Gemini

Lyle Schroeder

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Gemini

STRUCTURAL DESIGN CRITERIA

GEMINI SPACECRAFT (U)

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DATE	PAGES AFFECTED REVISED	ADDED	REMOVED	REMARKS	REVISED BY	APPROVED
6/9/62	REVISION "A" Report completely revised.				J. D. Shepherd	E. T. Akeroyd T. P. Brooks R. C. Goran
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	2.4.1, 2.6 3.2.1 3.2.2			Retyped due to changes on adjacent pages	J. D. Shepherd	E. T. Akeroyd
	3.1.1	3.1.2 3.1.3		Added Factors of Safety for Parachutes		T. P. Brooks
	2.3.2 3.6.2 3.6.3 3.6.6	3.6.1.1 3.6.1.2 3.6.1.3	3.6.1	Added drogue chute deploy weight, three parachute system and changed order		
		2.7.1 2.7.2 2.7.3 2.7.4 2.7.5 2.7.6 3.3.1 3.3.2	3.3	Added meteoroid environment and penetration criteria		R. C. Goran
	3.7.1 3.7.2 3.8			Added ejection seat, ballute and personnel parachute to Mode I abort criteria.		
	3.10			Revised control force criteria.		
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	1.3 1.4	... 1.1.2	...		J. D. Shepherd Sr. Grp. Engr. Loads	E.T. Akeroyd Chief Loads Engineer
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	3.1.1 3.1.2 3.1.3			Revised factors of safety for hatches		W.B. Ingalbart Project Strength Engineer
	3.6.1.2 3.6.1.3			Corrected pilot chute load for two chute system. Changed load and pull off angle for pilot chute in three chute system.		T. P. Brooks Section Manager Strength
	3.10			Added criteria for levers in general		R. C. Goran Chief Strength Engineer
	3.11.1 3.12 3.13.1 3.13.2	3.13.3 3.13.4		Revised shock and acceleration environment and vibration and acoustic environment to be consistent with the sources.		

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1.4 References

1.	MAC Report 8611	Gemini Spacecraft Performance Specification
2.	NASA TN D-595	A Reference Atmosphere for Patrick AFB, Cape Kennedy, Florida, Annual
3.	NASA TN D-610	Monthly and Annual Wind Distribution as a Function of Altitude for Patrick AFB, Florida
4.	NASA - Engineering Criteria Bulletin -- No. EC-1	Meteoroid Environment in Near-Earth, Cislunar, and Near-Lunar Space, dated 8 November 1963.
5.	NASA Letter GPO 00169	Gemini Landing Requirements - dated 21 June 1962
6.	MAC Report 8433	General Environmental Requirements for Model 133P
7.	MAC Report 8610	Gemini Spacecraft - Environmental Criteria Requirements, revision dated 10 July 1964
8.	Martin Letter MG-D964	Digital Printout of Trajectory 309, dated 15 May 1963
9.	Martin IDC Design Verification Trajectories	Digital Printout of Trajectory 333
10.	Lockheed Document LMSC/A377490	Design Launch to Orbit Trajectory for Gemini Mission, dated 29 May 1963
11.	MAC Report 9998	Gemini Ablation Shield Structural Capability and Performance Limits (to be published)
12.	Lockheed (LMSC) Interdepartmental Communication SW/80.571	Preliminary Gemini/Agena D Balance and Inertia Data, dated 11 December 1963

1.5 Summary

This report forms part of the Gemini Spacecraft Performance Specification (Reference 1) and gives the detail criteria for the design of the structure.

The structural design criteria as described herein for the Gemini spacecraft is applicable for all spacecraft as the missions are currently projected. It is planned that this report will be revised as new requirements arise either from changed or added missions or from more refined analysis of the current missions.

1.6 Introduction

The basic mission of the Gemini Spacecraft requires that it be launched into orbit with two astronauts in control using the Titan II ICBM launch vehicle, that a rendezvous maneuver be performed to bring the spacecraft adjacent to a previously launched Agena vehicle, that the spacecraft dock and become secured to the Agena vehicle, that the spacecraft perform maneuvers in orbit using the Agena for propulsion, and that the spacecraft be capable of remaining in orbit for two days for this mission and returning the astronauts safely to the earth's surface. An alternate mission for the Gemini Spacecraft and crew requires the capability for performing orbital missions of up to 14 days duration. The early spacecraft will deploy a parachute and will be designed to land only in the water. Later spacecraft will deploy a paraglider for landing at a predetermined prepared land site; however, these spacecraft will also be designed for landing in the water.

The criteria presented herein are based on Government specifications or parts thereof which are considered to be applicable to an orbital vehicle and on NASA/McDonnell experience in the development of the Mercury Spacecraft and the Mercury Spacecraft components.

The basic data, including vehicle arrangement, design weights and environment are presented in Part Two.

The criteria for loads, temperatures, operational phases, and various components are presented in Part Three.

The Target Docking Adapter criteria are presented in Appendix A.

PART TWO - BASIC DATA

2.1 Spacecraft Description

The Gemini Spacecraft general arrangement is shown in Figure 2.2. The spacecraft consists of two basic sections: the Re-entry Module and the Adapter Module.

The Adapter Module is made up of three sections. Adjacent to the launch vehicle is the Launch Vehicle Mating Section which remains with the launch vehicle after separation. Next is the Equipment Section which contains the propulsion, control and other equipment needed for orbit and rendezvous. This section is retained in orbit and is released prior to retrograde. Adjacent to the Re-entry Module is the Retrograde Section which contains the retrograde rockets required for de-orbiting. This section is released following Retrograde.

The Re-entry Module has an internal pressure vessel for the two astronauts. The pressure vessel is shaped so as to leave space between it and the outer conical shell for equipment. In the re-entry attitude, the forebody consists of an ablative type heat shield and the afterbody consists of heat resistant shingles. After re-entry, the early spacecraft will deploy a parachute and the Re-entry Module will land in water with the Z axis inclined to the vertical. On later spacecraft, a paraglider will be deployed and the vehicle maneuvered for landing by moving the center of gravity relative to the paraglider. This is accomplished by changing the lengths of the cables between the paraglider and the Re-entry Module. A tricycle ski type landing gear is provided for landing on a prepared designated landing site.

The Gemini Spacecraft is designed to be launched on a Titan II ICBM launch vehicle.

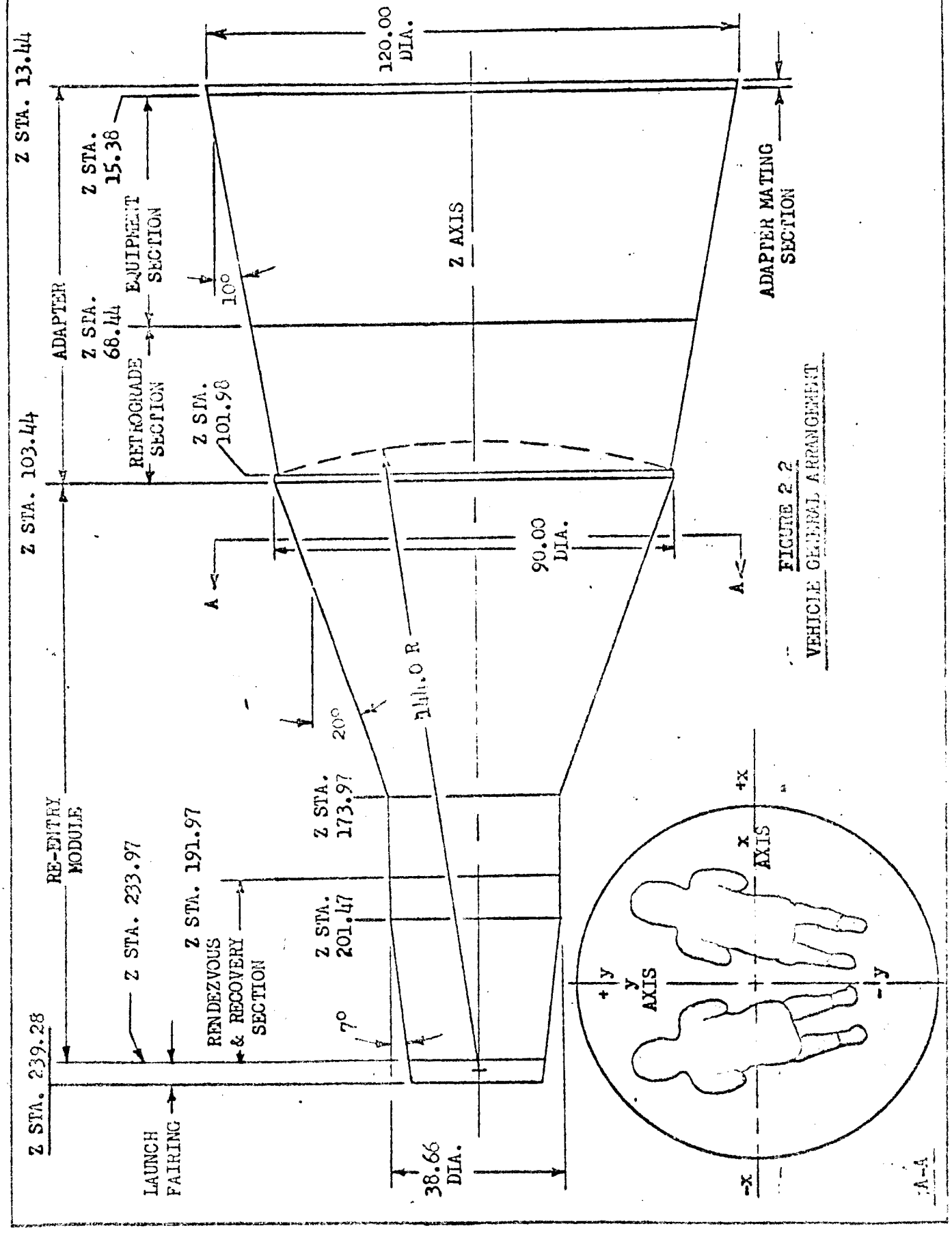

FIGURE 2.2
VEHICLE GENERAL ARRANGEMENT

2.3 Spacecraft Design Weights

The weights to be used for structural design loads are presented in the following sections which correspond to the phases for which the criteria is established. These weights are for structural design only. A range of weight or a maximum weight is shown for each phase. The weight within this range producing the most critical loads shall be used. The weight distribution used shall be that for a spacecraft meeting the nominal weight requirements. The extremes of the ranges or the maximum weight shall be covered by multiplying the distribution by the ratio of the extreme weight to the nominal weight. The nominal Design Launch Weight is 7,000 lbs. and the incremental weights correspond to this value.

2.3.1 Design Launch Weight (6500 to 7500 lbs.)

Spacecraft weight during boost phase from launch until just prior to separation from the launch vehicle.

2.3.2 Design Orbit Weight (6450 to 7450 lbs.)

Maximum spacecraft weight in orbit. It is the Design Launch Weight less the launch vehicle mating section and propellant utilized during separation.

2.3.3 Design Rendezvous Weight (6200 to 7200 lbs.)

Spacecraft weight for docking loads and maneuver loads after docking. It is the Design Orbit Weight less one half the orbit correction and rendezvous maneuvering propellant.

2.3.4 Design Retrograde Weight (4800 to 5800 lbs.)

Spacecraft weight just prior to retrograde firing. It is the Design Rendezvous Weight less the equipment section of the adapter.

2.3.5 Design Re-entry Weight (4050 to 5050 lb.)

Spacecraft weight upon re-entry into the atmosphere. It is the Design Retrograde Weight less attitude propellant and the retrograde section of the adapter.

2.3.6 Design Deployment Weights

2.3.6.1 Design Drogue Parachute Deployment Weight (4730 lb.)

Spacecraft weight at the time of deploying the drogue parachute and/or pilot parachute.

2.3.6.2 Design Main Parachute Deployment Weight (4400 lb.)

Spacecraft weight including parachute at the time of deploying the main parachute.

2.3.6.3 Design Paraglider Deployment Weight (3650 to 4650 lb.)

Spacecraft weight at the time of initiating the paraglider deployment sequence. It is the Design Re-entry Weight less attitude propellant, ablative material, pilot chute and paraglider housing.

2.3.7 Design Landing Weights

2.3.7.1 Design Parachute Landing Weight (4300 lb.)

Spacecraft weight impacting the water. It is the weight suspended under the parachute.

2.3.7.2 Design Paraglider Landing Weight (3300 to 4300 lb.)

Spacecraft weight during landing runout or impacting the water. It is also the weight suspended under the paraglider. It is the Design Paraglider Deployment Weight less paraglider and propellant jettison.

2.3.8 Design Flotation Weight (3300 to 4300 lb.)

This is the same as the maximum Landing Weight.

2.3.9 Design Water Recovery Weight (5500 to 6500 lb.)

This is the weight to be hoisted after a water landing. It is the Design Flotation Weight plus 2200 lb. of trapped water.

2.3.10 Design Abort Weight (4900 to 5900 lb.)

The spacecraft weight at the initiation of separation from the launch vehicle for an abort. It is the Design Launch Weight less the launch vehicle mating section and the equipment section of the Adapter.

2.3.11 Design Transportation Weight (6000 lb.)

The weight for hoisting, handling, and transporting as a unit.

2.4 Standard Atmosphere

All loads and temperature calculations shall be based on the atmosphere as defined in Reference (2). Density and pressure are shown on Figure 2.4.2 and Figure 2.4.3 as a function of altitude.

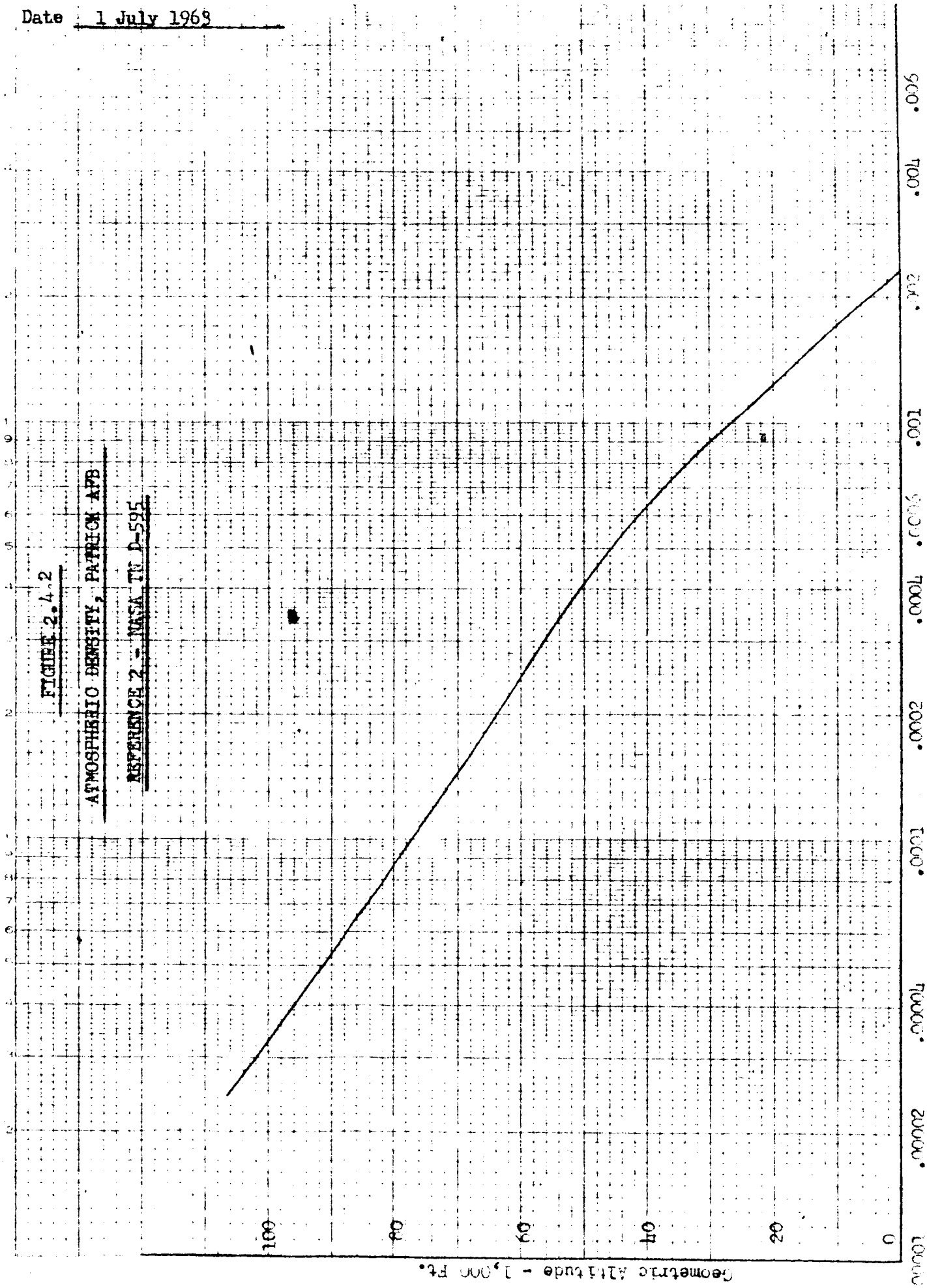

FIGURE 2.4.2

ATMOSPHERIC DENSITY, PATRICK AFB

REFERENCE 2 - NASA TN D-595

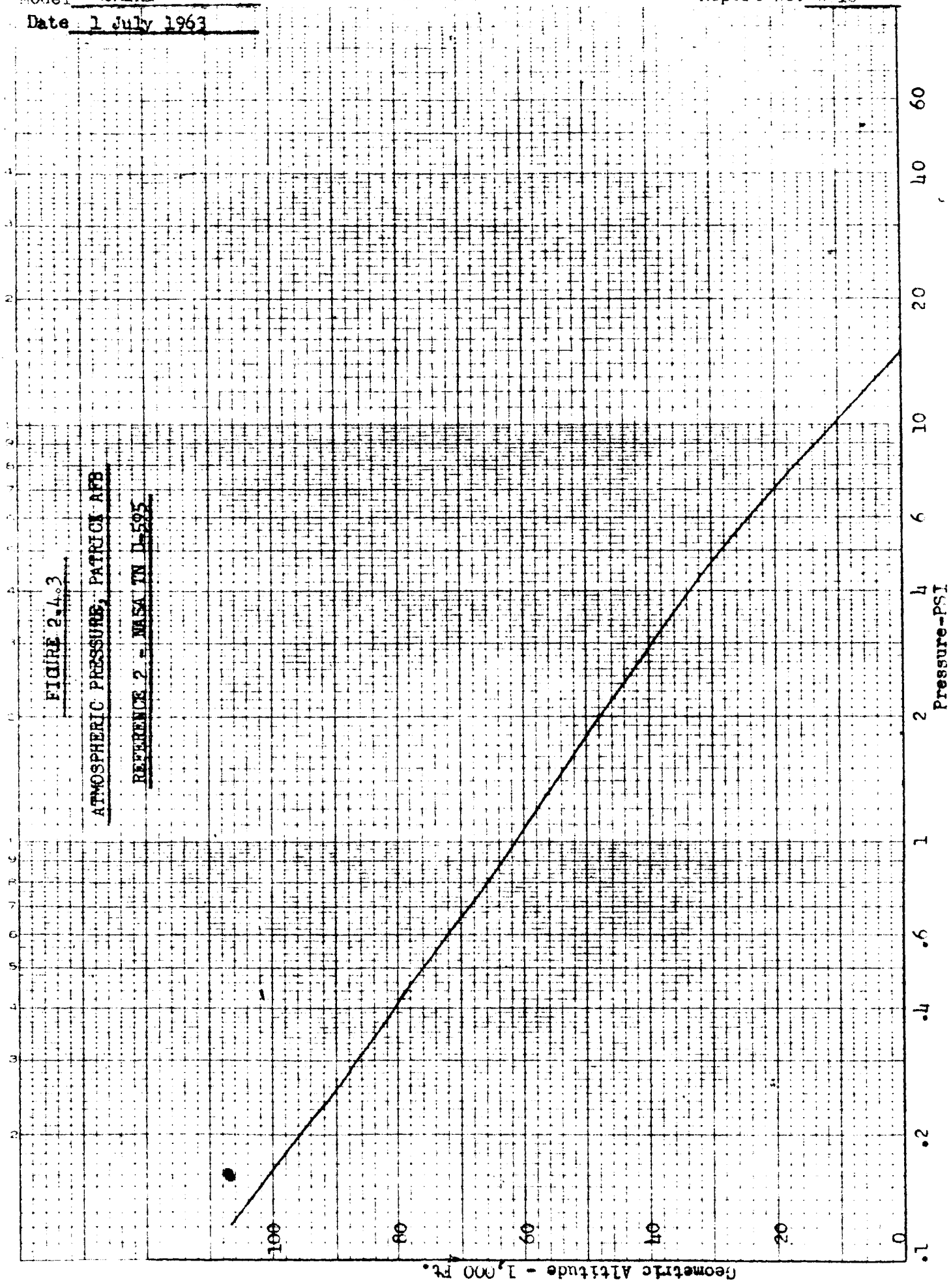

FIGURE 2.4.3

ATMOSPHERIC PRESSURE, PATRICK AFB

REFERENCE 2 - NASA TN D-595

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2.5 Wind Shear and Velocity

The wind shears and velocities to be considered for structural design purposes are as follows:

2.5.1 Nominal Wind Velocity

This is the median annual scalar velocity as given in Reference (3). It is shown on Figure 2.5.2.

2.5.2 Design Wind Shear

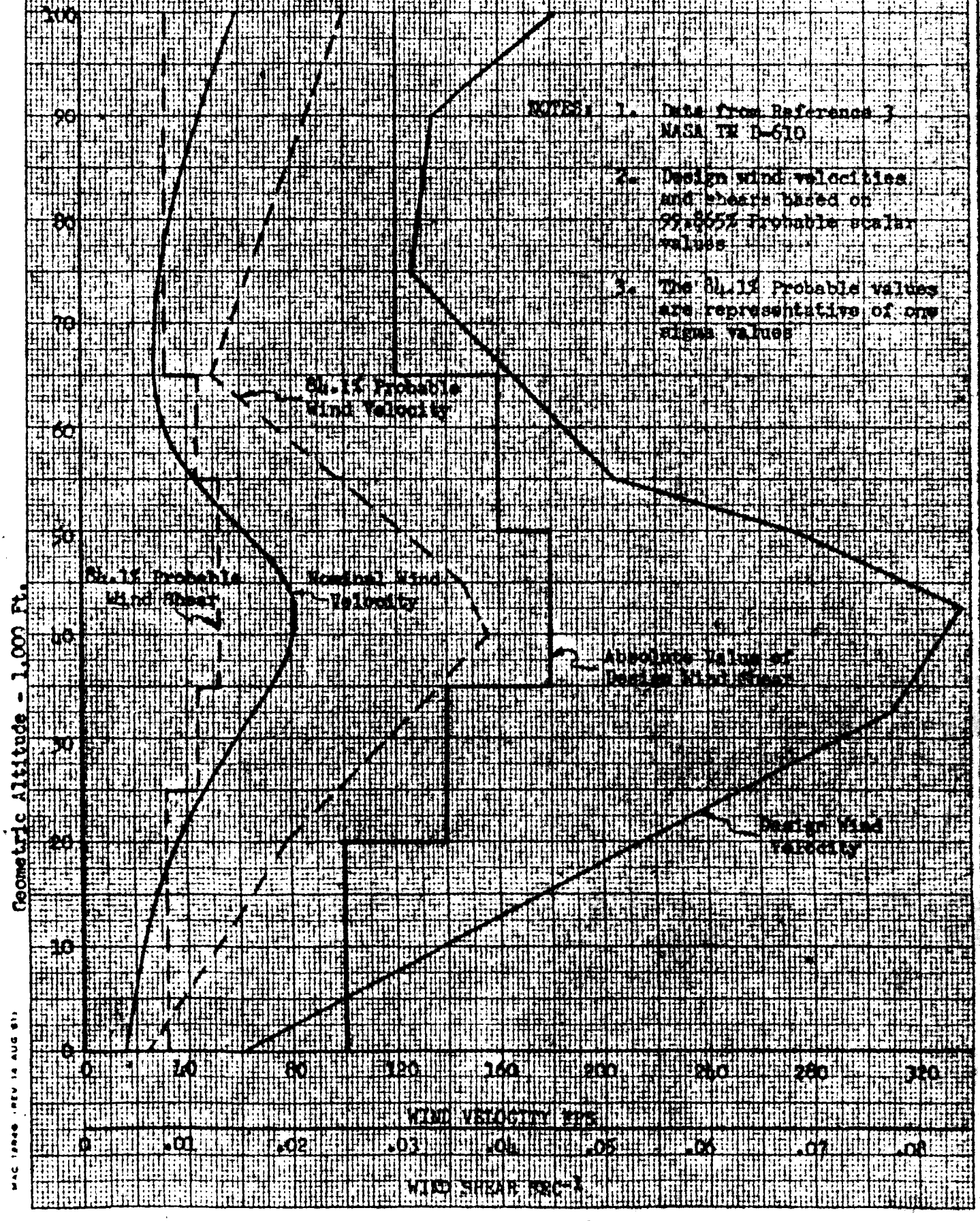
At any altitude a maximum shear not exceeding the design wind shear shown on Figure 2.5.2 shall be considered to exist. These values correspond approximately to those given in Reference (3) as the annual values with a 99.865% probability of not being exceeded. Approximate values for wind shears with an 84.1% probability of not being exceeded are also shown on Figure 2.5.2. These values are comparable to one sigma values for normal distributions.

2.5.3 Design Wind Velocity

Starting at the nominal wind velocity at any altitude a positive wind velocity gradient not exceeding the design wind shear given in 2.5.2 shall be considered to exist until the peak wind velocity is attained. The peak wind velocity shall not exceed the design wind velocity shown on Figure 2.5.2. This corresponds approximately to the annual scalar value given in Reference (3) which has a 99.865% probability of not being exceeded. Approximate values for wind velocities with an 84.1% probability of not being exceeded are also shown on Figure 2.5.2. These values are comparable to one sigma values for normal distributions.

FIGURE 2.5.2

WIND SHEARS AND VELOCITIES



2.6 Gusts

Strength shall be provided for loads associated with isotropic sharp edged gusts of 30 fps, equivalent airspeed, below 40,000 feet and 60 fps, true airspeed, above,40,000 feet neglecting penetration effects and with an alleviation factor of 1.0. Gusts shall be considered separately or in conjunction with the wind shears of Section 2.5. When combining the gust velocities with the wind shear requirements, the sum of the wind plus gust velocities shall not exceed the design wind velocity shown in Figure 2.5.2.

2.7 Meteoroid Environment

The meteoroid environemnt shall consist of the near-earth and cislunar sporadic meteoroids and the major meteoroid streams. The specific environment based on data from Reference (4) is defined in the following paragraphs.

2.7.1 Near Earth and Cislunar, Sporadic Meteroids Flux, Mass:

$$\text{LOG } N = -1.34 \log m - 10.423$$

Where N = number of impacts per square foot per day

m = mass in grams

Density: 0.5 gm/cc, all particle sizes

Average Geocentric Velocity: 30 km/sec, all particle sizes.

The flux-mass relationship stated above is shown graphically in Figure 2.7.3. The flux relation given above is an average of the monthly variations. For a particular period, the factors from Figure 2.7.4 are used. Since the sporadic meteroids are non-directional, the above criteria are applied to the surface area of the vehicle. All velocities are assumed to be directed normal to the target surface.

2.7.2 Near Earth and Cislunar, Meteoroid Streams Flux, Mass:

$$\text{LOG } N = -1.34 \log m - 2.68 \log V - 6.465 + \log F$$

Where N = number of impacts per square foot per day

m = mass in grams

V = geocentric velocity of the meteoroid stream (km/sec)

F = ratio of accumulative meteor stream flux to the sporadic meteor flux. The value of F, the period of activity, and the geocentric velocity of the major streams are shown in Table 2.7.5. The integrated value of F for any given date during the entire year is shown in Figure 2.7.6.

Density: 0.5 gm/cc, all particle sizes

Geocentric Velocity: As shown in Table 2.7.5. Since the meteoroid streams are directional, the above criteria are applied to the projected area of the vehicle. The vehicle shall be given the most critical orientation relative to the stream. That is, at any time, the largest projected area of the vehicle or component is to be used.

2.7.3 Shielding Factor

The shielding factor accounts for the fact that in the vicinity of the earth, meteoroids whose velocity vectors lie within a cone with the apex at the spacecraft and surface tangent to the surface of the earth have intercepted the earth's surface. Consequently, the spacecraft is effectively shielded from these meteoroids. To account for this, a near earth shielding factor of 0.5 shall be applied to the surface area of the spacecraft.

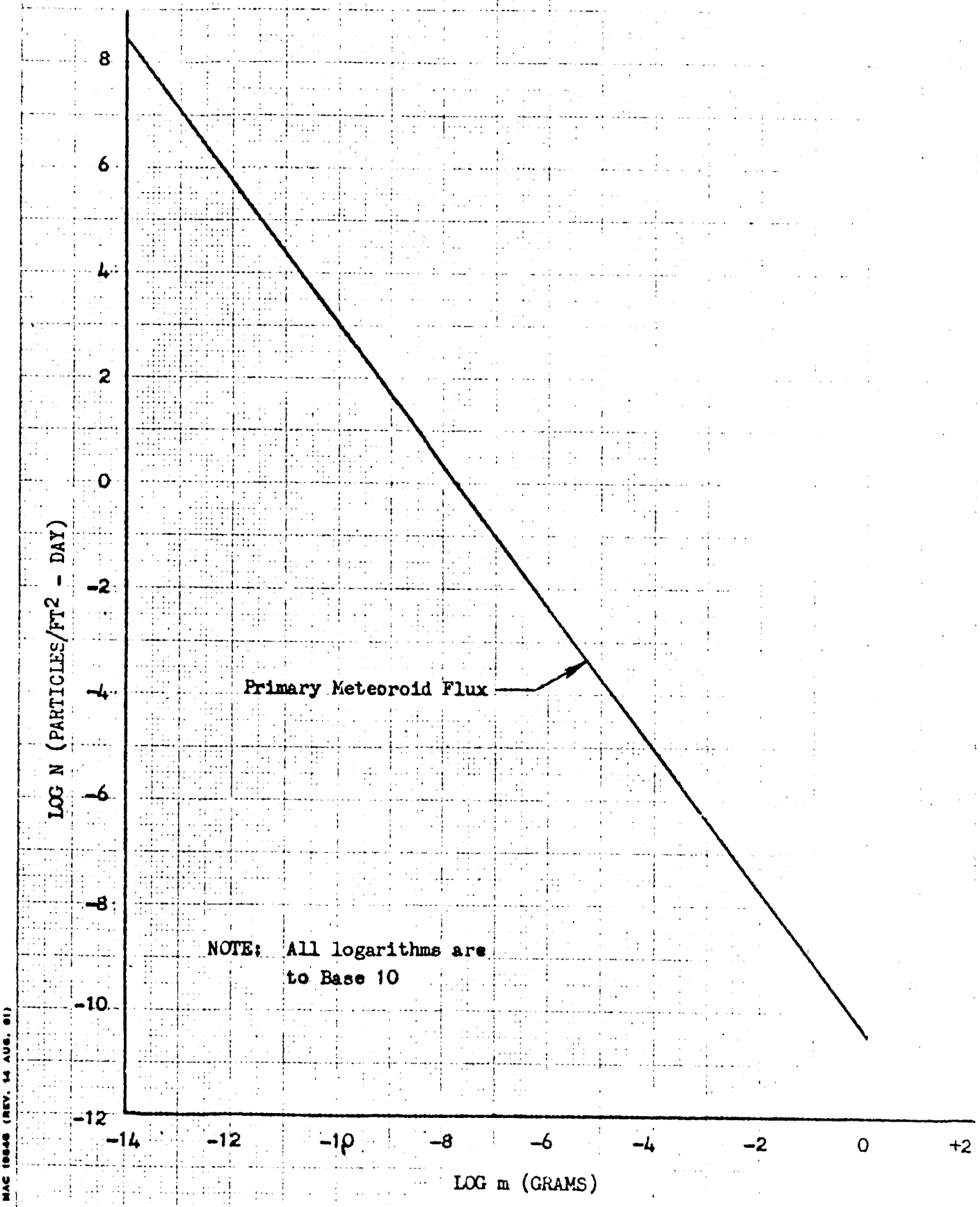


FIGURE 2.7.3 – NEAR-EARTH AND CISLUNAR METEOROID ENVIRONMENT

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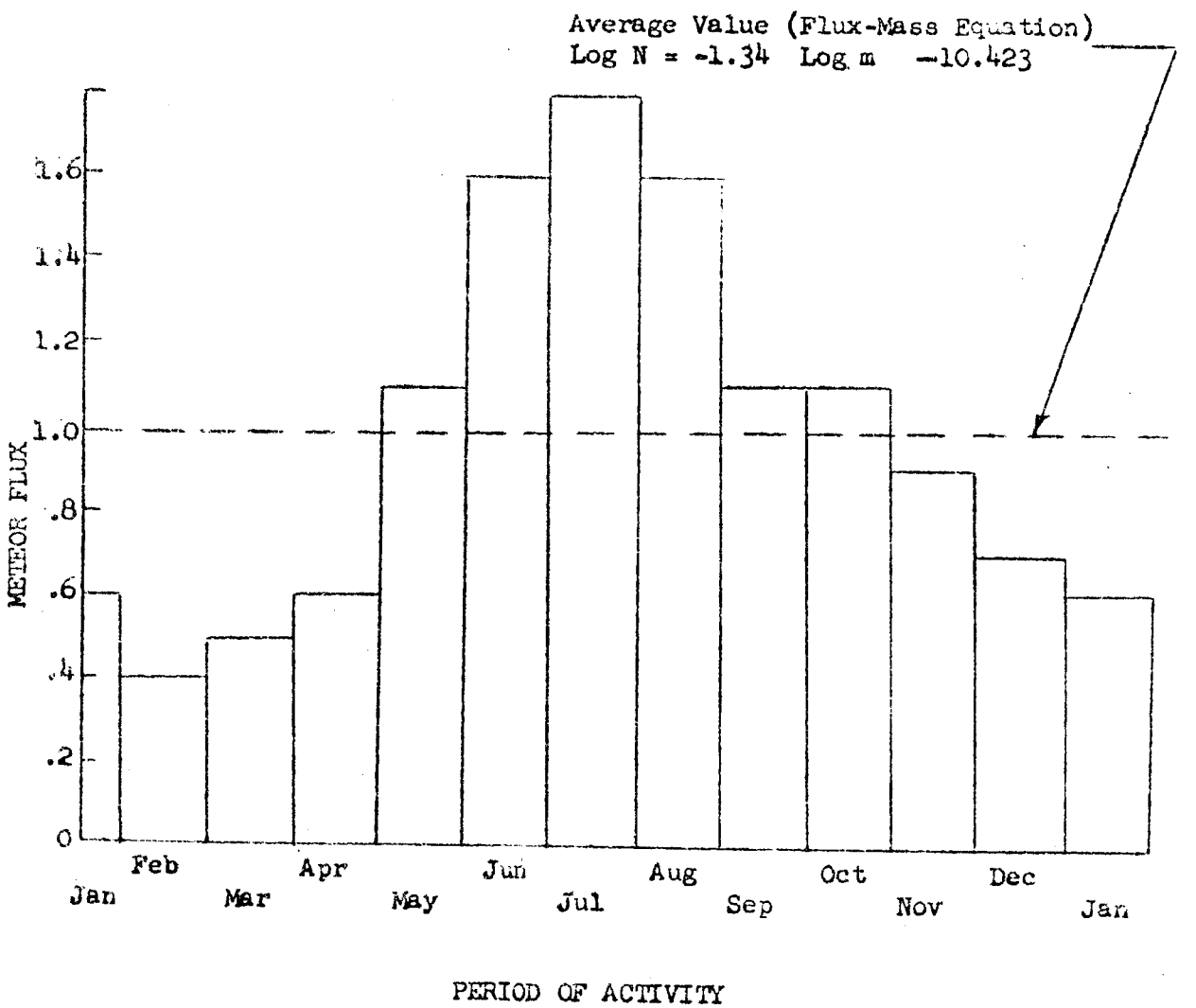


FIGURE 2.7.4—YEARLY SPORADIC METEOR FLUX.

Table 2.7.5

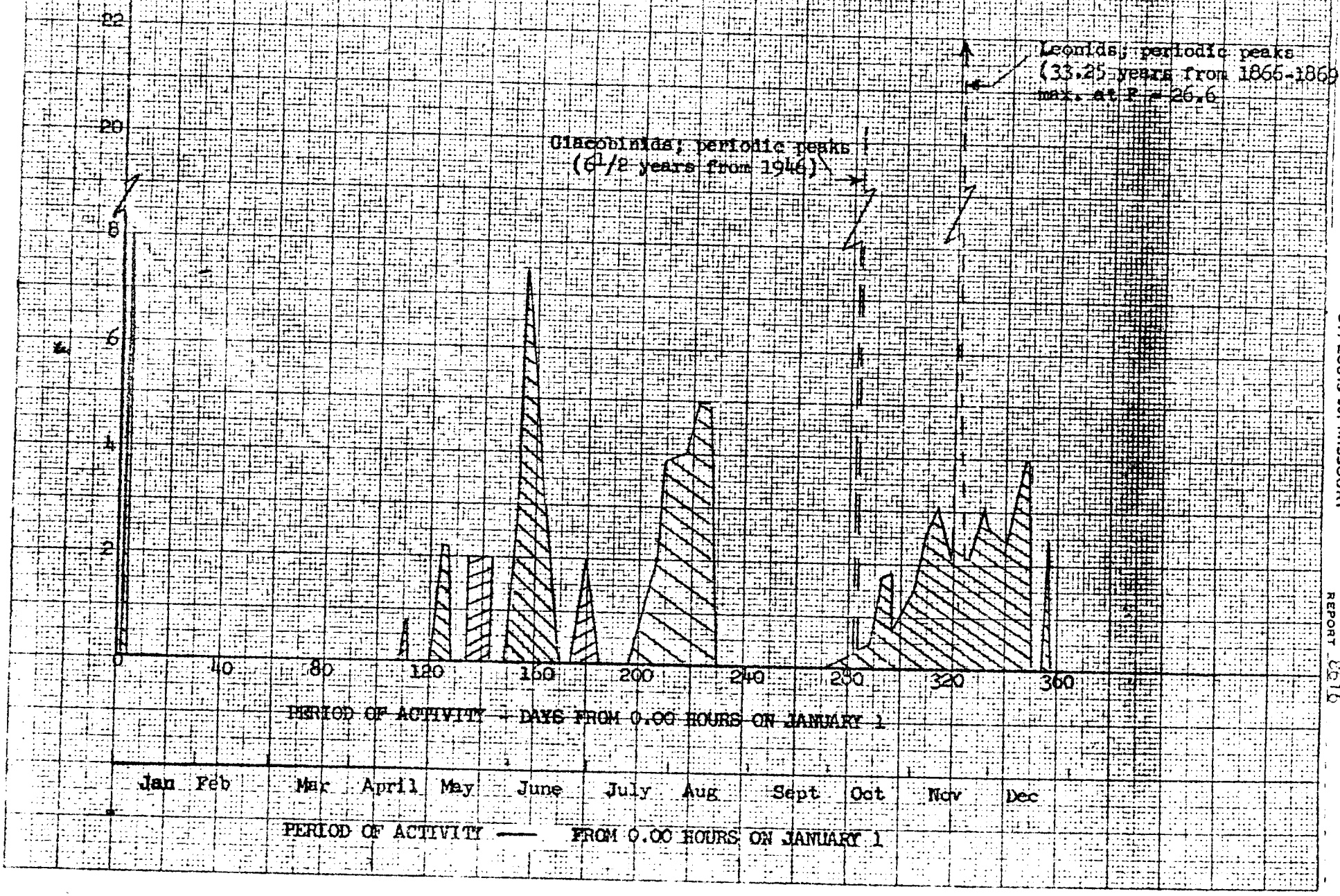
Periods of Activity, Relative Frequency, and Velocities for Major Meteor Streams

Name	Period of Activity	Date Max.	$F_{max.}$	Geocentric Velocity (km/sec)
Quadrantids	Jan 2-4	Jan 3	8.0	42
Lyrid	April 19-22	April 21	.85	48
η -Aquarid	May 1-8	May 4-6	2.2	64
O-Cetid	May 14-23	May 14-23	2.0	37
Arietid	May 29-June 19	June 6	4.5	38
ξ -Perseid	June 1-16	June 6	3.0	29
β -Taurids	June 24-July 5	June 28	2.0	31
δ -Aquarid	July 26-Aug 5	July 28	1.5	40
Perseid	July 15-Aug 18	Aug 10-14	5.0	60
Giacobinid*	Oct 9-10	Oct 10	20	23
Orionid	Oct 15-25	Oct 20-23	1.2	66
Arietid, Southern	Oct-Nov	Nov 5	1.1	28
Taurids, Northern	Oct 26-Nov 22	Nov 10	0.4	29
Taurids, Night	Nov		1.0	37
Taurids, Southern	Oct 26-Nov 22	Nov 5	0.9	28
Leonid*	Nov 15-20	Nov 16-17	0.9	72
Bielids	Nov 15-Dec 6		2.5	16
Geminid	Nov 25-Dec 17	Dec 12-13	4.0	35
Ursids	Dec 20-24	Dec 22	2.5	37

*All activity is taken to be periodic, annually except for Giacobinids which have a periodic peak every 6.5 years and the extra peak of the Leonids every 33.25 years.

F = ratio of maximum accumulative meteor stream flux to the sporadic meteor flux.

FIGURE 2.7.6
RATIO OF ACCUMULATIVE METEOROID STREAM FLUX TO THE SPORADIC METEOROID FLUX FOR A CALENDAR YEAR



PART THREE - CRITERIA

3.1 Limit and Ultimate Conditions

Limit loads and heating effects result from environmental conditions arising from normal mission, as described in this part, combined with the effects of any single malfunction. Abort or seat ejection shall not be considered a malfunction in this definition. In the case of malfunction of a single retrograde rocket, causing only three out of four to fire, maneuvering restriction shall be placed on the re-entry flight to preclude exceeding the structural limitations required for the nominal mission.

Ultimate loads are 1.36 times limit loads with the following exceptions:

1. For retrograde rocket pressures on the blast shield and water impact pressures on the re-entry module, ultimate design loads may be equal to limit loads and damage to the structure shall be acceptable provided that astronaut safety and flotation requirements are met.
2. A minimum margin of safety of 25% shall be maintained as defined below for the following elements or assemblies

   a. Landing Gear and Support Fittings - All joints where structural integrity could be dependent on a single bolt or pin and the design condition is defined by landing loads.

   b. Paraglider - All joints where structural integrity could be dependent on a single bolt or pin and the design condition is defined by paraglider deployment or maneuvering.

   c. Hatch Actuation - Hatch actuator, latching mechanism and all joints where structural integrity could be dependent on a single bolt or pin and the design condition is defined by hatch actuation.

3.1 Limit and Ultimate Conditions

Limit loads and heating effects result from environmental conditions arising from normal mission, as described in this part, combined with the effects of any single malfunction. Abort or seat ejection shall not be considered a malfunction in this definition. In the case of malfunction of a single retrograde rocket, causing only three out of four to fire, maneuvering restriction shall be placed on the re-entry flight to preclude exceeding the structural limitations required for the nominal mission.

Ultimate loads shall be limit loads multiplied by the Factor of Safety. The required factor of safety shall be 1.36 with the following exceptions:

(1) For retrograde rocket pressures on the blast shield and water impact pressures on the re-entry module, ultimate design loads may be equal to limit loads and damage to the structure shall be acceptable provided that astronaut safety and flotation requirements are met. The required factor of safety shall be 1.00.

(2) For the crew hatch and hatch actuator, the required factors of safety shall be 1.10 and 1.25 respectively, where the design condition is defined by hatch actuation with the actuator in tension, provided the load is substantiated by test.

(3) For the drogue parachute support structure, the minimum factor of safety shall be 1.36 for the normal mission and 1.10 for the case of a re-entry from the v-γ abort boundary (Figure 3.5.3) with a failure of the attitude control system (without rate damping). This value is used because of the improbable combination of events that would be required to reach this extreme condition.

3.1 Limit and Ultimate Conditions (continued)

(4) For the personnel parachute canopies, the required factor of safety shall be 1.10 based on the minimum failing strength. This value is used because of the improbability of all of the necessary circumstances combining to produce the design condition and the probability that the canopy strength does exceed the minimum. The minimum failure load of the C-9 type canopy, which is being used by the direction of NASA, is 5,000 pounds and the nominal capability is 6,500 pounds. In order to reach the design condition for deployment of the personnel parachute, ejection must occur at a particular altitude during launch and the seat must be oriented in a unique attitude. In this attitude, the barostat, which initiates parachute deployment, senses ram pressure superimposed on static pressure causing premature parachute deployment.

In addition to the required factors of safety listed above a minimum margin of safety of 25% shall be maintained as defined below for the following elements or assemblies.

(a) Landing Gear and Support Fittings - All joints where structural integrity could be dependent on a single bolt or pin and the design condition is defined by landing loads.

(b) Paraglider and Support Fittings - All joints where structural integrity could be dependent on a single bolt or pin and the design condition is defined by paraglider deployment or maneuvering.

(c) Hatch Actuation System - Hatch actuator, latching mechanism and all joints where structural integrity could be dependent on a single bolt or pin and the design condition is defined by hatch actuation.

3.1 Limit And Ultimate Conditions (continued)

Ultimate heating effects are those obtained by increasing limit temperatures 200°F or heat inputs by 15% whichever is critical for re-entry and increasing limit temperature 100°F for boost, except for items inside the pressure vessel. Structure inside the pressure vessel which is not attached to the skin and has no significant thermal mass shall be designed for 250°F ultimate. Ultimate design conditions are either ultimate loads combined with limit heating effects or ultimate heating effects combined with limit loads.

Deformations resulting from aero-thermal elastic effects at limit conditions shall not effect adversely the aerodynamic or functional characteristics of the vehicle. Nonsurvivable failure shall not occur under ultimate loads or under limit loads for the unique situations where ultimate loads equal limit loads.

The design shall be based on a service life of one nominal mission. The Re-entry Module shall be re-usable after a minimum amount of refurbishment and replacement of certain critical items.

All conditions in this report are limit conditions unless otherwise specified.

3.2 Boost Phase

The boost phase is defined as including all operations and environments encountered from the instant of launch vehicle engine thrust initiation until shutdown and separation of the last stage (injection into orbit). The calculations for the design structural loads and temperatures will be based on the two boost phase trajectories presented in Figures and Tables from 3.2.3 to 3.2.8. The trajectories are essentially vertical for the first 20 seconds; then a gravity turn is maintained throughout the remaining 129 seconds of first stage burning. The maneuvering required for orbital control is accomplished during the active second stage flight. The first trajectory is a nominal launch with insertion at an altitude of 87 nautical miles. The second trajectory is off-nominal in that the first stage thrust was assumed to be decreased by 3% along with a -42.8°/hr. pitch gyro drift. The second trajectory results in critical boost phase temperatures. These trajectories are from References (8) and (9).

Loads during the boost phase shall be based upon (1) the effects of winds and gusts as specified in Paragraphs 2.5 and 2.6, (2) a momentary guidance failure that results in a 10 degree angle of attack at any altitude with restoring moments from maximum thrust vector deflection, or (3) a launch vehicle malfunction which results in divergent angles of attack. The launch vehicle malfunction conditions shall be investigated and design loads defined so as to preclude re-entry module structural failure prior to the completion of the ejection or abort operation. The effects of combining either the momentary guidance failure condition or the launch vehicle malfunction condition with the 84.1% (representative of one sigma values) wind shear inputs, as derived from Reference (3), shall be

3.2 Boost Phase (Continued)

considered. However, the considerations for the momentary guidance failure condition shall be limited by the thrust vector restoring moment available to maintain stable flight. For the malfunction conditions, the boost phase shall be considered ended when both astronauts have passed clear of the open hatch structure during ejection or when the connection between the spacecraft and the booster has been severed during retrograde rocket abort. Booster engines may or may not be shut down for ejection, but they must be shut down for the retrograde rocket abort.

The altitude ranges through which ejection abort or retrograde rocket abort conditions must be considered are presented in Section 3.7.

Atmospheric density variation shall be considered by increasing dynamic pressure in the nominal trajectory by 5%.

Revised 9 June 1962
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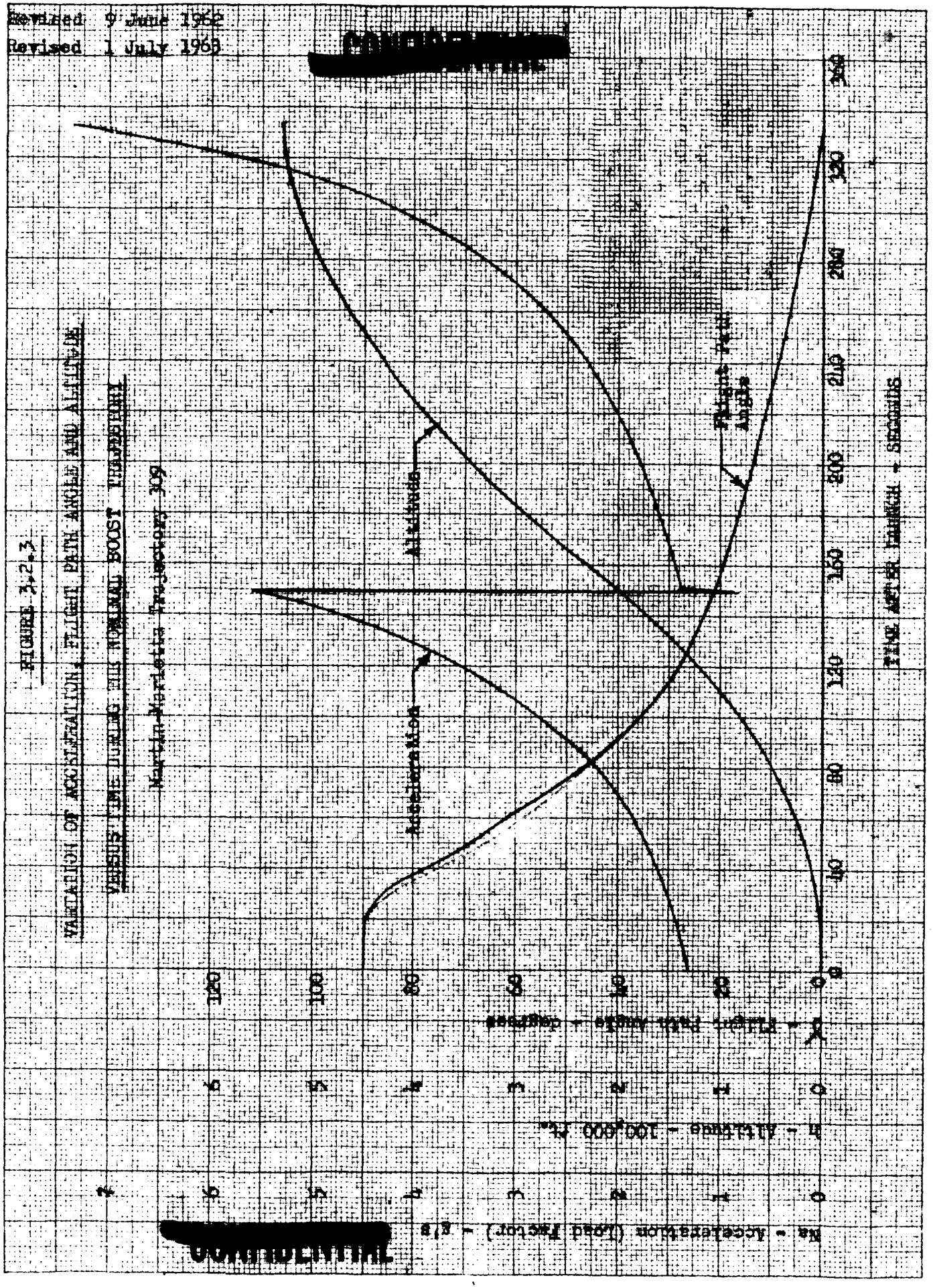

FIGURE 3.2.3

VARIATION OF ACCELERATION, FLIGHT PATH ANGLE AND ALTITUDE VERSUS TIME DURING ETR NOMINAL BOOST TRAJECTORY

Martin-Marietta Trajectory 309

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FIGURE 3.2.4

VARIATION OF VELOCITY, MACH NUMBER, AND DYNAMIC PRESSURE

VERSUS TIME DURING THE NOMINAL BOOST TRAJECTORY

Martin-Marietta Trajectory 309

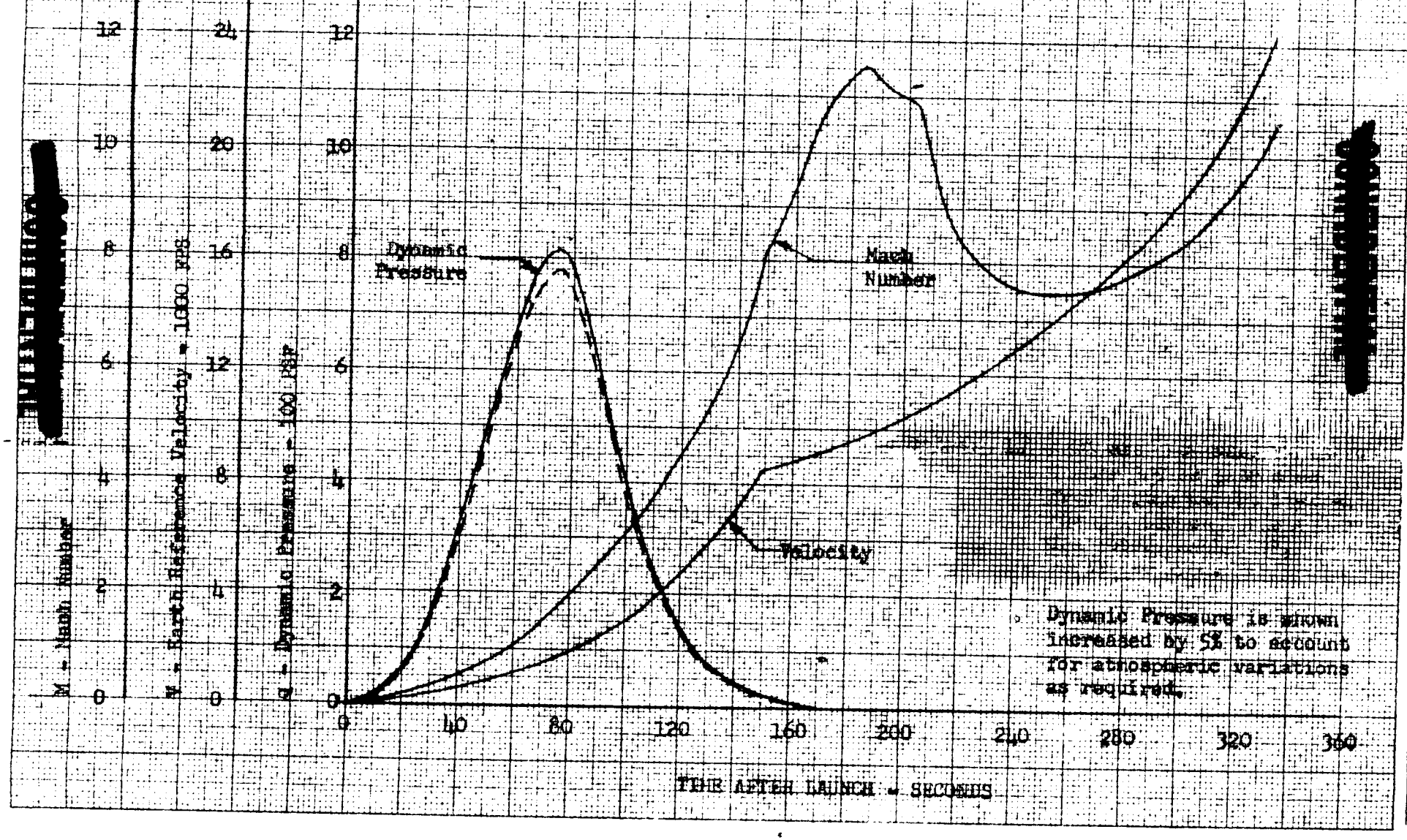

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TABLE 3.2.5

NOMINAL BOOST TRAJECTORY FOR INSERTION OF GEMINI SPACECRAFT
AT AN ALTITUDE OF 87 NAUTICAL MILES
TABULATED DATA

Time t Seconds	Acceleration Na g's	Velocity V ft./sec.	Altitude h feet	Flight Path Angle Degrees	Mach. Number M	Dynamic Pressure q p.s.f.	Dynamic Pressure (5% Increased Density) q p.s.f.
0	1.29	0	32	90.0	0	0	0
10	1.36	106	543	89.8	.09	13	14
20	1.44	236	2,232	89.7	.21	61	63
30	1.53	393	5,354	86.0	.35	153	160
40	1.64	586	10,162	77.7	.53	294	310
50	1.76	825	16,865	69.2	.77	475	498
60	1.85	1,116	25,626	61.5	1.07	649	681
70	1.98	1,457	36,461	54.4	1.47	759	795
75	2.09	1,657	42,648	51.1	1.72	777	816
80	2.21	1,881	49,350	47.7	1.99	742	780
90	2.49	2,412	64,292	41.4	2.53	565	593
100	2.80	3,063	81,183	35.6	3.13	384	403
110	3.14	3,842	100,092	31.4	3.82	235	247
120	3.56	4,759	121,338	28.2	4.53	137	144
130	4.09	5,842	145,165	25.5	5.37	76	80
140	4.78	7,133	171,841	23.3	6.45	41	43
148.5	5.59	8,442	196,986	21.6	8.05	25	26
149.7	.88	8,535	200,725	21.4	8.23	22	23
150	.88	8,540	201,686	21.3	8.25	21	22
151	1.39	8,568	204,786	21.2	8.35	19	20
160	1.45	8,880	232,128	19.6	9.34	7	7
170	1.52	9,257	261,321	18.0	10.86	2	2
180	1.60	9,666	289,249	16.4	11.42	0	0
190	1.69	10,110	315,889	14.9	11.31	0	0
200	1.78	10,591	341,219	13.5	11.00	0	0
210	1.89	11,112	365,210	12.1	9.58	0	0
220	2.01	11,679	387,831	10.8	8.45	0	0
230	2.15	12,293	409,046	9.6	7.87	0	0
240	2.31	12,961	428,812	8.4	7.57	0	0
250	2.50	13,690	447,078	7.4	7.46	0	0
260	2.72	14,489	463,788	6.3	7.47	0	0
270	2.97	15,369	478,870	5.3	7.57	0	0
280	3.29	16,345	492,240	4.4	7.76	0	0
290	3.68	17,437	503,830	3.5	8.03	0	0
300	4.17	18,672	513,460	2.7	8.40	0	0
310	4.81	20,088	520,961	1.8	8.88	0	0
320	5.69	21,749	526,047	1.0	9.50	0	0
330	6.96	23,747	528,337	0.2	10.32	0	0
332.3	7.35	24,279	528,414	0	10.55	0	0

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Trajectory 309

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Figure 3.2.6

Variation of Acceleration, Flight Path Angle, and Altitude Versus Time for the Maximum Heating (Off-Nominal) Boost Trajectory

Martin-Marietta Trajectory 333

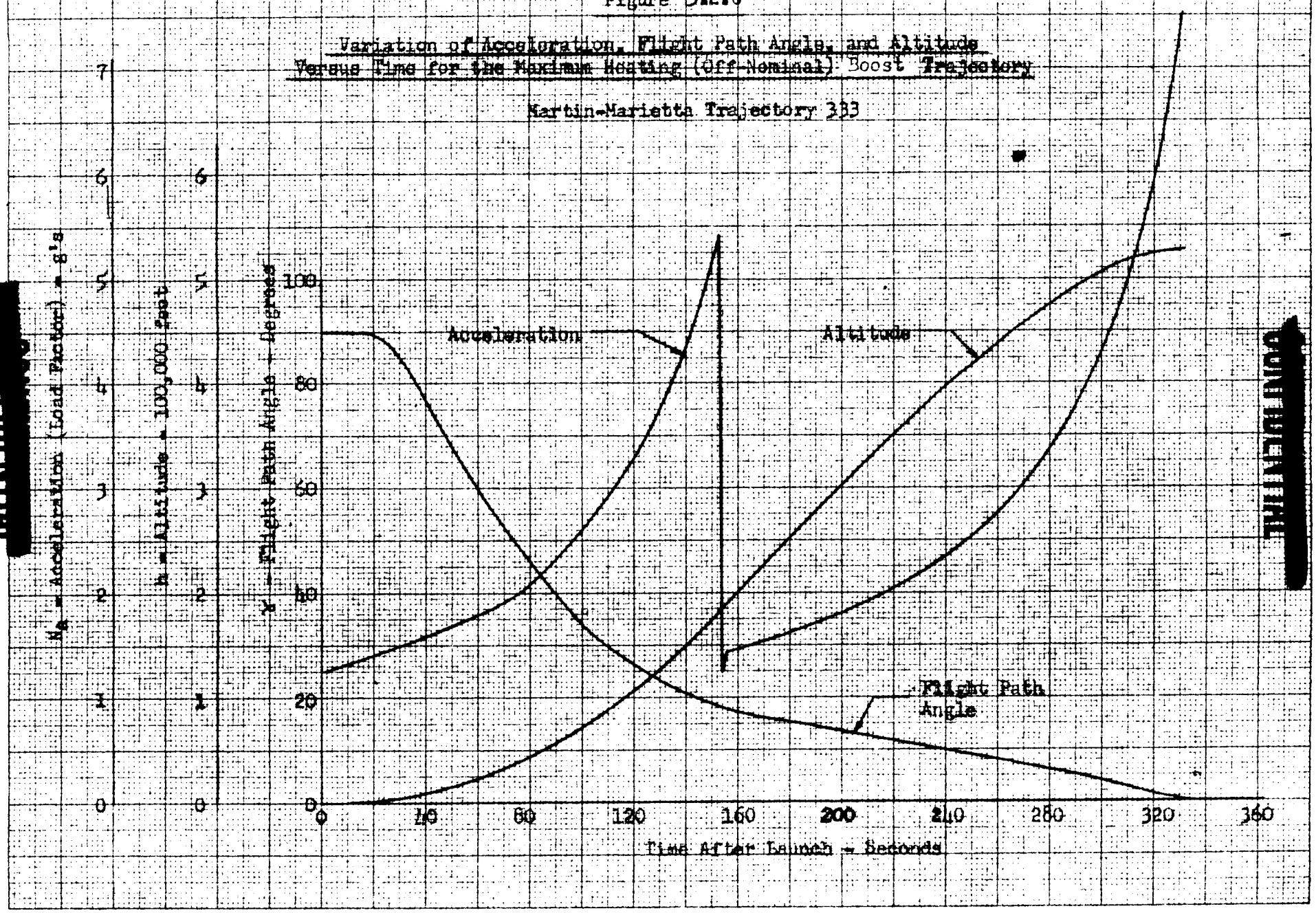

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Figure 3.2.7

Variation of Velocity, Mach Number and Dynamic Pressure
Versus Time for the Maximum Heating (Off-Nominal) Boost Trajectory

Martin-Marietta Trajectory 333

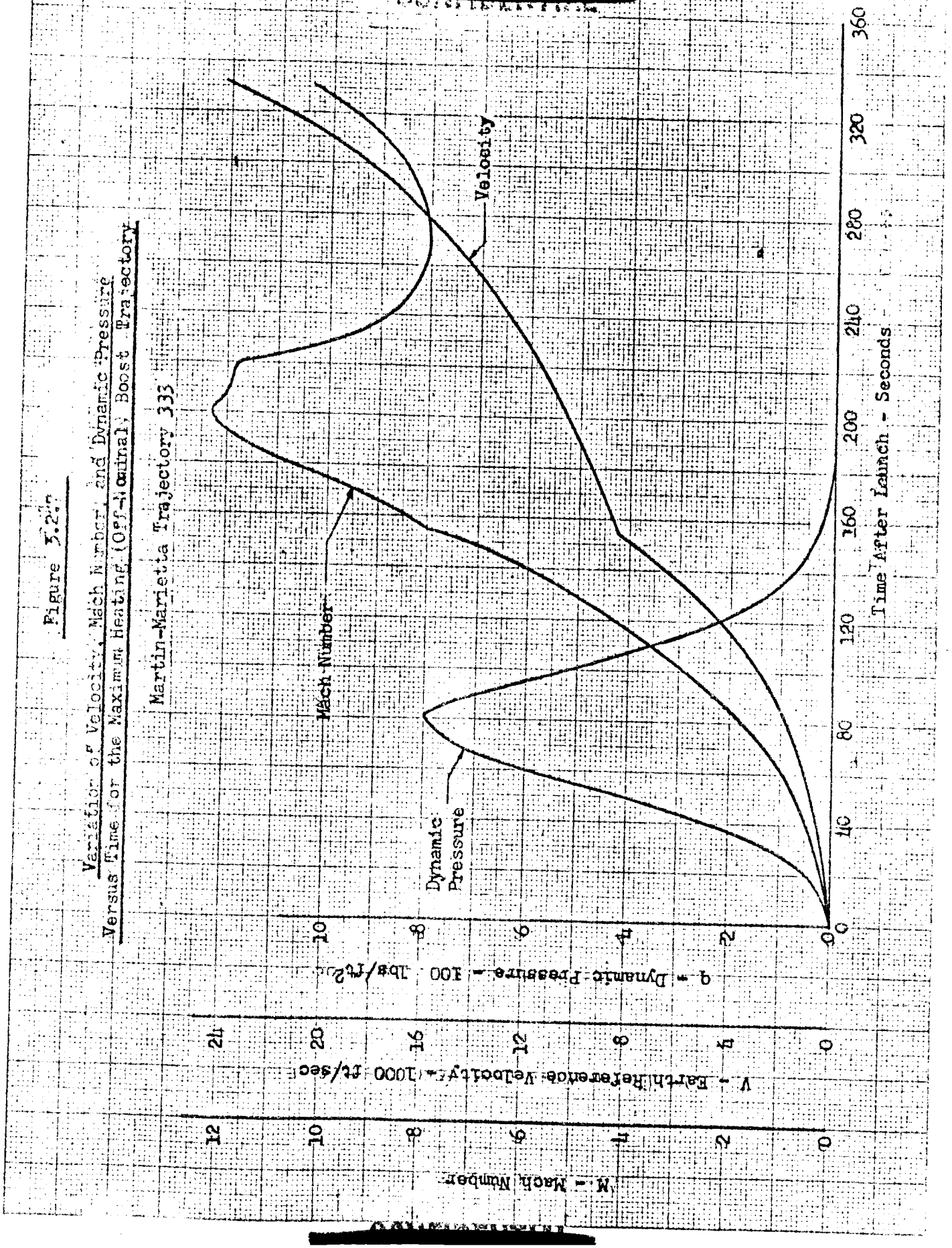

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TABLE 3.2.8

MAXIMUM HEATING (OFF-NOMINAL) BOOST TRAJECTORY
FOR INSERTION OF GEMINI SPACECRAFT AT AN ALTITUDE OF 87 NAUTICAL
MILES - TABULATED DATA

Time t Seconds	Acceleration Load Factor Na g's	Velocity V ft./sec.	Altitude h feet	Flight Path Angle Degrees	Mach. Number M	Dynamic Pressure q p.s.f.
0	1.26	0	0	90	0	0
10	1.32	93	478	89.8	.08	10
20	1.40	208	1,963	89.7	.18	48
30	1.48	349	4,726	85.2	.31	123
40	1.58	524	9,000	76.4	.48	245
50	1.69	744	14,977	67.6	.69	409
60	1.79	1,017	22,812	59.8	.96	596
70	1.88	1,337	32,550	52.8	1.32	737
80	2.07	1,727	44,113	46.2	1.80	794
80.76	2.09	1,800	45,000	45.2	1.85	800
90	2.32	2,213	57,422	39.6	2.35	694
100	2.60	2,814	72,272	33.6	2.91	518
110	2.92	3,535	88,704	29.4	3.57	363
120	3.27	4,382	107,071	26.2	4.30	226
130	3.74	5,376	127,439	23.4	5.05	134
140	4.32	6,549	149,882	21.0	5.99	77
150	5.11	7,952	174,533	18.9	7.19	46
153.1	5.41	8,443	182,608	18.3	7.77	40
154	1.24	8,532	185,051	18.2	7.90	38
155	1.42	8,560	187,704	18.0	7.98	35
160	1.46	8,744	200,757	17.1	8.43	23
170	1.53	9,125	226,265	16.1	9.47	9
180	1.62	9,533	251,512	15.3	10.77	3
190	1.70	9,976	276,538	14.4	11.78	1
200	1.80	10,458	301,277	13.6	12.28	0
210	1.91	10,980	325,646	12.7	11.92	0
220	2.04	11,547	349,545	11.8	11.73	0
230	2.18	12,164	372,861	10.9	9.81	0
240	2.35	12,836	395,465	9.9	8.86	0
250	2.54	13,570	417,207	9.0	8.35	0
260	2.76	14,376	437,913	8.0	8.1	0
270	3.03	15,264	457,373	7.1	8.0	0
280	3.36	16,250	475,342	6.1	8.1	0
290	3.77	17,354	491,519	5.0	8.2	0
300	4.28	18,605	505,512	3.9	8.5	0
310	4.97	20,043	516,807	2.8	8.9	0
320	5.91	21,734	524,733	1.6	9.5	0
330	7.29	23,782	528,376	.3	10.3	0
331.4	7.54	24,113	528,460	0	10.4	0

Martin-Marietta
Trajectory 333

3.3 Orbital Phase

The orbital phase is defined as beginning with the termination of thrust on the last launch vehicle stage and ending with the initiation of retro-thrust to re-entry the atmosphere.

3.3.1 Orbital Phase Maneuver Loads

The loading conditions encountered during this phase are the result of the rendezvous maneuver, the actual docking and coupling with the Agena vehicle in orbit and the orbital maneuvers performed after docking using the Agena vehicle for propulsion. The design loads for this phase shall consider maximum thrust and control power for conditions before and after docking. The criteria for the actual docking maneuver and for the composite Gemini-Agena vehicle are covered in paragraph 3.4.

3.3.2 Orbital Phase Meteoroid Hazard Analysis

The meteoroid environment presented in Section 2.7 shall be used to establish the structural reliability of the orbiting vehicle in resisting meteoroid penetration. The penetration equation that shall be used for finite sheet thickness is as follows:

$$\frac{t_i}{dp} = 3.42K \left\{\frac{\rho_p}{\rho t} \cdot \frac{V_p}{Ct}\right\}^{2/3}$$

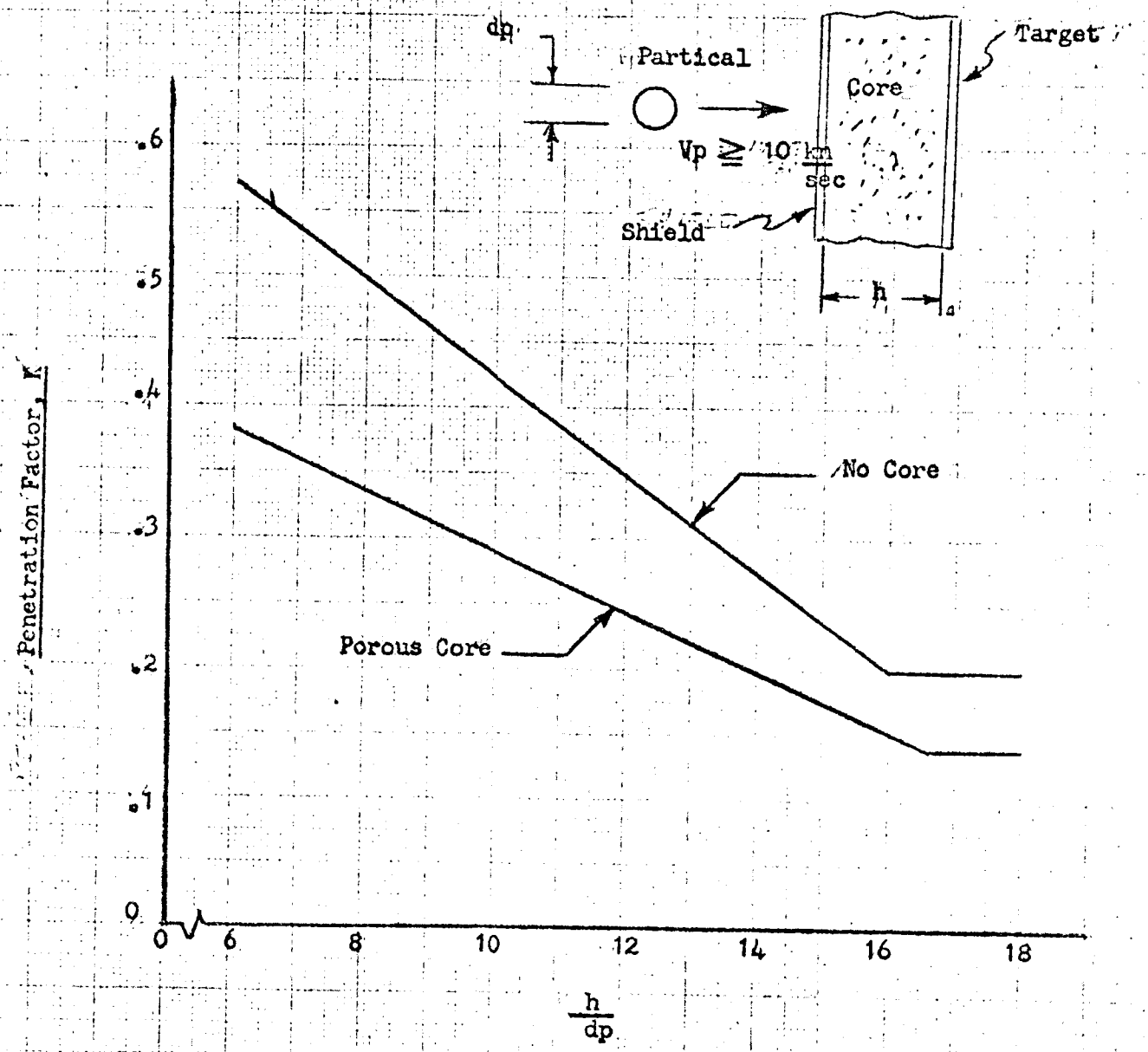
and $$t_i = t_t + t_s \left(\frac{\rho s \; Cs}{\rho t \; Ct}\right)$$

Definition of Symbols

t_i = effective target sheet thickness that will just resist penetration, cm.
K = double-wall penetration factor from Figure 3.3.2 (Note: $t_i = t_t$ and K = 1.0 for single sheet targets)
t_t = target sheet thickness, cm
t_s = shield sheet thickness, cm
ρp = density of particle, gm/cc
ρt = density of target material, gm/cc
ρs = density of shield material, gm/cc
Vp = velocity of particle, km/sec
Ct = speed of sound in target material, km/sec
Cs = speed of sound in shield material, km/sec
d_p = diameter of particle, cm

FIGURE 3.3.2

PENETRATION FACTOR, K, FOR DOUBLE WALL CONSTRUCTION



3.4 Rendezvous Phase

Design docking loads shall be those associated with any combination of 1.5 feet per second maximum relative longitudinal velocity, a ± .5 feet per second maximum relative lateral velocity, a ± 10 degree per second maximum relative rolling velocity, mismatch angles up to 10 degrees, and a maximum lateral mismatch distance of ± 1.0 ft. Interaction effects of the Gemini and Agena control systems shall be considered.

After docking is completed, the combined Gemini-Agena vehicle shall be designed to maneuver in orbit. The variations in weight of the two vehicles; the maximum Agena thrust and gimbal angle; and any dynamic effects including those due to gimbal rate, thrust build up and decay rates, and flexibility of the Gemini/Target Docking Adapter/Agena structure shall be considered in the design.

Refer to Appendix A for criteria pertaining to the Target Docking Adapter during the rendezvous phase.

3.5 Re-Entry Phase

The re-entry phase criteria covers re-entry from both orbital and abort conditions. From orbital conditions, this phase is initiated by applying retrograde thrust and begins at the instant the spacecraft is separated from the retrograde section of the adapter. From abort conditions, the re-entry phase begins at the instant the spacecraft is separated from the retrograde section of the adapter after completion of the abort operation.

Structural design for re-entry from boost phase abort shall be based on the boost phase abort boundary shown in Figure 3.5.3. The boundary defines combinations of flight path angle and velocity at which an abort may be initiated so as not to exceed the design maximum re-entry load factor of 15 g's and/or the design maximum heating rate of 70 BTU/ft.^{2}/sec., and also defines the velocity range in which maximum lift must be used so as not to exceed the design maximum re-entry load factor of 15 g's. A summary of the five trajectories selected as representative of the most critical combinations of loads and heating rates is shown in Table 3.5.4. The variation of the significant parameters with time from 400,000 ft. are shown on Figures 3.5.5 through 3.5.19.

Structural design for re-entry from orbit shall be based on re-entries using lift varying from zero to maximum from anywhere in the 161 nautical mile circular orbit and from the 87 to 161 nautical mile elliptical orbit except for the region 60° prior to and after perigee where only re-entries using zero lift will be used for design. The re-entry module shall be designed for all structural loading and temperature conditions encountered during these re-entries.

Plots of total heat which is an indicator for the structural temperatures are shown on Figure 3.5.20 for re-entries from the design orbits. A summary of some design trajectories are shown in Table 3.5.21. The variations of the significant parameters with time from 400,000 feet are shown on Figures 3.5.22 thru 3.5.39.

Restrictions imposed by off-design conditions such as re-entries from perigee with other than zero lift, re-entries from dispersed orbits, and cases where only 3 out of 4 retrograde rockets are fired will be defined by an operating boundary or envelope. Reference 11 discusses the ablation shield performance limits for off-design conditions.

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FIGURE 3.5.3

GEMINI SPACECRAFT

BOOST PHASE ABORT BOUNDARIES

DEFINED BY RE-ENTRY LOAD FACTOR AND HEATING RATE

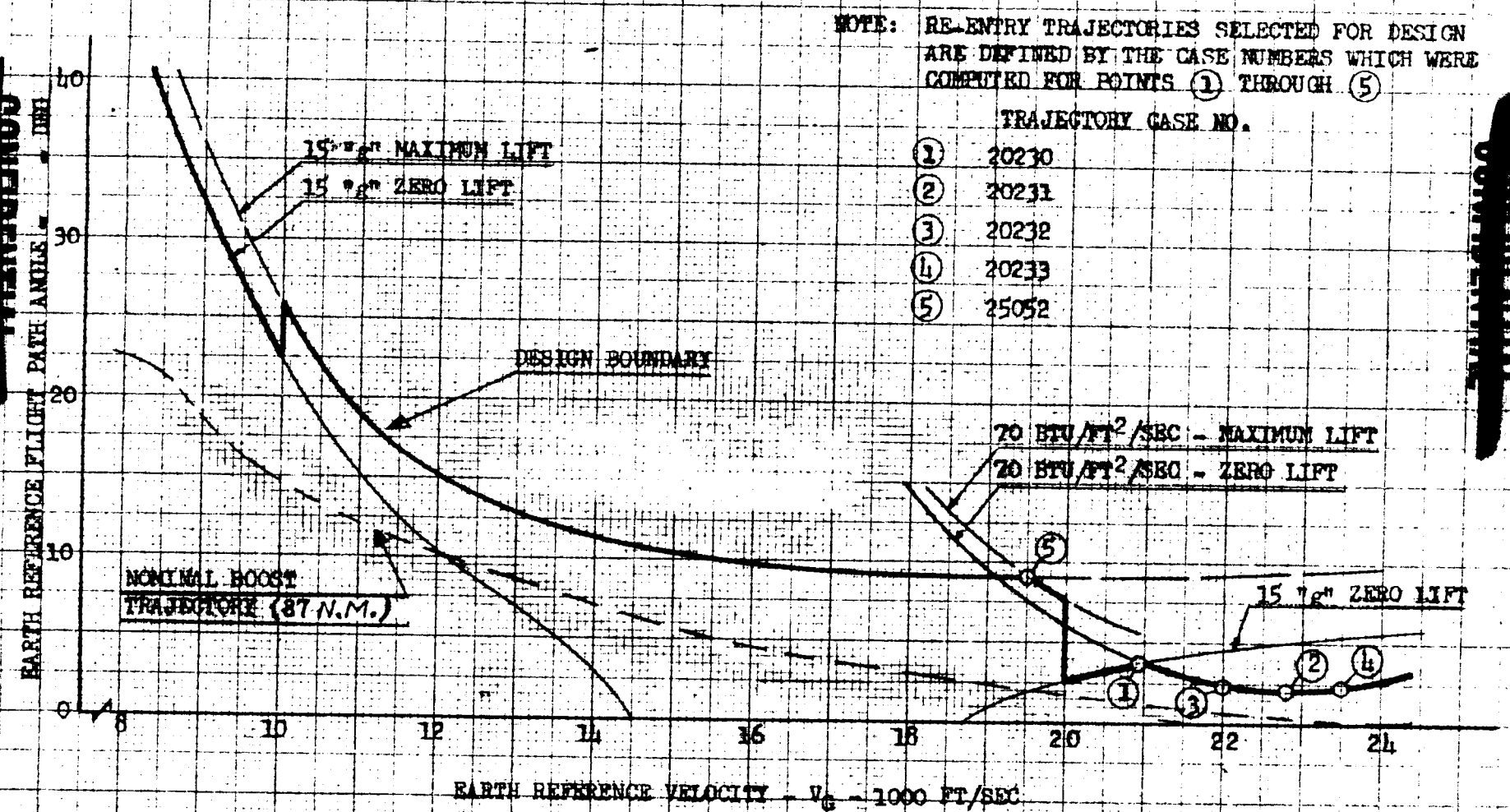

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TABLE 3.5.4

SUMMARY OF DESIGN RE-ENTRY CONDITIONS FROM BOOST PHASE ABORT

Point No.	Case No.	Abort Conditions				Re-entry Design Parameters (Note 2)					
		Time from Lift-off Sec.	Altitude Ft.	Velocity FPS	γ Deg	Lift	Time from Lift-off Sec.	Time from 400,000 ft. Sec.	Long. Load Factor	Heating Rate BTU/Ft.^{2}/Sec	Total Heat BTU/Ft.^{2}
1	20230	313.7	500,928	20,930	3.65	Zero	838.7	112.3	-7.20	70.53	2151
							858.7	132.3	-14.96	30.71	3304
							873.7	147.3	-8.73	3.99	3510
							998.7	272.3	-1.07	.006	3543
2	20231	323.82	501,234	22,800	2.0	Zero	1043.82	169.32	-5.89	70.42	3580
							1073.82	199.32	-11.04	25.29	5177
							1088.82	214.32	-8.07	5.78	5380
							1173.82	299.32	-1.19	.015	5436
3	20232	319.75	501,171	22,000	2.3	Zero	884.75	142.85	-6.32	70.77	2856
							909.75	167.85	-12.57	28.96	4261
							924.75	182.85	-8.63	5.50	4485
							1094.75	352.85	-1.04	.004	4530
4	20233	327.14	501,209	23,500	2.25	Zero	1507.14	187.99	-5.12	70.20	4168
							1542.14	222.99	-10.00	26.39	6048
							1557.14	239.99	-8.12	7.39	6289
							1662.14	342.99	-1.12	.011	6352
5	25052	304.5	500,000	19,500	9.0	Max.	969.5	71.2	-9.71	71.38	1439
							979.5	81.2	-14.91	48.10	2074
							1014.5	116.2	-3.28	1.48	2527
							1079.5	181.2	-1.20	.03	2550

NOTES:
1. Re-entry weight 5050 lb.
2. For each case under Re-entry Design Parameters, the first line is maximum heating rate, the second is maximum load factor, the third is approximately 99% of maximum total heat, and the fourth is maximum total heat.

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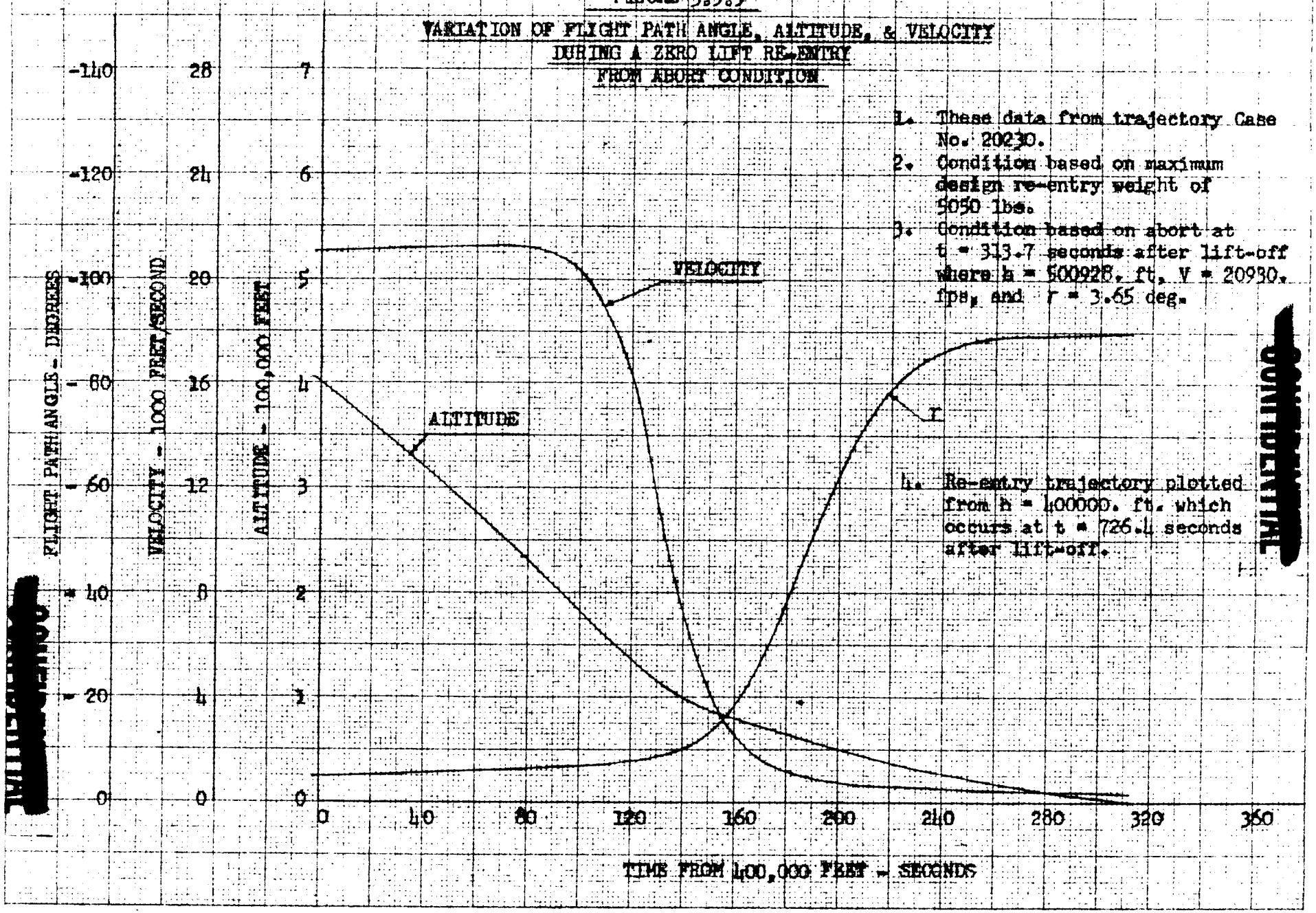
FIGURE 3.5.5

VARIATION OF FLIGHT PATH ANGLE, ALTITUDE, & VELOCITY
DURING A ZERO LIFT RE-ENTRY
FROM ABORT CONDITION



1. These data from trajectory Case No. 20230.
2. Condition based on maximum design re-entry weight of 5050 lbs.
3. Condition based on abort at t = 313.7 seconds after lift-off where h = 500928. ft, V = 20930. fps, and γ = 3.65 deg.
4. Re-entry trajectory plotted from h = 400000. ft. which occurs at t = 726.4 seconds after lift-off.

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[redacted]

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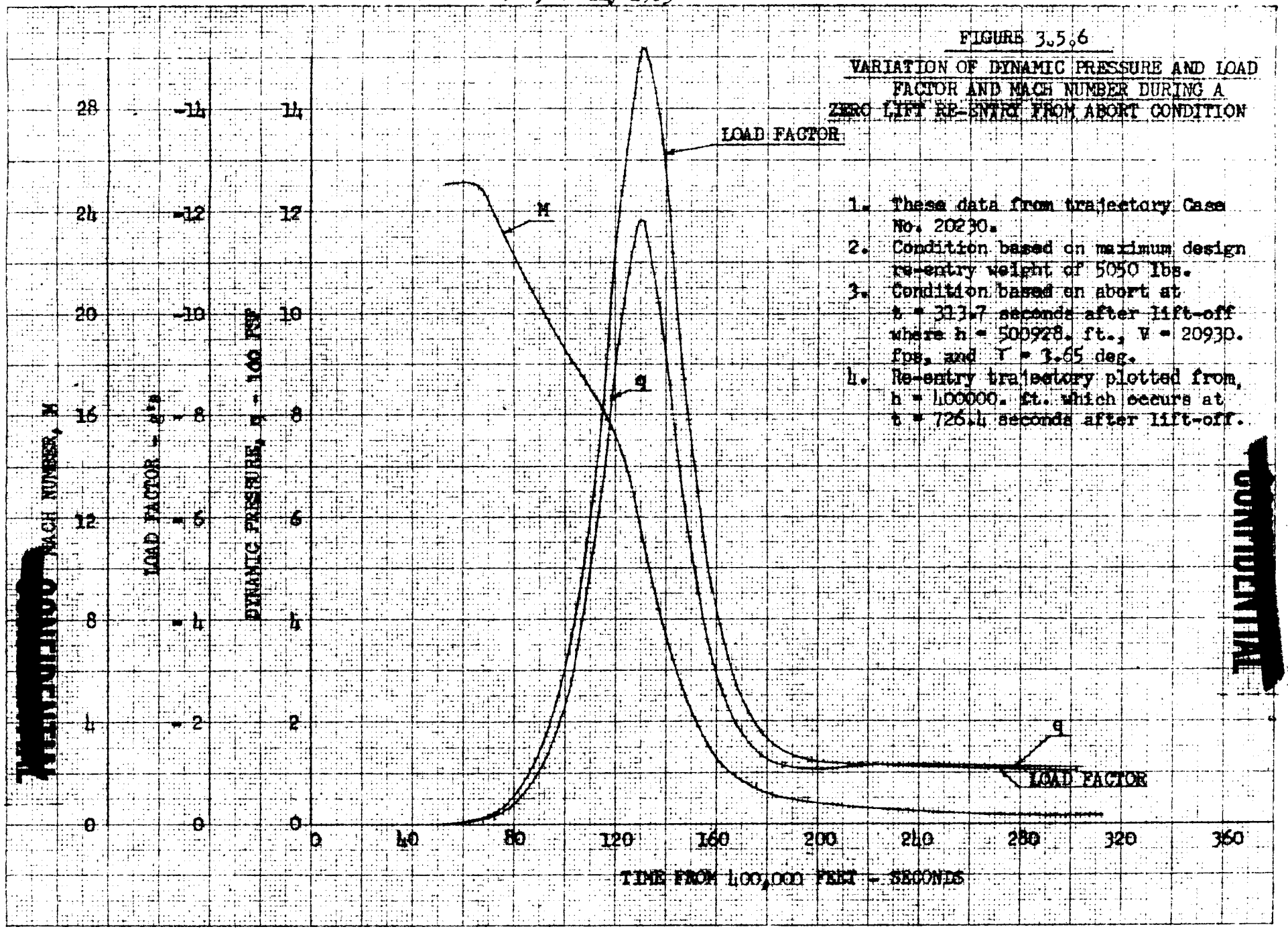

FIGURE 3.5.6

VARIATION OF DYNAMIC PRESSURE AND LOAD FACTOR AND MACH NUMBER DURING A ZERO LIFT RE-ENTRY FROM ABORT CONDITION

1. These data from trajectory Case No. 20230.
2. Condition based on maximum design re-entry weight of 5050 lbs.
3. Condition based on abort at t = 313.7 seconds after lift-off where h = 500928. ft., V = 20930. fps, and γ = 3.65 deg.
4. Re-entry trajectory plotted from, h = 400000. ft. which occurs at t = 726.4 seconds after lift-off.

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Revised 9 June 1962, 1 July 1963

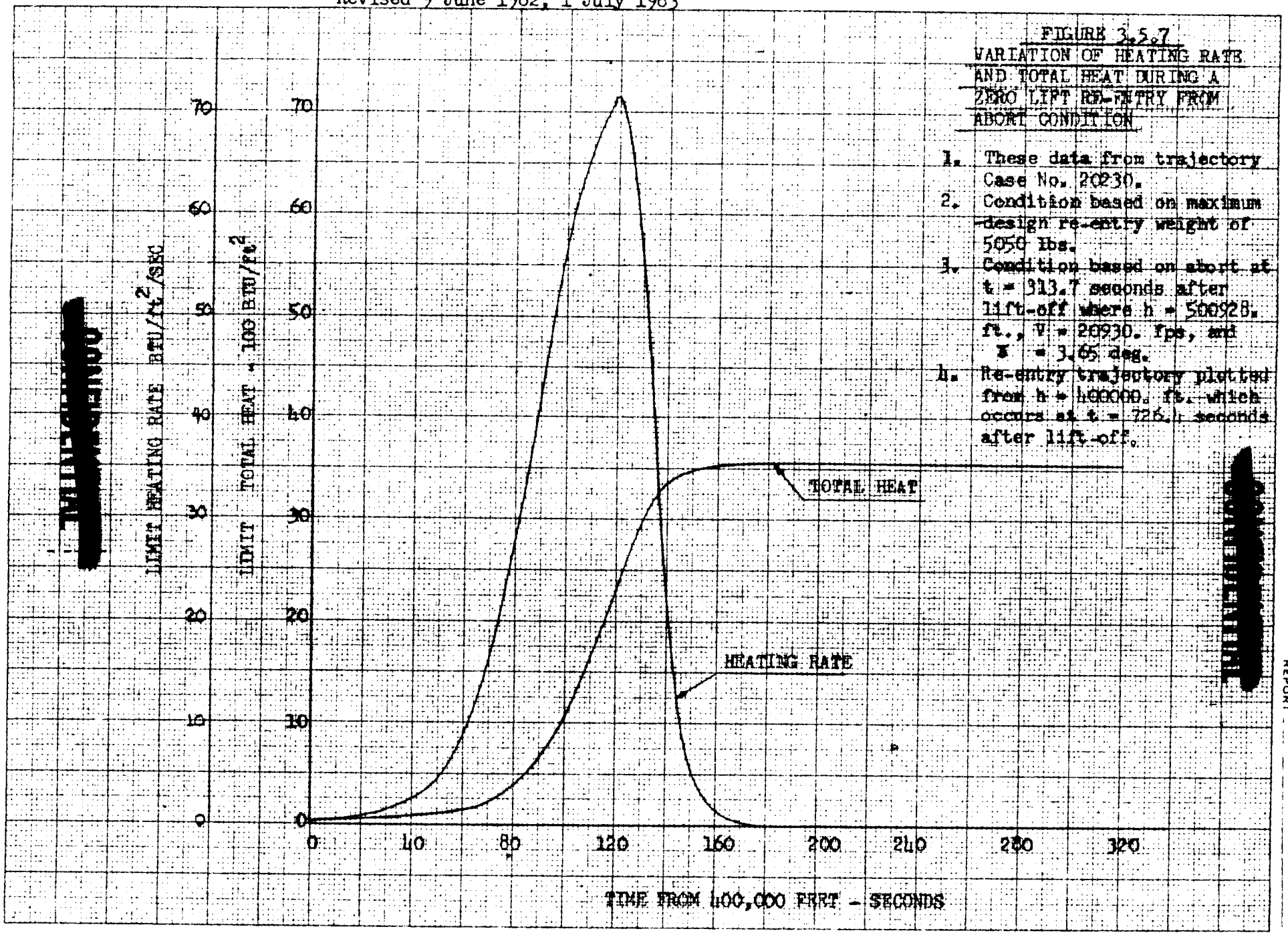

FIGURE 3.5.7
VARIATION OF HEATING RATE AND TOTAL HEAT DURING A ZERO LIFT RE-ENTRY FROM ABORT CONDITION

1. These data from trajectory Case No. 20230.
2. Condition based on maximum design re-entry weight of 5050 lbs.
3. Condition based on abort at t = 313.7 seconds after lift-off where h = 500928. ft., V = 20930. fps, and γ = 3.65 deg.
4. Re-entry trajectory plotted from h = 400000. ft. which occurs at t = 726.1 seconds after lift-off.

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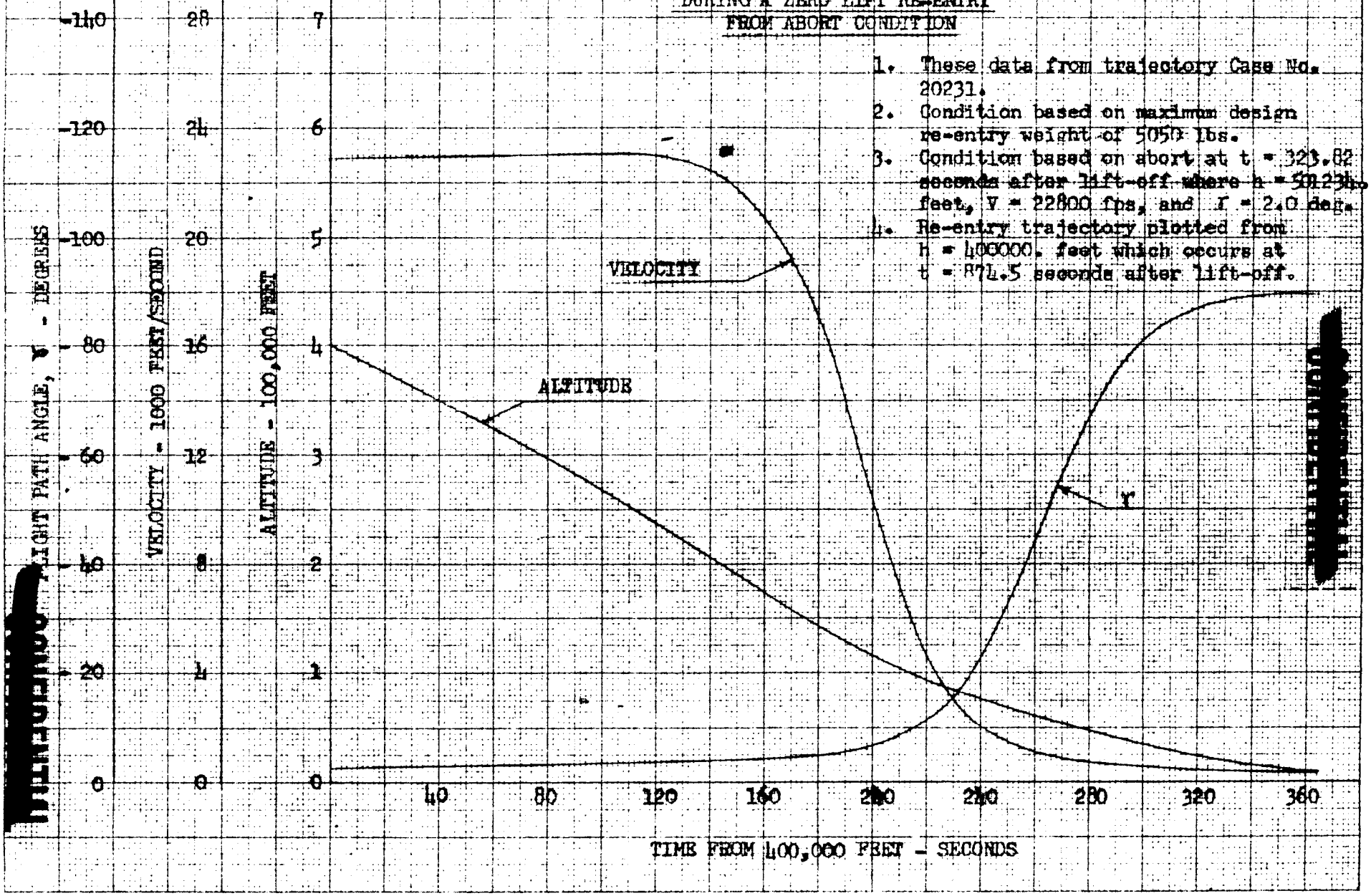
FIGURE 3.5.8
VARIATION OF FLIGHT PATH ANGLE, ALTITUDE, AND VELOCITY
DURING A ZERO LIFT RE-ENTRY
FROM ABORT CONDITION

1. These data from trajectory Case No. 20231.
2. Condition based on maximum design re-entry weight of 5050 lbs.
3. Condition based on abort at t = 323.82 seconds after lift-off where h = 50234. feet, V = 22800 fps, and γ = 2.0 deg.
4. Re-entry trajectory plotted from h = 400000. feet which occurs at t = 874.5 seconds after lift-off.



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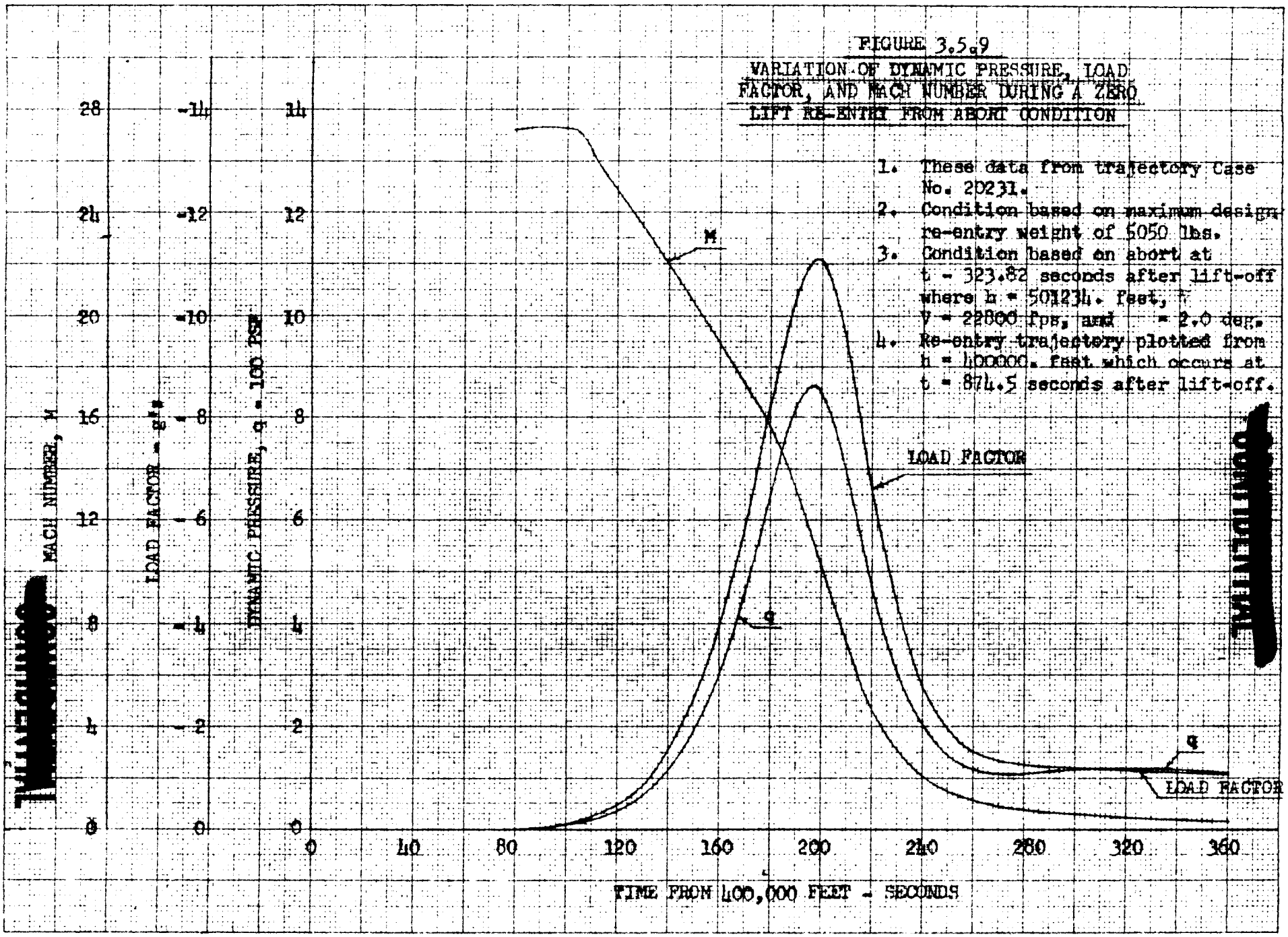

FIGURE 3.5.9

VARIATION OF DYNAMIC PRESSURE, LOAD FACTOR, AND MACH NUMBER DURING A ZERO LIFT RE-ENTRY FROM ABORT CONDITION

1. These data from trajectory Case No. 20231.
2. Condition based on maximum design re-entry weight of 5050 lbs.
3. Condition based on abort at t - 323.82 seconds after lift-off where h = 501234. feet, V = 22800 fps, and = 2.0 deg.
4. Re-entry trajectory plotted from h = 400000. feet which occurs at t = 874.5 seconds after lift-off.

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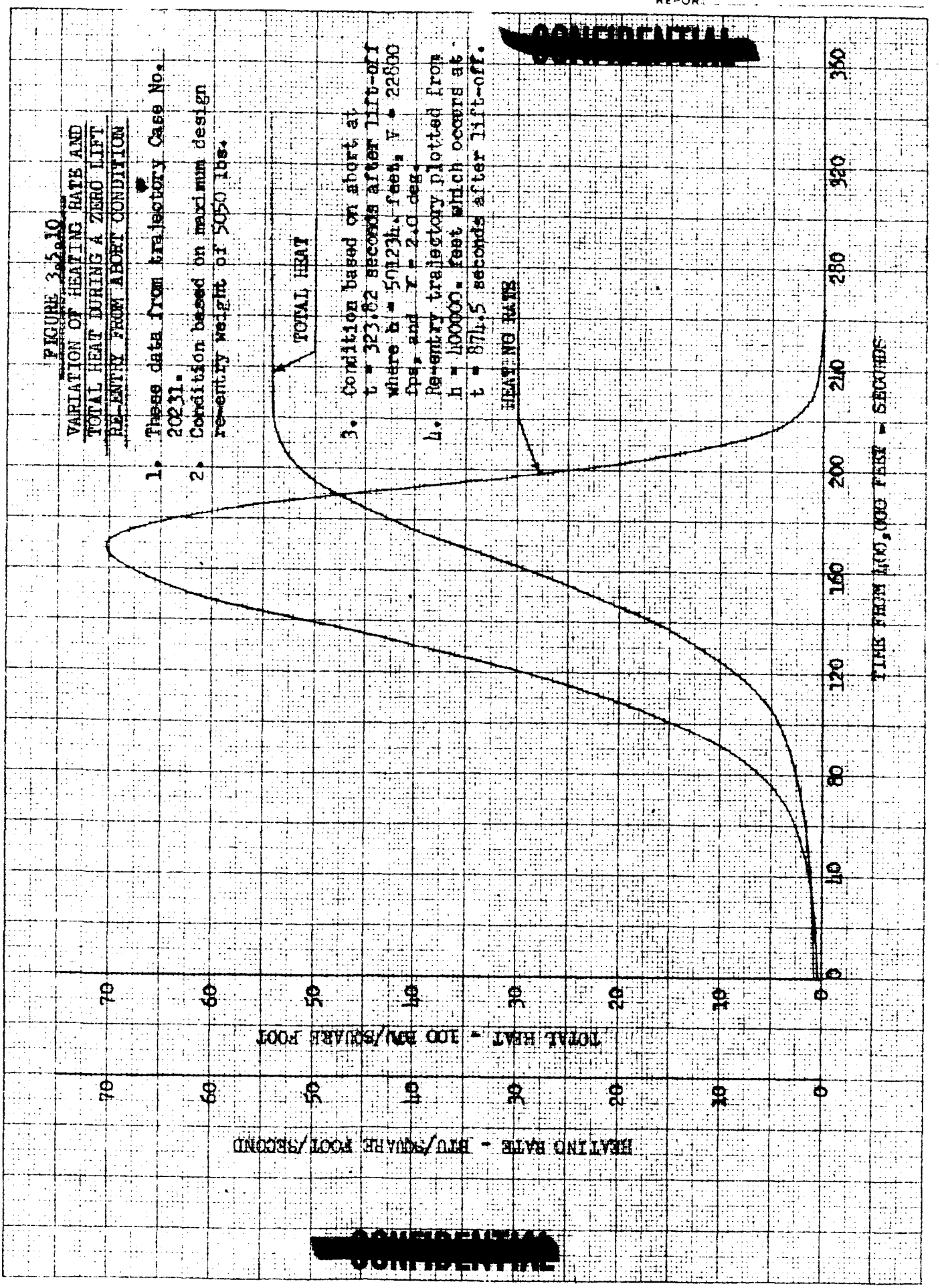
[redacted]

[redacted]



PIGURE 3.5.10
VARIATION OF HEATING RATE AND TOTAL HEAT DURING A ZERO LIFT RE-ENTRY FROM ABORT CONDITION

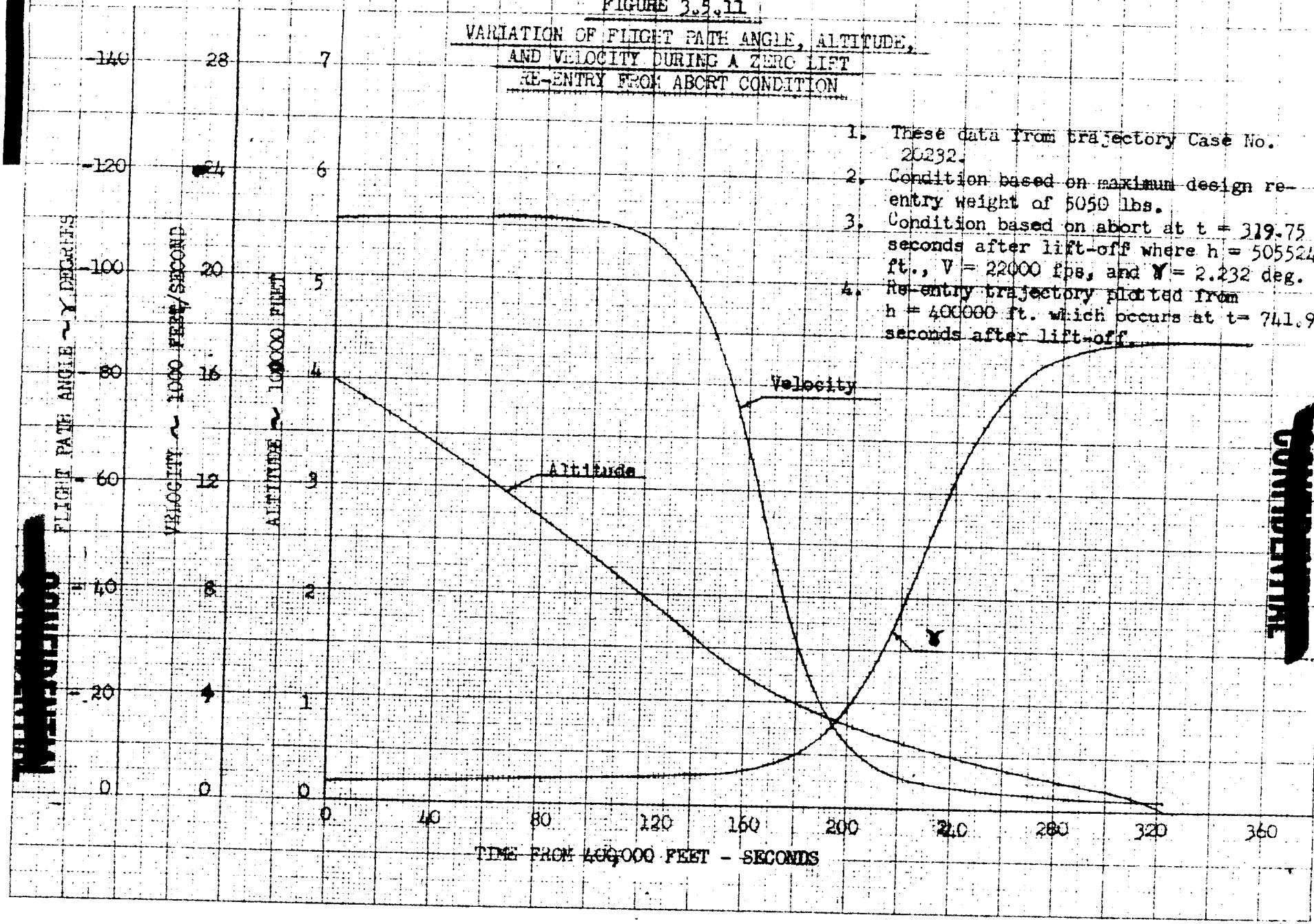
1. These data from trajectory Case No. 20231.
2. Condition based on maximum design re-entry weight of 5050 lbs.
3. Condition based on abort at t = 323.82 seconds after lift-off where h = 501234. feet, V = 22600 fps, and γ = 2.0 deg.
4. Re-entry trajectory plotted from h = 400000. feet which occurs at t = 874.5 seconds after lift-off.

[redacted]

[redacted]

FIGURE 3.5.11

VARIATION OF FLIGHT PATH ANGLE, ALTITUDE, AND VELOCITY DURING A ZERO LIFT RE-ENTRY FROM ABORT CONDITION



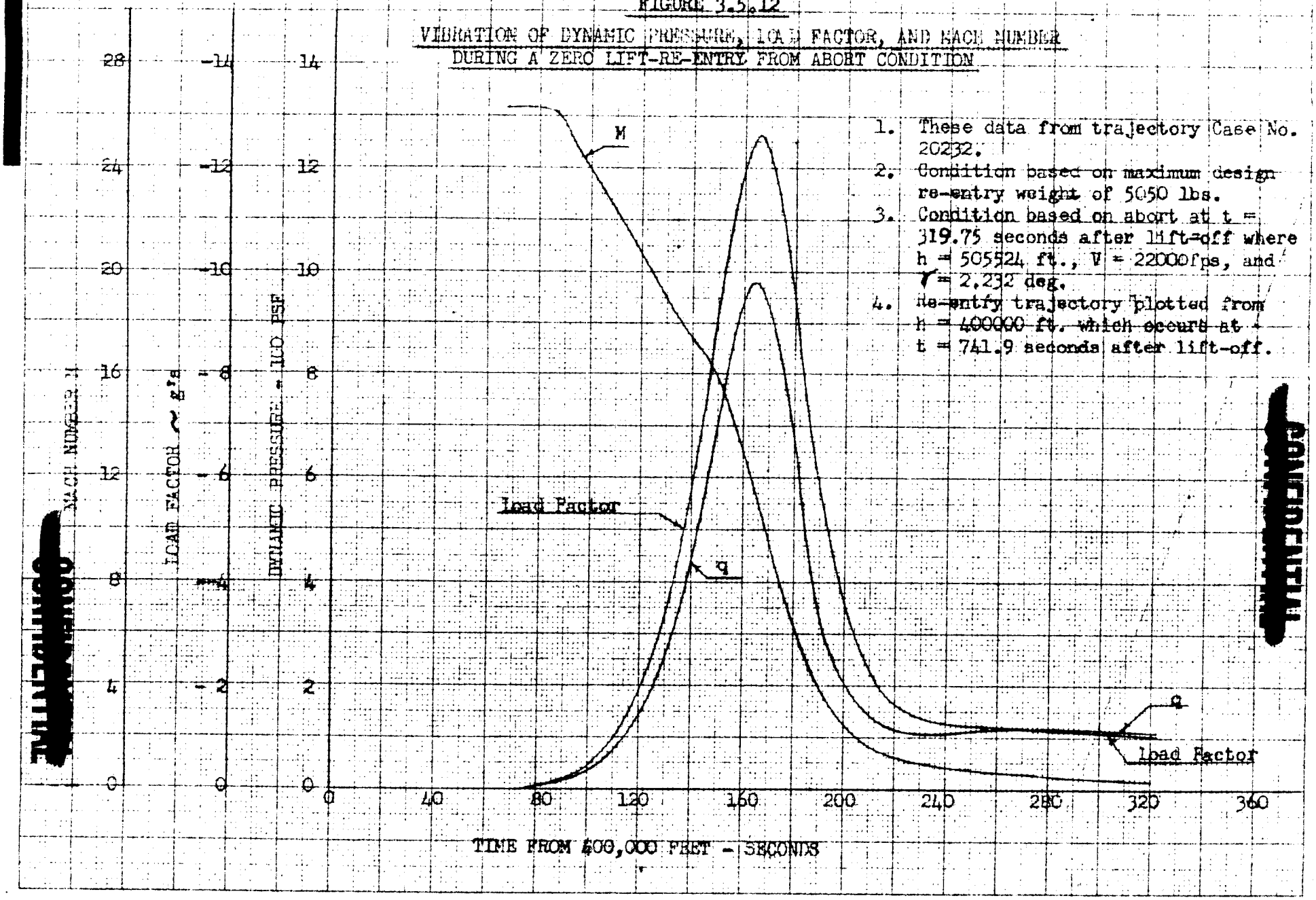
1. These data from trajectory Case No. 20232.
2. Condition based on maximum design re-entry weight of 5050 lbs.
3. Condition based on abort at t = 319.75 seconds after lift-off where h = 505524 ft., V = 22000 fps, and γ = 2.232 deg.
4. Re-entry trajectory plotted from h = 400000 ft. which occurs at t = 741.9 seconds after lift-off.

[redacted]

[redacted]

FIGURE 3.5.12

VIBRATION OF DYNAMIC PRESSURE, LOAD FACTOR, AND MACH NUMBER DURING A ZERO LIFT-RE-ENTRY FROM ABORT CONDITION



1. These data from trajectory Case No. 20232.
2. Condition based on maximum design re-entry weight of 5050 lbs.
3. Condition based on abort at t = 319.75 seconds after lift-off where h = 505524 ft., V = 22000 fps, and γ = 2.232 deg.
4. Re-entry trajectory plotted from h = 400000 ft. which occurs at t = 741.9 seconds after lift-off.

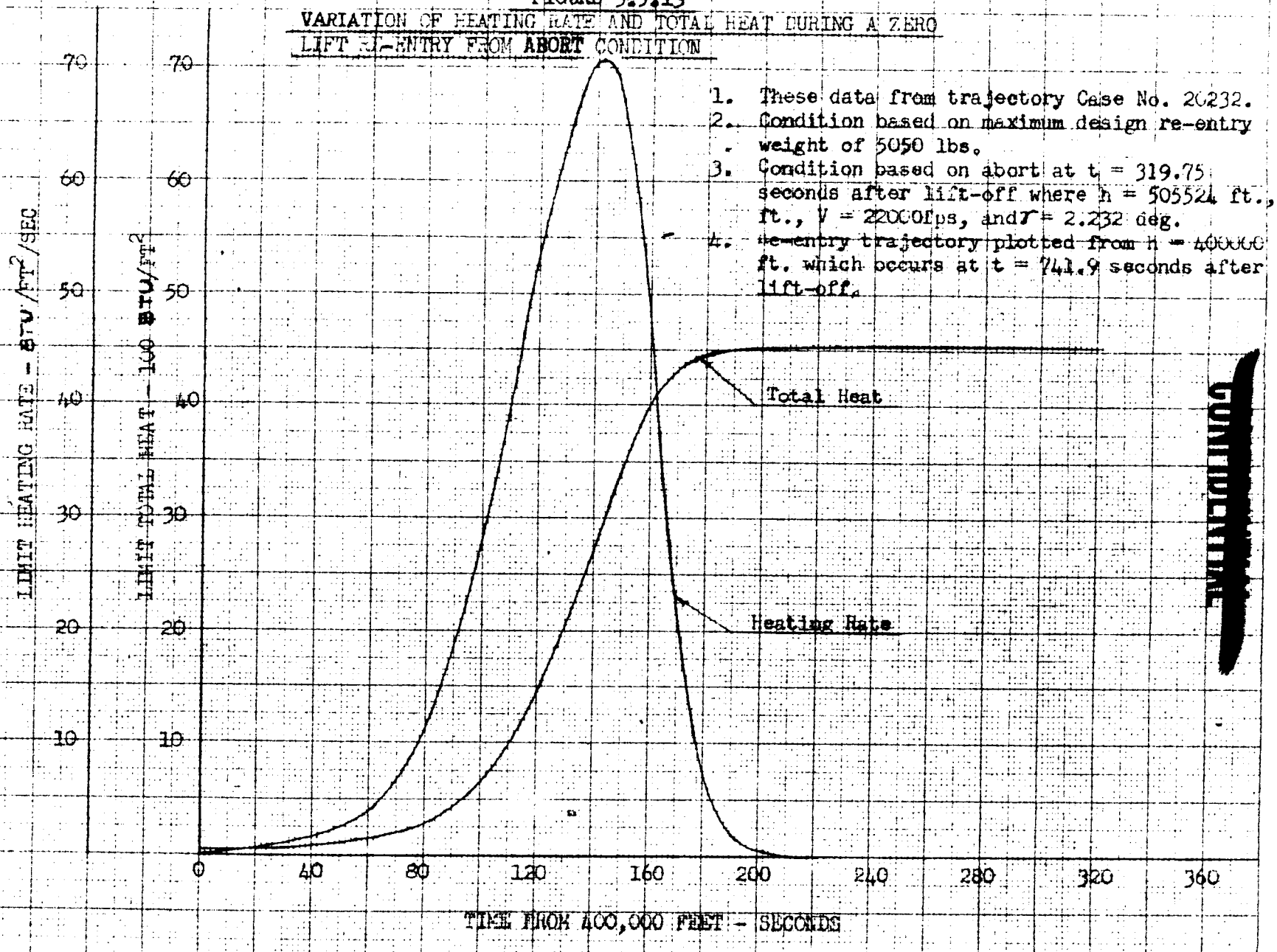
[redacted]

[redacted]

[redacted]

FIGURE 3.5.13

VARIATION OF HEATING RATE AND TOTAL HEAT DURING A ZERO LIFT RE-ENTRY FROM ABORT CONDITION



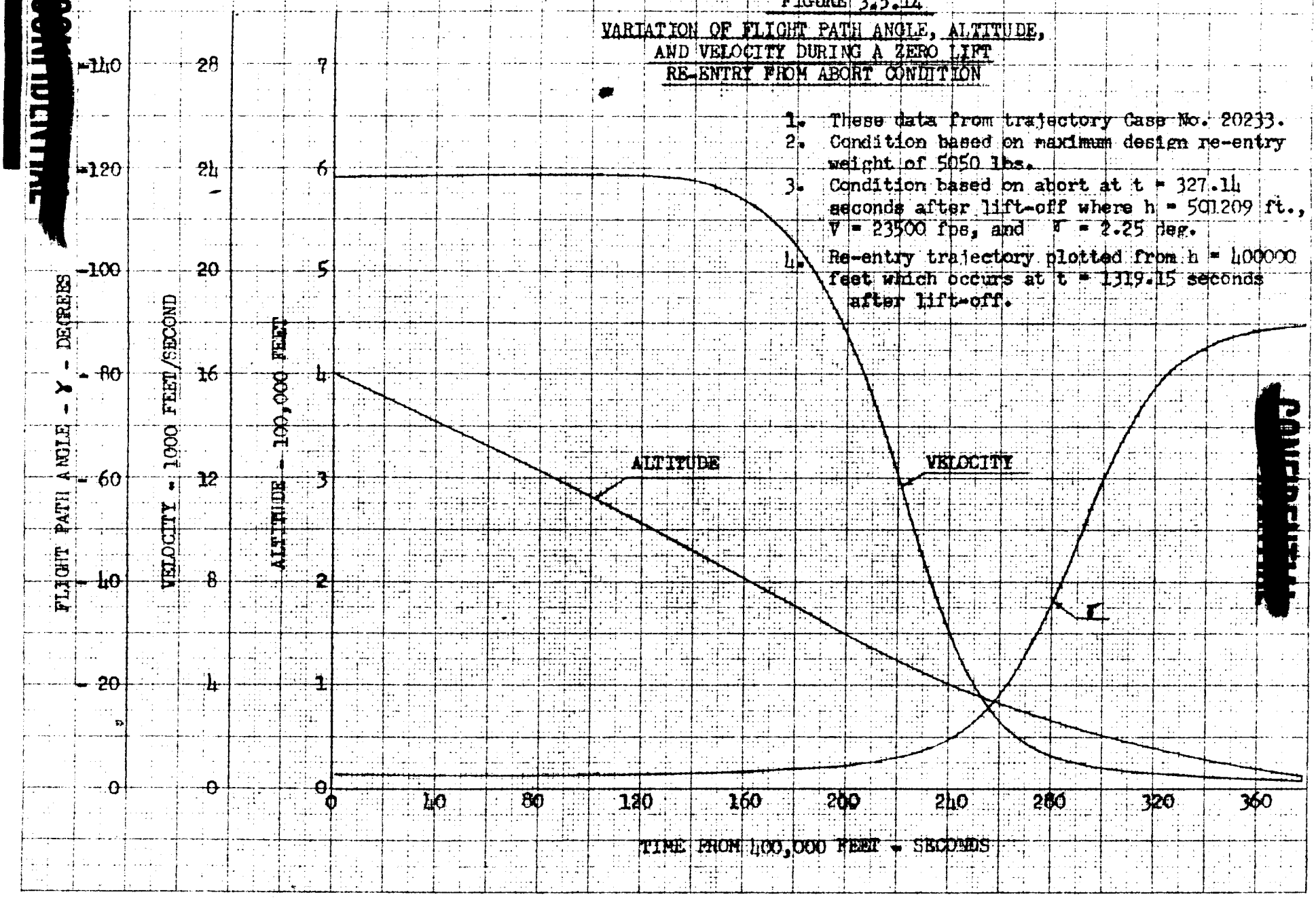
[redacted]

[redacted]

[redacted]

FIGURE 3.5.14

VARIATION OF FLIGHT PATH ANGLE, ALTITUDE, AND VELOCITY DURING A ZERO LIFT RE-ENTRY FROM ABORT CONDITION



[redacted]

[redacted]

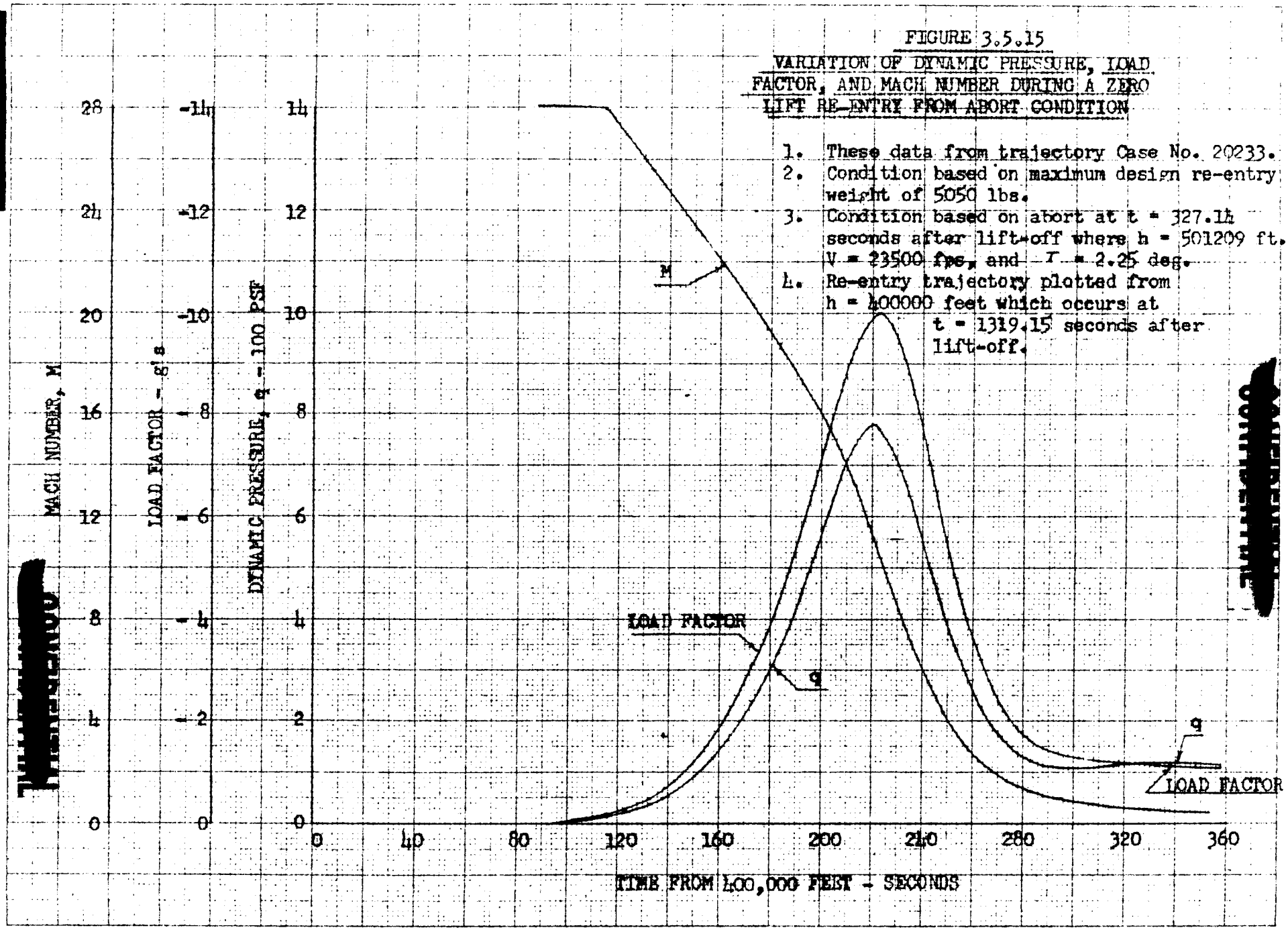


FIGURE 3.5.15

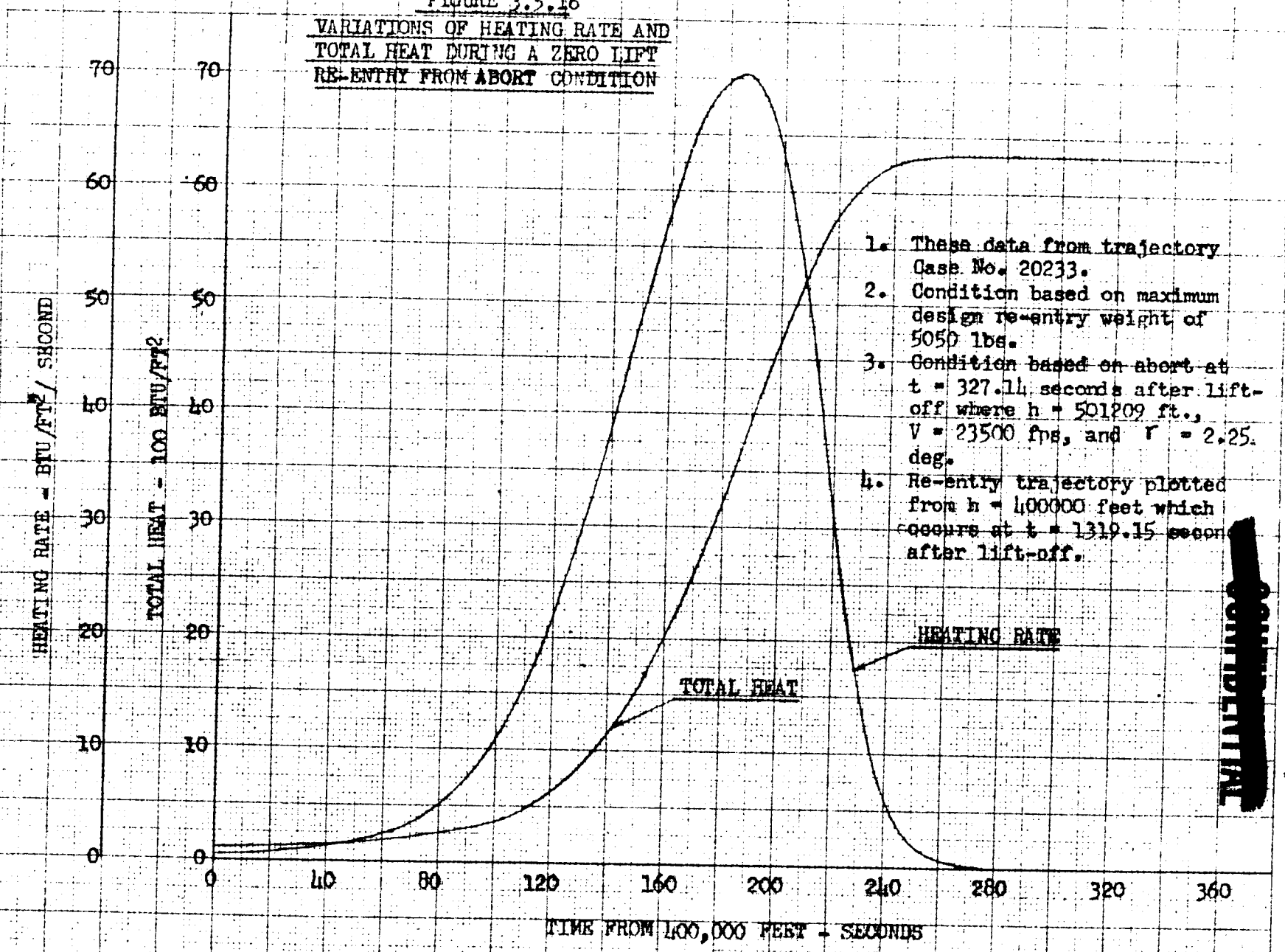
VARIATION OF DYNAMIC PRESSURE, LOAD FACTOR, AND MACH NUMBER DURING A ZERO LIFT RE-ENTRY FROM ABORT CONDITION

1. These data from trajectory Case No. 20233.
2. Condition based on maximum design re-entry weight of 5050 lbs.
3. Condition based on abort at t = 327.14 seconds after lift-off where h = 501209 ft., V = 23500 fps, and γ = 2.25 deg.
4. Re-entry trajectory plotted from h = 400000 feet which occurs at t = 1319.15 seconds after lift-off.

[redacted]

[redacted]

FIGURE 3.5.16
VARIATIONS OF HEATING RATE AND TOTAL HEAT DURING A ZERO LIFT RE-ENTRY FROM ABORT CONDITION

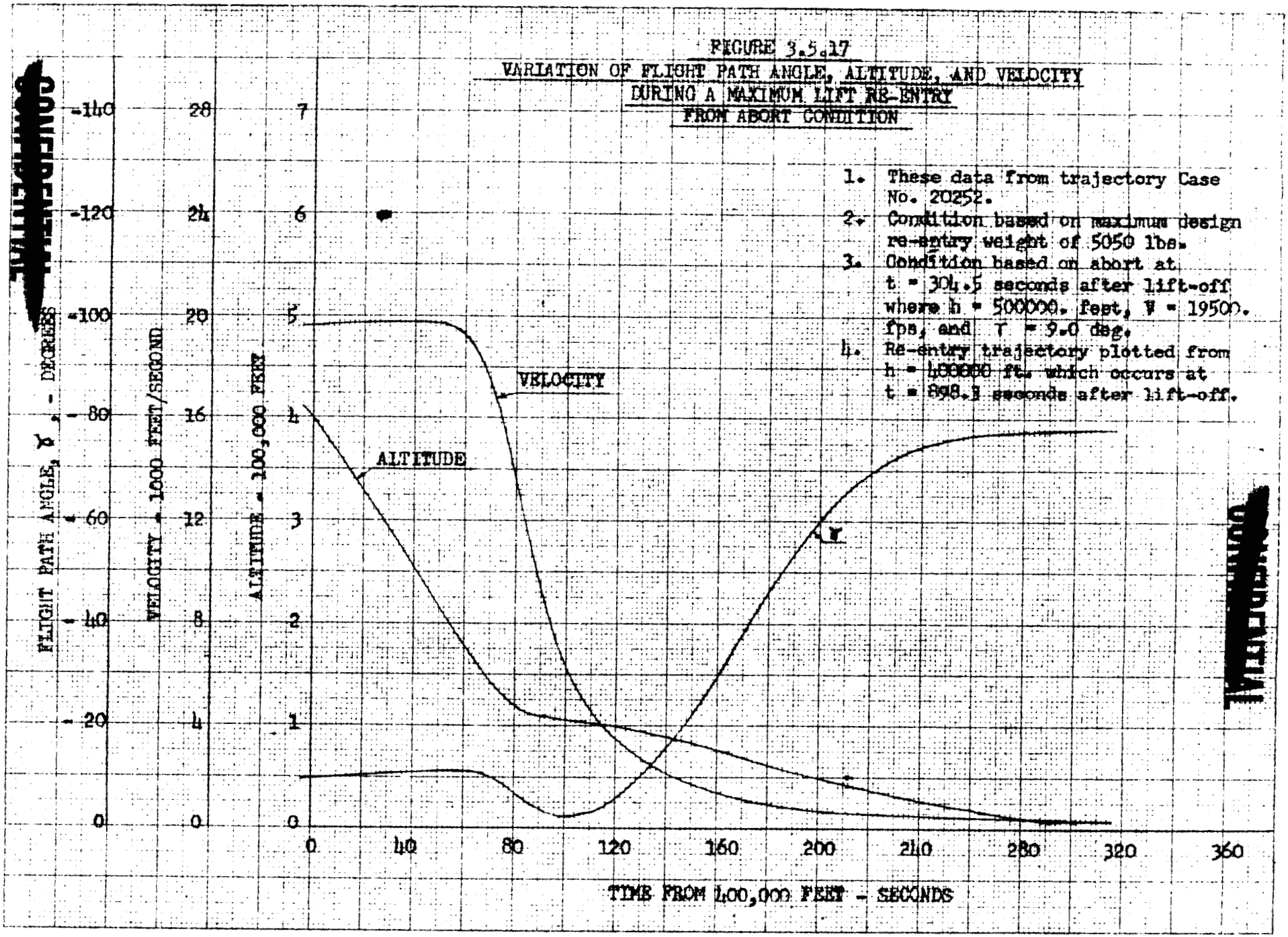


1. These data from trajectory Case No. 20233.
2. Condition based on maximum design re-entry weight of 5050 lbs.
3. Condition based on abort at t = 327.14 seconds after lift-off where h = 501209 ft., V = 23500 fps, and γ = 2.25 deg.
4. Re-entry trajectory plotted from h = 400000 feet which occurs at t = 1319.15 seconds after lift-off.

[redacted]

[redacted]

FIGURE 3.5.17
VARIATION OF FLIGHT PATH ANGLE, ALTITUDE, AND VELOCITY
DURING A MAXIMUM LIFT RE-ENTRY
FROM ABORT CONDITION



1. These data from trajectory Case No. 20252.
2. Condition based on maximum design re-entry weight of 5050 lbs.
3. Condition based on abort at t = 304.5 seconds after lift-off where h = 500000. feet, V = 19500. fps, and γ = 9.0 deg.
4. Re-entry trajectory plotted from h = 400000 ft. which occurs at t = 898.3 seconds after lift-off.

[redacted]

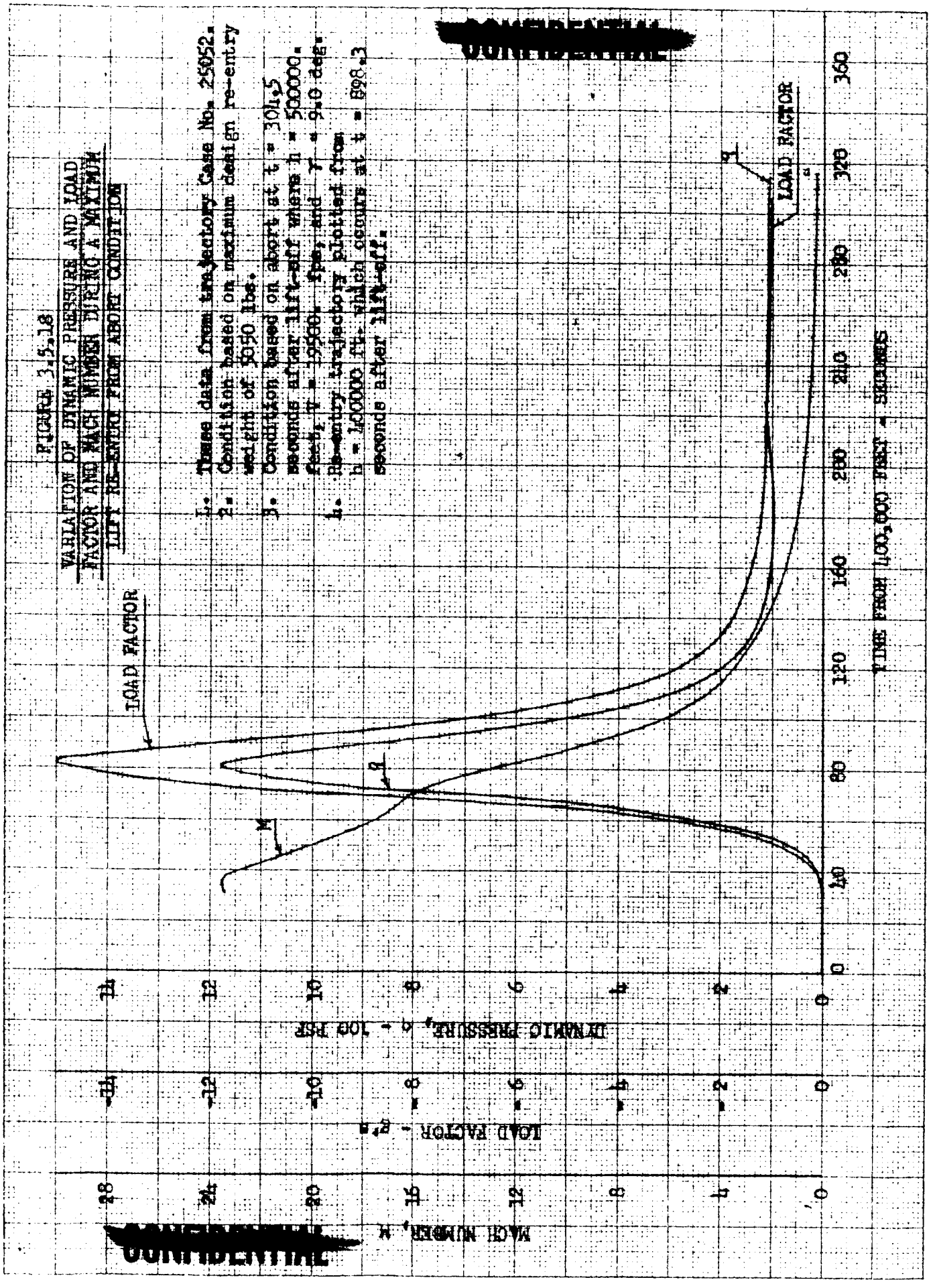
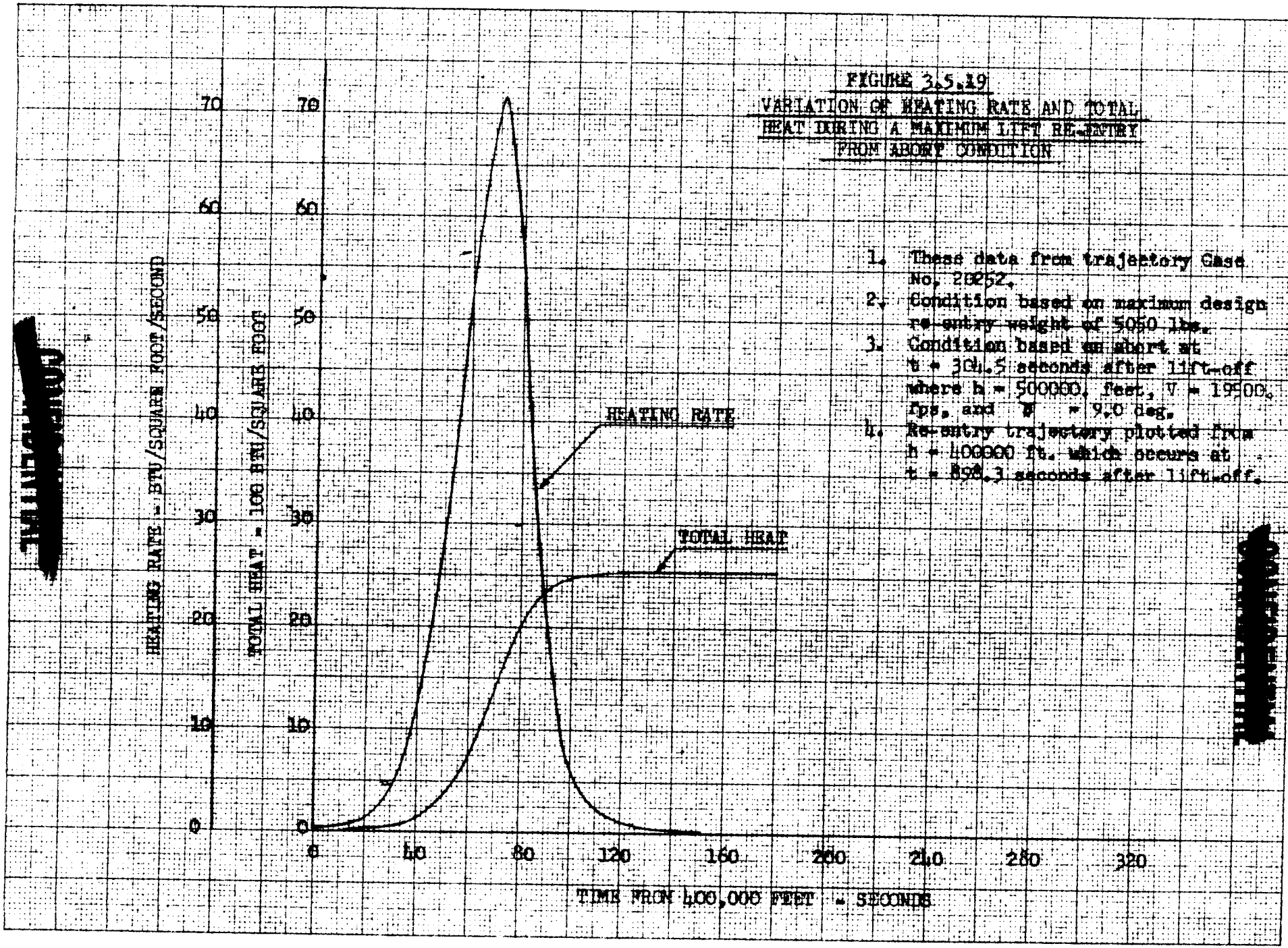


FIGURE 3.5.18
VARIATION OF DYNAMIC PRESSURE AND LOAD FACTOR AND MACH NUMBER DURING A MAXIMUM LIFT RE-ENTRY FROM ABORT CONDITION
1. These data from trajectory Case No. 25052.
2. Condition based on maximum design re-entry weight of 5050 lbs.
3. Condition based on abort at t = 304.5 seconds after lift-off where h = 500000. feet, V = 19500. fps, and γ = 9.0 deg.
4. Re-entry trajectory plotted from h = 400000 ft. which occurs at t = 898.3 seconds after lift-off.
LOAD FACTOR
q
M
LOAD FACTOR
TIME FROM 400,000 FEET - SECONDS
DYNAMIC PRESSURE, q - 100 PSF
LOAD FACTOR - n
MACH NUMBER, M
CONFIDENTIAL
CONFIDENTIAL

FIGURE 3.5.19
VARIATION OF HEATING RATE AND TOTAL HEAT DURING A MAXIMUM LIFT RE-ENTRY FROM ABORT CONDITION



1. These data from trajectory Case No. 20252.
2. Condition based on maximum design re-entry weight of 5050 lbs.
3. Condition based on abort at t = 304.5 seconds after lift-off where h = 500000. feet, V = 19500. fps, and γ = 9.0 deg.
4. Re-entry trajectory plotted from h = 400000 ft. which occurs at t = 898.3 seconds after lift-off.

[redacted]

[redacted]

Revised 1 May 1964

[redacted]

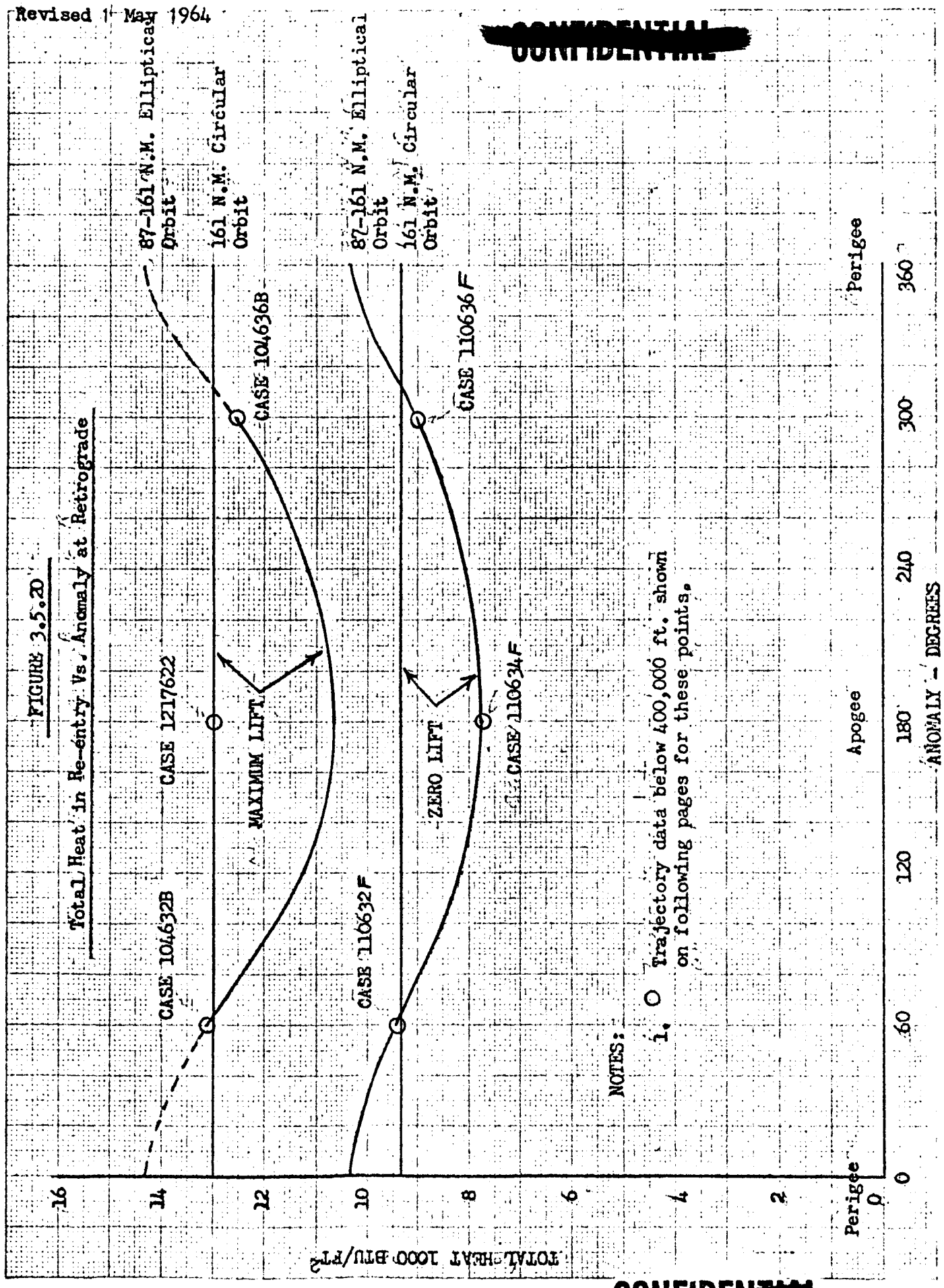


FIGURE 3.5.20

Total Heat in Re-entry Vs. Anomaly at Retrograde

NOTES:

1. ○ Trajectory data below 400,000 ft. shown on following pages for these points.

[redacted]

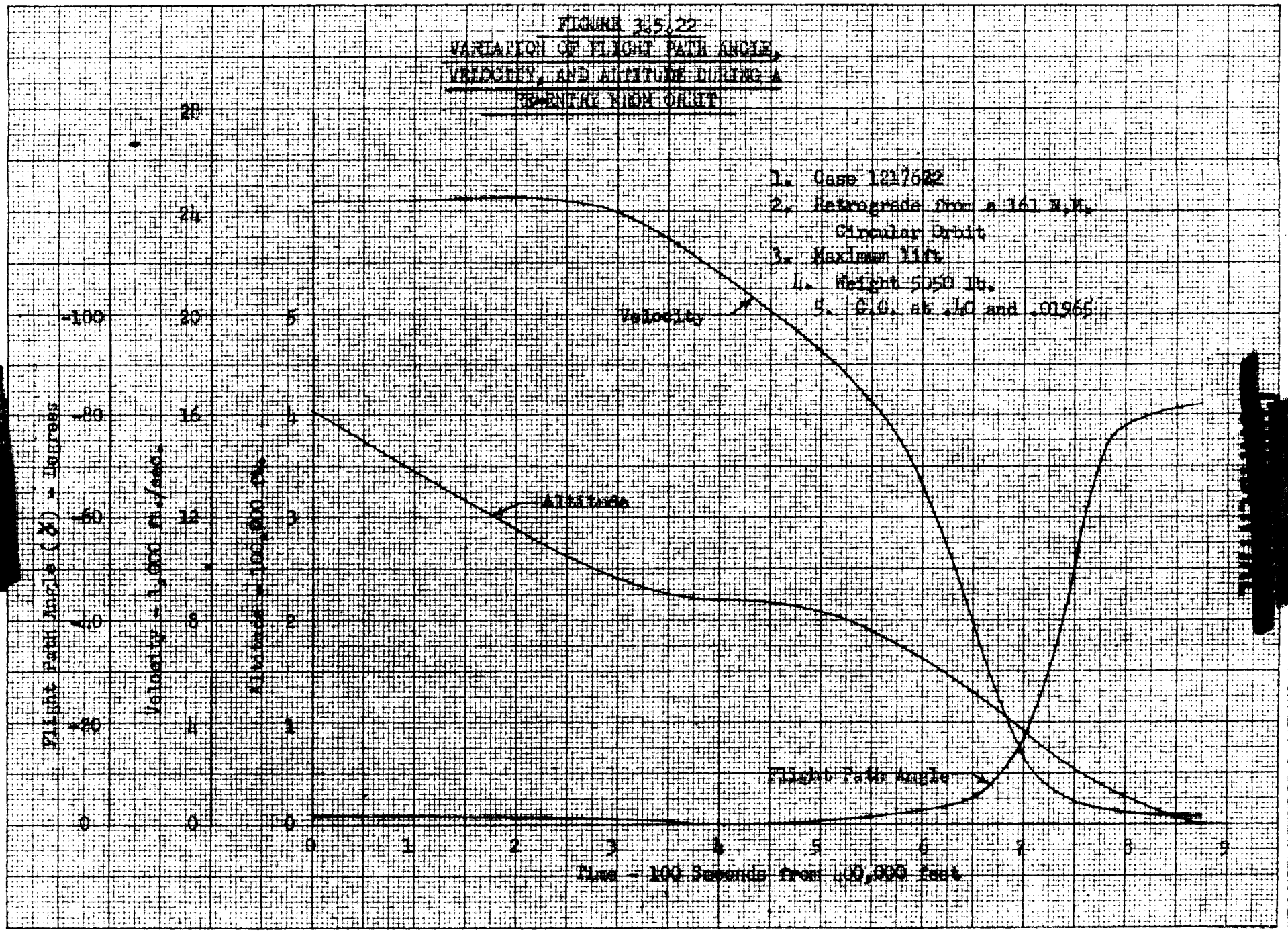
TABLE 3.5.21
SUMMARY OF DESIGN RE-ENTRY CONDITIONS FROM CIRCULAR AND ELLIPTICAL ORBITS

CASE NO.	RETROGRADE CONDITIONS					400,000 Ft.	RE-ENTRY DESIGN PARAMETERS (SEE NOTE)					
	POINT IN ORBIT	RETRO ANGLE DEG.	ATTITUDE FT.	VELOCITY FPS	γ DEG.	γ DEG.	LIFT	TIME FROM RETRO. SEC.	TIME FROM 400,000 FT SEC.	LONG. LOAD FACTOR	HEATING RATE BTU/FT^2 SEC.	TOTAL HEAT BTU/FT^2
1217622	161 N.M. Circular	-16	978,252	23,984	0	-1.408	Max.	1710 2010 2075	355 655 720	- .82 -3.78 -1.78	38.92 6.65 .15	4667 12850 12986
104632B	87-161 N.M. 60° after Perigee	-16	633,433	24,266	.561	-1.091	Max.	1690 1980 2035	410 700 755	- .79 -3.77 -2.26	37.37 6.81 .33	5151 13000 13138
104636B	87-161 N.M. 60° before Perigee	-16	632,522	24,266	-.560	-1.268	Max.	905 1185 1275	355 635 725	- .91 -3.81 -1.15	40.26 7.09 0	4677 12359 12509
110632F	87-161 N.M. 60° after Perigee	-16	633,433	24,266	.561	-1.091	Zero	1705 1770 1960	425 490 680	-2.70 -7.77 -1.04	53.69 17.19 0	6681 9409 9638
110634F	87-161 N.M. at Apogee	-16	988,278	23,825	.030	-2.013	Zero	1335 1390 1580	250 305 495	-3.27 -8.35 -1.04	63.25 22.22 0	4829 7557 7858
110636F	87-161 N.M. 60° before Perigee	-16	632,522	24,266	-.560	-1.268	Zero	920 985 1175	370 435 625	-2.71 -7.83 -1.04	55.47 18.30 0	6060 8900 9146

Re-Entry Weight 5050 lbs., C.G. at Z/D = .40 and Y/D = .01965

[redacted]

FIGURE 3.5.22
VARIATION OF FLIGHT PATH ANGLE, VELOCITY, AND ALTITUDE DURING A RETROGRADE NION ORBIT



[redacted]

[redacted]

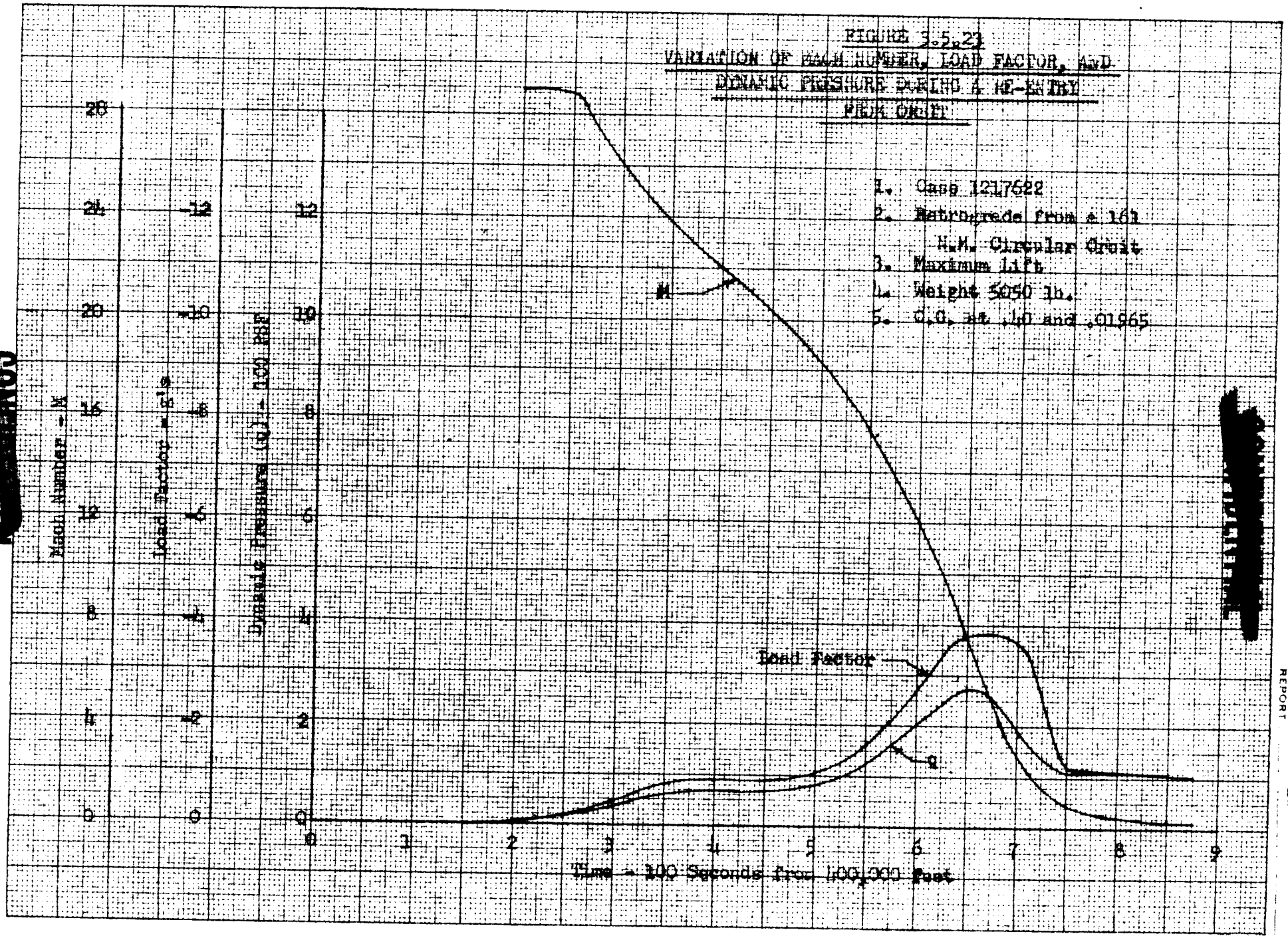
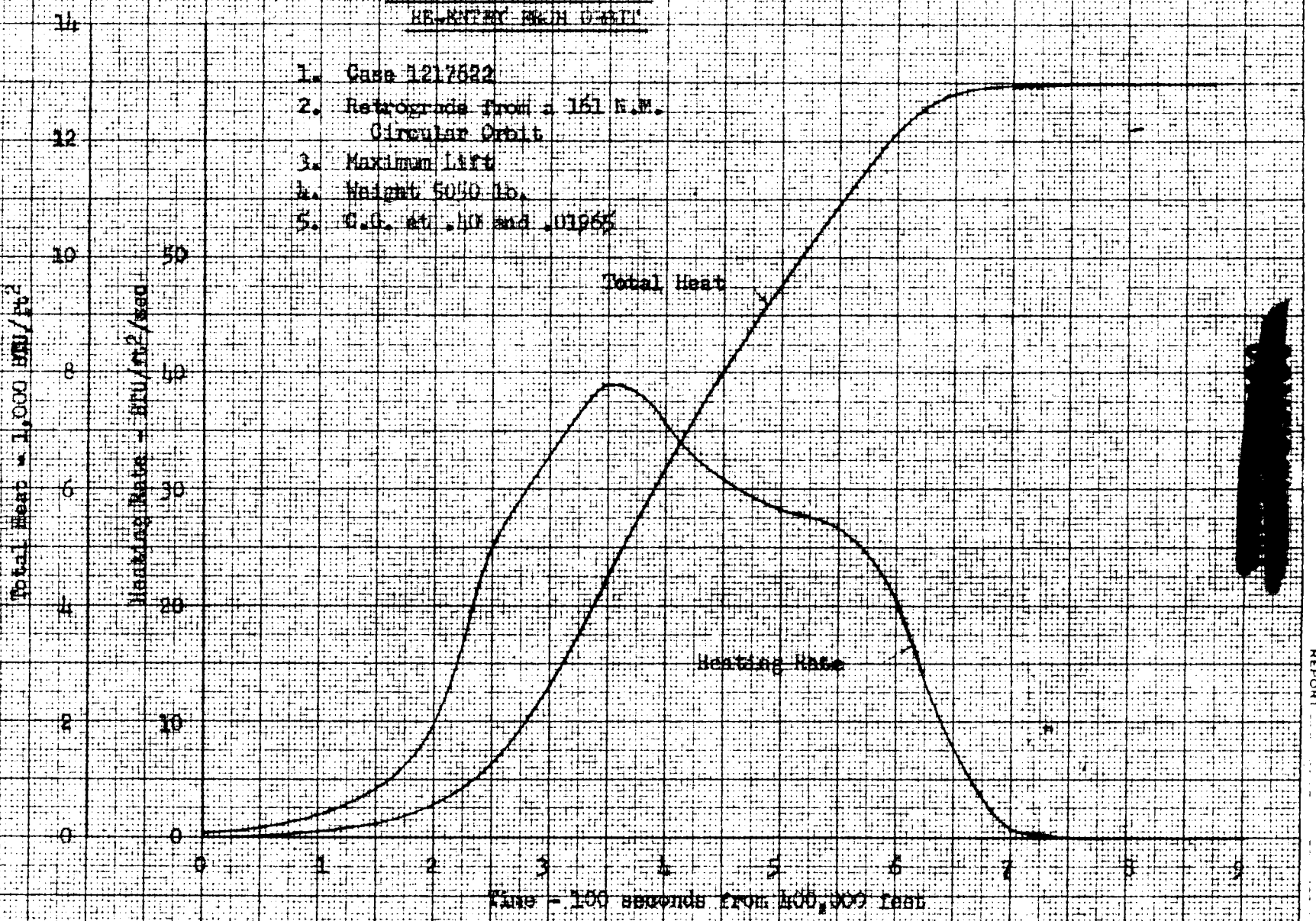


FIGURE 3.5.23
VARIATION OF MACH NUMBER, LOAD FACTOR, AND DYNAMIC PRESSURE DURING A RE-ENTRY FROM ORBIT

[redacted]

FIGURE 3.5.24
VARIATION OF TOTAL HEAT AND
HEATING RATE DURING A
RE-ENTRY FROM ORBIT



[redacted]

[redacted]

[redacted]

1. Case 1046328
2. Retrograde from an 87-161 N.M. elliptical orbit at 60° after perigee
3. Maximum lift
4. Weight 5050 lbs.
5. C.G. at .40 and .01965

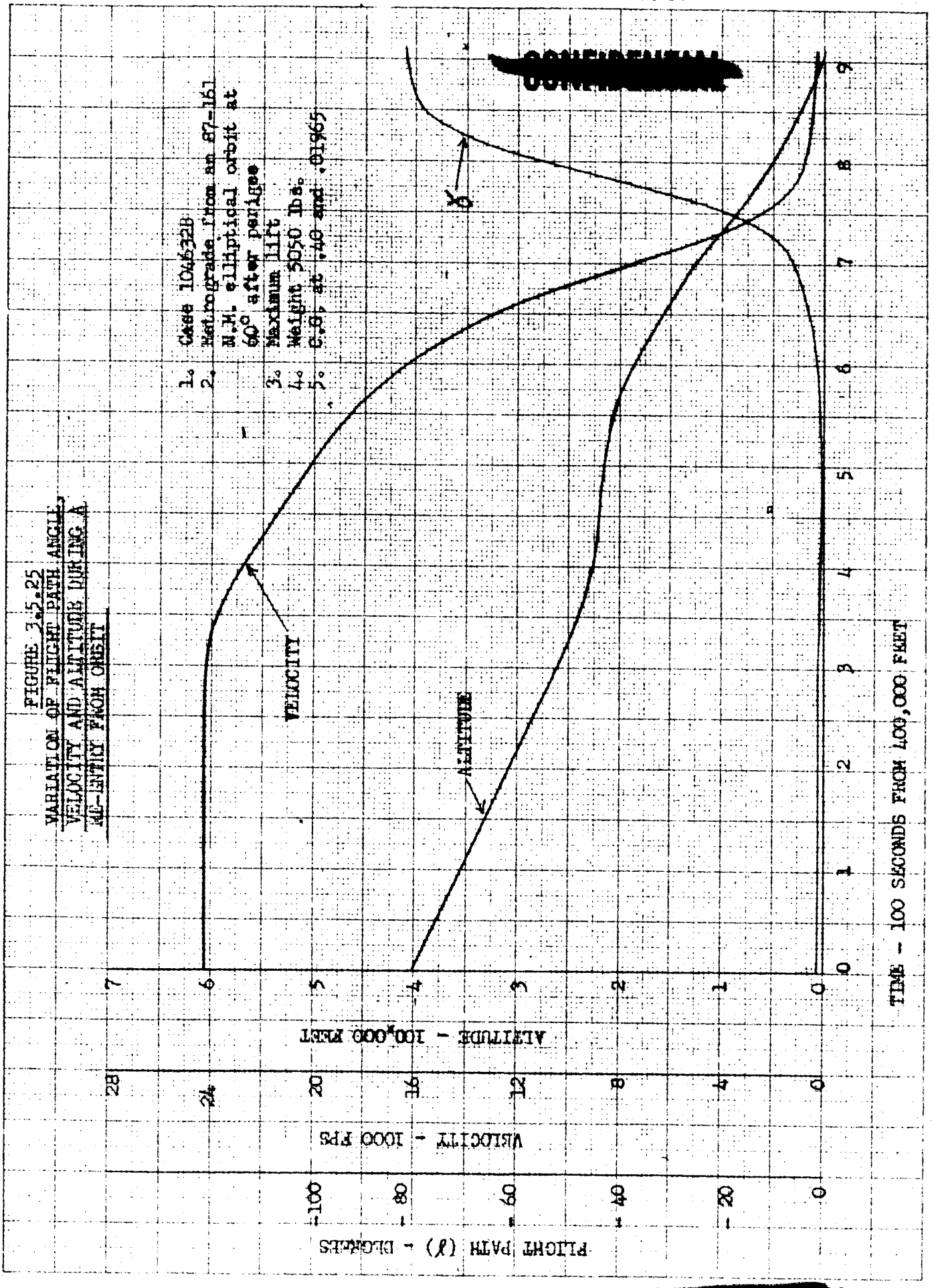


FIGURE 3.5.25
VARIATION OF FLIGHT PATH ANGLE, VELOCITY AND ALTITUDE DURING A RE-ENTRY FROM ORBIT

[redacted]

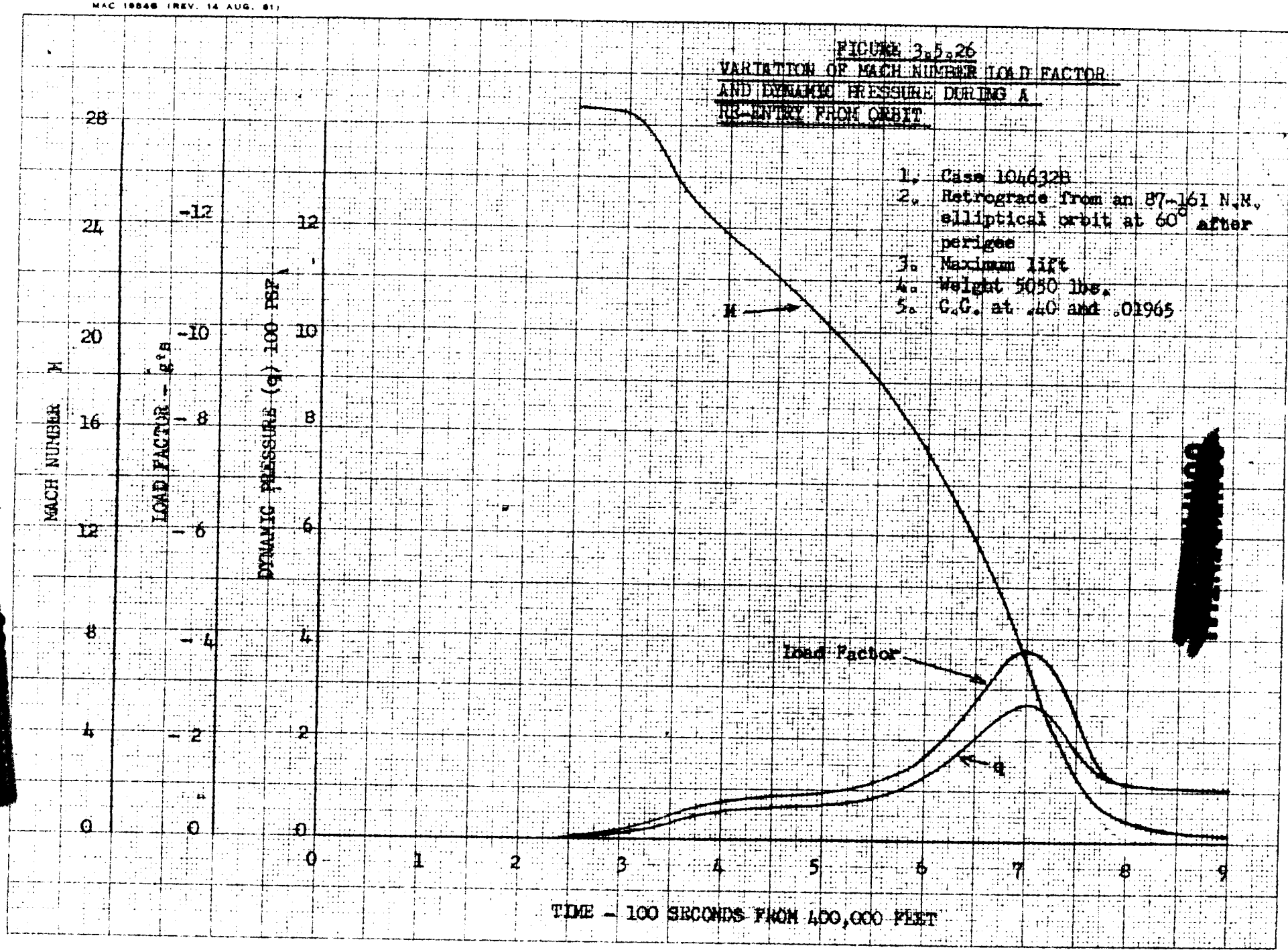


FIGURE 3.5.26
VARIATION OF MACH NUMBER LOAD FACTOR AND DYNAMIC PRESSURE DURING A RE-ENTRY FROM ORBIT

[redacted]

[redacted]

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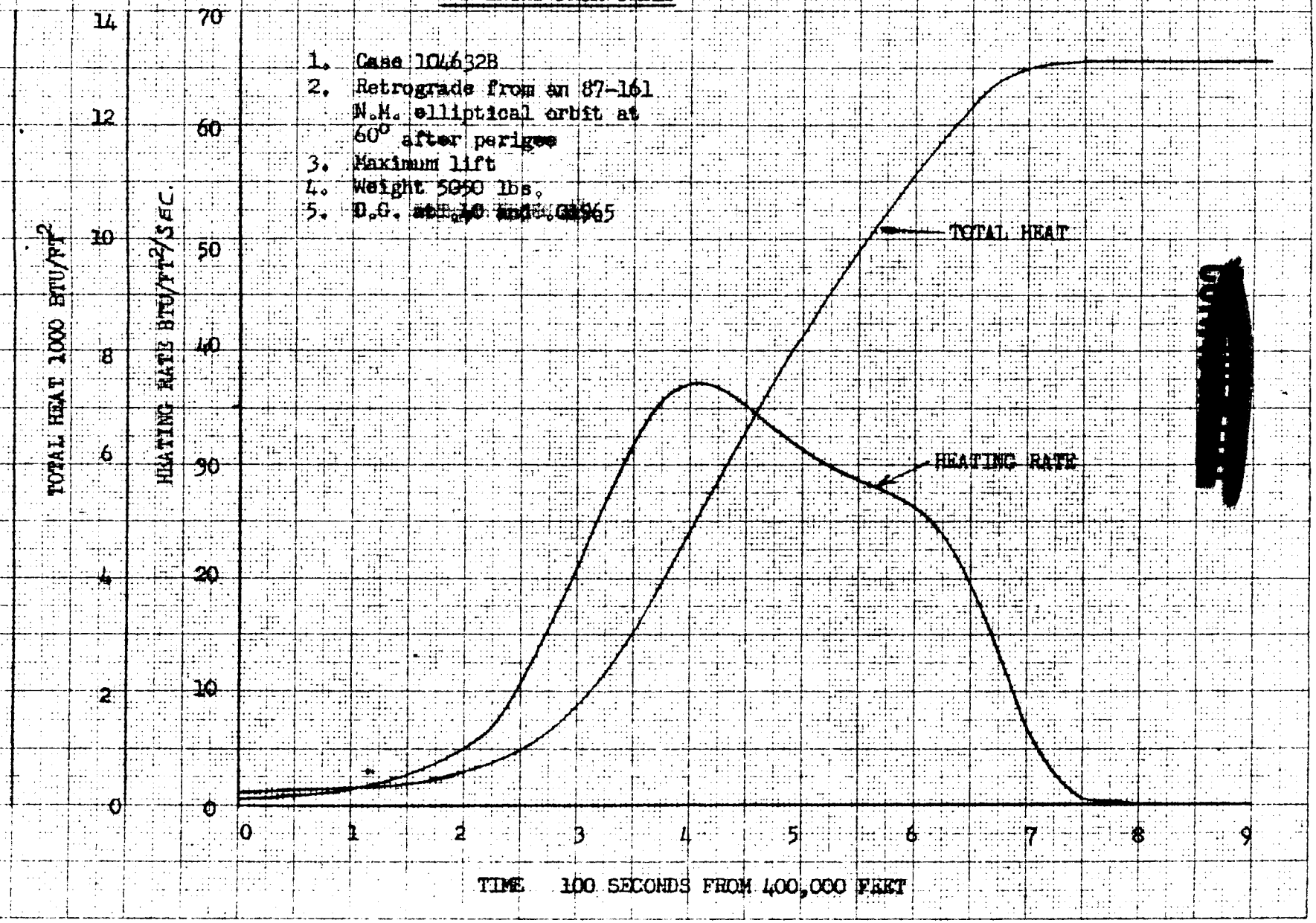
FIGURE 3.5.27
VIBRATION OF TOTAL HEAT AND
HEATING RATE DURING A
RE-ENTRY FROM ORBIT



[redacted]

104636B

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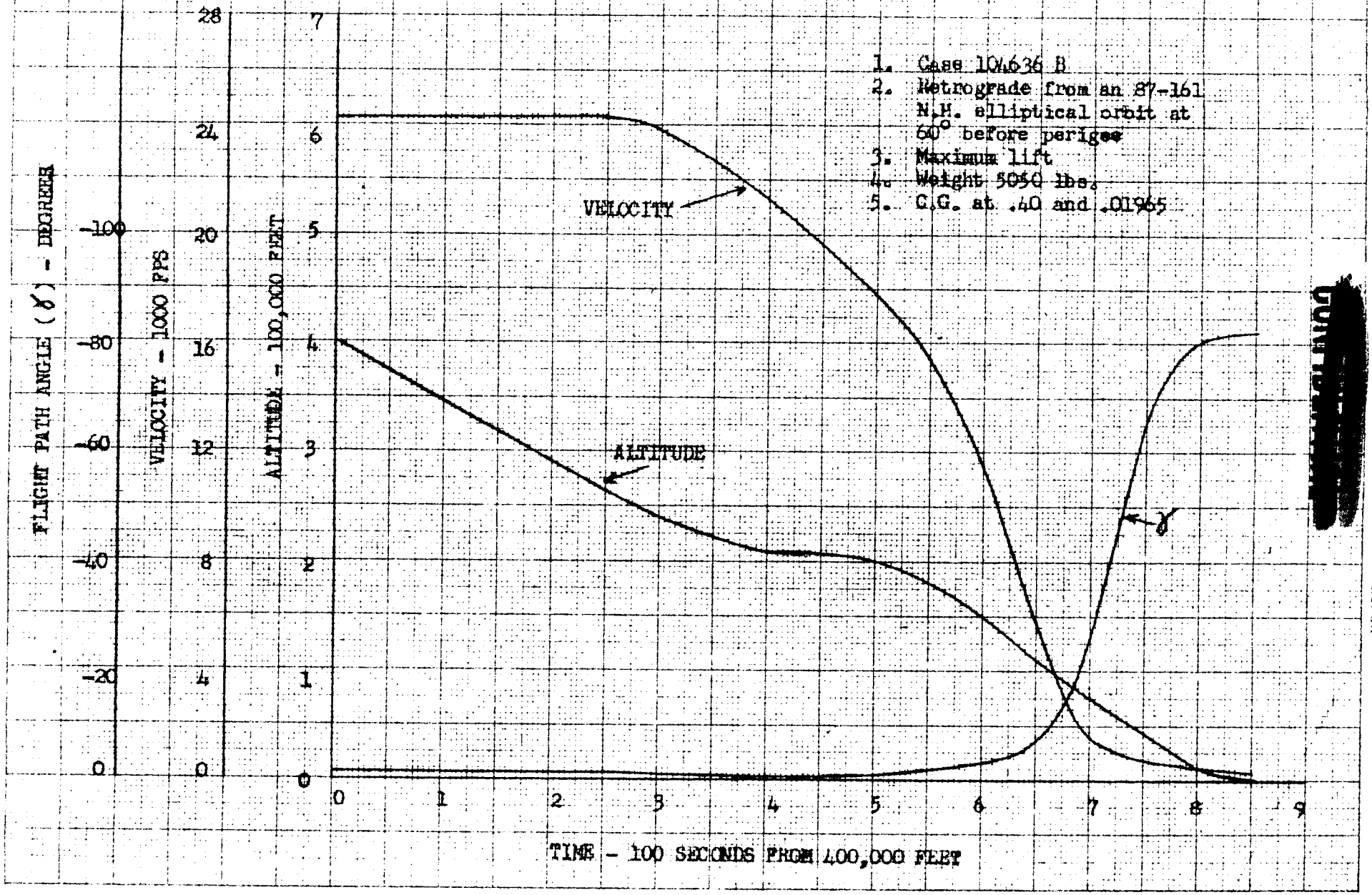
[redacted]

FIGURE 3.5.28
VARIATION OF FLIGHT PATH ANGLE VELOCITY AND ALTITUDE DURING A RE-ENTRY FROM ORBIT



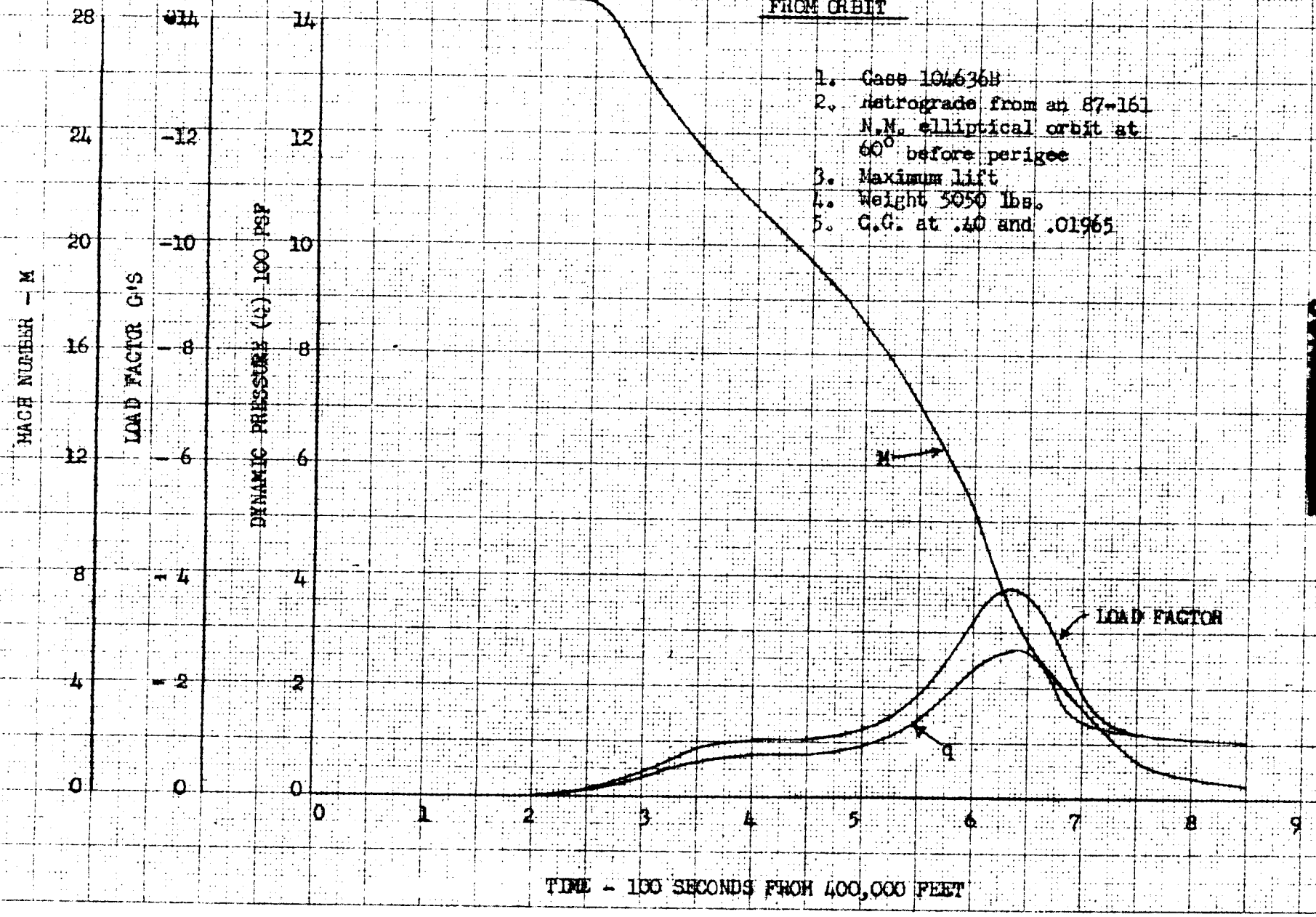
[redacted]

MAC 18646 (REV. 14 AUG. 61)

[redacted]

FIGURE 3.5.29
VARIATION OF MACH NUMBER, LOAD FACTOR, AND DYNAMIC PRESSURE DURING A RE-ENTRY FROM ORBIT

1. Case 104636B
2. Retrograde from an 87-161 N.M. elliptical orbit at 60° before perigee
3. Maximum lift
4. Weight 5050 lbs.
5. C.G. at .40 and .01965



[redacted]

104636B

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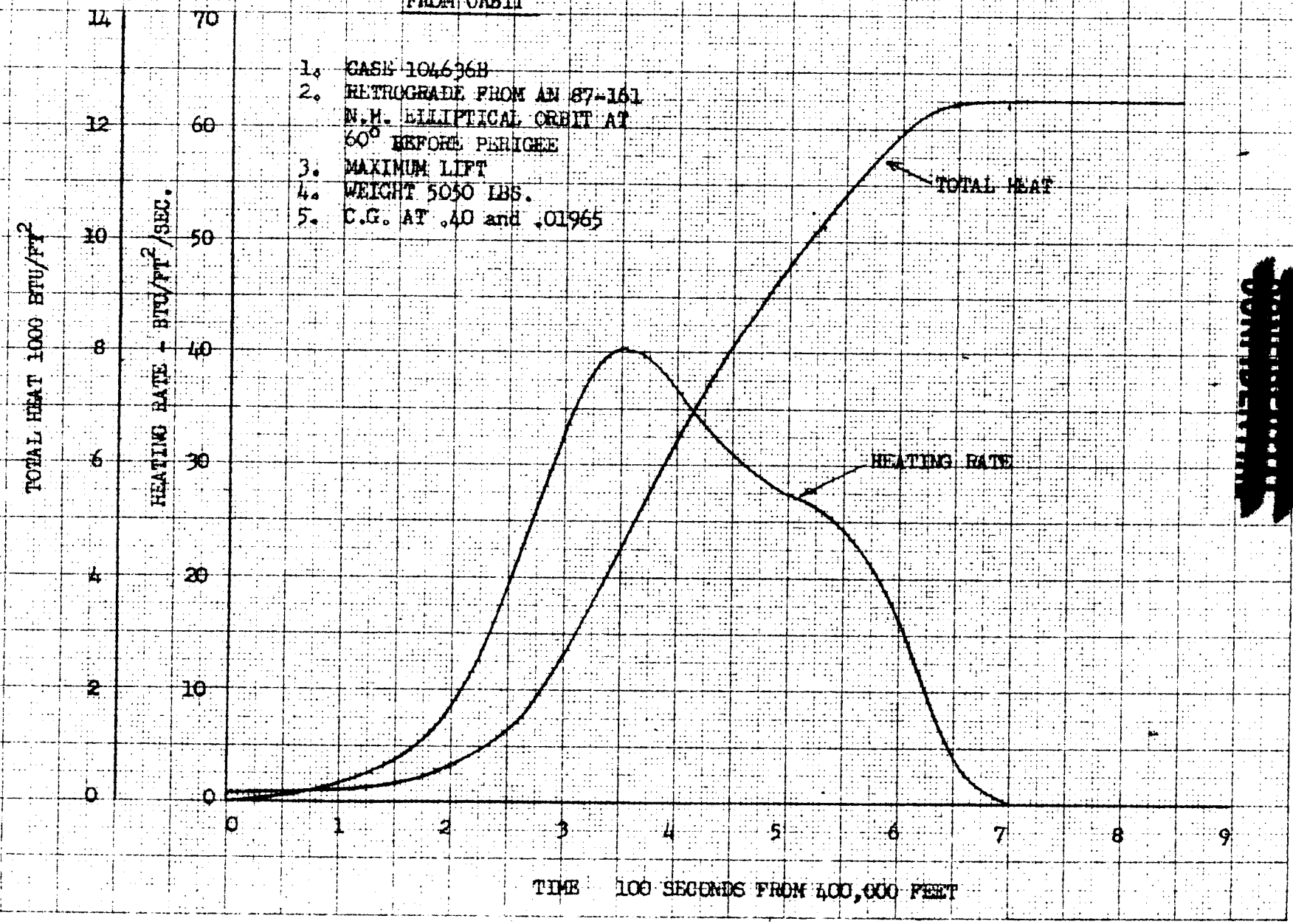
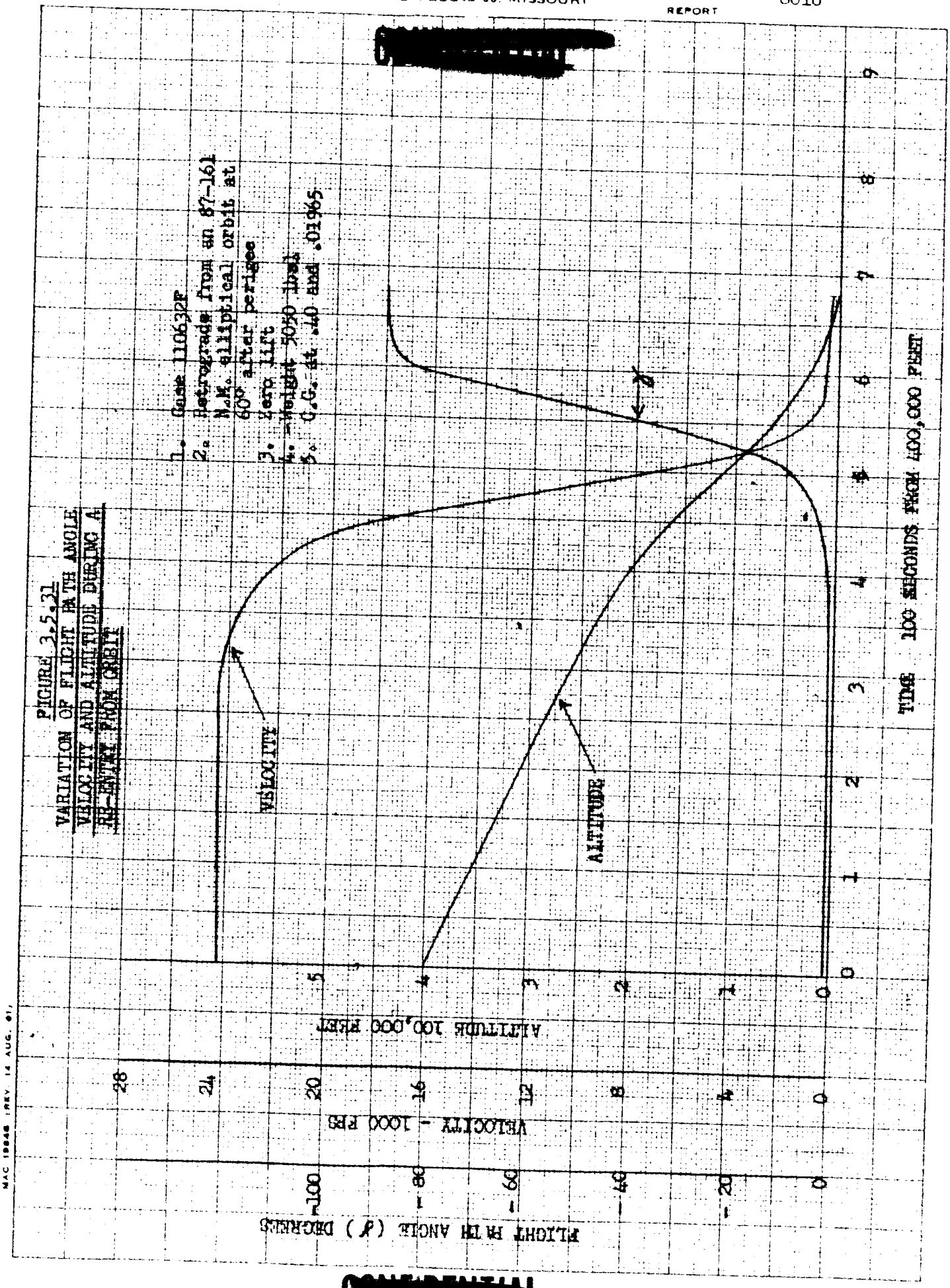
[redacted]

FIGURE 3.5.30
VARIATION OF TOTAL HEAT AND HEATING RATE DURING A RE-ENTRY FROM ORBIT



[redacted]

[redacted]



PICTURE 3.5.31
VARIATION OF FLIGHT PATH ANGLE
VELOCITY AND ALTITUDE DURING A
RE-ENTRY FROM ORBIT

110632F

[redacted]

MAC 19848 (REV. 14 AUG. 61)

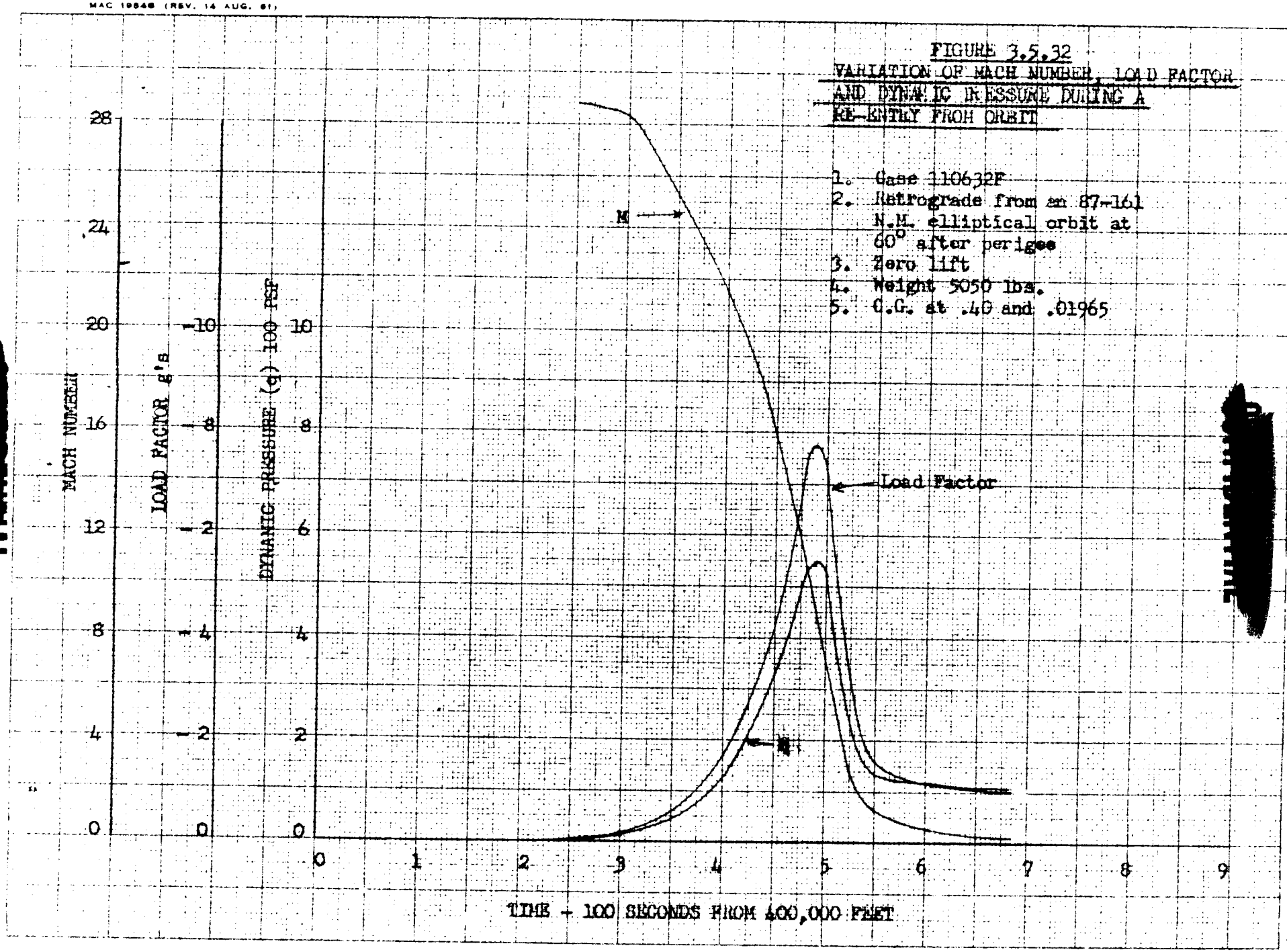


FIGURE 3.5.32
VARIATION OF MACH NUMBER, LOAD FACTOR AND DYNAMIC PRESSURE DURING A RE-ENTRY FROM ORBIT

1. Case 110632F
2. Retrograde from an 87-161 N.M. elliptical orbit at 60° after perigee
3. Zero lift
4. Weight 5050 lbs.
5. C.G. at .40 and .01965

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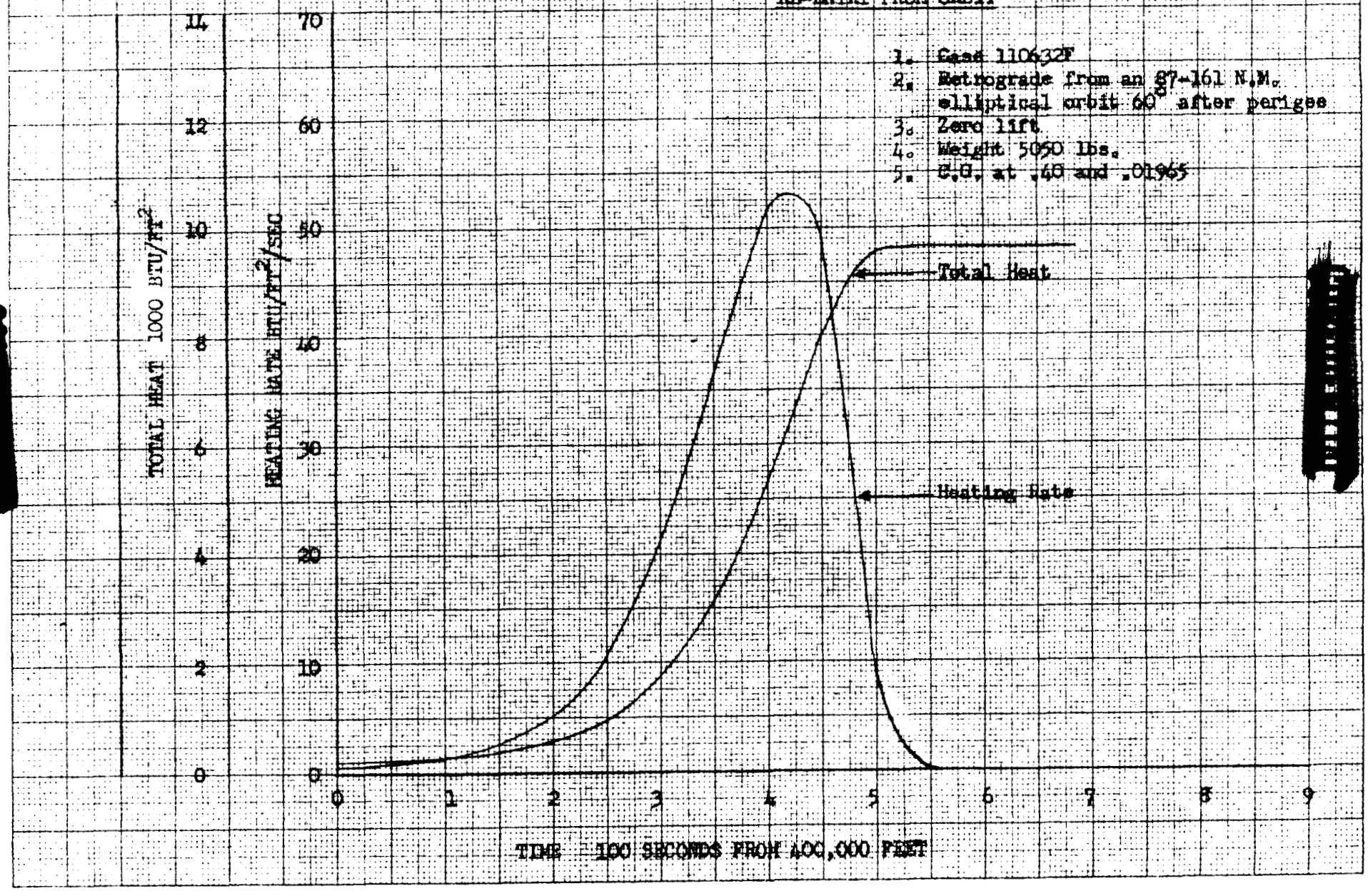
[redacted]

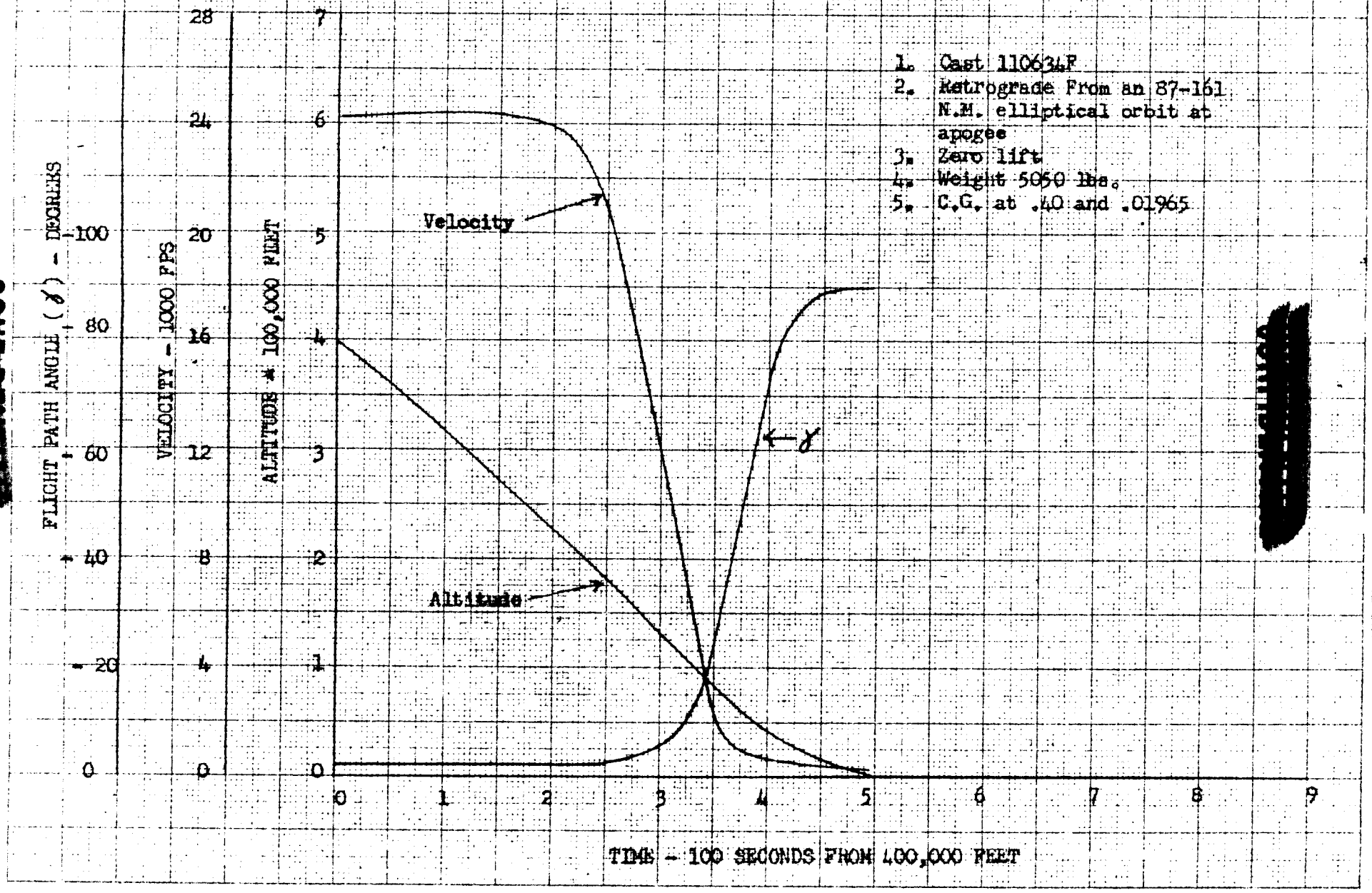
FIGURE 3.5.33
VARIATION OF TOTAL HEAT AND HEATING RATE DURING A RE-ENTRY FROM ORBIT



[redacted]

MAC 10646 (REV. 14 AUG. 61)

FIGURE 3.5.34
VARIATION OF FLIGHT PATH ANGLE VELOCITY, AND ALTITUDE DURING A RE-ENTRY FROM ABORT



[redacted]

[redacted]

110634F

MAC 19848 (REV. 14 AUG. 61)

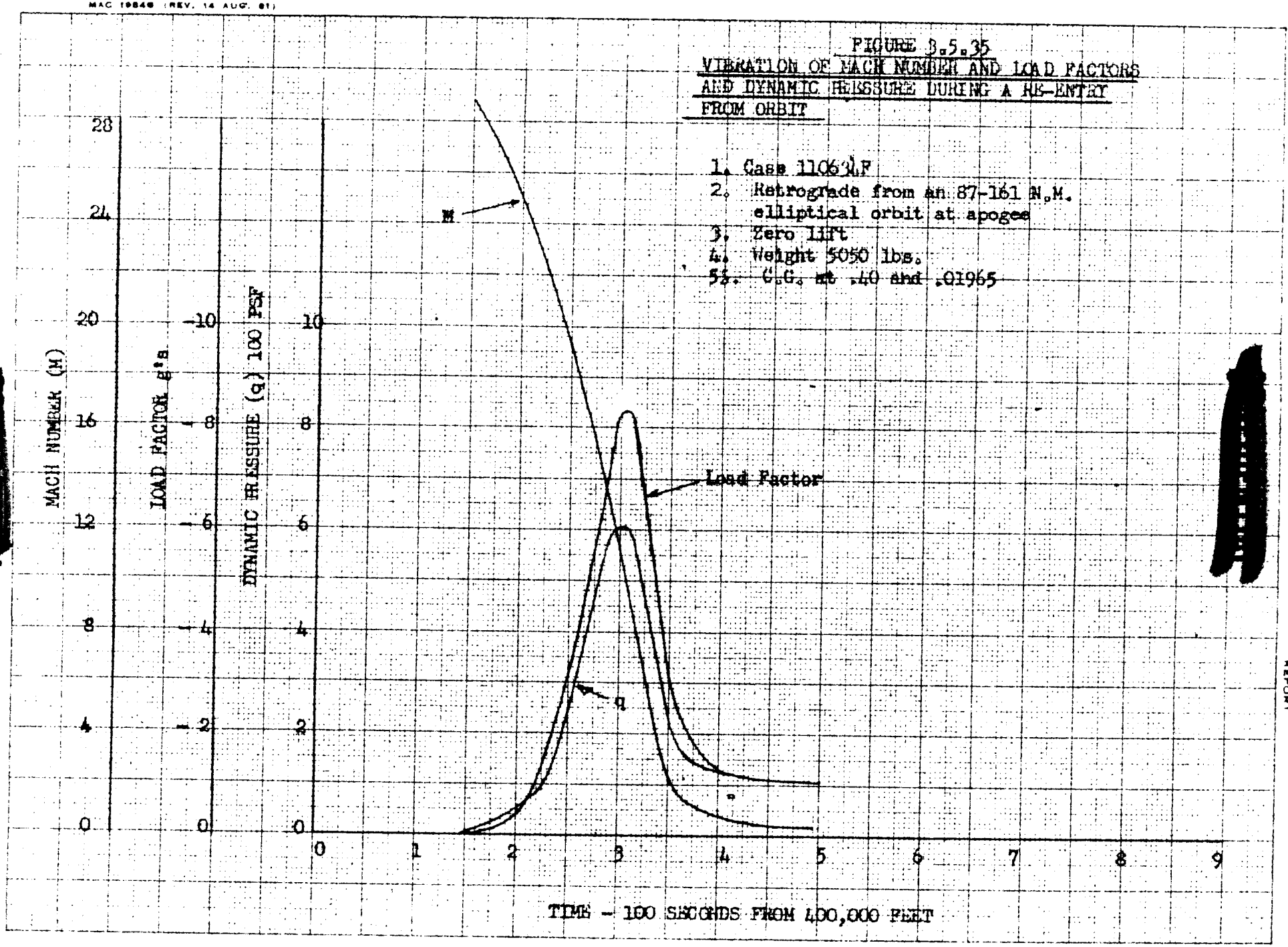


FIGURE 3.5.35
VIBRATION OF MACH NUMBER AND LOAD FACTORS AND DYNAMIC PRESSURE DURING A RE-ENTRY FROM ORBIT

1. Case 110634F
2. Retrograde from an 87-161 N.M. elliptical orbit at apogee
3. Zero Lift
4. Weight 5050 lbs.
5. C.G. at .40 and .01965

[redacted]

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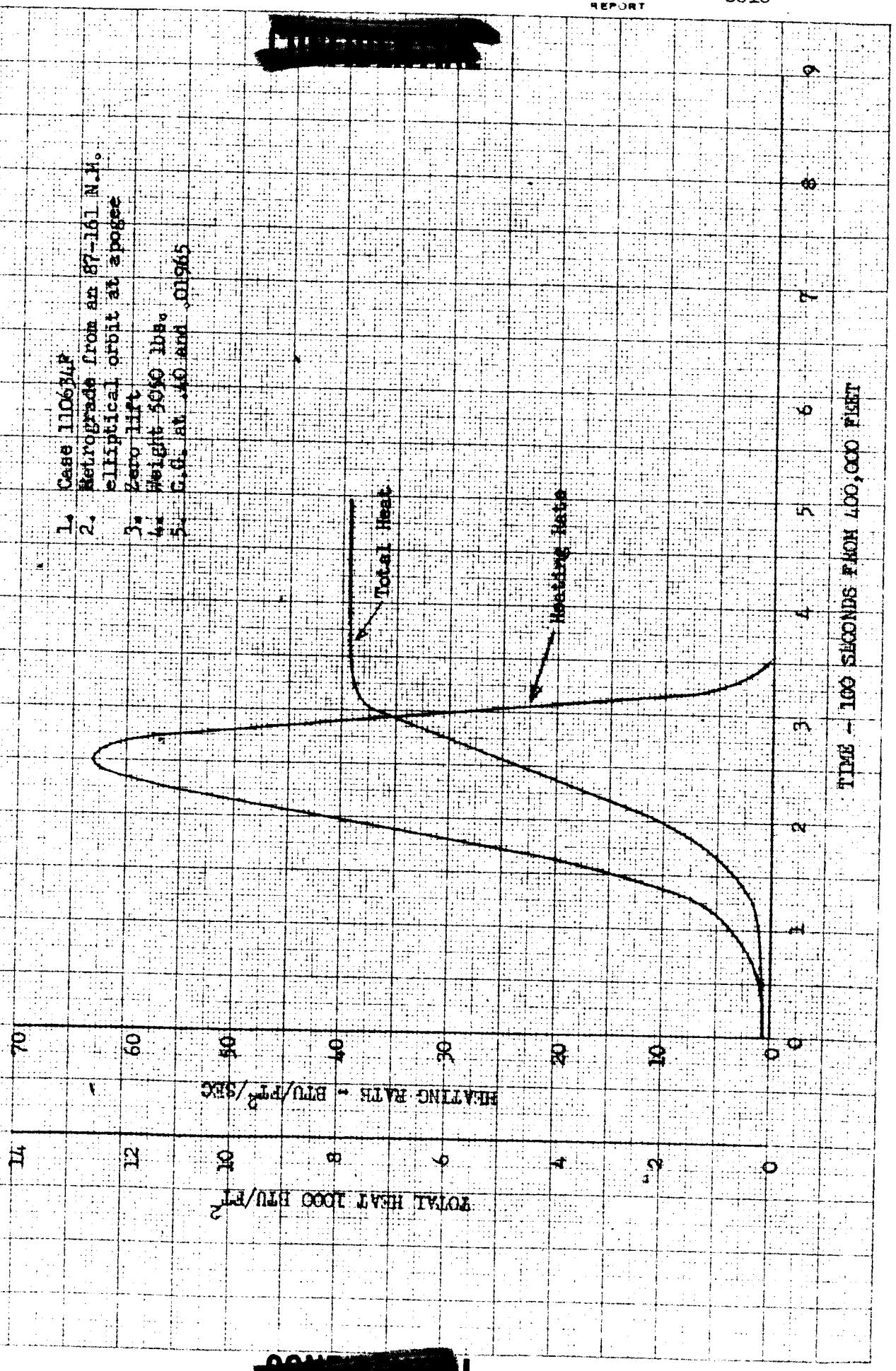
[redacted]

FIGURE 3.5.36
VARIATION OF TOTAL HEAT AND HEATING RATE DURING A RE-ENTRY FROM ORBIT

1. Case 110634F
2. Retrograde from an 87-161 N.M. elliptical orbit at apogee
3. Zero lift
4. Weight 5050 lbs.
5. C.G. at .40 and .01965



110634F

[redacted]

[redacted]

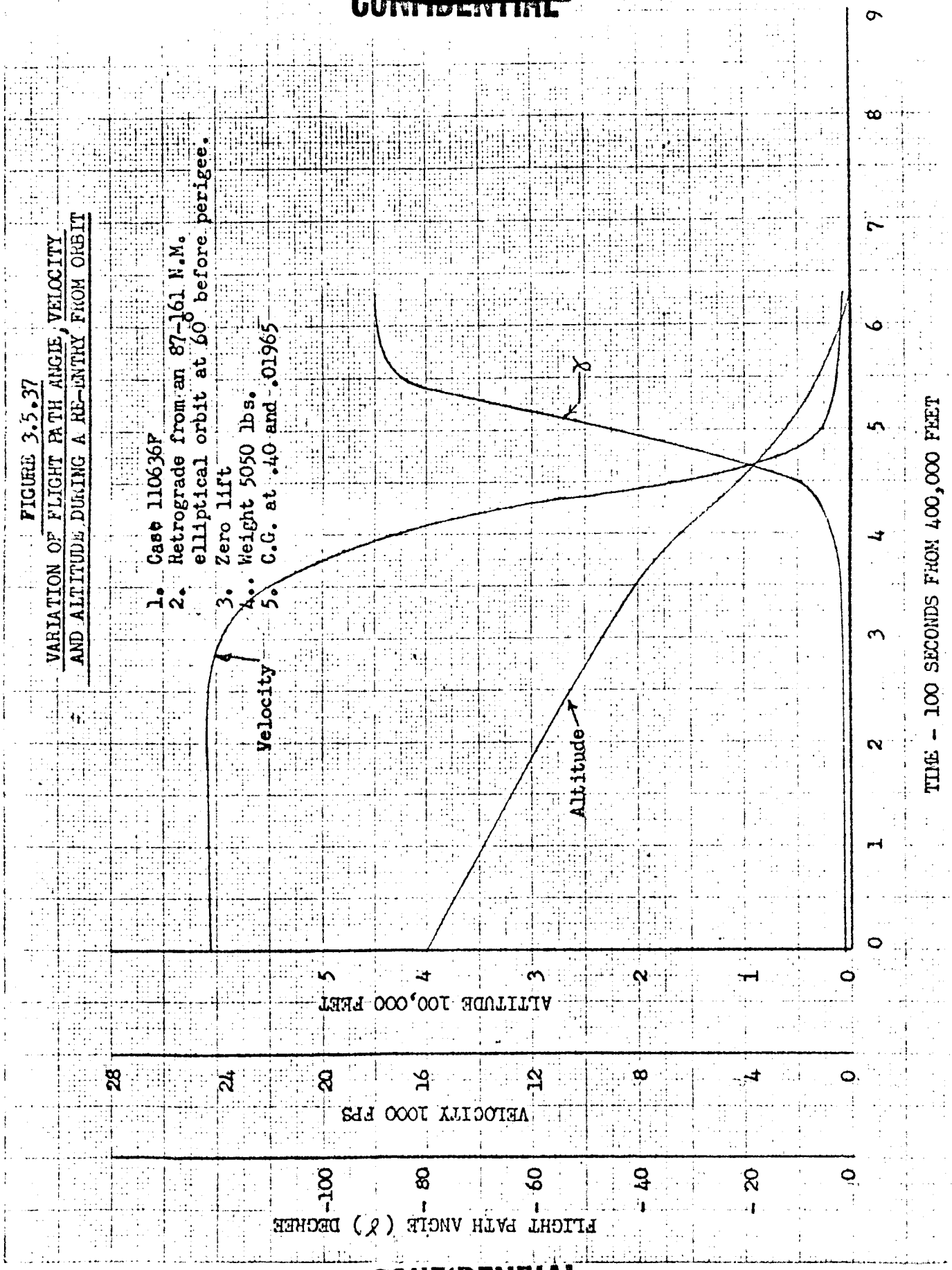


FIGURE 3.5.37
VARIATION OF FLIGHT PATH ANGLE, VELOCITY AND ALTITUDE DURING A RE-ENTRY FROM ORBIT

[redacted]

MAC 19846 (REV. 14 AUG. 61)

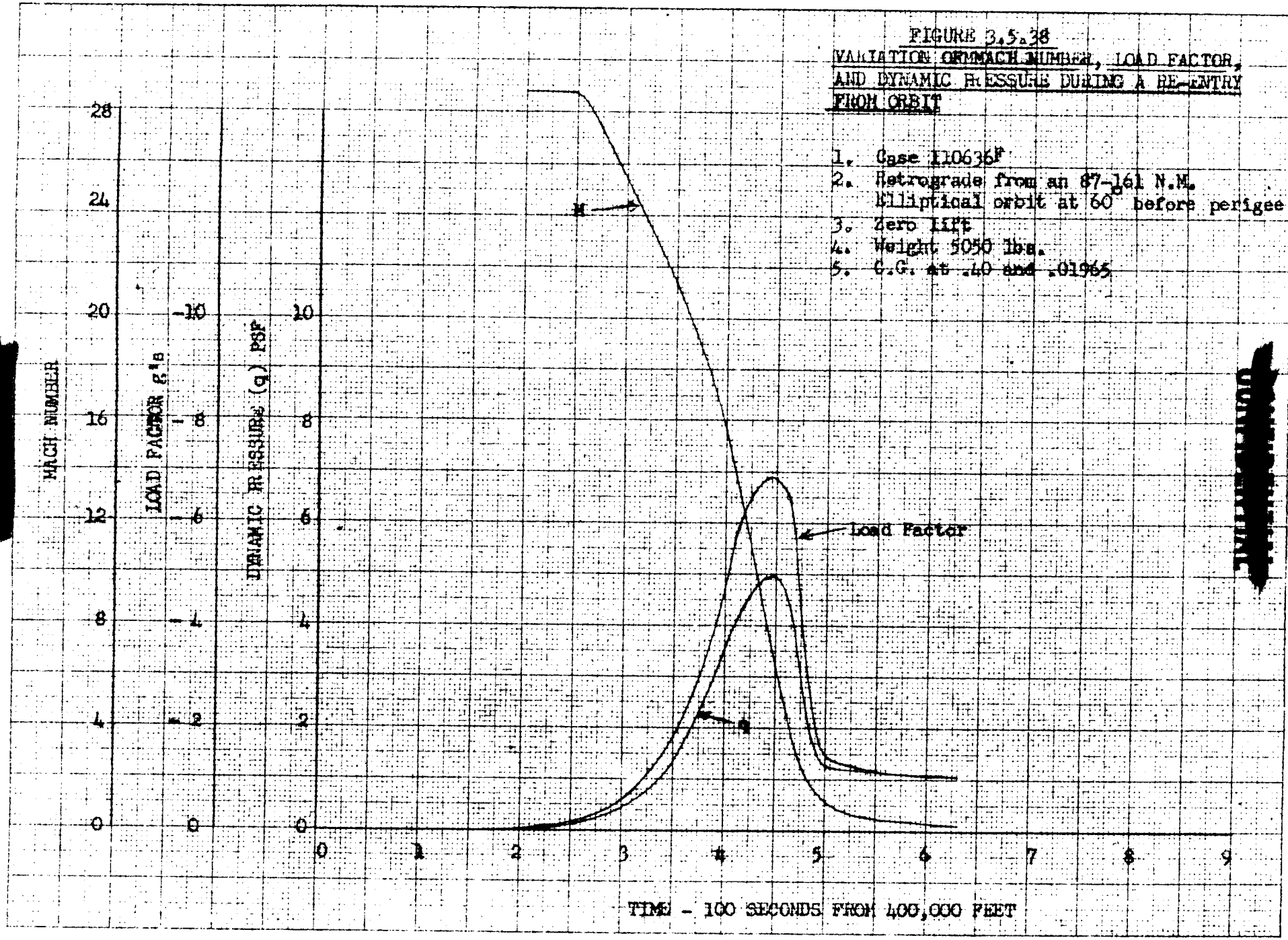


FIGURE 3.5.38
VARIATION OF MACH NUMBER, LOAD FACTOR, AND DYNAMIC PRESSURE DURING A RE-ENTRY FROM ORBIT

1. Case 110636P
2. Retrograde from an 87-161 N.M. Elliptical orbit at 60° before perigee
3. Zero Lift
4. Weight 5050 lbs.
5. C.G. at .40 and .01965

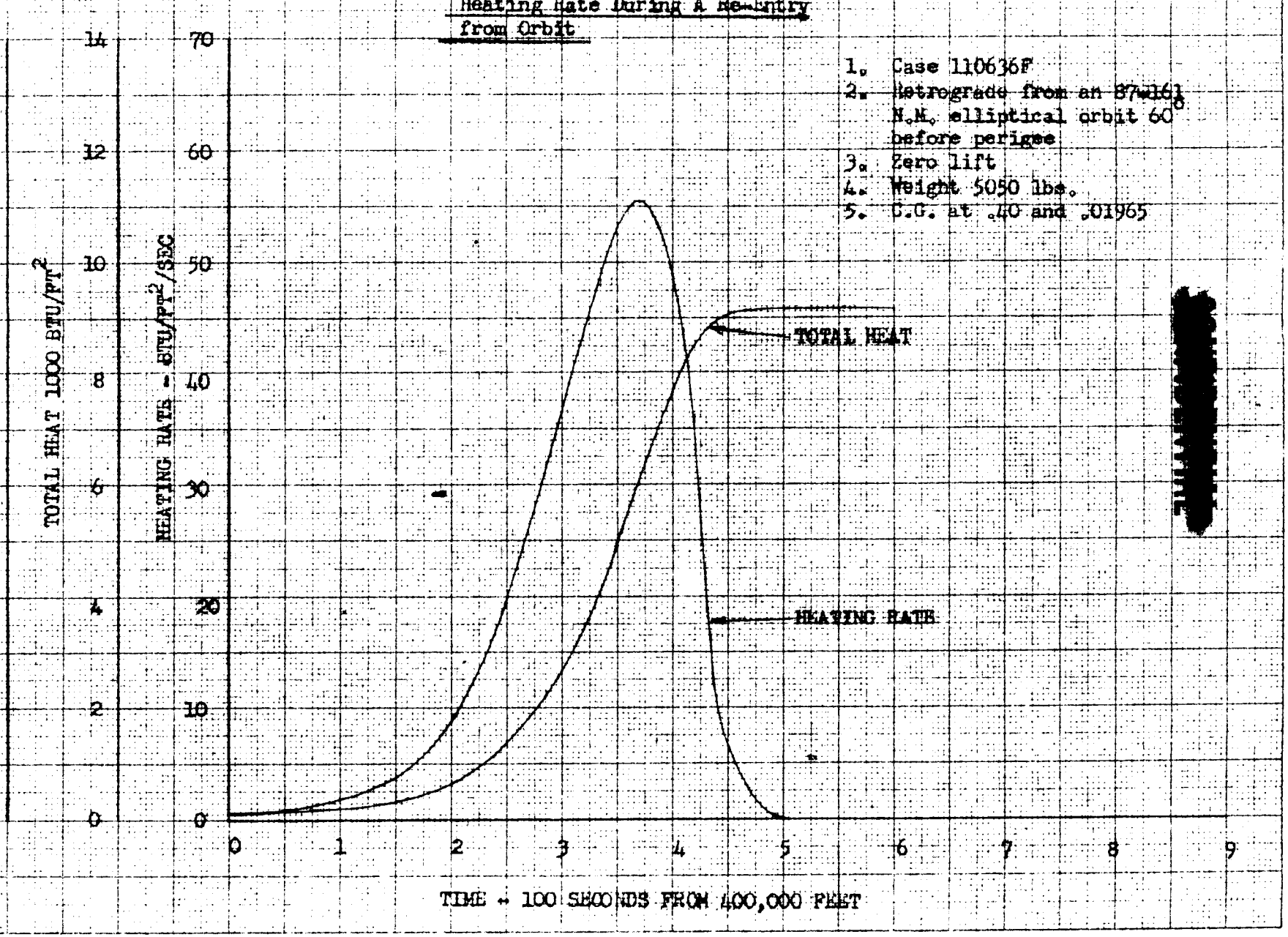
[redacted]

[redacted]

MAC 19848 (REV. 14 AUG. 61)

FIGURE 3.5.39

Variation of Total Heat and Heating Rate During A Re-Entry from Orbit



[redacted]

3.6 Landing Phase

The landing phase is defined as including all operations starting from the initiation of the recovery system deployment until the re-entry module is safely on the ground or water. It covers drogue parachute deployment, pilot parachute deployment, main parachute or paraglider deployment, steady state or maneuvering descent, and surface impact considerations. All mass items shall be designed for the shock and acceleration requirements of Section 3.12.

3.6.1 Parachute Landing Phase

Parachute landing system criteria are described herein for both the two parachute system being used on the early spacecraft and the three parachute system being developed for later spacecraft.

The basic two parachute system consists of a ring-sail pilot parachute 18 feet in diameter and a ring-sail main parachute 84 feet in diameter. The three parachute system consists of a conical type drogue parachute 8.3 feet in diameter in addition to the identical ring-sail pilot and main parachutes used in the basic system. The design loads for each parachute in the sequence are based on deployment at the spacecraft terminal free-fall dynamic pressure of 120 psf. This condition is consistent with normal fully controlled re-entries both from orbit and from the abort boundaries. For a normal fully controlled re-entry, the spacecraft re-enters heat shield first with pitch and yaw rate damping operative. Designing all parachutes for this dynamic pressure insures that in case of failure or malfunction of the drogue or pilot chutes, the remaining system can effect a safe recovery. The criteria for both systems are summarized in Table 3.6.1.

Table 3.6.1

Parachute Landing System

Configuration	Parachute	Type	Dia. Feet	Re-Entry	Deploy Altitude Feet	Reefed	Limit (2) Load (3) Lbs. (4)	Pull Off Angle Max.
Two Chute System	Pilot	Ring Sail	18	All	10,600	Yes	3,000	45°
	Main	Ring Sail	84	All	(1)	Yes	16,000	90°
Three Chute System	Drogue	Con-ical	8.3	Normal Abort	50,000 40,000	Yes Yes	3,500 3,500	90° 90°
	Pilot	Ring Sail	18	All	10,600	Yes	4,700 (5)	20°
	Main	Ring Sail	84	All	(1)	Yes	16,000	90°

Notes:

(1) Two seconds after pilot parachute deployment

(2) Nominal limit loads are based on a dynamic pressure (q) of 120 psf.

(3) Ultimate loads are 1.36 times limit loads

(4) Parachutes shall be qualified at a dynamic pressure 1.50 times that used for design ($q_{qual.} = 1.50 \times q_{design} = 180$ psf).

(5) The increased load on the structure for this condition over the two-chute system is due to the fact that the drogue chute is still attached when the pilot chute is deployed. The design limit load for the pilot parachute is 3,000 pounds.

In the basic two parachute system, the pilot parachute is deployed in a reefed condition at an altitude of 10,600 ± 750 feet. After a delay of approximately two seconds, the Rendezvous and Recovery Section is separated from the spacecraft. The reefed pilot chute pulling the R & R Section away stretches the main parachute lines deploying the main parachute in a reefed condition. After a short delay, the main parachute is then disreefed. The pilot chute is disreefed 6 seconds after deployment to reduce the probability of recontact of the R & R Section and the main parachute canopy and/or the spacecraft.

In the three parachute system, the drogue parachute has been added to ensure spacecraft stability below an altitude of 50,000 feet. It is deployed in a reefed condition at an altitude of 50,000 feet after re-entry from orbit and at 40,000 feet after re-entry from launch aborts. At an altitude of 10,600 ± 750 feet, the pilot parachute is deployed in a reefed condition with the drogue chute attached in tandem. After a delay of approximately two seconds, the R & R Section is separated from the spacecraft and the remaining portion of the sequence is identical to that for the two parachute system. The probability of recontact is further reduced because of the added drag of the drogue chute in tandem with the pilot chute. For the case where rate damping has been lost or degraded, the dynamic pressure at the drogue parachute deployment altitude can reach 145 psf. The increased loads for this condition will be absorbed with the reduced factor of safety noted in Section 3.1.

Spacecraft employing parachutes are designed for water landings only. To minimize water impact loads, the main parachute suspension bridles have a provision for rotating the spacecraft to a position where the spacecraft Z axis is inclined at an angle of 55° relative to the parachute axis. Water impact loads shall be those resulting from a vertical velocity of 30 fps combined with a horizontal velocity from winds up to 51 fps plus the effect of parachute swing. The effect of parachute swing shall be considered either as a horizontal velocity increment of

14 fps with zero swing angle or a zero horizontal velocity increment with 15 degree swing angle. A maximum wave slope of 9 degrees shall be considered. The strength requirement for water landing capability is limited to the capability to remain afloat for at least 36 hours per Reference (5) in a flotation attitude in which the hatches are on the upper surface.

3.6.2 Paraglider Landing Phase

The paraglider landing system consists of a 8.3 foot diameter conical type drogue parachute, the paraglider, and the skid type landing gear.

The drogue parachute is deployed at an altitude of 60,000 feet following re-entry from orbit and at an altitude of 35,000 feet following a retrograde abort. The design dynamic pressure shall be 125 psf. The drogue parachute is attached to the Rendezvous and Recovery Section and is jettisoned with this section.

Paraglider deployment begins at an altitude of 50,000 feet following re-entry from orbit. The paraglider assembly is uncovered by the jettisoning of the Rendezvous and Recovery Section. It then goes through a sequence of partial deployment, inflation of the stiffening members, and finally release to the glide configuration. Loads during the deployment sequence shall be investigated and strength provided for all critical cases.

In the glide configuration, the spacecraft shall be designed to maneuver using the available rate and amount of center of gravity travel in both the longitudinal and lateral directions. The nominal range of paraglider angle of attack shall be 25 to 40 degrees with provision for 20 to 45 degrees.

The spacecraft landing gears shall be designed for the landing parameters defined below using the weight defined in subsection 2.3.7.2. Wing lift shall be considered in a rational manner. For landing gear loading conditions, the factor

3.6.2 Paraglider Landing Phase (Continued)

of safety shall be applied to the total energy to be absorbed in the vertical direction. The total energy shall include the kinetic energy due to sink speed and the potential energy based on gear compression allowing for wing lift. The loads computed on this basis will be ultimate design loads for the landing conditions.

Landing Gear Design Parameter	Maximum	Minimum
Sink speed (limit)	10.0 fps	0
Horizontal airspeed	100 fps	50 fps
Pitch attitude at contact (angle measured between the surface and the spacecraft Z axis)	0° (nose up)	-15° (nose down)
Yaw angle	$\pm 25^\circ$	0°
Roll attitude	$\pm 10^\circ$	0
Coefficient of friction (at impact)	0.50	0.20

The values quoted in the preceding table are extreme values for each of the individual parameters. These parameters shall be combined in a rational manner for the landing gear design conditions. Alowable combinations of yaw and roll attitudes at touchdown are shown in Figure 3.6.4.

Sink speeds for yawed and rolled attitudes shall be determined from the equation:

$$V_{unsym.} = V_{sym.} R_\psi R_\phi$$

where R_ψ and R_ϕ are the yaw and roll sink speed factors from Figure 3.6.5.

Lateral force components on the gear shall be determined by using the designation coefficients of friction except where applicable test data indicates that this is not valid: however, design features of the contact surface penetration, or other special tendencies shall be accounted for in a rational manner.

FIGURE 3.6.4

ALLOWABLE COMBINATIONS OF YAW AND ROLL

Spacecraft landing gears shall be designed for any combination of yaw and roll angles within this boundary.

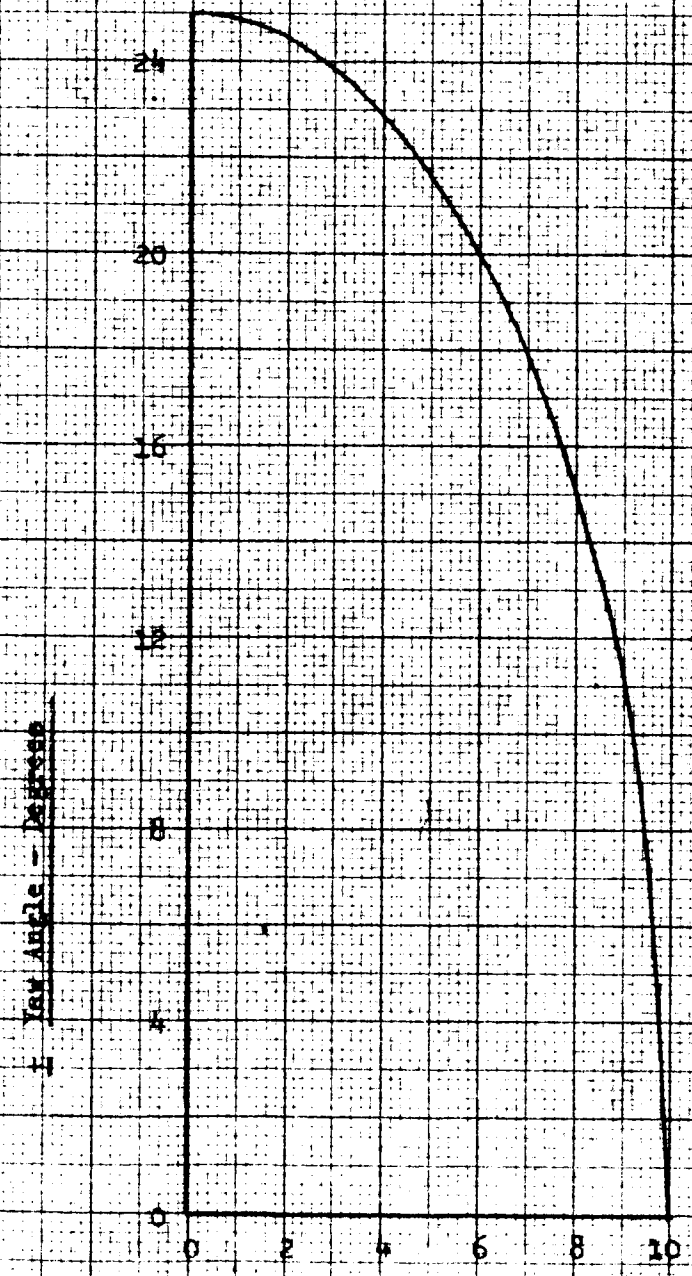
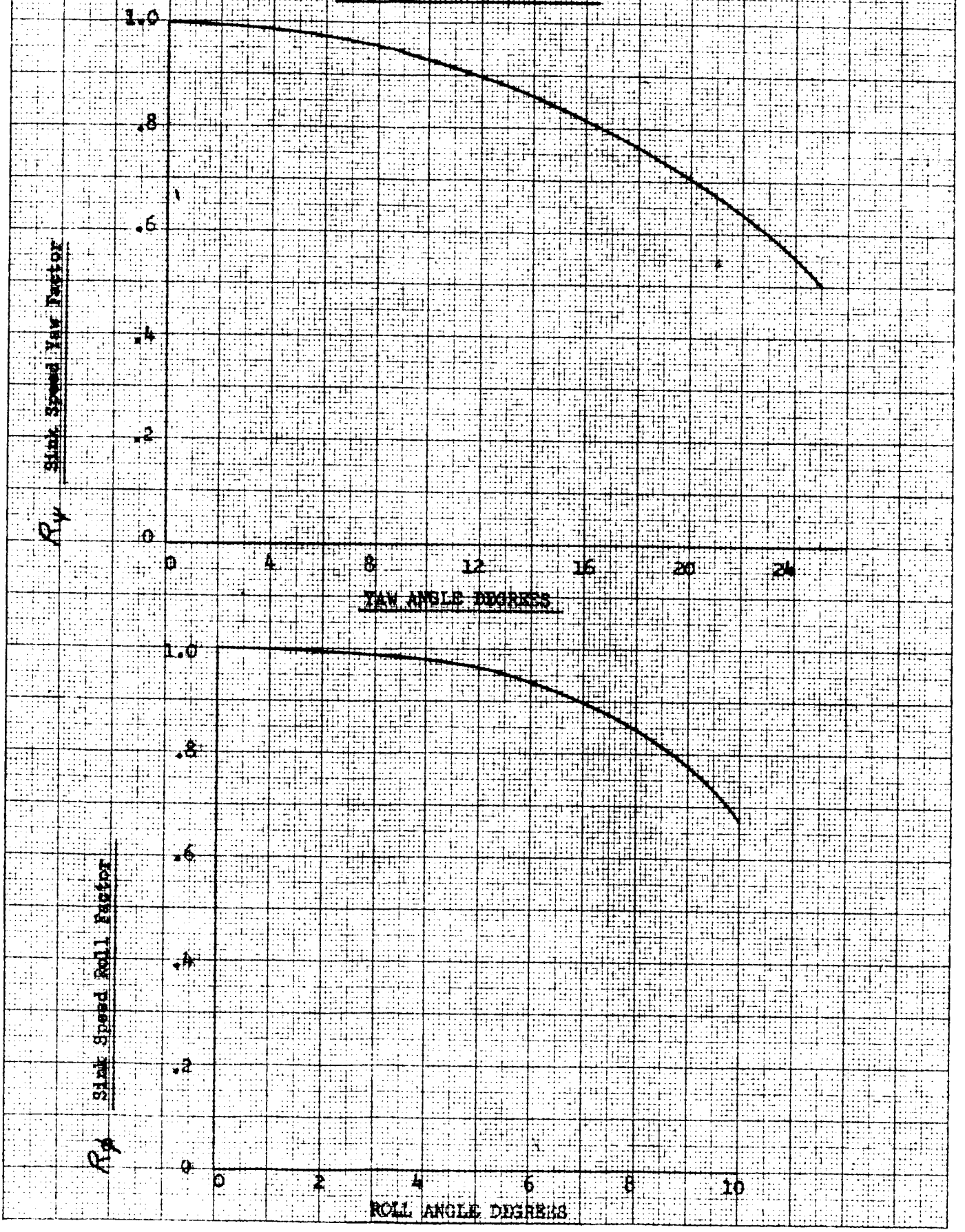


FIGURE 3.6.5
VARIATION OF SINK SPEED FACTORS
WITH YAW AND ROLL ANGLES



3.6.2 Paraglider Landing Phase (Continued)

Paraglider water landings shall be based on the same sink speeds, airspeeds, and attitudes as ground landings. Wave slopes up to 9° shall be considered. The strength requirement for water landing capability is limited to the capability to remain afloat for at least 36 hours per Reference (5) in a flotation attitude in which the hatches are on the upper surface.

3.7 Abort Phase

The abort phase is defined as including all operations required to return the astronauts safely to earth subsequent to a malfunction in the launch vehicle or the spacecraft which requires termination of the mission.

3.7.1 Mode I (Ejection) Abort Phase

Mode I aborts are accomplished with the use of the ejection seats. This mode of escape is used off-the-pad only after removal of the erector tower and during the boost phase up to an altitude fo 70,000 feet. The ejection seats may also be used for escape below an altitude of 60,000 feet following re-entry from orbit and below an altitude fo 35,000 feet following re-entry after Abort Mode II. The re-entry module surrounding the astronauts must maintain structural integrity with the hatches open until both astronauts are clear. The design loads for this phase shall consider that the time interval required to complete the ejection cycle (from the initial detection of the failure including allowances for both human and system reaction times, system operating time, and programmed delays) is short enough to preclude the vehicle reaching a structurally catastrophic condition. The hatch operating loads and the ejection seat loads for Mode I launch phase aborts shall be based on a total angle of attack of 15° in pitch and/or yaw. The overall vehicle loads for Mode I aborts shall not exceed the strength capabilities required by launch phase criteria defined in Section 3.2.

The Mode I abort sequence may be initiated by either astronaut pulling his "D-ring". The hatches are opened by pyrotechnically powered actuators. The seats are then propelled along guide rails by separate pyrotechnically powered catapults. Just prior to leaving the rails, sustainer rockets attached to each of the seats are fired propelling the seats clear of the spacecraft. The ejection seats shall be designed for all forces resulting from these operations.

3.7.1 Mode I (Ejection) Abort Phase (Continued)

Seat-man separation is programmed to occur 1.03 seconds after the seat leaves the rails. Five seconds after the seat leaves the rails, a ballute attached to each astronaut's backboard is deployed. The design load for the ballute shall be 3,750 pounds ultimate. The ballute is jettisoned at a pressure altitude of 7,500 ± 700 feet.

The barostat controlling the personnel parachute deployment is armed at seat-man separation and the parachute is deployed 2.3 seconds after the barostat senses a pressure altitude of 5700 ± 600 ft. The design load for the personnel parachute shall be 5000 pounds ultimate. The ratio of ultimate to limit load is defined in Section 3.1.

3.7.2 Mode II (Retrograde Salvo) Abort Phase

Mode II Aborts are accomplished by terminating booster thrust, severing the adapter structure at Z Station 68.44, and firing the retrograde rockets in salvo. This mode of abort is used at altitudes between 70,000 ft. and 522,000 feet. After burn-out of the retrograde rockets the retrograde section is jettisoned, the re-entry module is turned to its normal re-entry attitude, and following re-entry the landing system is deployed.

The spacecraft shall be designed for all loads occurring during and after separation from the launch vehicle. The time for completing the Mode II Abort cycle shall include allowances for both human and system reaction times, system operating times, and programmed delays. Abort re-entry trajectories are shown in Section 3.5.

3.7.3 Mode III (Separation) Abort Phase

The Mode III Abort Phase covers abort requirements during the remaining portion of the boost trajectory above 522,000 feet. Aborts during this period are accomplished by using the normal mission separation, re-entry and landing

3.7.3 Mode II(Separation)Abort Phase (Continued)

sequences. The Mode III Abort sequence is as follows: The launch vehicle thrust is terminated, the adapter structure is severed at the normal separation plane, the rendezvous maneuvering system is fired to provide separation velocity, the retrograde rockets may or may not be fired as required to attain the desired abort re-entry trajectory, and following re-entry the landing system is deployed.

The abort boundaries are shown on Figure 3.5.3.

3.8 Hoisting and Transportation

The hoisting limit load factor is 2.0 for the spacecraft during pre-launch operations and 3.0 for the capsule plus trapped water during recovery after a water landing. The vehicle as packaged for shipping shall be designed for the following ultimate accelerations applied to supporting fixtures separately.

Transportation by aircraft with the accelerations in the carrier aircraft axes:

6.0g Vertical (upward)

2.25g Lateral (±)

3.0g Longitudinal (aft)

The spacecraft Z axis shall be parallel to the aircraft longitudinal axis and the other spacecraft axes shall be oriented as determined by the design of the shipping fixtures.

3.9 Pressurization

For structural design the cabin pressure shall be considered to be 12.0 psi ultimate (burst) and 3.0 psi ultimate (collapsing).

The cabin leakage rate shall be measured at sea level using nitrogen at a temperature of 70°F and a cabin pressure of 5.1 psig. The allowable leakage rate shall be 61.023 cubic inches per minute (1000 cc/min.) for Spacecraft No. 1 and No. 2 and 30.511 cubic inches per minute (500 cc/min.) for Spacecraft No. 3 and up.

3.10 Controls

The design loads for control handles, levers, and knobs shall be as follows:

Primary Controls	Limit Load With Reaction At: Stops	Limit Load With Reaction At: Switches or Valves
Attitude Control Grip		
Pitch Moment	133 in. lb.	Sufficient to create 100 lbs. minimum at switches
Side (Roll)	100 lb.	
Twist (Yaw)	133 in. lb.	
(Pitch and yaw loads are referenced to Grip Pivot axis and Side Loads are referenced to center of grip.)		
Abort Handle		
Side	$50x\left(\frac{1 + \text{lever length}}{3}\right)$ lb.	Sufficient to create 100 lbs. minimum at switches
Fore/Aft	50 lb. min. to 150 lb. max.	
(Loads are referenced to center of knob.)		
Maneuvering Handle		
Vertical, Side and	$50x\left(\frac{1 + \text{lever length}}{3}\right)$ lb.	Sufficient to create 100 lbs. minimum at switches
Fore/Aft	50 lb. min. to 100 lb. max.	
(Loads are referenced to center of knob in unstowed position.)		
Environmental Controls		
Levers	$50x\left(\frac{1 + \text{lever length}}{3}\right)$ lb. 50 lb. min. to 100 lb. max.	3x pilot operating but not less than 70 lb. or less than that sufficient to create 100 lbs. minimum at valves.

Other Controls - General		
Levers	$50x\left(\frac{1 + \text{lever length}}{3}\right)$ lb. 50 lb. min. to 150 lb. max.	Not applicable
(Loads are referenced to center of grip or knob.)		
Push-Pull Handles	100 lbs.	Not applicable
(Loads are referenced to center of knob or ring.)		
Rotating Knobs	Not applicable	100 in. lbs.
(Load is not applicable to knobs operating electrical switches.)		

3.11 Seats and Harness

The landing load factors shown on Figure 3.11.2 shall apply to the harness used to restrain the astronauts, to the seats, and to the attachment of the seats to the primary structure. The magnitude of the design ultimate inertia load vector is consistent with an acceleration of 40 g's but is terminated as a function of direction such that the components do not exceed the values shown in Figure 3.11.2. The seats, seat equipment, and harness shall also be designed for ejection free flight conditions, stabilization device loads and personnel parachute loads as applicable.

3.12 Shock and Acceleration Environment

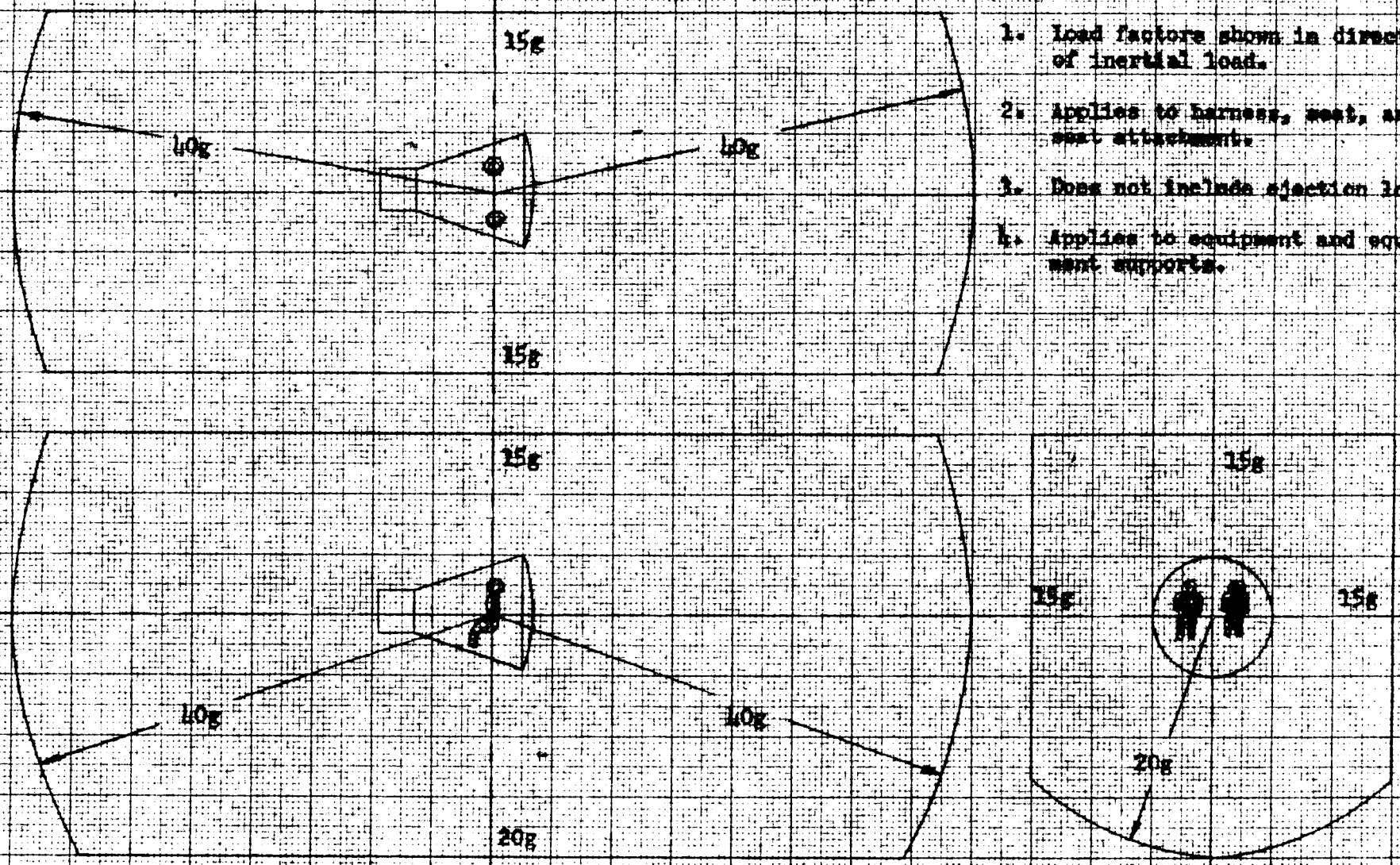
The shock and acceleration environments for design of the spacecraft equipment and support structure as defined in Reference (6) and (7) are summarized in Table 3.12. For the special case of parachute water landing with RCS fuel tanks full, the tank supports shall be designed for the actual accelerations resulting from water impact. The limit values to be used, as estimated for the spacecraft c.g. from model test data, are $n_z = 12.0$, $n_y = 1.6$, $\ddot{\theta} = 110$ radians/second^2 for heat shield first landings, and $n_z = -4.6$, $n_y = 12.0$, $\ddot{\theta} = 110$ radians/second^2 for cone first landings.

3.13 Vibration and Acoustic Environment

The vibration and acoustic environments to be used for the design of the spacecraft equipment and support structure as described in References (6) and (7) are defined in Table 3.13 and in Figures 3.13.2 and 3.13.4.

FIGURE 3.11.2

ULTIMATE LOAD FACTORS FOR DESIGN OF SEATS, HARNESSES, AND EQUIPMENT IN THE MAIN CABIN SECTION



Notes:

1. Load factors shown in direction of inertial load.
2. Applies to harness, seat, and seat attachment.
3. Does not include ejection loads.
4. Applies to equipment and equipment supports.

TABLE 3.12
SHOCK AND ACCELERATION ENVIRONMENT
FOR EQUIPMENT AND EQUIPMENT SUPPORT DESIGN

Section	Station		Operational Phases					Transportation, Handling, and Storage
			Shock	Accelerations				Shock and Acceleration
	From	To		Launch	Abort		Re-entry	
Rendezvous & Recovery Module	191.97	239.00	Not Applicable	Longitudinal S/C Axis: 1.0 g's to 7.25 g's linearly with time over 333 seconds.	11.6 g's for 1.0 seconds parallel to S/C Z axis - Note (4)	7.25 g's for 1.0 second in any direction-Note (4)	15.0 g's along the longitudinal axis and 4.5 g's along the lateral or normal axis in any direction in the roll plane simultaneously, 30 seconds duration.	Shock - 15.0 g's in any direction, 11 milliseconds duration. Acceleration - 6.0 g's in any direction, 1 minute duration. These factors applicable only to equipment mounted in the proper shipping configuration.
Reaction Control Module and Upper Cabin Section	160.00	191.97	Landing - 30 g's along the S/C longitudinal axis and 30 g's along either the S/C vertical or lateral axis, 11 millisecond duration					
Main Cabin Section	103.44	160.00	Landing - Use ultimate values shown in Figure 3.11.2					
Adapter Retrograde Section	68.44	103.44	Not Applicable					
Adapter Equipment Section	13.44	68.44	Not Applicable		4.0 g's for 1.0 second any direction		Not Applicable	

NOTES:
1. Ultimate load is 1.36 times limit load.
2. Satisfactory performance is required during and/or after limit load application whichever is appropriate. This applies to all environments except landing shock. Only rescue equipment is required to give satisfactory performance after being subjected to the landing shock environment. No equipment shall tear loose from its mounts and internal parts shall be contained under application of ultimate loads.
3. Testing to the critical lateral load factor shall be as stated in Reference (6) or as called out in the SCD.
4. Does not combine with other load factors.
5. The load resulting from a positive longitudinal load factor is directed aft.

TABLE 3.13

VIBRATION AND ACOUSTIC ENVIRONMENT FOR

EQUIPMENT DESIGN

Vibration

Sinusoidal - Equipment having major resonances which can be determined accurately by conducting sinusoidal sweep tests shall be designed to withstand vibration in accordance with the table below.

Frequency	Level
10 - 15 cps	0.3 inches double amplitude
15 - 100 cps	± 3 g
100 - 500 cps	± 5 g
500 - 2000 cps	± 8 g

Random - Equipment items such as hermetically sealed instruments, mechanically complex electronic equipment and other mechanically complex items, where accurate determination of resonances cannot be achieved by sinusoidal testing, shall be designed to withstand the random vibration environment as defined by one of the spectrums given in Figures 3.13.3 and 3.13.4.

Note : Testing to the critical vibration or acoustic environment shall be as stated in Reference (6) or as called out in the SCD.

TABLE 3.13 (Cont.)

VIBRATION AND ACOUSTIC ENVIRONMENT FOR

EQUIPMENT DESIGN

Acoustic Noise

Frequency Octave Bands (cps)	-----	Sound Pressure Level DB RE: .0002 Dynes/cm^2			
		Cabin Area	Equipment Bay	Inside Adapter	External
37.5 to 75	--	120	122	124	131
75 to 150	--	123	128.5	131	139
150 to 300	--	127	135	139	148
300 to 600	--	130	140	149	159
600 to 1200	--	128.5	138.5	152	162
1200 to 2400	--	125	135	146	157.5
2400 to 4800	--	121.5	131.5	140	153
4800 to 9600	--	118	128	134	148.5
Over - All	--	135	145	155	165

Note: Testing to the critical vibration or acoustic environment shall be as stated in Reference (6) or as called out in the SCD.

FIGURE 3.13.3
GEMINI SPACECRAFT VIBRATION PSD
(RE-ENTRY MODULE REGION)

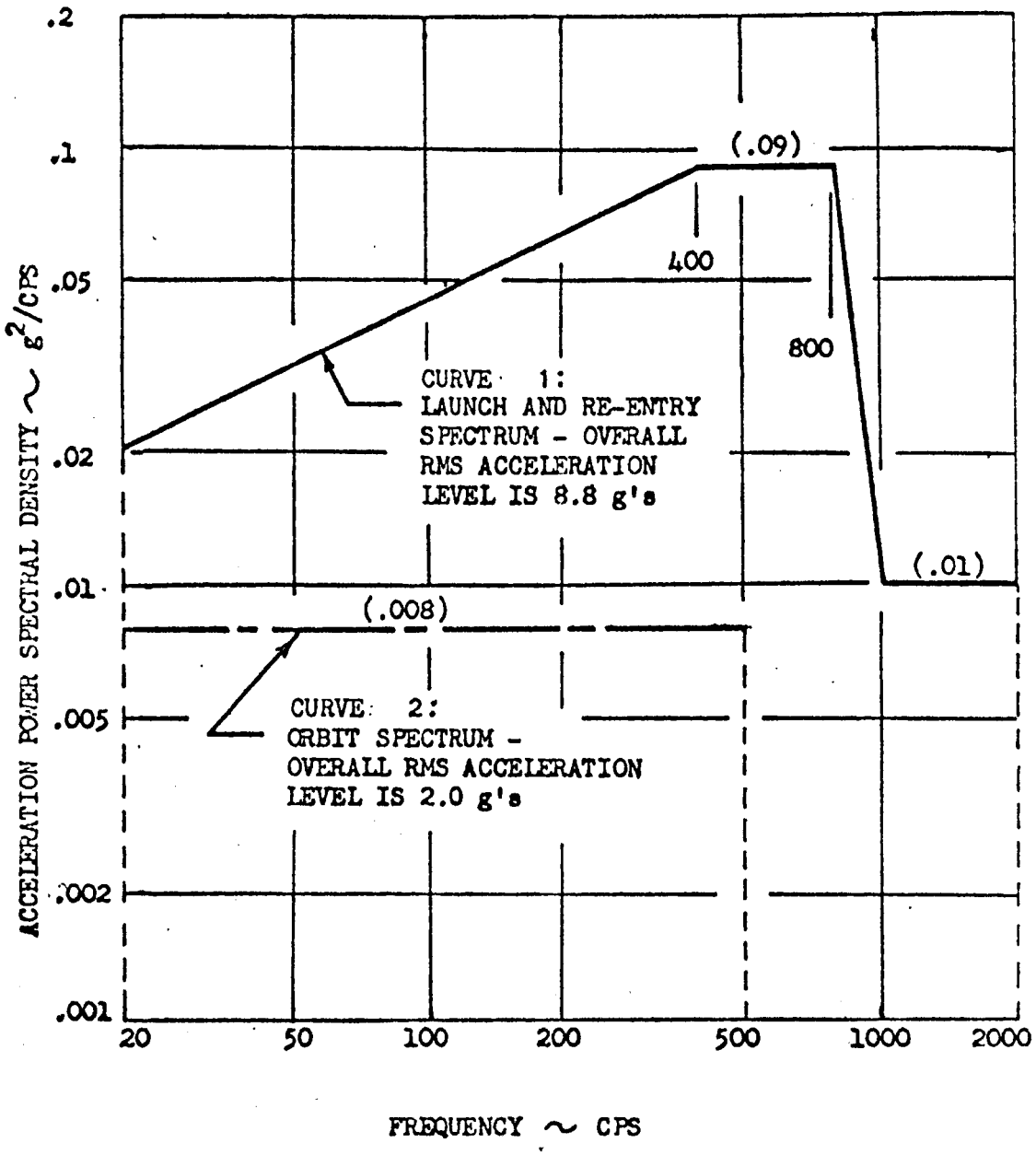
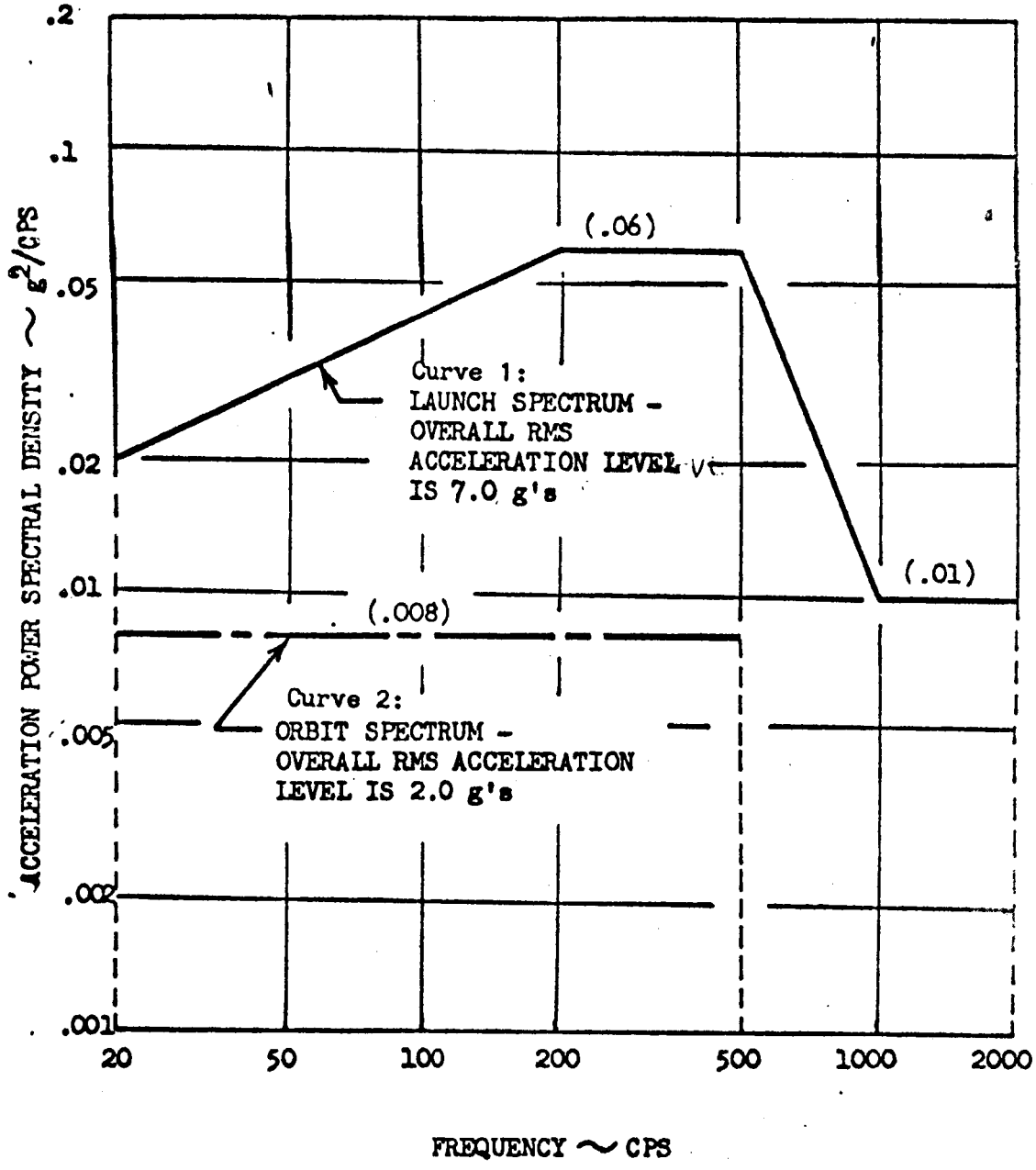


FIGURE 3.13.4

GEMINI SPACECRAFT VIBRATION PSD
(ADAPTER BLAST SHIELD REGION)



APPENDIX A

TARGET DOCKING ADAPTER CRITERIA

A-1.1 Boost Phase

The Agena/Target Docking Adapter/Ascent Shroud boost phase configuration shown on Figure A-1.2 is boosted using an Atlas for the first stage. The boost phase trajectory is shown on Figure A-1.3 The portion of the adapter forward of TDA Sta. 23.795 is protected by the Ascent Shroud which is jettisoned after leaving the earth's atmosphere. Boost phase loads on the shroud are transmitted to the adapter at TDA Sta. 23.795 and these plus the other adapter loads are transmitted to the Agena at TDA Sta. 0.00. For the boost phase, ultimate loads are 1.[illegible] times limit loads. Loads on the adapter during the boost phase shall be compatible with the design loads of the Agena. The adapter weight shall be between [illegible] and [illegible] pounds.

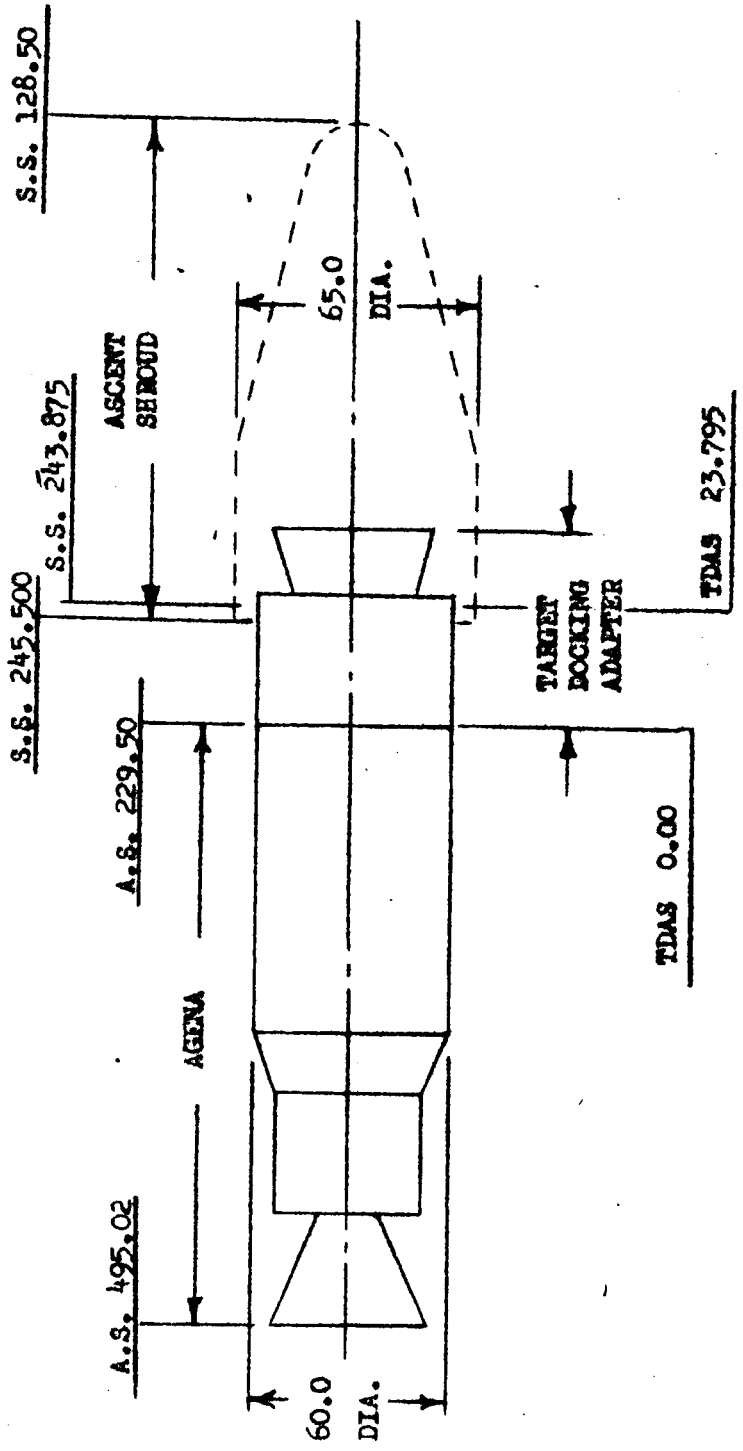


FIGURE A-1.2

BOOST PHASE CONFIGURATION

A.S. = AGENA STA.
TDAS = TARGET DOCKING ADAPTER STA.
S.S. = SHROUD STA.

FIGURE A-1.3

ATLAS-AGENA-TARGET DOCKING ADAPTER

BOOST PHASE TRAJECTORY

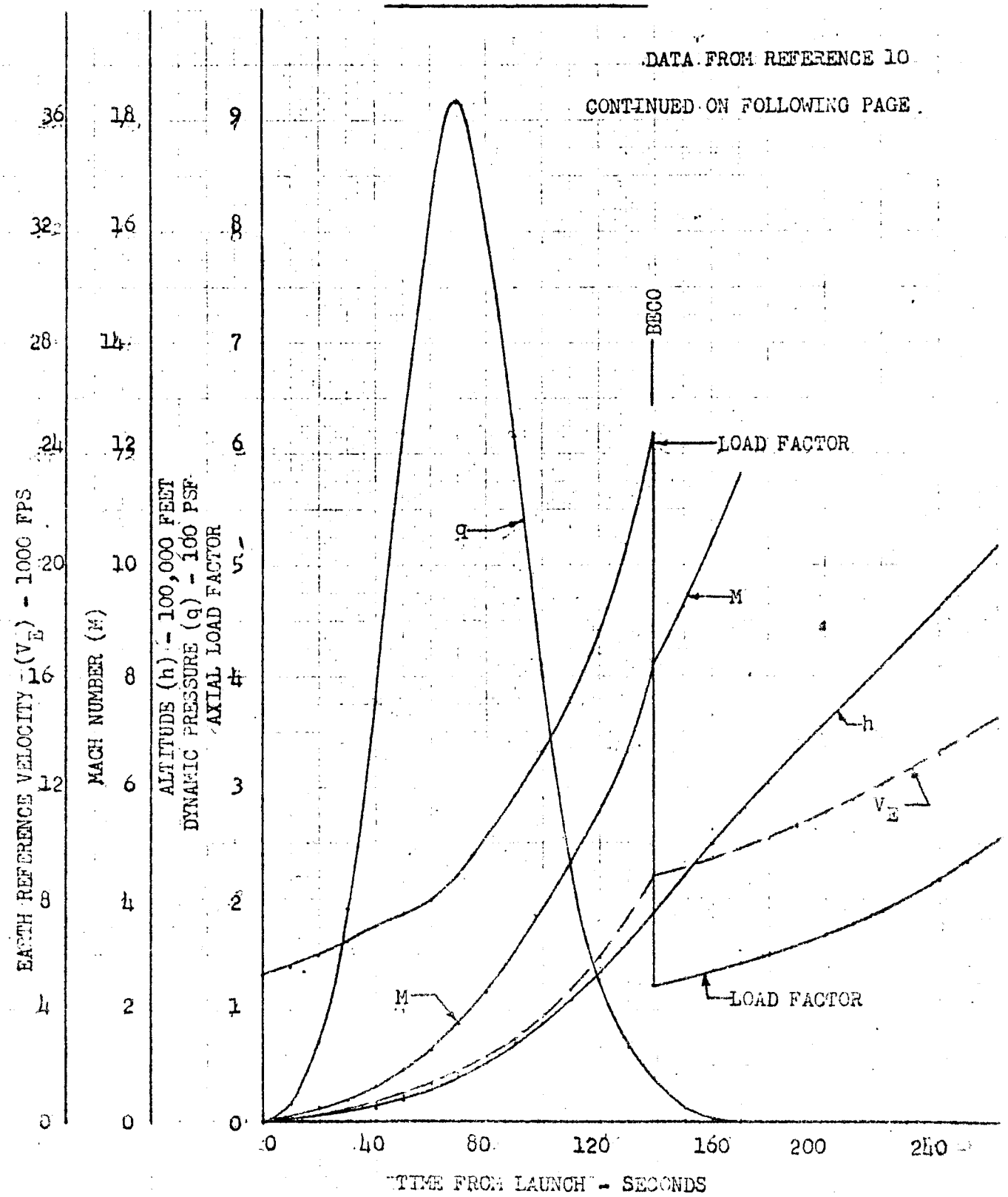


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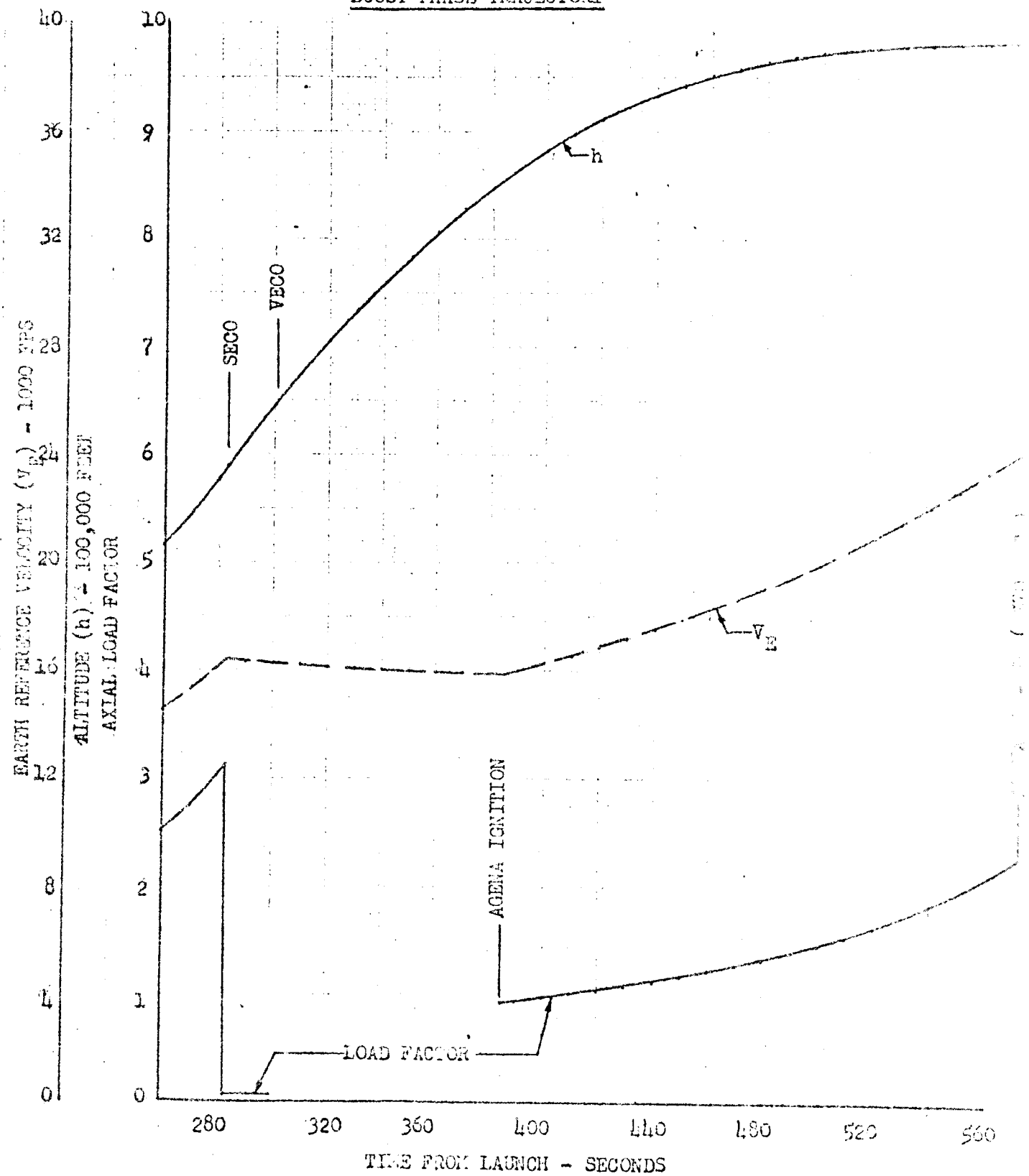
FIGURE A - 1.3

ATLAS - AGENA - TARGET DOCKING ADAPTER

BOOST PHASE TRAJECTORY



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