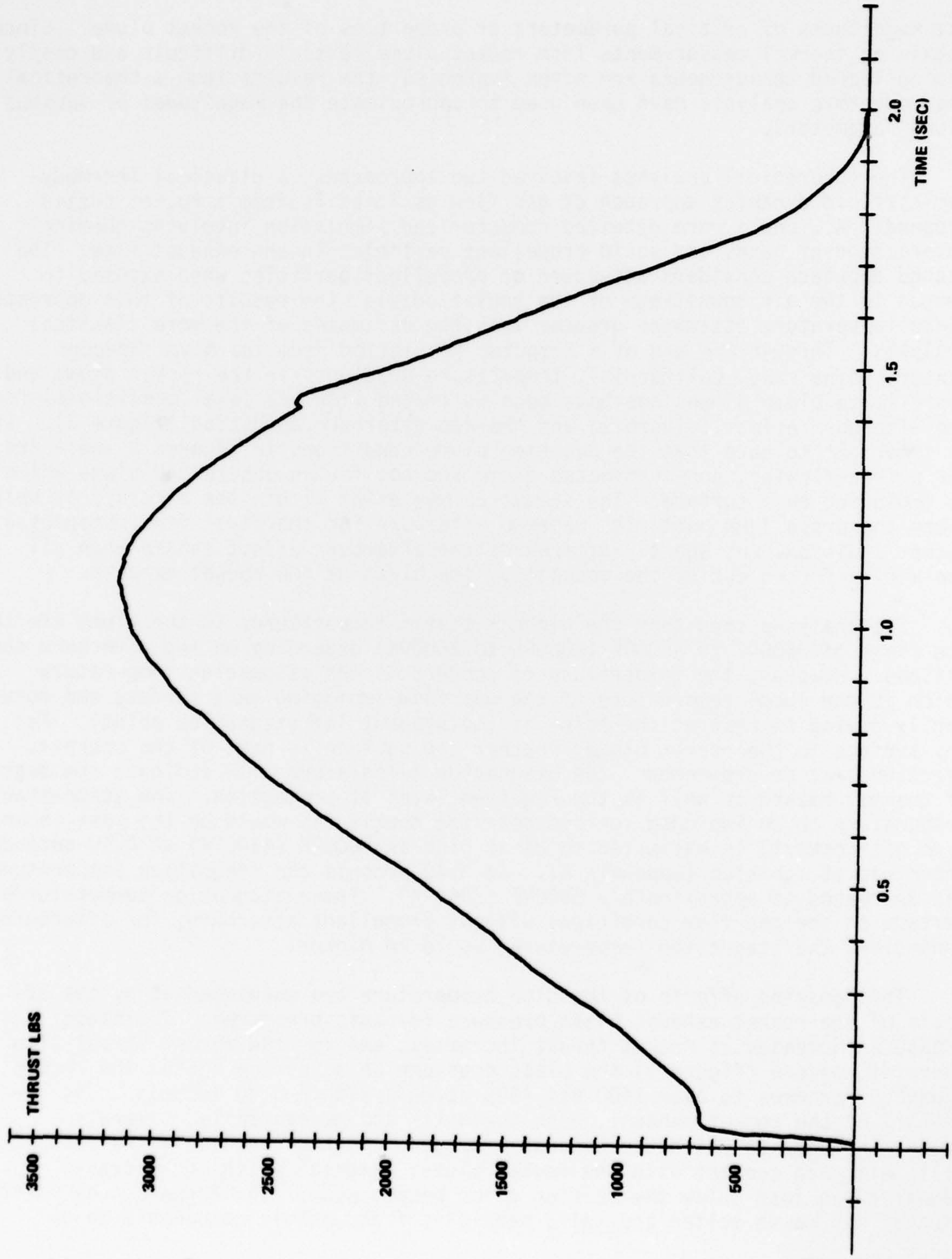

FIGURE 1 - Thrust versus Time Profile for Experimental Rocket Motor

the magnitudes of critical parameters or properties of the rocket plume. Since obtaining thermal measurements from rocket plume tests is difficult and costly and collected measurements are often imprecise, the results from a theoretical thermodynamic analysis have been used to approximate the magnitudes of various plume parameters.

The theoretical analyses involved two approaches, a classical thermodynamics/fluid dynamics approach of gas flow as it exits from a rocket nozzle (appendix A), and a more detailed computerized simulation involving chemical interaction of gases and solid propellant particles in the exhaust flow. The second approach considers afterburn of propellant particles when exposed to oxygen in the air downstream of the rocket nozzle; the results of this approach yield temperature estimates greater than the estimates of the more classical analysis. Through the aid of a computer simulation from the Naval Weapons Center, China Lake, California[2], temperature gradients in the rocket plume and approximate plume dimensions have been estimated (for sea level conditions) for the afterburn effect (figure 2) and the non-afterburn condition (figure 3). It is important to note that the depicted plume conditions in figures 2 and 3 are for a free-flowing, non-obstructed plume and not for an obstructed plume which is impinging on a surface. The situation may exist within the aircraft cockpit where the propellant particles undergo afterburn for the first few instances of rocket ignition, but shortly afterwards the afterburn effect ceases when all the air is forced out of the cockpit by the blast of the rocket exhaust.

The analyses show that the highest static temperatures in the plume are in the range of 3600°F to 4500°F (2000°K to 2500°K) depending on the afterburn conditions. However, the temperature of concern is the stagnation temperature which is the local temperature of the gas flow impinging on a surface and momentarily coming to rest at the point of impingement (or stagnation point). For any surface in the rocket plume, whether the surface is part of the cockpit, ejection seat or crewmember, the stagnation temperature will indicate the degree of thermal hazard as well as the required level of protection. The stagnation temperature on an impinged surface near the nozzle (as would be the case in an aircraft cockpit) is estimated to be as high as 7600°F (4500°K) at 0.10 seconds after rocket ignition (appendix A). At 0.40 seconds the stagnation temperature has decreased to approximately 5000°F (3060°K). These stagnation temperatures pertain to the gas flow conditions without propellant afterburn; for afterburn conditions the stagnation temperatures would be higher.

The damaging effects of the high temperature are supplemented by the effects of the rocket exhaust blast pressure (dynamic pressure). The blast pressure increases as rocket thrust increases, and for the thrust versus time curve of concern (figure 1) the blast pressure on a surface behind the rocket exhaust increases to over 1400 psi (995 nt/cm^2) within 0.40 seconds. The intensity of the rocket exhaust, both thermally and mechanically, severely erodes surfaces on which the exhaust impinges. Most materials in the cockpit will melt upon contact with the rocket plume; material which is heated to temperatures just below the melting point become weak. The force of the rocket exhaust may cause molten and solid particles from cockpit components to be

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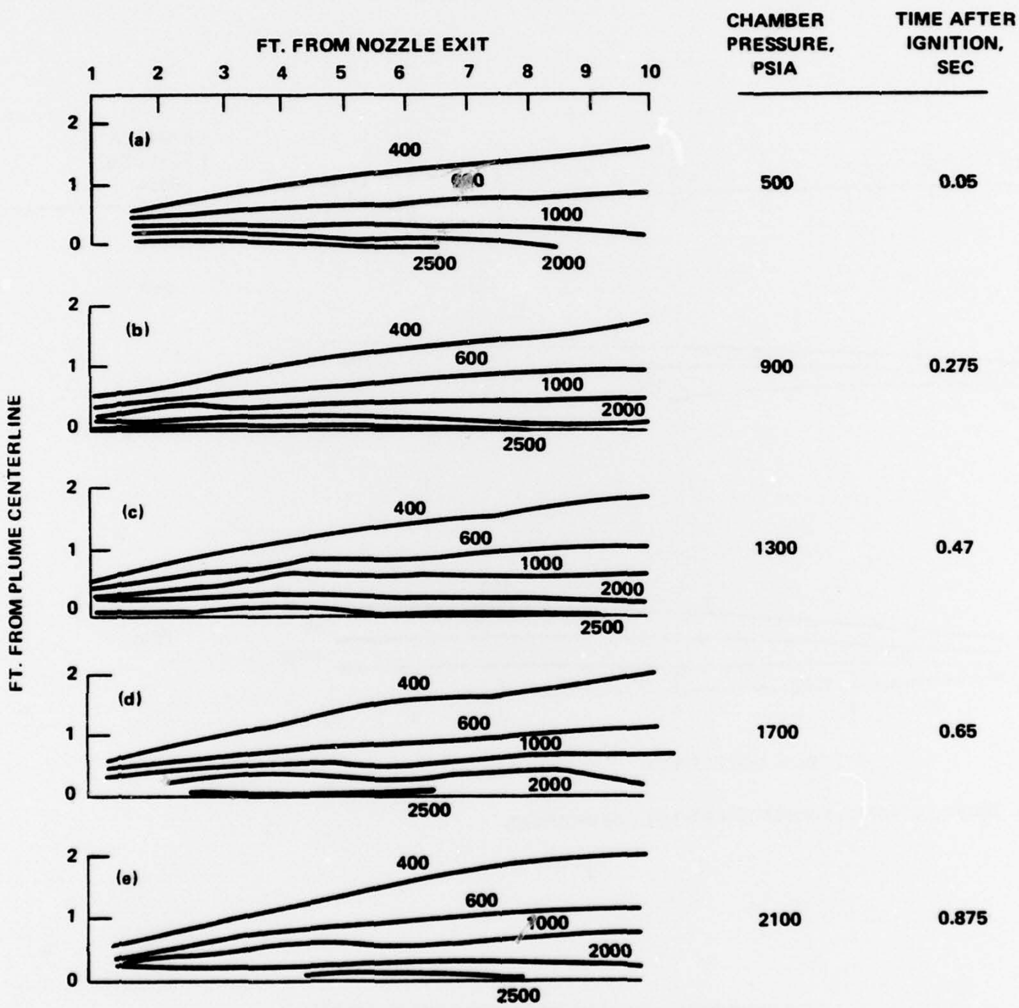
[2] *Victor, A.C.; Computer Analysis of Exhaust Plume Properties for Vertical Seeking Ejection Seat Rocket Motor, NWC-TM-3469, Naval Weapons Center, China Lake, Ca., May 1978.*



TEMPERATURE GRADIENTS ARE KELVIN DEGREES.

FIGURE 2 - Temperature Gradients Across Rocket Plume with Propellant Afterburn

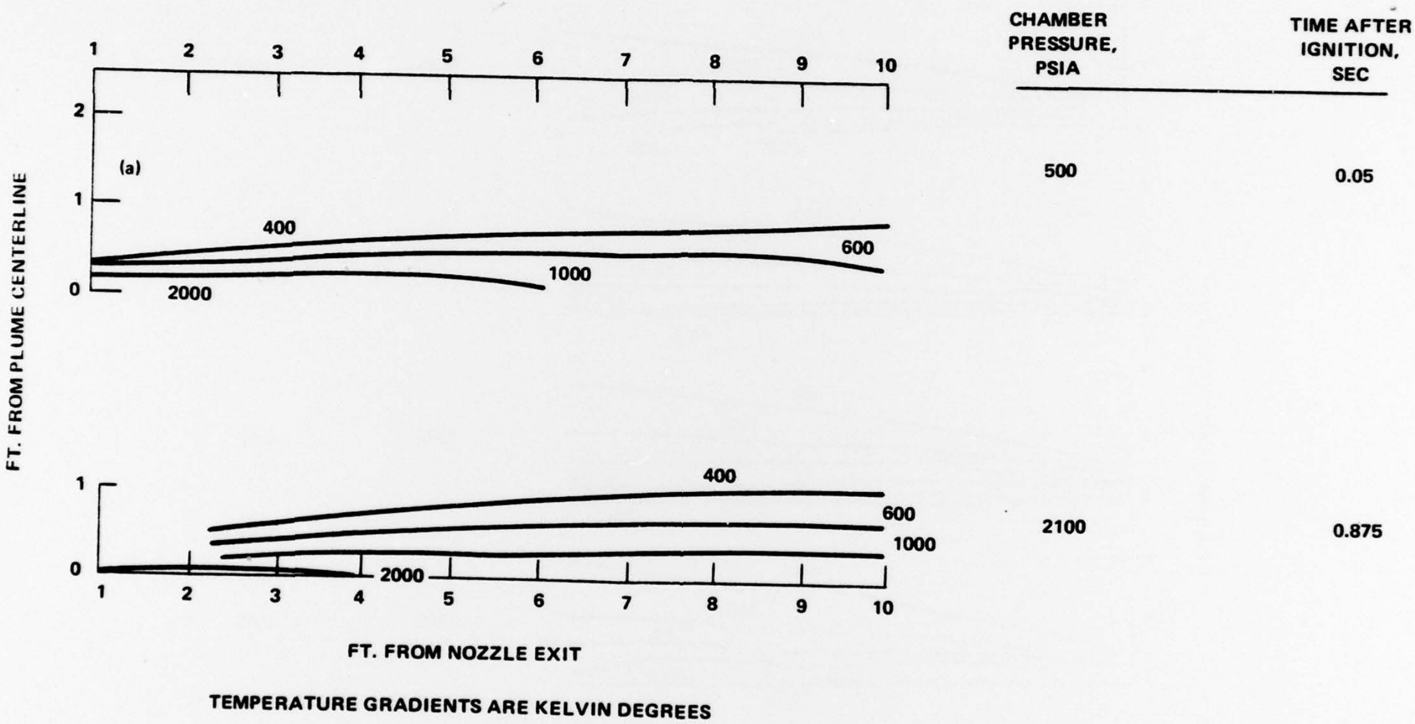


FIGURE 3 - Temperature Gradients Across Rocket Plume without Propellant Afterburn

spewed about the cockpit. Unburnt propellant particles also circulate about the cockpit. These hot particles may splatter on the crewman and partially unburnt particles may undergo renewed burning when the crewmember is ejected out of the oxygen-starved cockpit and into the oxygen-laden atmosphere.

Effects of lower ambient pressures due to ejection at higher altitudes and effects of aerodynamic turbulence in the cockpit due to high airspeeds should be minimal with regard to significant changes to the rocket plume dynamics.

Protecting the crewmember from the effects of the rocket plume presents a difficult engineering problem. The problem is especially difficult because design considerations such as cost, weight, maintainability and interaction with other systems and structures on the seat and in the aircraft must also be addressed.

FEASIBILITY OF DESIGN ALTERNATIVES

The various design approaches for protecting the crewmember from the rocket plume which were investigated for feasibility were as follows:

1. Plume venting,
2. Plume quenching, and
3. Plume containment.

Each of these methods of protection were compared to the standard method of the catapult assist. The advantages and disadvantages of these methods are discussed in the following text.

PLUME VENTING

If the rocket plume were allowed to flow freely and not reflect off any surfaces, then the rocket plume hazard would be eliminated. To achieve this condition in current aircraft cockpits is extremely difficult and very costly. The plume would have to be vented through the cockpit floor. This would imply a hole with a diameter almost as large as the diameter of the rocket plume. The most probable approach would be to detonate a line charge to cut the hole in the floor. Secondly, obstacles under the floor would have to be cleared away, otherwise the plume would be reflected back into the cockpit. Retrofitting aircraft cockpits for this purpose is complex and costly with the possibility that a safety hazard would still exist because of the use of detonation charges. Perhaps, the use of the rocket plume venting would be more feasible if cockpit design allowances were incorporated in a new aircraft; of course, this would mean that a rocket propelled ejection seat was already designated for use in the aircraft.

PLUME QUENCHING

This design approach consisted of flooding the cockpit prior to rocket ignition with a flame retardant gas such as carbon dioxide or a halogenated compound. This gas would prevent afterburn of propellant particles. However, this approach would only lower the temperatures to non-afterburn conditions, and, therefore, the temperatures would still be much too high for the safety of the ejecting crewmember.

PLUME CONTAINMENT

This approach involved containment of the rocket plume in an enclosed container. However, the thermodynamics of fluid flow into an enclosed volume implies an energy buildup in the volume; hence, temperatures and pressures are greater than stagnation temperatures and pressures resulting from free flow impingement conditions. Conservative estimates are that temperatures in the container would be greater than 6000°F and internal static pressures would be over 1000 psi (690 nt/cm^2). Most structural metals melt at temperatures less than 3000°F and become very weak at temperatures near the melting point. Some metals such as tungsten (2½ times heavier than steel and 7 times heavier than aluminum) with melting point of approximately 6000°F, tantalum (2 times heavier than steel) with a melting point of 5200°F, and molybdenum (1½ times heavier than steel) with a melting point of 4800°F are the most thermally resistant metals, but they are usually unsuitable for use in aircraft because of their weight.

A structure fabricated of almost any material can be designed to protect the crewmember from the rocket plume by containing the plume if the structure had enough mass; hence, it could absorb the heat and withstand the blast pressures generated as the seat was propelled out of the cockpit. Only the interior surface of the containing structure would melt. The amount of mass to be used would depend on the thermal and structural properties of material. However, the mass required to offer the proper protection could result in an unacceptable weight increase.

To complicate the design, the containing volume would have to be located behind the rocket nozzle and under the seat, but for normal aircraft flight the seat bottom is usually positioned a couple of inches above the cockpit floor. Therefore, the containment structure would have to be extendable to contain the rocket plume as the seat is propelled up the seat guide rails during ejection.

Thus, due to cockpit dimensional constraints and seat/aircraft weight limitations, development of a practical containment structure will be very difficult.

Additional areas of uncertainty indicating the risks of plume containment are:

- There may be an aerodynamic stability problem and ejected weight problem if the plume containment structure were ejected with the seat.

- There may be reliability problems involving the plume containment structure in the seat/aircraft ejection separation, especially if the structure was attached to the seat but was to stay with the aircraft after ejection.

- There is a possibility that the buildup of pressure in the containment volume will become equal to the chamber (internal) pressure of the rocket motor and seriously inhibit rocket exhaust flow (i.e., degrade thrust performance).

Hence, the plume containment method is considered to be a high risk development program, that is, the benefits of this method are doubtful and RDT&E (Research, Development, Test and Evaluation) needed to settle this issue would be very costly and time consuming.

A COMMENT ON EXPOSURE TIME TO HIGH TEMPERATURE

For the experimental rocket motor in this investigation, the crewmember was exposed to hot gases in the cockpit for approximately 0.40 seconds. This duration can be lowered to approximately 0.20 seconds to 0.30 seconds by increasing the rocket thrust during the beginning of the rocket burn. An increase in thrust would increase blast pressure and decrease stagnation temperature; unfortunately, the temperature would still be several thousand degrees farenheit.

It has been estimated that destruction of skin tissue will occur instantaneously when skin temperature reaches 161°F (72°C)[3]. Thus, the exposure time to the extreme temperatures near and within the rocket plume needs to be only a fraction of a second before skin damage occurs.

Current protective clothing, although helpful, will not offer complete protection against the intense temperature and pressure effects of the rocket plume. To provide sufficient protective clothing would create discomfort which would be unacceptable to the crewmember.

CONCLUSION

To eliminate the use of the catapult on ejection seats and to implement an ejection seat propelled solely by a rocket motor into current aircraft cockpits would require costly modification of the cockpit; such modification would not significantly reduce aircraft weight nor guarantee improved crewmember safety. Trade-offs combining weight limitations, reliability/maintainability, ejection seat performance, and cost indicate that the catapult is still a recommended propulsion technique for current aircraft cockpits.

Perhaps, for future aircraft designs provisions may be made — cost effectively — before production to accommodate a rocket (non-catapult) ejection seat in a newly-designed cockpit. At such a time, the benefits of an ejection seat propelled solely by a rocket system may be realized.

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[3] *Stoll, A.M. and M.A. Chianta; Heat Transfer through Fabrics as Related to Thermal Injury, Transactions of the New York Academy of Sciences, Series II, Vol. 33, No. 7, Pp 649-670, Nov 1971.*

APPENDIX A

ANALYSIS OF ROCKET PLUME PROPERTIES

The following analysis of rocket plume properties will identify the magnitude of certain critical parameters which affect the engineering design necessary to protect the crewmember from the rocket plume hazard.

The rocket plume will impinge upon a cockpit surface near the rocket nozzle as shown in figure A-1. Stagnation properties of the gas flow ill be calculated on this surface and free flow properties will also be calculated. Properties will be calculated for 0.1 second intervals up to 0.5 second.

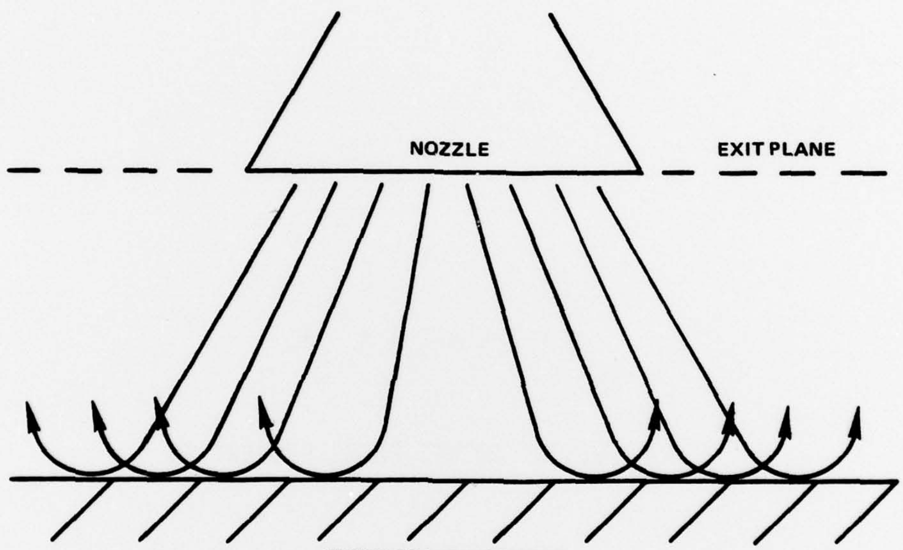

FIGURE A-1 - Rocket Plume Surface Impingement

The following assumptions are made:

1. Ambient pressure is sea level atmospheric pressure.

2. No shocks occur at the nozzle and exit pressure equals ambient pressure.

The basic applicable equation to be used for this analysis is the fundamental continuity equation (or conservation of mass equation) for fluid flow which is given in equation (1).

$$\sum F = \int \int V \frac{(\rho V A)}{g_c} \tag{1}$$

where

F = *Rocket thrust* (lb_f, *nt*)

V = Velocity of fluid (ft/sec, m/sec)

ρ = Fluid density (lbm/ft^3, Kg/m^3)

A = Exit area (ft^2, m^2)

g_c = Gravitational constant ($32.2 \frac{lbm - ft}{lb_f - sec^2}$, $1.0 \frac{Kg - m}{nt - sec^2}$)

For a given nozzle area, equation (1) can be simplified

$$F = \frac{\rho A V^2}{g_c} \tag{2}$$

Other equations of interest are

$$\dot{m} = \rho A V \tag{3}$$

where

$\dot{m}$ = mass flow (lbm/sec, Kg/sec)

and

$$F = \dot{m} (V/g_c) \tag{4}$$
$$= \dot{m} I_{sp}$$

where

I_{sp} = specific impulse of propellant $\left(\frac{lb_f\text{-sec}}{lbm}, \frac{nt\text{-sec}}{Kg}\right)$

It is known that I_{sp} for the rocket propellant is $241 \frac{lb_f\text{-sec}}{lbm}$ $\left(\frac{2362\ nt\text{-sec}}{Kg}\right)$

From equation (4)

$$V = I_{sp}\, g_c$$

$$= 241 \frac{lb\text{-sec}}{lbm} \times 32.2 \frac{lb_m - ft}{lb_f - sec^2}$$

$$= 7660 \frac{ft}{sec} \quad (2328\ m/sec)$$

V and I_{sp} are assumed constant throughout rocket burn.

The thrust-time curve for the rocket is known from experimental testing. This curve is shown in figure A-2.

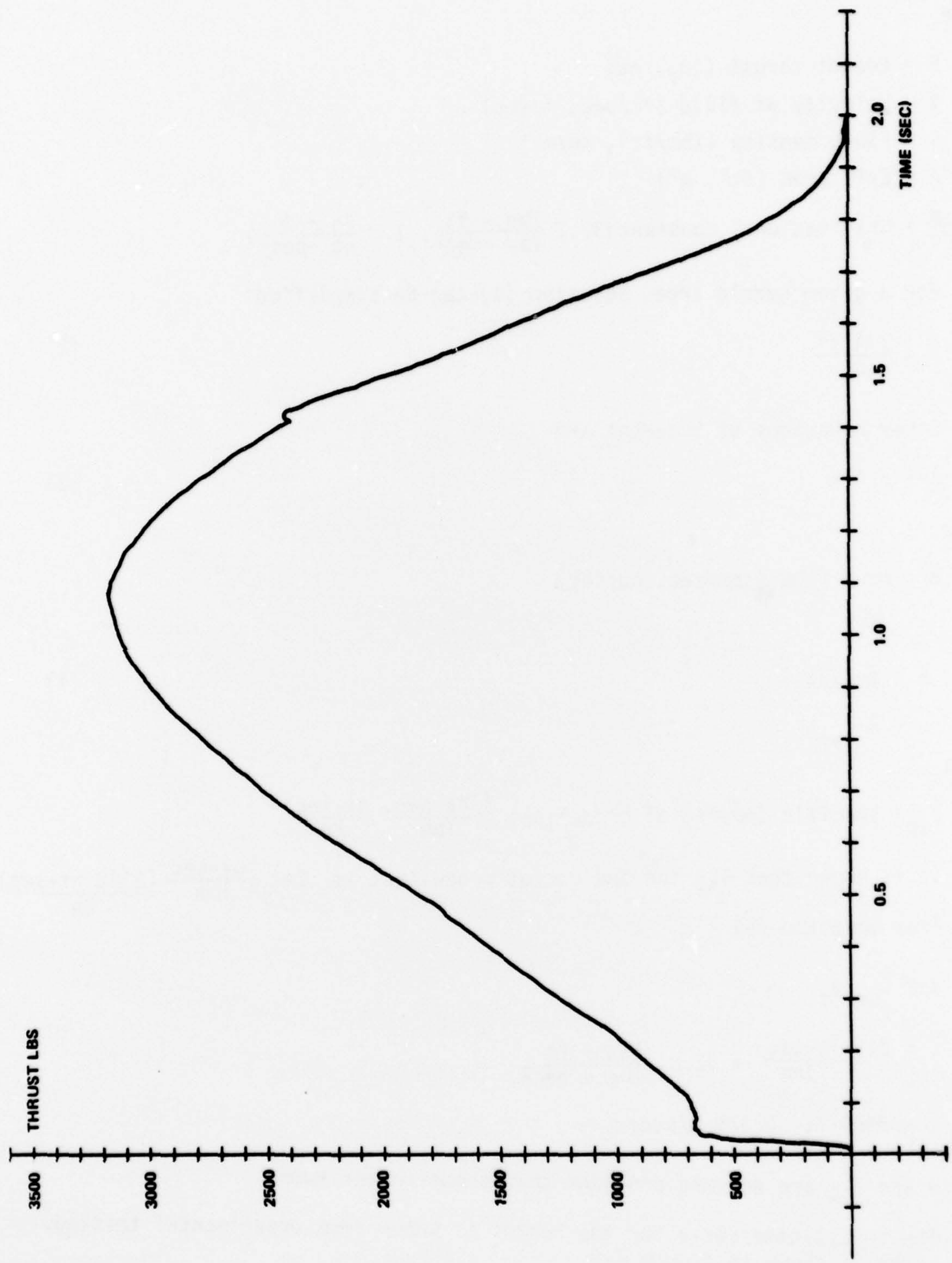

FIGURE A-2 - Thrust versus Time Profile for Experimental Rocket Motor

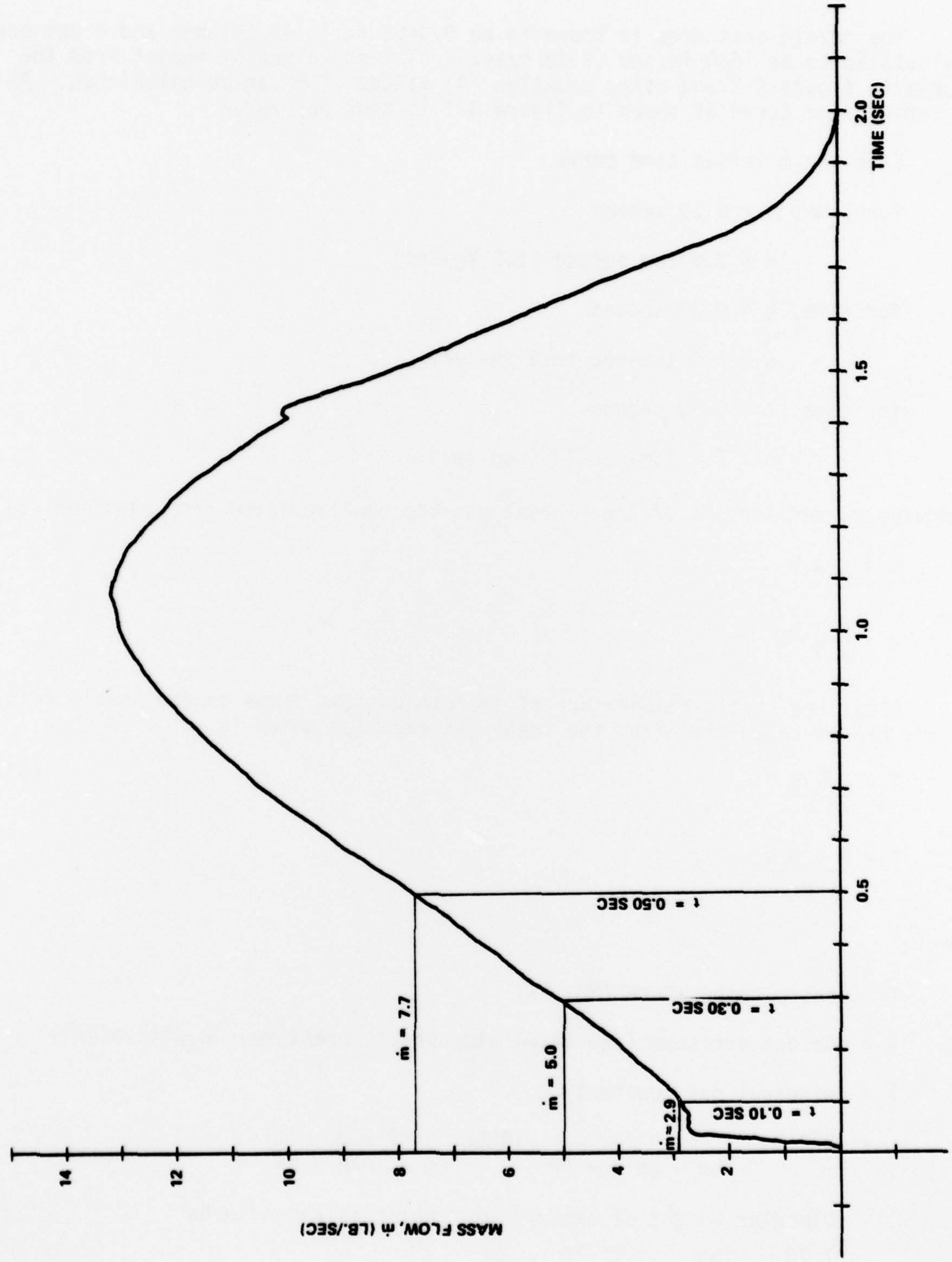

FIGURE A-3 - Mass Flow versus Time Profile for Experimental Rocket Motor

The nozzle exit area is known to be 0.0469 ft.^{2} (43.54 cm^2) and V has been calculated to be 7660 ft/sec (2328 m/sec). Taking values of thrust from the curve in figure A-2 and using equation (4) values of $\dot{m}$ can be calculated. An $\dot{m}$ versus time curve as shown in figure A-3 is thus generated.

From the $\dot{m}$ versus time curve

for time, t = 0.10 second

$$\dot{m} \approx 2.9 \text{ lbm second (1.3 Kg-sec)}$$

for time, t = 0.30 second

$$\dot{m} \approx 5.0 \text{ lbm-sec (2.3 Kg-sec)}$$

for time, t = 0.50 second

$$\dot{m} \approx 7.7 \text{ lbm-sec (3.5 Kg-sec)}$$

Knowing $\dot{m}$, density, ρ, of the exhaust gas can be calculated from equation (3),

$$\dot{m} = \rho A V$$

or

$$\rho = \frac{\dot{m}}{A V}$$

Also, the static temperature of the exhaust gas close to the nozzle exit plane can be calculated from the ideal gas equation which is

$$P = \frac{\rho \bar{R} T}{M.W.} \tag{5}$$

or

$$T = \frac{P \times M.W.}{\rho \bar{R}}$$

where

T = static temperature (^oR, ^oK)

P = ambient pressure (sea level atmospheric pressure, lb_f/ft^2, nt/m^2)

$\bar{R}$ = universal gas constant

$$= 1545.33 \ \frac{ft - lbf}{lbm - mole - {}^oR} \left(8288 \ \frac{nt - m}{Kg - mole - {}^oK}\right)$$

M.W. = molecular weight of exhaust gas which is known to be

$$27.70 \ \frac{lbm}{lbm - mole} \left(27.70 \ \frac{Kg}{Kg - mole}\right)$$

Stagnation temperature, T_O, of flow impinging a surface close to the nozzle exit plane is given by

$$T_o = \frac{V^2}{2\, g_c\, C_p} + T \tag{6}$$

where

$$C_p = \frac{\gamma R}{\gamma - 1} \tag{7}$$

C_p = heat capacity at constant pressure $\left(\frac{BTU}{lbm - {}^oR}, \frac{nt - m}{Kg - {}^oK}\right)$

γ = gas constant pertaining to compressibility (dimensionless)

R = $\overline{R}$/M.W.

The gas constant, γ, which is usually assumed constant for computational purposes does, in reality, vary with large temperature fluctuations. However, in the analysis γ is assumed to be constant and equal to 1.26.

The dynamic pressure (wind blast pressure), q, of the rocket exhaust can be calculated from the following equation

$$q = P\, \{[1 + (\frac{\gamma - 1}{2})\, M^2]^{\gamma/(\gamma - 1)} - 1\} \tag{8}$$

where

$$\text{Mach Number, } M = \frac{V}{\sqrt{\gamma\, g_c\, R\, T}}$$

and M is dimensionless.

From the previous equations and figures, the values listed in table A-1 are calculated for t = 0.10, 0.20, 0.30, 0.40, and 0.50.

TABLE A-1

CALCULATED ROCKET PLUME PROPERTIES
(SEA LEVEL CONDITIONS, NO PROPELLANT AFTERBURN)

Time, t	Thrust, F		Mass Flow, $\dot{m}$		Density, ρ	
sec	lb_f	nt	lb_m/sec	Kg/sec	lb_m/ft^3	Kg/m^3
0.10	700	3100	2.9	1.3	0.0080	0.129
0.20	950	4200	3.9	1.8	0.0109	0.178
0.30	1200	5330	5.0	2.3	0.0140	0.226
0.40	1550	6860	6.4	2.9	0.0178	0.286
0.50	1850	8210	7.7	3.5	0.0214	0.345

Static Temp., T		Stag. Temp., T_o		Mach No., M	Blast Pressure, q	
^oR	^oK	^oR	^oK	(dimensionless)	lb_f/in^2	nt/m^2
4740	2630	8100	4500	2.34	184	127
3780	1930	6850	3800	2.72	370	255
2710	1505	6985	3380	3.09	719	495
2130	1180	5505	3060	3.49	1447	998
1770	985	5145	2860	3.83	2574	1771