NASA TECHNICAL NOTE

NASA TN D-2943

NASA TN D-2943

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A COLD GAS, SHORT DURATION TECHNIQUE FOR HIGH ALTITUDE, UNDEREXPANDED JET EXHAUST IMPINGEMENT STUDIES

by A. J. Gopin and E. L. Margolin

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Huntsville, Ala.

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION • WASHINGTON, D. C. • NOVEMBER 1965

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DEFINITION OF SYMBOLS

Symbol	Definition
γ	Specific heat ratio c_p/c_v
c_p	Specific heat at constant pressure
c_v	Specific heat at constant volume
ϵ	Nozzle expansion area ratio
$\dot{m}$	Mass rate of flow
V	Velocity
DPS	Dual plane separation
A	Area
ρ	Density
P	Pressure
$\dot{P}$	Rate of change of pressure with time
R	Specific gas constant
T	Absolute temperature
M	Mach number
a	Velocity of sound
MW	Molecular weight

SUBSCRIPTS

AV	Average

DEFINITION OF SYMBOLS (Concluded)

SUBSCRIPTS	Definition
E	Exit
FS	Full scale
C	Chamber
M	Model
∞	Ambient
MAX	Maximum
T	Throat
cr	Critical
o	Total

A COLD GAS, SHORT DURATION TECHNIQUE FOR HIGH ALTITUDE, UNDEREXPANDED JET EXHAUST IMPINGEMENT STUDIES

SUMMARY

A short duration, cold flow test technique which experimentally duplicates the pressure effects of underexpanded free hot rocket exhaust jets impinging upon adjacent surfaces is described. The technique lends itself to any type of high altitude test cell with a minimal amount of simulated pressure altitude decay. The theory to justify the use of cold gases to simulate hot rocket exhaust is given, and short duration runs, i.e., as low as fifteen milliseconds, are shown to be feasible and within the state of the art.

The application of these two techniques - cold gas and short run times - permits the use of non-cryopumped vacuum tanks, thereby significantly reducing the cost of testing when compared to continuous hot or cold flow test methods now in common use.

The hardware required to use these techniques is also described. The device was developed for use in determining the exhaust impingement forces on the Saturn S-IC/S-II interstage during its separation from the Saturn S-II stage. While the model description and test results are applicable only to this particular test, the basic principles of the test technique and apparatus are applicable to any experimental investigation of free jet impingement.

SECTION I. INTRODUCTION

In the scale model investigation of the free jet impingement forces exerted upon the S-IC/S-II aft interstage during separation from the Saturn S-II stage, an extremely severe set of requirements had to be met. It was required that a large number of tests be run at simulated pressure altitudes over 76 km (250,000 ft) with high mass injection rates while simulating the prototype exhaust plume characteristics. Since high mass injection rates and high simulated altitudes are directly antagonistic, it was apparent that, if a test was run at any but the shortest flow durations, altitude decay would be severe. The use of a cryopumping facility - if one had been available - to reduce altitude decay would have been extremely costly. Furthermore, the use of the prototype O_2-H_2 mixture as the test gas would have increased model hardware costs and created facility problems in the handling of hydrogen gas.

To avoid these problems, a radically simple application was made of cold flow and short duration test techniques. The result was greatly reduced cost, near constant altitude simulation, and a significant increase in the speed of performing the investigation.

The principles used in this test are directly applicable to the more general problem of free jet impingement with the added advantages just described.

Before continuing with the description of the test techniques, some attention should be given to previous work in the investigation of free jet impingement at high altitudes. One of the earliest of these high altitude investigations was performed by Bauer and Schlumpf [1] in an ejector type test facility. While the use of an ejector maintains a constant test cell pressure, despite the continuous addition of a working fluid, the simulated altitude is usually limited to relatively low values. Recent studies by Piesik [2] (also see Ansley and Barebo [3, 4] on impingement of the Apollo reaction control rockets and measurements of stage separation forces by Binion and Heron [5]) were performed at simulated pressure altitudes above 61 km (200,000 ft) using cryopumped test facilities. Fergus and Gall [6] also used a cryopumped facility in their investigations. Unfortunately, the ability to maintain a constant pressure altitude with all but the smallest volumes of gases being continuously injected into a cyropumped facility is severely limited. Notably, Piesik was required to present impingement pressure as a function of testing time due to the altitude decay. In his investigation of jet effects on lunar surfaces, Stitt [7] resorted to using the base pressures behind a cone-cylinder-flare body in a supersonic wind tunnel.

The problems associated with these facilities are obvious. Occupancy in high speed wind tunnels and cryopumping facilities is expensive, and both ejectors and wind tunnels are limited to operation at relatively low simulated altitudes. Finally, cryopumping facilities usually produce excessive altitude decay with all but extremely low values of continuous mass flow injection. Shock tunnels might be used with the added advantage of near exact gas simulation, but with the disadvantages inherent in making shock tube measurements, as well as long delays between runs to replace diaphragms and re-evacuate the test cell.

Hence, a dilemma exists for investigators concerned with jet impingement at very high altitudes, above 61 km (200,000 ft) either in terms of cost or lack of altitude stability. The new test technique described in the body of this report solves these problems with the added advantage that a simple vacuum tank may be used as the test cell.

SECTION II. DISCUSSION

To deal with the problems inherent in high altitude impingement testing while keeping costs at a minimum, the following design criteria were established for the Saturn S-II stage separation test:

(a) A simple vacuum tank was to be used as the test cell.
(b) A pressure altitude of about 87.2 km (286,000 ft) was to be maintained at a nearly constant value throughout a test run.
(c) Cold gas was to be used to simulate the O_2-H_2 prototype mixture.

A. GAS SIMULATION

To properly simulate prototype impingement forces with a cold gas mixture, it was necessary to specify a mixture that would reproduce the momentum distribution of the exhaust gas. The use of gas mixtures to simulate properties of hot gases has been discussed by various authors, such as Chapman [10] and Templemeyer [11]. However, neither author considered gas mixtures for exhaust gas impingement studies. Proof of the momentum simulation is as follows:

$$\text{Momentum} = \dot{m}V = V^2 A\rho ,$$

since

$$\rho = P/RT \tag{1}$$

$$M = V/a = V/\sqrt{\gamma RT} \, ; \quad V^2 = M^2 \gamma RT \, . \tag{2}$$

Substituting equations (1) and (2), we can then write

$$\text{momentum} = V^2 A\rho = M^2 \gamma RT \cdot A \cdot P/RT = M^2 \gamma AP.$$

Equating momentum per unit area at a discrete point in the exhaust flow fields of the model and the prototype (assuming the same flow direction), we obtain

$$\gamma PM^2 \Big|_{FS} = \gamma PM^2 \Big|_{M} \, .$$

To simulate the momentum everywhere throughout the flow is theoretically impossible since the momentum distribution is a function of Mach number, specific heat ratio, and flow direction. A perfectly matched momentum distribution requires that the specific heat ratio history through the nozzle and exhaust flow field be perfectly matched. Unfortunately, such a duplication is not possible. Therefore, the nozzle exit plane was selected as the unique position at which to simulate γ in an attempt to achieve a reasonable compromise between γ simulation and minimal flow distortion. Since the Saturn S-II test model (1/50 scale) exactly duplicated chamber pressure and nozzle geometry, the exit plane similarity condition could be written as

$$\gamma_E \Big|_{M} = \gamma_E \Big|_{FS} \, .$$

Thus the entire problem is reduced to simulating γ at the nozzle exit plane.

Characteristics of several gas mixtures which were considered for the Saturn S-II separation tests as well as the O_2/H_2 prototype mixture are shown in Table I. All of the mixtures were capable of simulating the average exit plane value of $\gamma = 1.28$, as specified by the manufacturer of the prototype engines. The variation of γ through the nozzle was determined by a computer program. Values were determined for a given mass fraction at ten-percent intervals and faired curves drawn through the calculated points. Figures 1 through 6 give the calculated results of pressure, temperature and γ at the nozzle exit plane for three different gas mixtures. A mixture of argon and sulfur hexafluoride (Tables I and II) was selected for the following reasons:

(a) It resists condensation quite well.
(b) It has a high molecular weight. As explained later, this yields improved facility performance.

The simulation of a low γ with a cold gas requires the use of multiatomic gases, which is almost synonymous with heavy gases. Such gases are all quite readily condensed at the temperature reached in the nozzles assuming ambient stagnation temperature. Therefore, the gas was preheated to provide higher than nominal stagnation temperatures of 450° F and thus prevent condensation.

TABLE I

	CO_2/Freon 12 (56% CO_2)	N_2/SF_6 (36% N_2)	A/SF_6 (32% A)	H_2/H_2O *
γ_C	1.18	1.18	1.15	1.15
γ_E	1.28	1.28	1.28	1.28 (avg.)
P_E	2.0 psia	1.9 psia	2.04 psia	
T_E	-125° F	-140° F	-112° F	
T_{cr}	233° F	113° F	113° F	
Corrosivity	None	None	None	
Toxicity	None	None	None	
Flammability	None	None	None	
Molecular Weight	63	57.96	79.07	12.12
Exit Mach No.	4.245	4.278	4.232	

Note: Mach number based on pressure and specific heats ratio in exit plane. The critical temperature for the mixture is the highest value for either constituent.
T_E is based on T_C = 450° F

* Main products of combustion of the J-2 engine

TABLE II

Properties of Argon - Sulfur Hexafluoride Mixture

	A	SF_6	Mixture
Mass Fraction	0.32	0.68	1.0
Mole Fraction	0.632	0.368	1.0
Molecular Weight	39.94	146	79.07
γ_C	1.667	1.069	1.15
γ_E	1.667	1.084	1.28

The advantages of as high a molecular weight (MW) as possible are shown as follows:

$$\dot{m}_{MAX} = \sqrt{\frac{\gamma}{R}\left(\frac{2}{\gamma+1}\right)^{\frac{\gamma+1}{\gamma-1}}} \cdot \frac{P_o}{\sqrt{T_o}} \ .$$

For constant values of γ, P_o and T_o,

$$\dot{m}_{MAX} \sim \sqrt{\frac{1}{R}} \sim \sqrt{MW} \ .$$

The change in test cell pressure as a function of time is

$$\dot{P} = \frac{\dot{m}RT}{V} \sim \frac{\dot{m}T}{V(MW)}$$

or

$$\dot{P} \sim \frac{\sqrt{MW}}{MW} \sim \frac{1}{\sqrt{MW}} \ .$$

Therefore, the higher the gas molecular weight, the longer the run time before the vacuum facility simulated altitude falls below the desired lower limit.

However, there is one distinct disadvantage in using a heavy, cold gas (T_c = 910° R): The cold gas mixture produces a larger mass flow than the same nozzle flowing a hot gas. Since $\dot{m}_{MAX} \sim \sqrt{\frac{MW}{1}}$, the use of a low temperature gas with a high molecular weight significantly increases the flow rate. As an example, the flow rate of a 1/50th scale model of the Saturn S-II stage using the prototype combustion products is 0.953 lbm/sec. Correcting for temperature and molecular weight, we get

$$\dot{m}_{MAX,M} = \dot{m}_{MAX,FS} \sqrt{\frac{T_{FS}}{T_M} \cdot \frac{MW_M}{MW_{FS}}}$$

$$\dot{m}_{MAX,M} = 0.953 \frac{lbm}{sec} \sqrt{\frac{5900}{910} \cdot \frac{79.07}{12.12}}$$

$$\dot{m}_{MAX,M} = 0.953 \frac{lbm}{sec} (6.5)$$

$$\dot{m}_{MAX,M} = 6.19 \text{ lbm/sec} .$$

This six-fold increase in flow rate due to cold gas seriously impairs the altitude capability of high altitude facilities in common use. Despite the extremely short duration of each S-II separation test run (15 - 30 milliseconds), this high mass flow rate required the use of an extremely large vacuum cell to limit altitude decay.

However, since most jet impingement investigations are done with a single nozzle (the S-II has a cluster of five rocket engines), flow rates would normally be low enough to allow the use of a moderately sized vacuum tank.

B. SHORT DURATION TESTING

The nominal pressure altitude for the Saturn S-II interstage separation is 87.2 km (286,000 ft); therefore, 76.2 km (250,000 ft) was set as the lower limit for testing at simulated pressure altitudes. This range may appear large, but is actually quite conservative because at such altitudes the S-II nozzles are highly underexpanded (P_∞/P_C = .474 $\times 10^{-6}$ at 76.2 km (250,000 ft)), and any change in altitude produces changes only in the extreme boundary regions of the jet exhaust. When compared to the overall impingement forces exerted by the jet exhaust at these altitudes, these boundary variations are completely negligible. This property of expansion of highly underexpanded exhaust jets is generally of great advantage in impingement testing, since altitude is not extremely critical, and a certain amount of altitude decay in the test cell can be tolerated. It is emphasized that this effect is limited to highly underexpanded exhaust jets, and only when altitude variation is not great enough to significantly alter the free jet flow field.

The use of short duration test techniques (15 to 30 milliseconds) served a twofold purpose in the Saturn S-II separation test. First, the short flow durations kept test cell altitude decay to a minimum, thereby reducing the pump-down time between runs. It is estimated that the use of short duration runs in the Saturn S-II separation test reduced the between-run, pump-down time from approximately one hour to twelve minutes. Needless to say, both running costs and the overall test period were reduced by a fifth. Secondly, impingement force measurements did not have to be interpreted through an altitude-time function, greatly simplifying the data reduction.

C. TEST FACILITY

When the test device was designed, it was not known that runs as short as 20 milliseconds could be readily performed and produce satisfactory results. Therefore, the test cell was selected on the basis of maximum internal volume. Use of a large test cell would permit runs of greater duration without having excessive altitude decay. As pointed out previously, the large mass injection rate also dictated the need for a large tank.

One facility which amply met the requirements was the 60-foot vacuum sphere at Langley Research Center, Hampton, Virginia, which is shown in Figure 7. This cell could be evacuated to simulate pressure altitudes well over 91.5 km (300,000 ft). Evacuation would be continuously maintained by oil diffusion pumps so that after each test run a minimal amount of time was required to reach simulated prototype altitudes. After each 30 millisecond test shot, the pressure altitude fell from 91.5 km (300,000 ft) to approximately 83.8 km (275,000 ft), and from five to fifteen minutes was required to re-evacuate the sphere to 91.5 km (300,000 ft), depending on external atmospheric conditions and the leakage in the system. Generally, twelve runs could be obtained per hour for extended periods of time, a very large number compared with ejector cells, cryopumping facilities, and shock tubes. Furthermore, the 60-foot sphere was relatively low in operating cost, as are all facilities of this type.

However, the use of short duration flows created problems in the working fluid control system as well as in the measurement of the interstage forces. The latter problem was met by designing a fast response, static force balance. The technology involved in the design of such a force balance has been standard for a number of years.

To control the working fluid for extremely short flow times, a high speed, electronically controlled, hydraulically operated valve was especially manufactured from standard parts. The disc type of valve had a travel time (to open or to close) of 5 milliseconds, establishing a minimum test run time of 10 milliseconds, and was operated by hydraulic pressure of 3000 psi. With proper hardware modifications, an increase in hydraulic pressure would have reduced the travel time. The valve was seated between the model Saturn S-II engine cluster and a heated plenum chamber of working fluid ($A-SF_6$), which was large enough so that, during the run, chamber pressure decay was negligible. The plenum chamber in turn was fed by gas from cylinders through a compressor. This last item was needed since the test gas was supplied in 650 psi cylinders, which was less than the pressure required in the plenum chamber. Since the bottled gas for this specific test series was very near the condensation point, as supplied, it was also necessary to submerge the cylinders in a hot water bath at 160°F to ensure that only the gaseous phase existed in the cylinders.

The valve actuator operation was controlled by a remote electronic timer. This device allowed the valve to operate (i.e., to remain fully open and to close) for a period of time from 10 to 150 milliseconds in increments of 10 milliseconds. It could also be manually operated for any period of time. However, it appears that manual operation could not give runs of less than 50 milliseconds, and run time was not very repeatable. The manual mode of operation was used to evacuate the plenum chamber before charging it with test gas, by keeping it open during the test cell evacuation period.

To eliminate the necessity of opening the test cell between runs, the model interstage and its force measuring devices were designed so that they could be positioned by remote control. Figure 8 presents a schematic of the test hardware and gas supply system.

The forces on the interstage were determined using a force balance system, the output of which was recorded on a variable speed oscillograph. A typical data trace is shown in Figure 9. Some aspects of this trace are discussed in the next section.

D. RESULTS

Since the purpose of this report is to introduce the application of cold flow, short duration testing for high altitude jet impingement studies, complete Saturn S-II separation test data are not presented. However, some examples of the test results are given in the following section along with a more detailed description of the test program. Complete details of the test and the test results can be found in References 8 and 9.

Figure 9 shows that the pressure trace, measured just upstream of the nozzle throat, reached a peak value within the five millisecond response time of the valve. Within two milliseconds of the valve reaching the full open position, the traces from the force balance members indicate that the full impingement forces have stabilized. This shows that there is no problem in establishing a steady state flow field within seven milliseconds. As previously noted, the small variations in the test cell pressure have no effect on the impingement forces. Similarly, the small drop in chamber pressure (approximately 20 psi) had no effect on the force measurement. During the steady state portion of the test period (30 milliseconds), the data traces are steady and easily read despite the extremely short run times, and the data were perfectly repeatable. In fact, any variation in the data reduction was primarily because of error in reading the traces.

These results confirm that short duration test techniques can be successfully employed in high altitude jet impingement studies. This simplifies data reduction and significantly reduces the cell evacuation time between runs.

Another important point is that, by simulating the exit plane specific heats ratio with a cold gas, the impingement pressures produced by a hot gas can be duplicated because the free jet momentum profile has been approximated. While this is an assumption, the authors contend it is a good one. Such a contention is partly based on previous experimental work [12]. This test established that free jet impingement pressures would be predicted by assigning a Newtonian pressure coefficient throughout the free jet flow field. Local jet properties were determined by the method of characteristics. Thus, despite the variations in flow properties throughout the nozzle because of specific heat ratio variations, the momentum profile remained a function of only the specific heat ratio.

Further credence is given to this opinion in that only small changes in the value of the specific heats ratio occur beyond the exit plane of highly underexpanded nozzles. In the case of the prototype engine, this is partially because frozen composition occurs well forward of the nozzle exit plane.

As a part of the S-II tests, the effects of changes in specific heat ratio and of changes in engine chamber pressure were determined and are presented in the following section. Specific heat ratio variations were obtained by varying the test gas mixture or by using a different test gas.

Because of the complexity of a five-engine-exhaust flow field, it is difficult to exactly predict the impingement forces. However, the effects of specific heat ratio and chamber pressure variations are qualitatively predictable, and did alter the impingement forces in the expected direction.

E. SATURN S-II DUAL PLANE SEPARATION

To assure a reliable separation of the Saturn S-II from the S-IC at a minimal payload penalty, a technique called "dual plane separation" (DPS) will be used. Under the concept of DPS, the first plane separation occurs at station 0, 16 inches forward of the S-II rocket cluster exit plane, thereby ensuring a safe separation of the S-IC stage. Approximately 30 seconds later, with the J-2 engines at full thrust and at an altitude where aerodynamic effects are essentially zero, the S-IC/S-II interstage, which weighs approximately 9000 pounds, is separated at station 196. The separation sequence is presented diagrammatically in Figure 10. The combination of S-II thrust plus drag on the interstage due to the impingement of the J-2 exhaust plumes will accelerate the interstage away from the S-II stage.

Since the outboard engines of the J-2 cluster are capable of deflecting approximately 7 degrees in pitch or yaw, there is the possibility that the rocket exhaust impingement forces on the interstage will be extremely asymmetric. Furthermore, this asymmetry will be aggravated if one of the outboard engines fails to ignite. Impingement forces of this type could conceivably accelerate the interstage laterally into the engines of the S-II stage.

The dual plane separation tests were undertaken to experimentally determine these asymmetric forces so that the trajectory of the aft interstage with respect to the Saturn S-II stage could be determined. These trajectories for various engine deflection patterns would thereby establish whether or not the DPS mode of separation would be successful.

The DPS model test furnished static forces and moments which were used to determine the trajectory of the interstage. The aft interstage was supported by a pair of diametrically opposed force balances which in turn were mounted on a remotely operated, movable carriage. In this manner the exhaust forces on the model interstage could be measured at any interstage position with respect to the J-2 rocket cluster. With a map of impingement forces and moments as a function of interstage position, the trajectory of the interstage with respect to the Saturn S-II stage could be readily calculated.

The DPS tests were performed for seven different nozzle deflection patterns, as shown in Figures 11 and 12. The nozzle position could be varied by inserting a mounting block machined to give each nozzle the proper cant angle. The model Saturn S-II base and engine cluster is shown in Figure 13. Notice that the model is equipped with a base heat shield so that base pressure as near prototype as possible is simulated. The entire test apparatus as mounted in the Langley 60-foot vacuum sphere is presented in Figure 14. By comparing Figure 14 with Figure 15, the various hardware components can be identified.

Figure 16 defines the reference system for the following data curves. Some typical data curves for the nominal 8-degree engine-out case are shown in Figures 17, 18 and 19. The data in this form were then used to calculate the trajectory shown in Figure 20. Notice that collision occurs with the "dead" engine in the null position.

The digital computer program used to calculate the interstage trajectory from the DPS test data used two-body (i.e., upper stage and interstage) three-degrees-of-freedom equations of motion. The primary purpose is to determine the relative position of a critical point on each body at any particular time during a separation sequence. After physical separation occurs, the two-body equations of motion are used to define the relative motion of the two stages.

In addition to varying the nozzle deflection patterns, several test series were performed at off-design values of specific heat ratio (γ) and chamber pressure (P_c).

The specific heat ratio was varied by employing different test gases. These investigations at p_c = 750 psia and 500 psia were performed for the 8-degree engine-out case with the nominal ($\gamma = 1.28$) gas mixture. The data are shown in Figures 21, 22, 23 and Figures 24, 25 and 26, respectively. By comparing these with the nominal 8-degree engine-out case, Figures 17, 18 and 19, the following general statements can be made:

1. Drag force increased with an increase in chamber pressure.

2. Normal force increased with an increase in chamber pressure.

3. Moment was only slightly sensitive to chamber pressure, either increasing or decreasing depending on the interstage position.

Similarly, investigations were performed for $\gamma = 1.18$ and $\gamma = 1.40$ for the 8-degree engine-out case at the nominal chamber pressure. The data are shown in Figures 27, 28, 29 and Figures 30, 31 and 32, respectively. By comparison with the nominal case, it can be seen that the specific heat ratio can be critical in the design of this type of test. Generally, there was little variation in forces and moments because of a change in specific heat ratio from $\gamma = 1.18$ to $\gamma = 1.28$, but there was a significant variation from $\gamma = 1.28$ to $\gamma = 1.40$.

Although changes in γ result in modifications of the momentum distribution in the exhaust flow field, very small variations have little effect on the impingement forces and moments. Thus, minor errors in the gas mixtures used to simulate γ are negligible.

A more detailed description of the DPS test and a complete set of test data can be found in Reference 9.

SECTION III. CONCLUSIONS AND RECOMMENDATIONS

The use of cold gas simulation and short duration testing is an extremely productive, highly repeatable method for the investigation of high altitude impingement of jet exhaust plumes. While this technique lends itself to any type of altitude test facility, it is not limited to only the more exotic and more costly test cells. Since nearly zero development time went into the production of the high speed valve used to control the working fluid, it would seem that a valve with lead and lag times of less than 5 milliseconds could be developed, if required. This would permit the use of a smaller test cell. Either total forces or pressures on specific surfaces can be determined using this technique, and experimental investigations of the effect of variation in engine operating conditions such as chamber pressure and mixture ratio can be readily performed.

In the application of cold gas and short duration techniques to impingement studies, the following recommendations are made:

1. If the plenum chamber is not regulated at a constant pressure, it must be voluminous enough that, during the test run, chamber pressure decay will not be excessive.

2. The test cell volume must be selected so that altitude decay during a run will not excessively alter the free jet exhaust shape. Both of these criteria can be evaluated by applying the method of characteristics to describe the nozzle and free jet flow fields.

3. If many test runs are needed to obtain the force or pressure data required, impingement surfaces should be movable by remote control to eliminate the need to open the test cell between runs.

4. The leaks in the gas supply system must be kept at a negligible level.

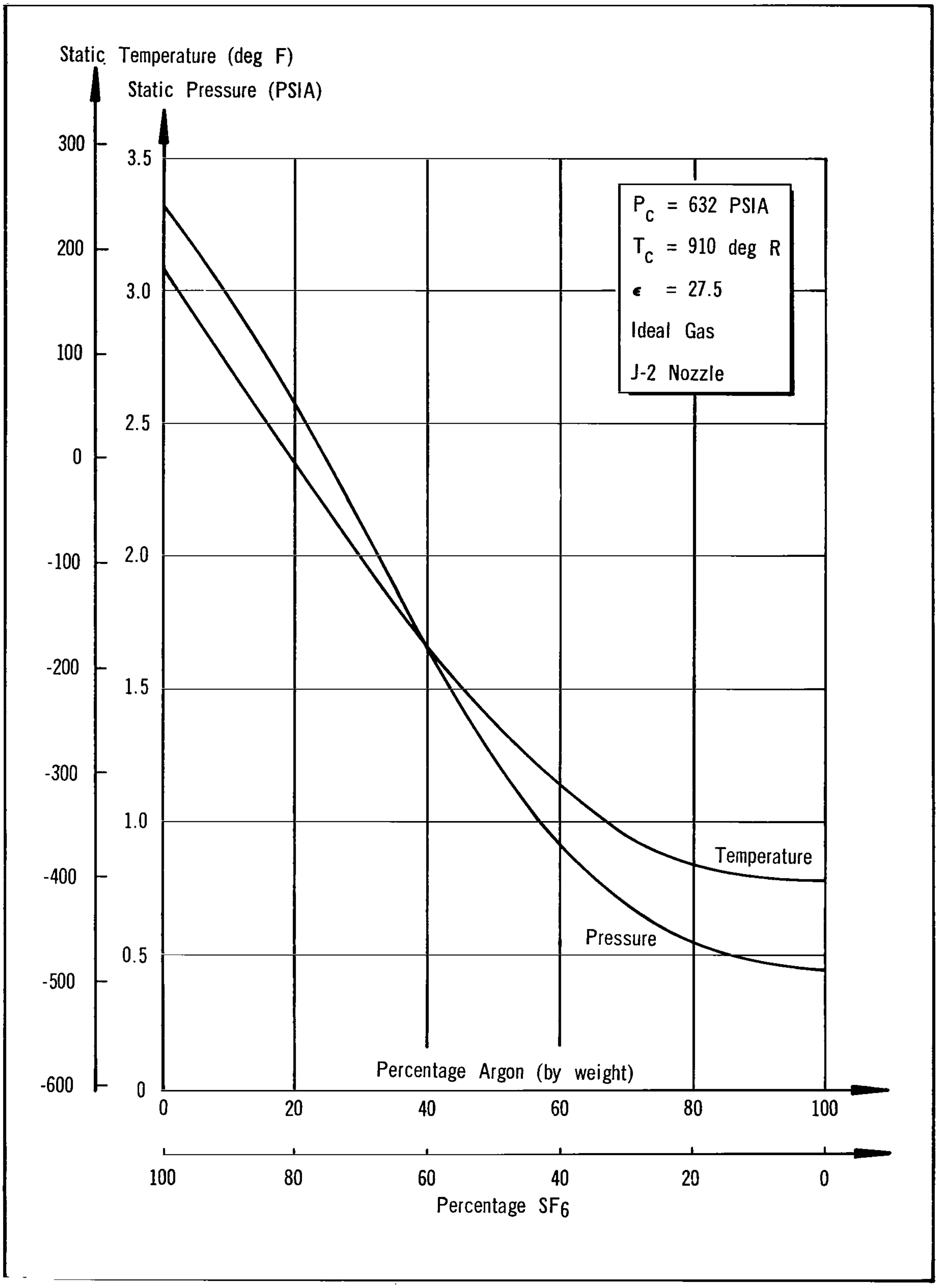

FIGURE 1. EXIT PLANE CONDITIONS ($A-SF_6$)

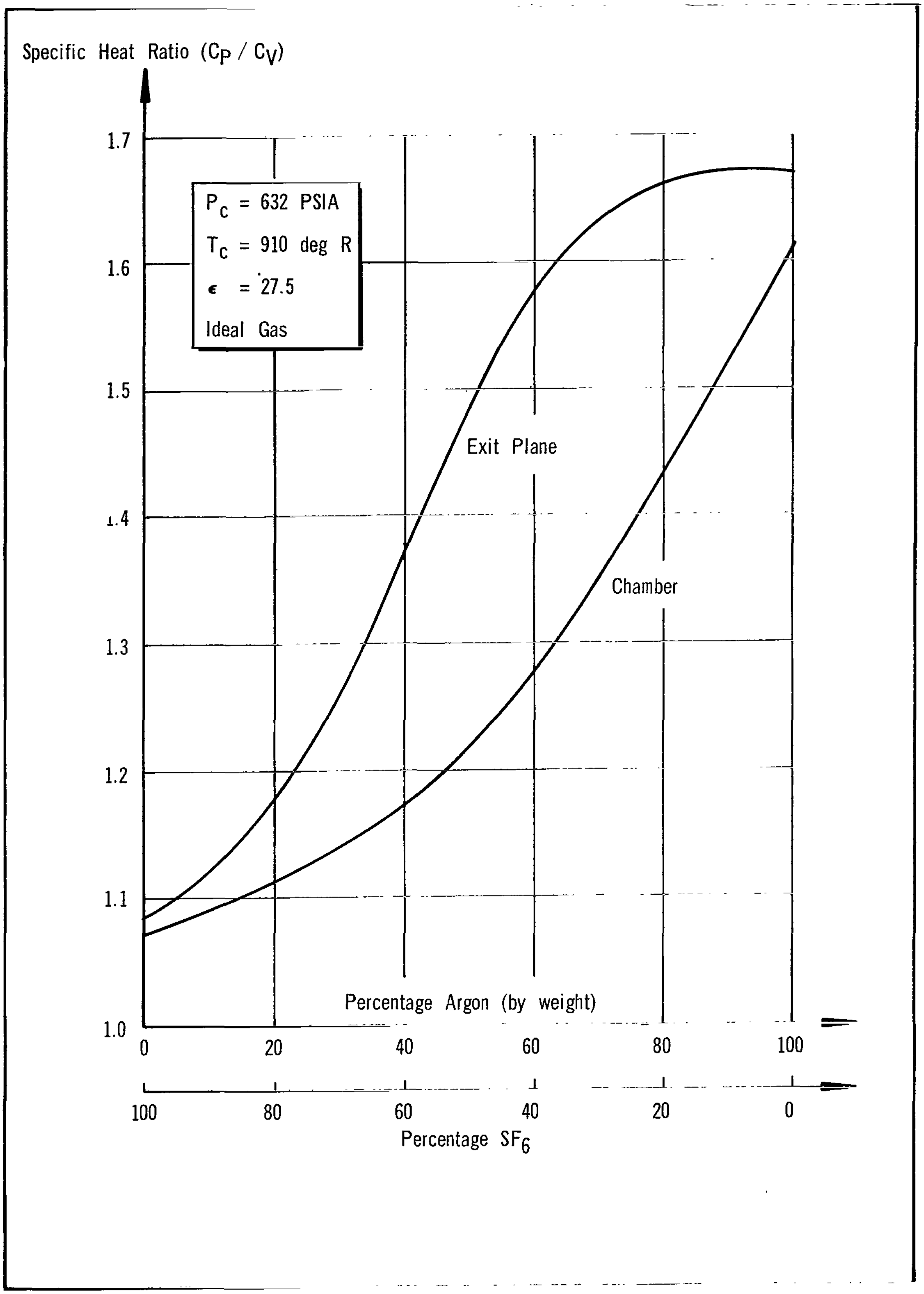

FIGURE 2. VARIATION OF SPECIFIC HEAT RATIO IN THE J-2 NOZZLE (A-SF_6)

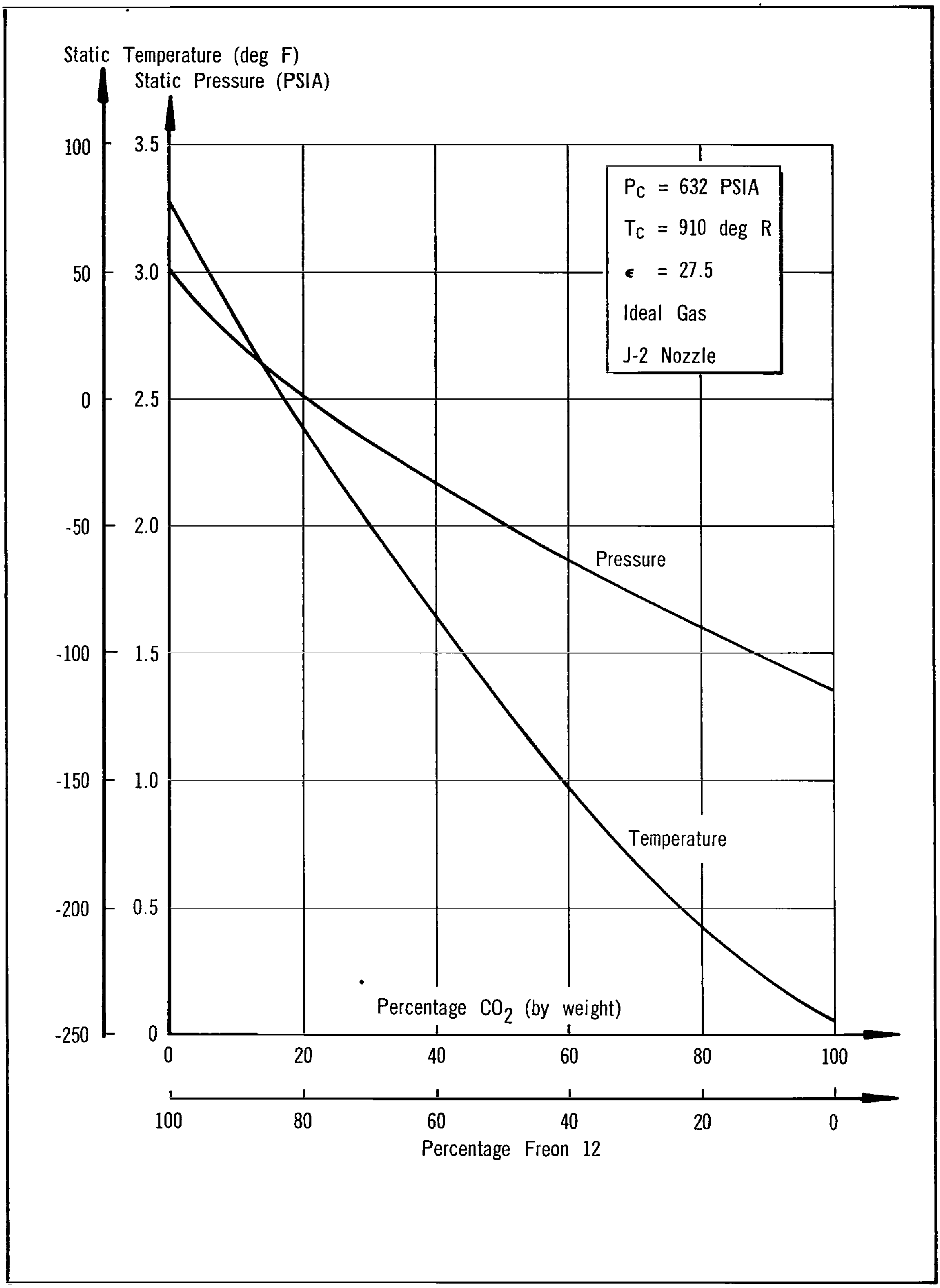

FIGURE 3. EXIT PLANE CONDITIONS (CO_2 - FREON 12)

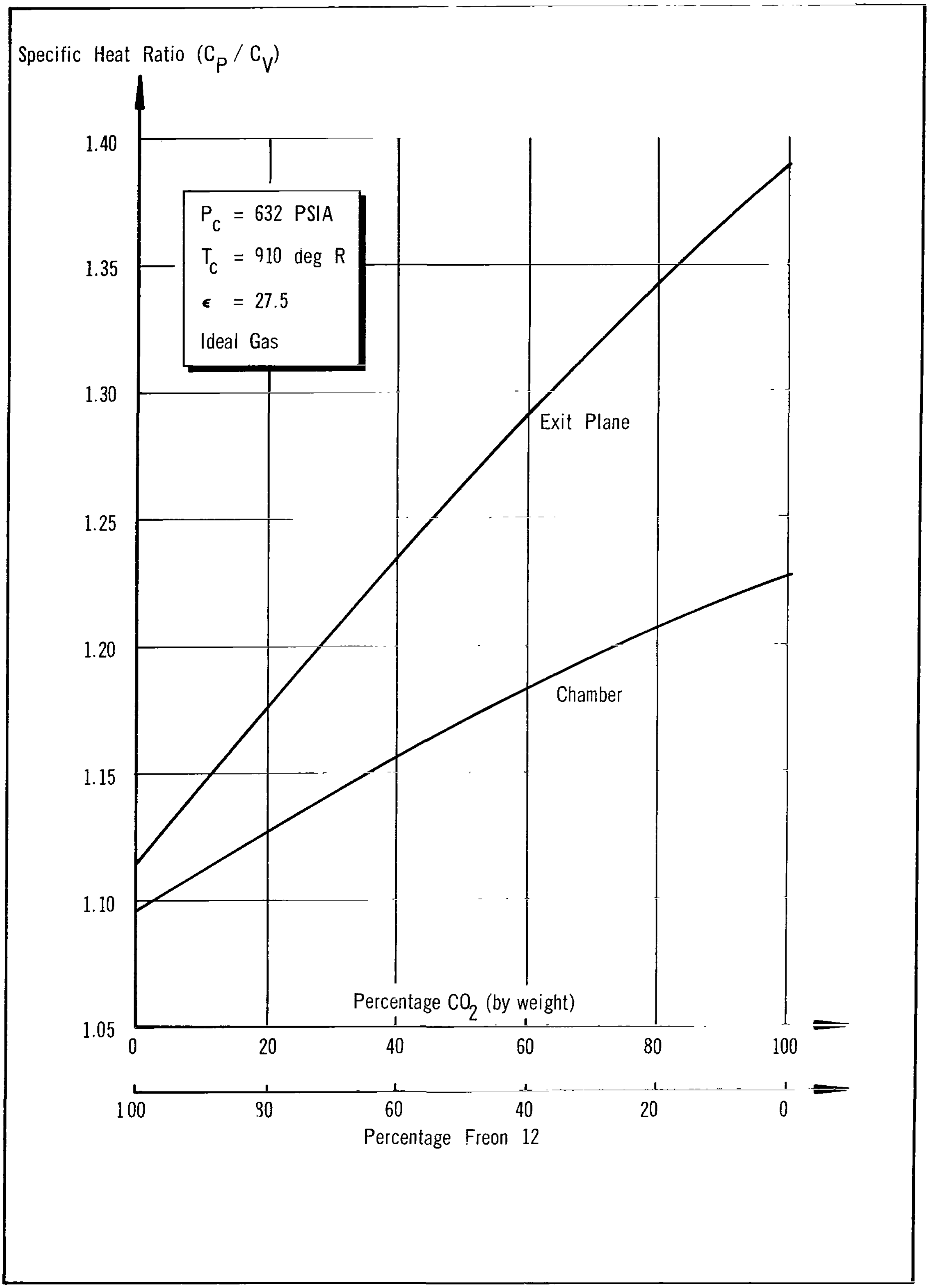

FIGURE 4. VARIATION OF SPECIFIC HEAT RATIO IN THE J-2 NOZZLE (CO_2 - FREON 12)

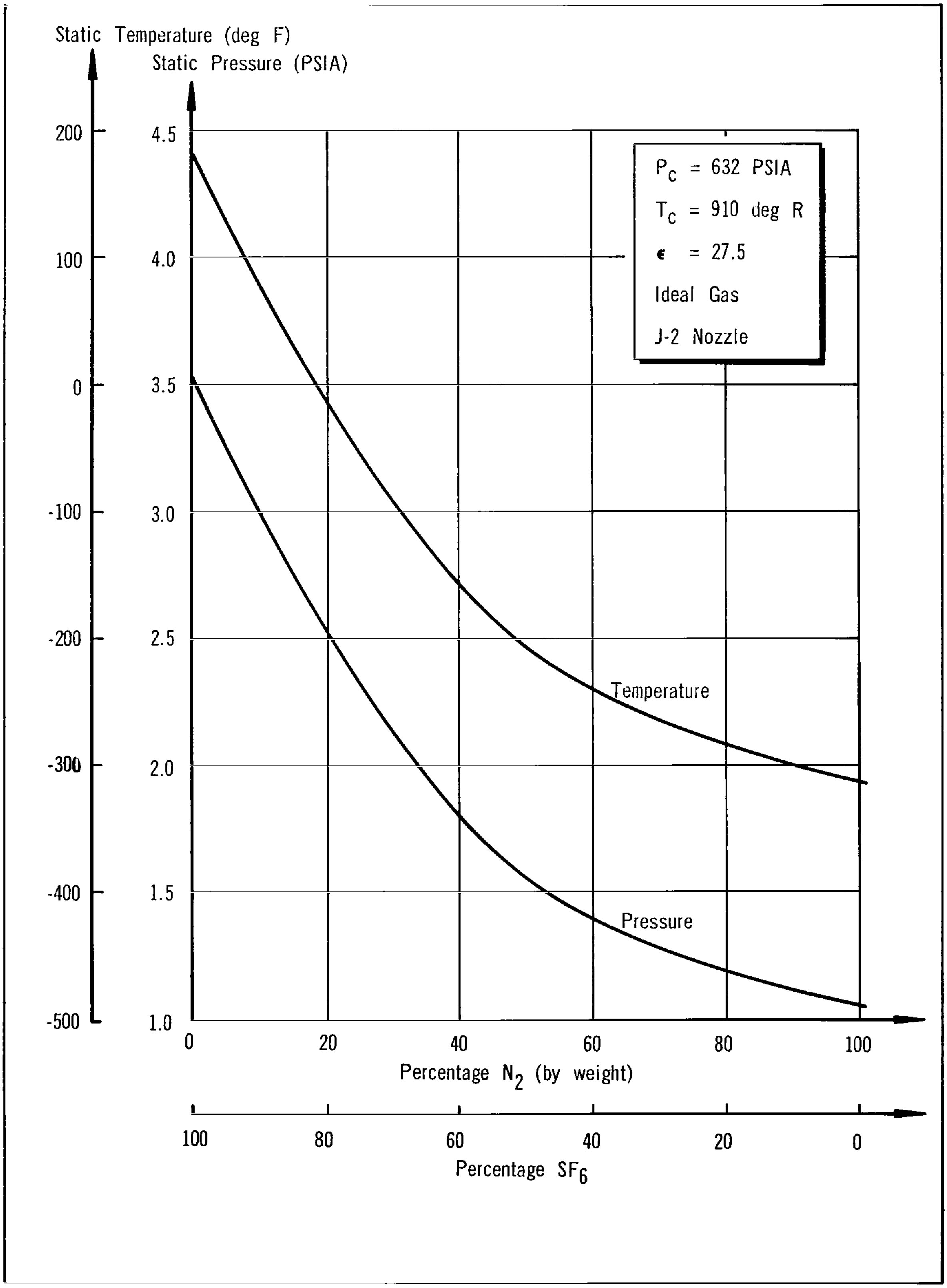

FIGURE 5. EXIT PLANE CONDITIONS (N_2 - SF_6)

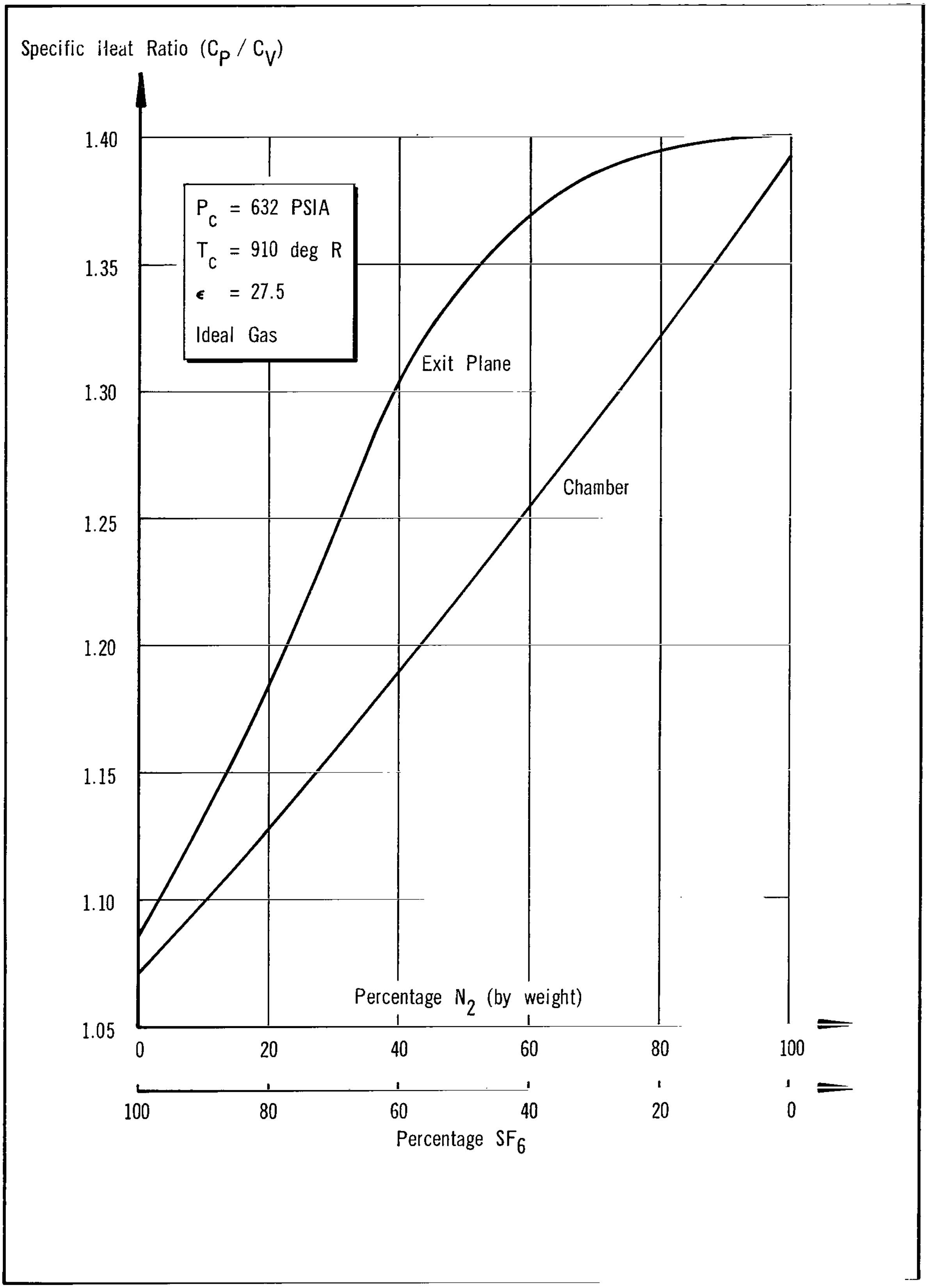

FIGURE 6. VARIATION OF SPECIFIC HEAT RATIO IN THE J-2 NOZZLE (N_2 - SF_6)

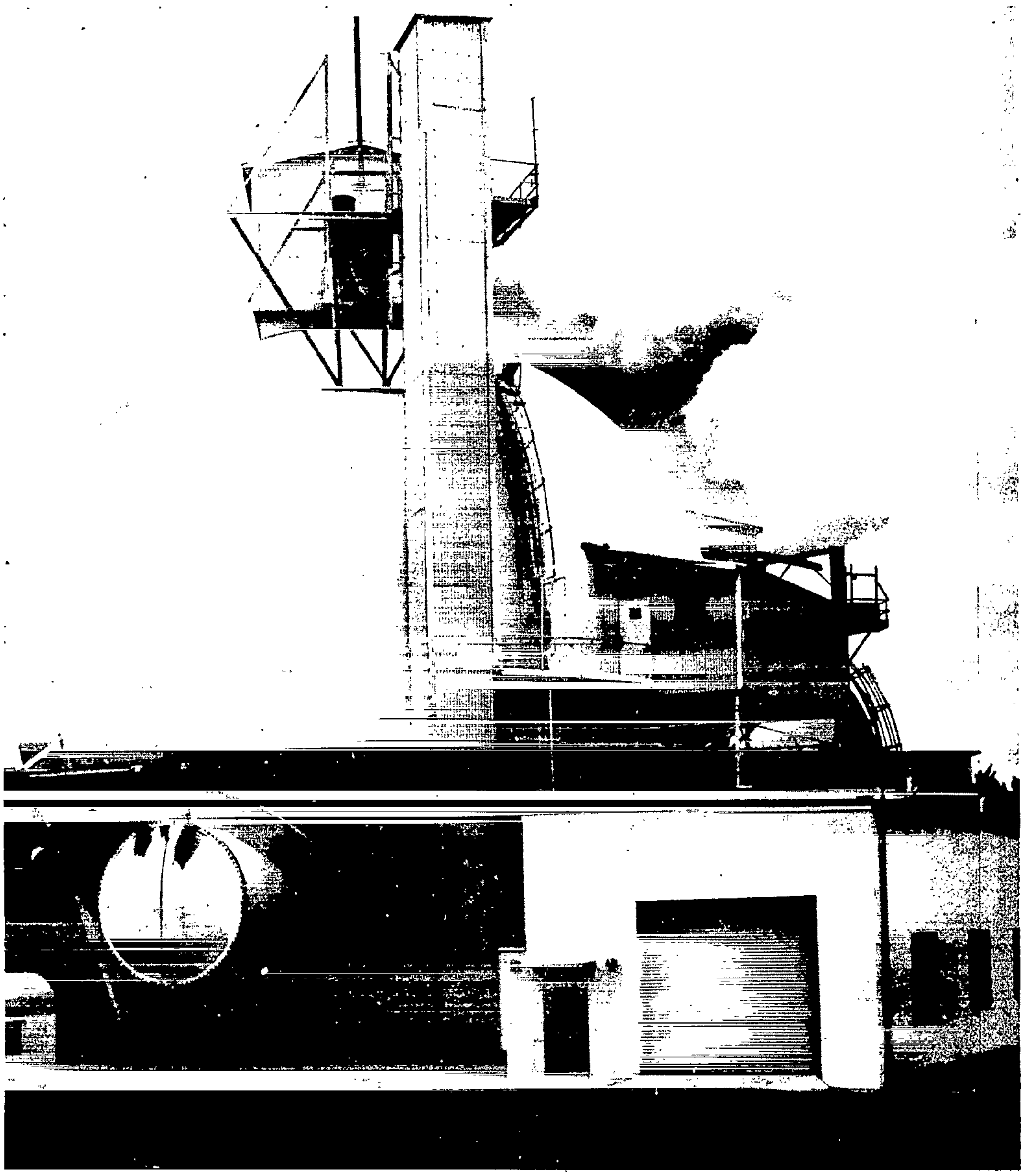

FIGURE 7. 60' VACUUM SPHERE LANGLEY RESEARCH CENTER

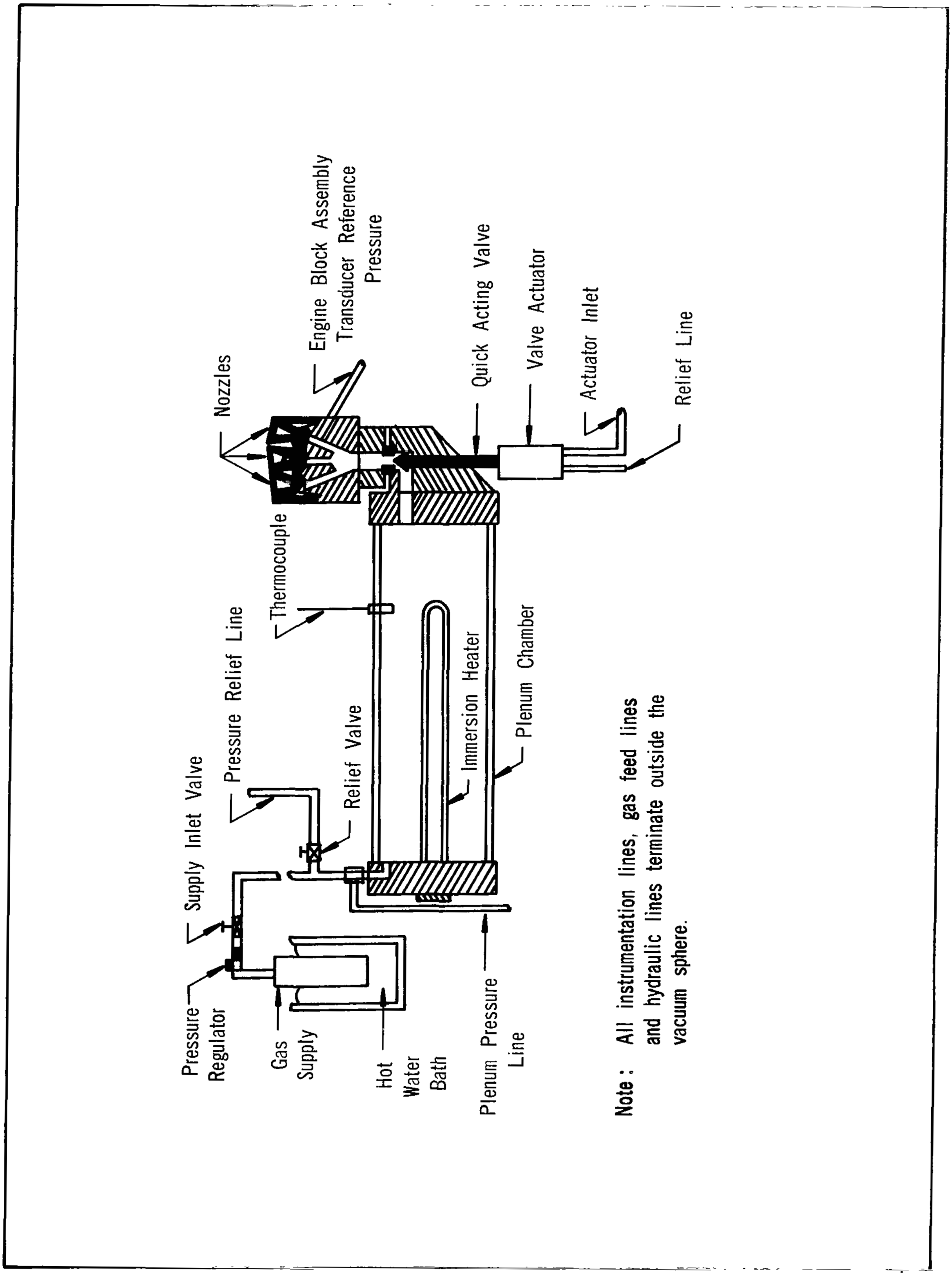

FIGURE 8. SCHEMATIC OF SATURN S-II DPS TEST HARDWARE

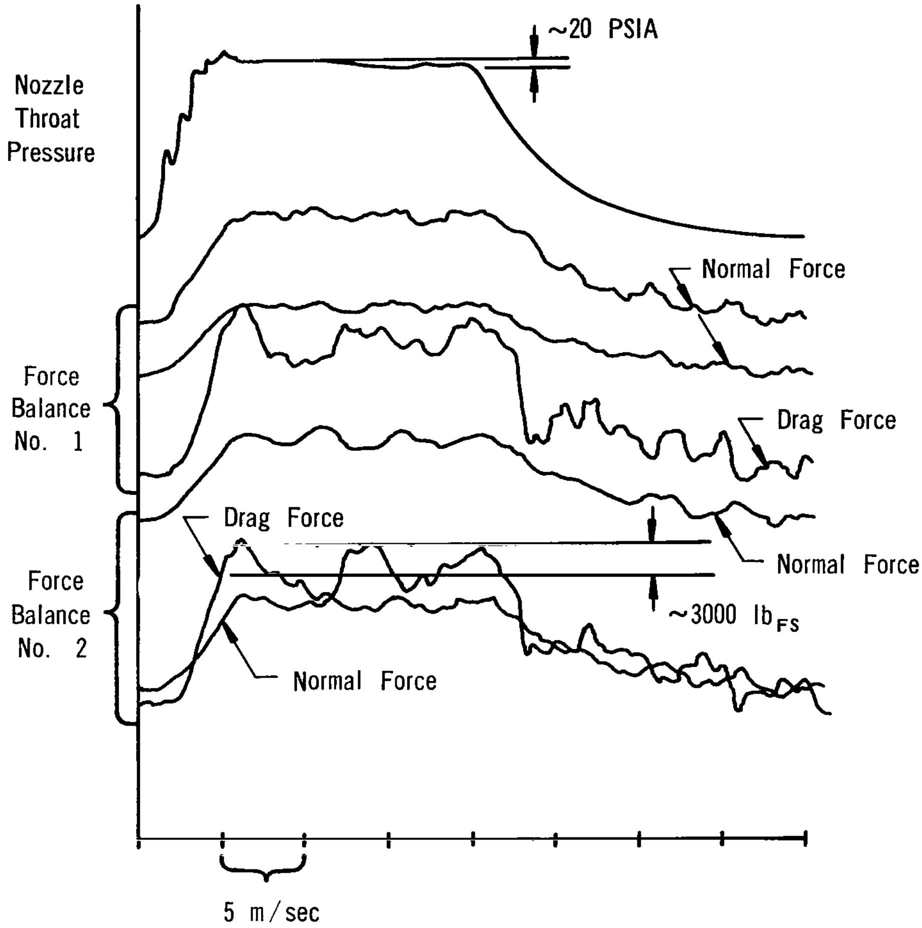

FIGURE 9. OSCILLOGRAPH OUTPUT

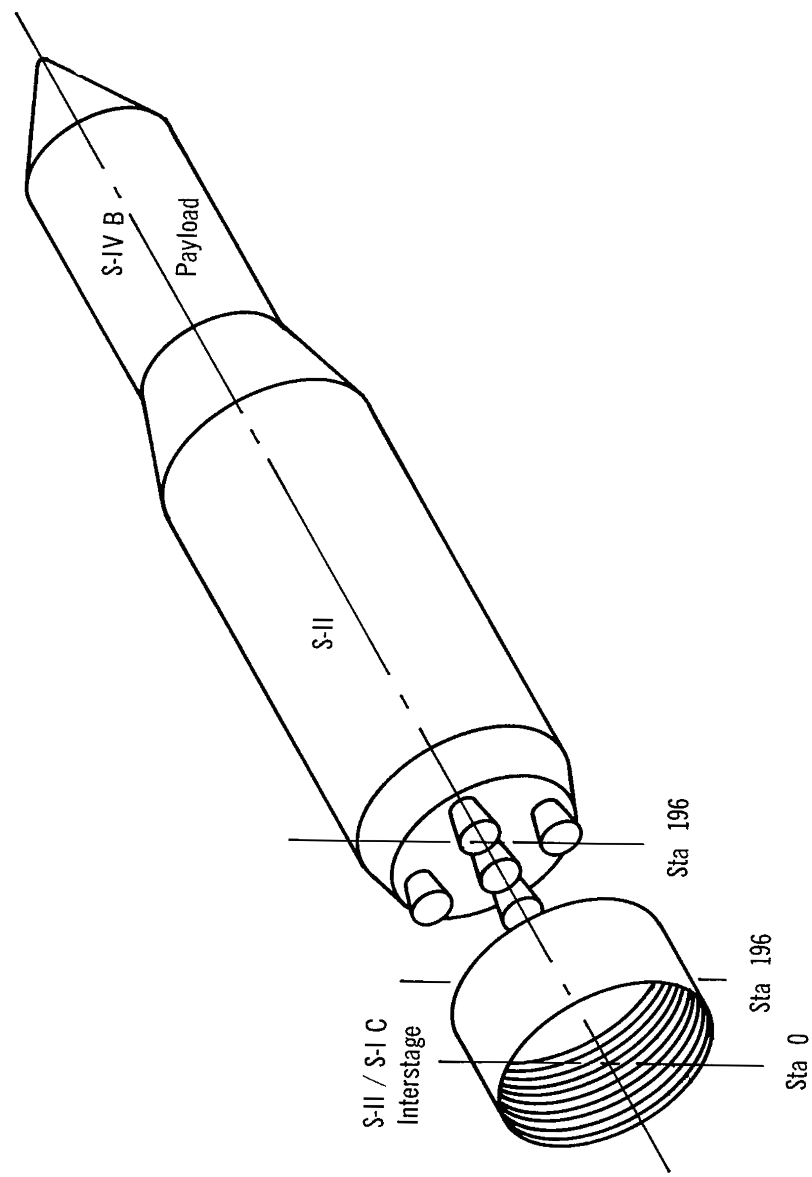

FIGURE 10. SEPARATION DIAGRAM

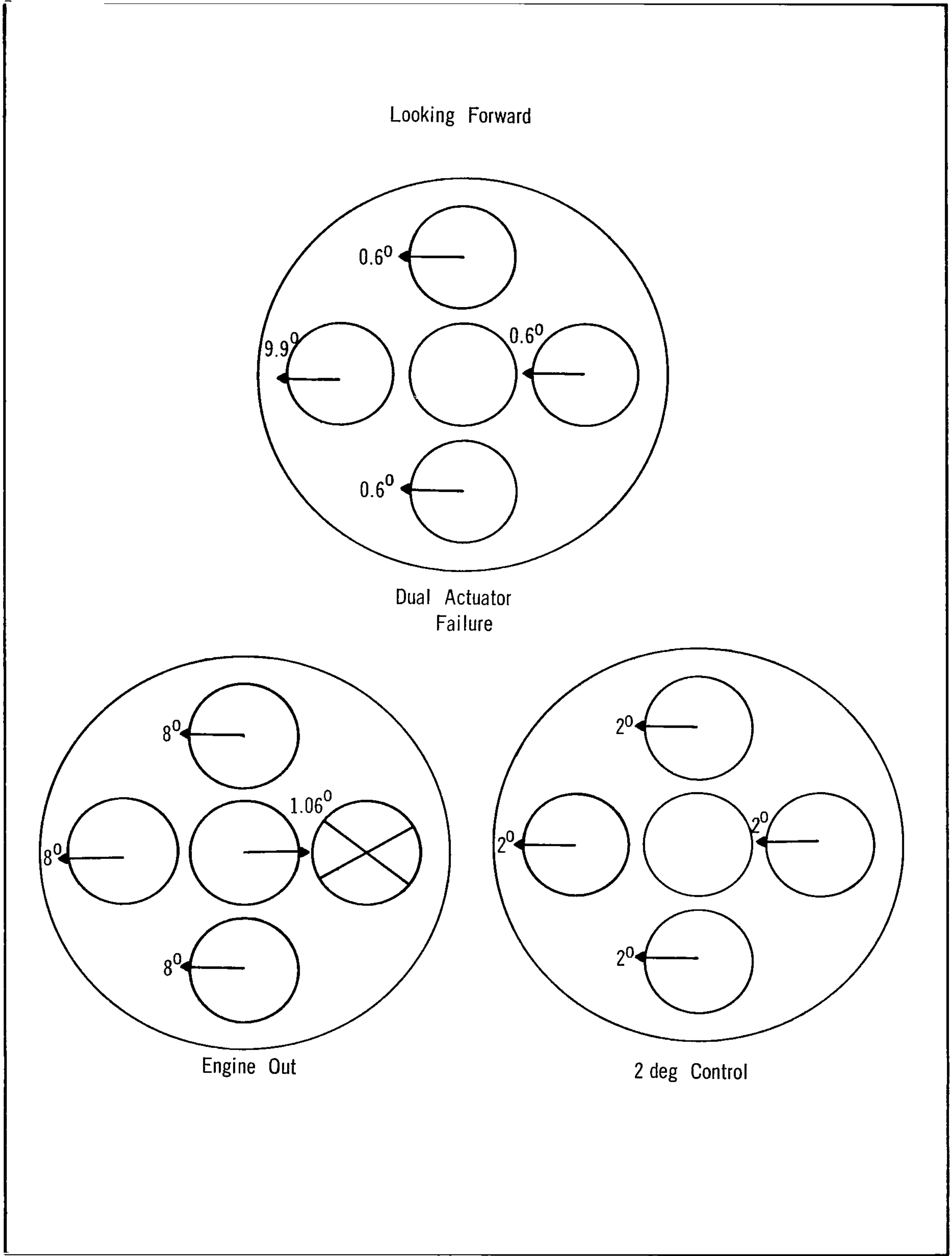

FIGURE 11. SCHEMATIC OF ENGINE DEFLECTION PATTERNS

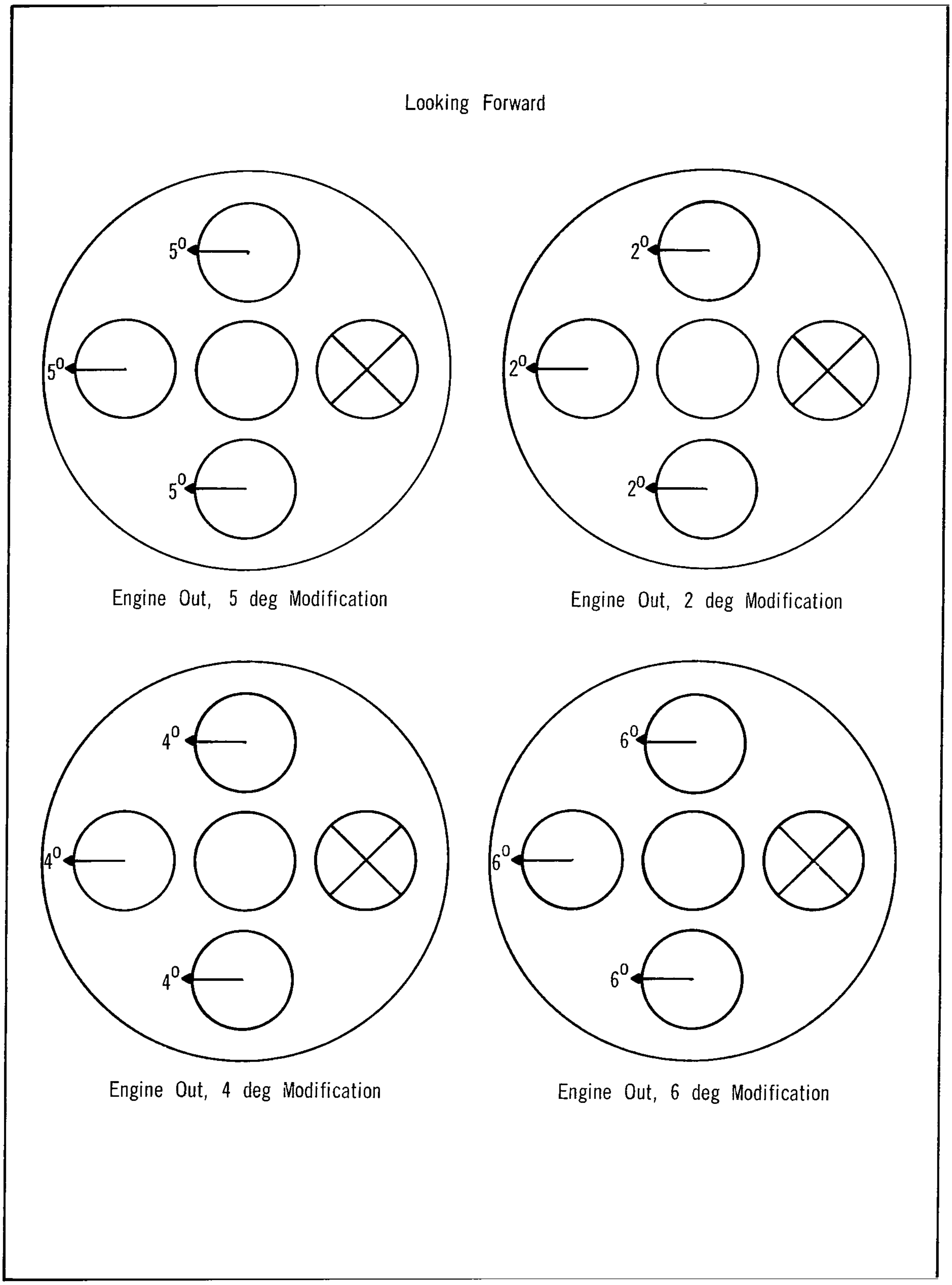

FIGURE 12. SCHEMATIC OF ENGINE DEFLECTION PATTERNS

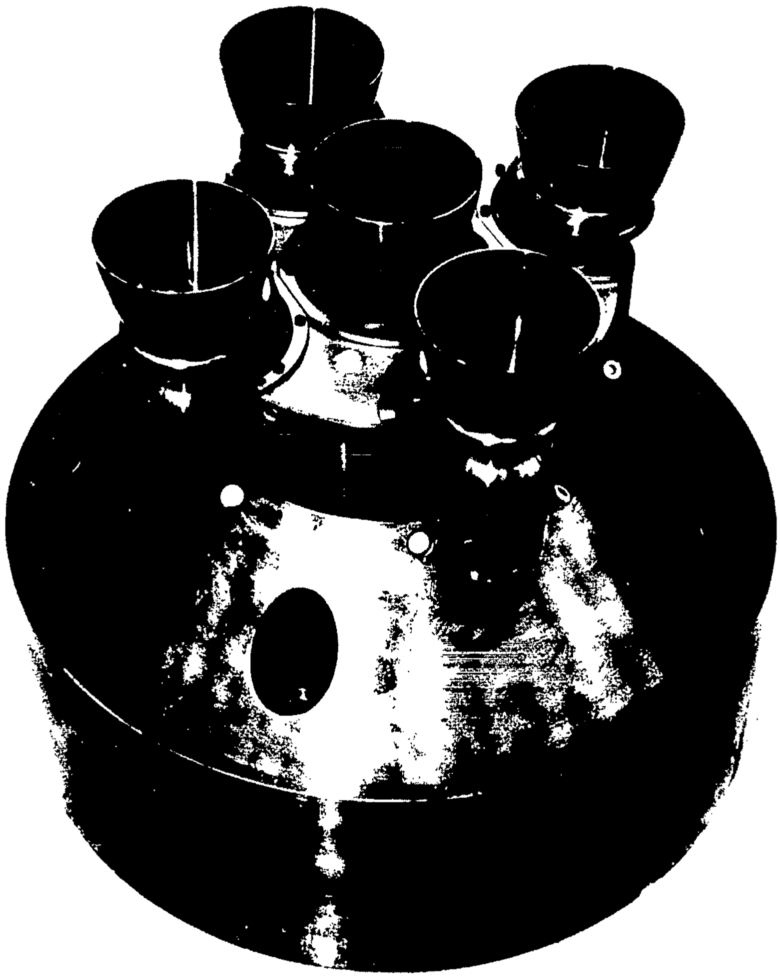

FIGURE 13. SATURN S-II MODEL BASE

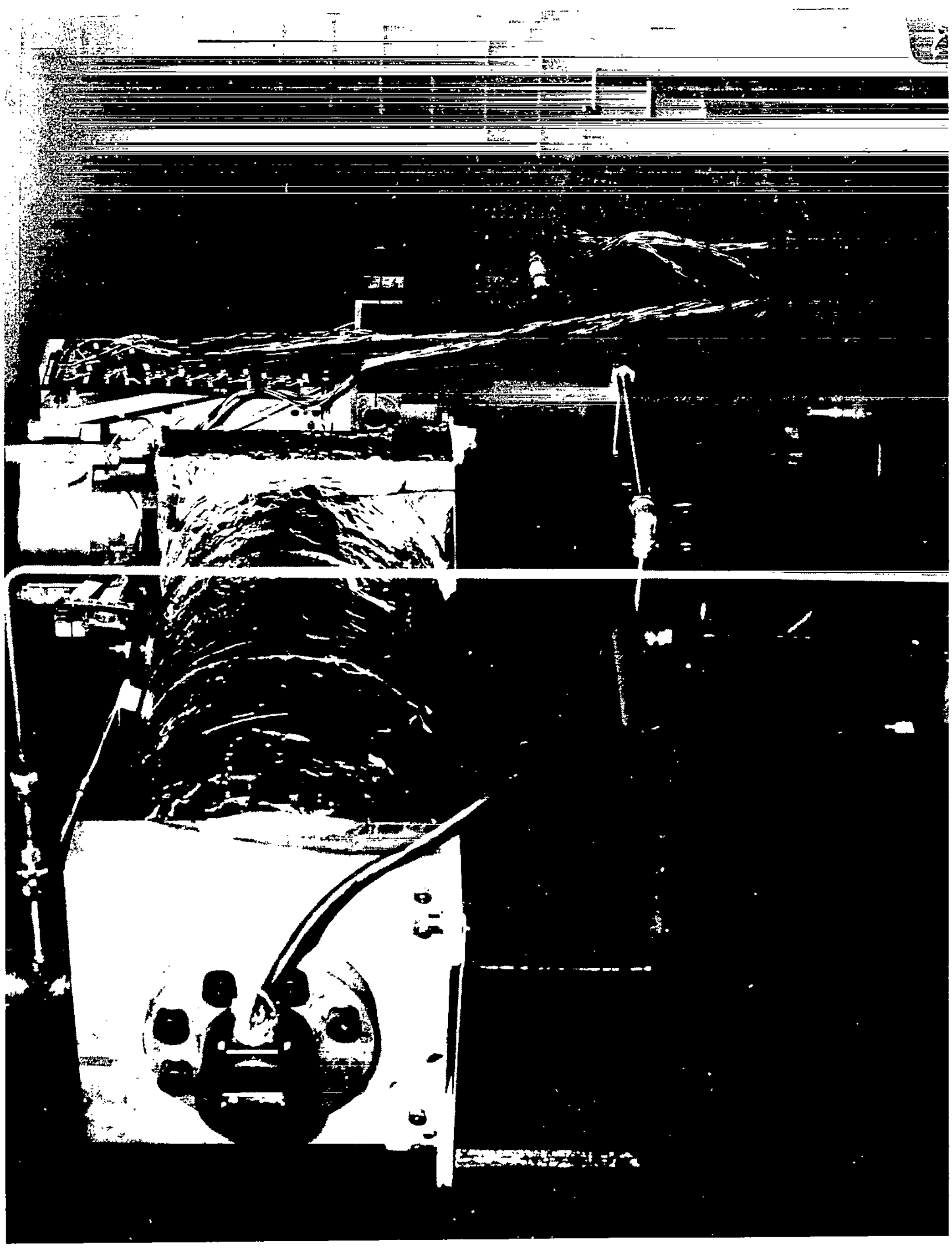

FIGURE 14. DPS TEST APPARATUS

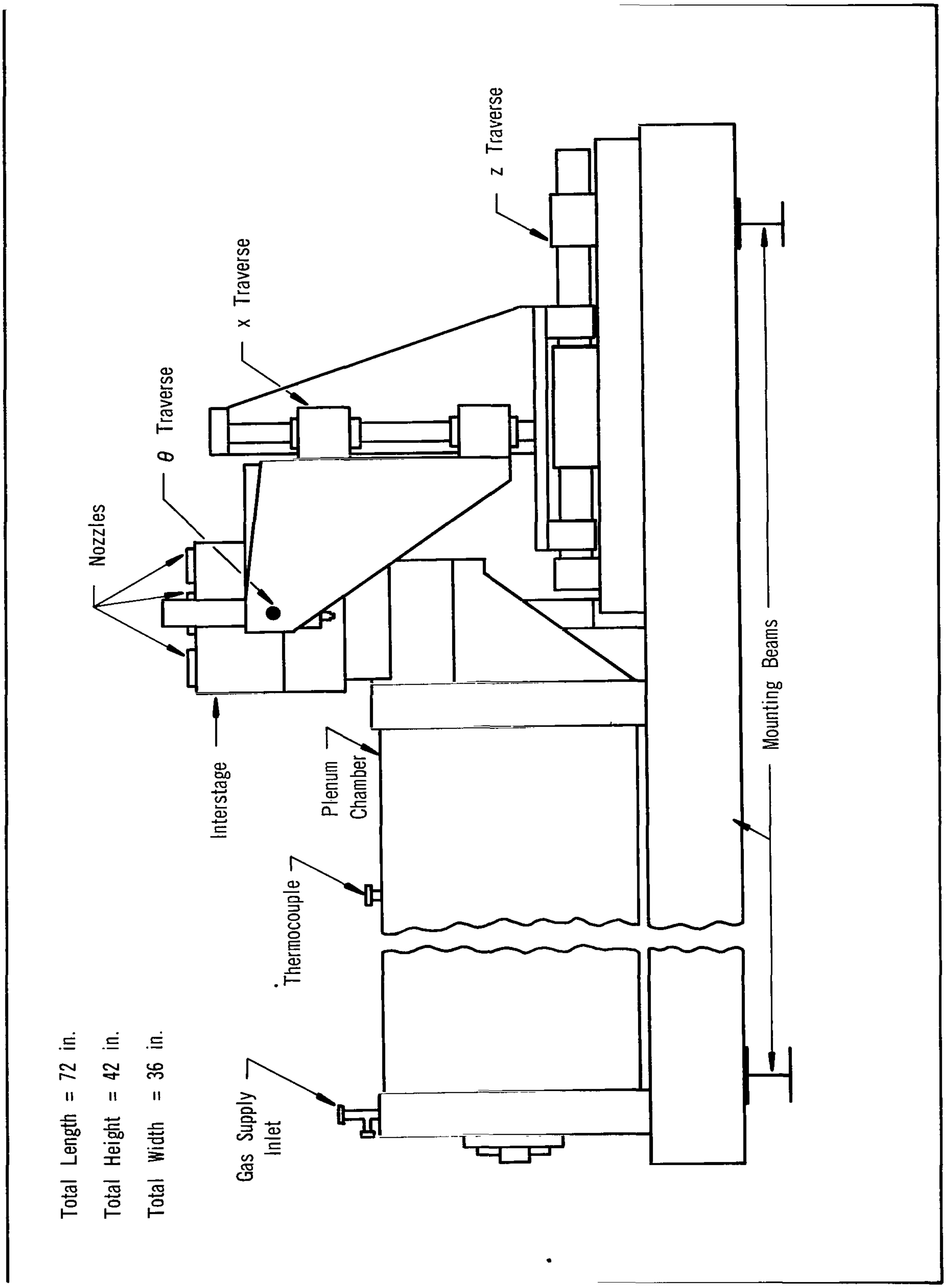

FIGURE 15. DUAL PLANE SEPARATION TEST SETUP

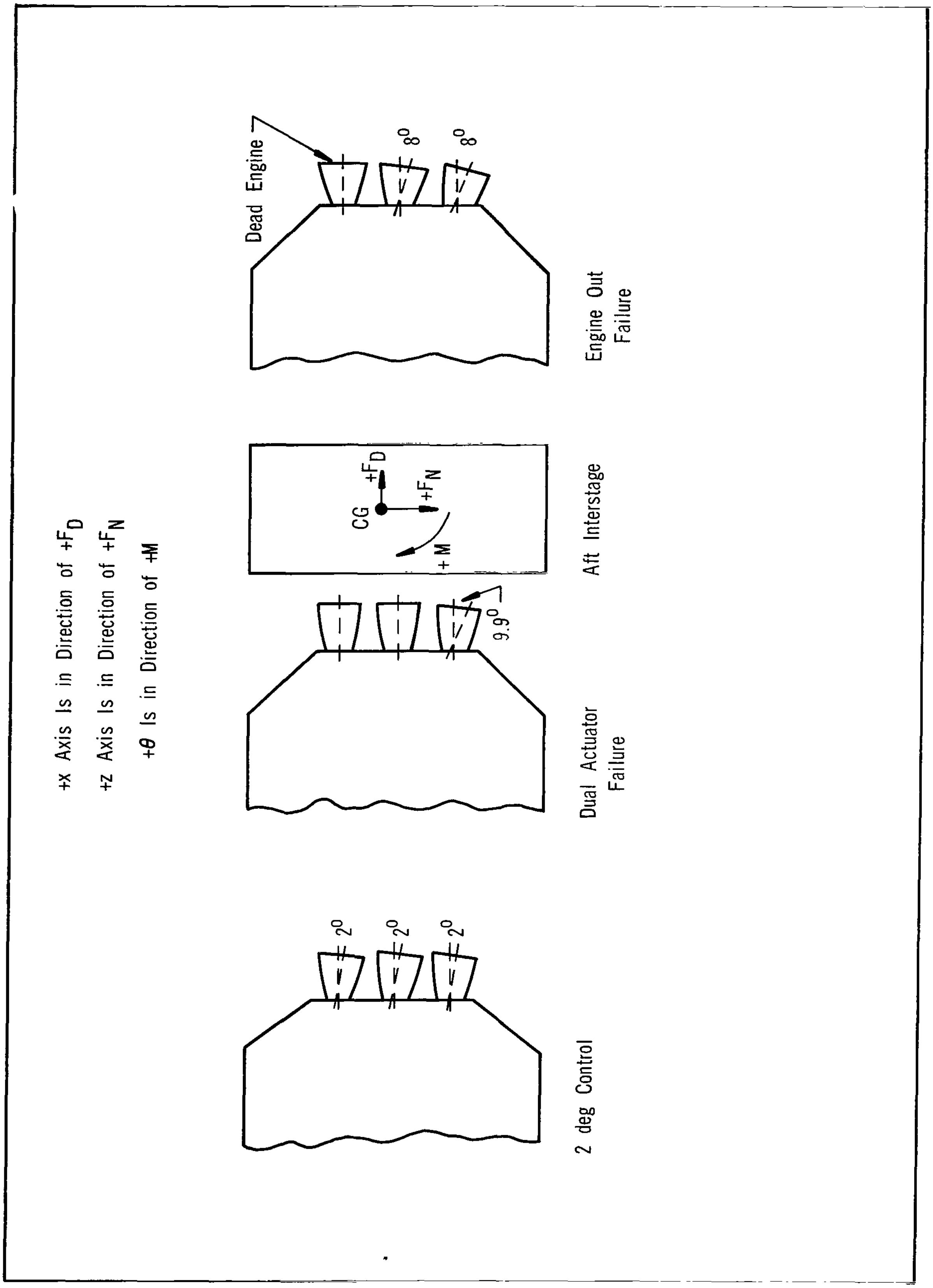

FIGURE 16. REFERENCE SYSTEM

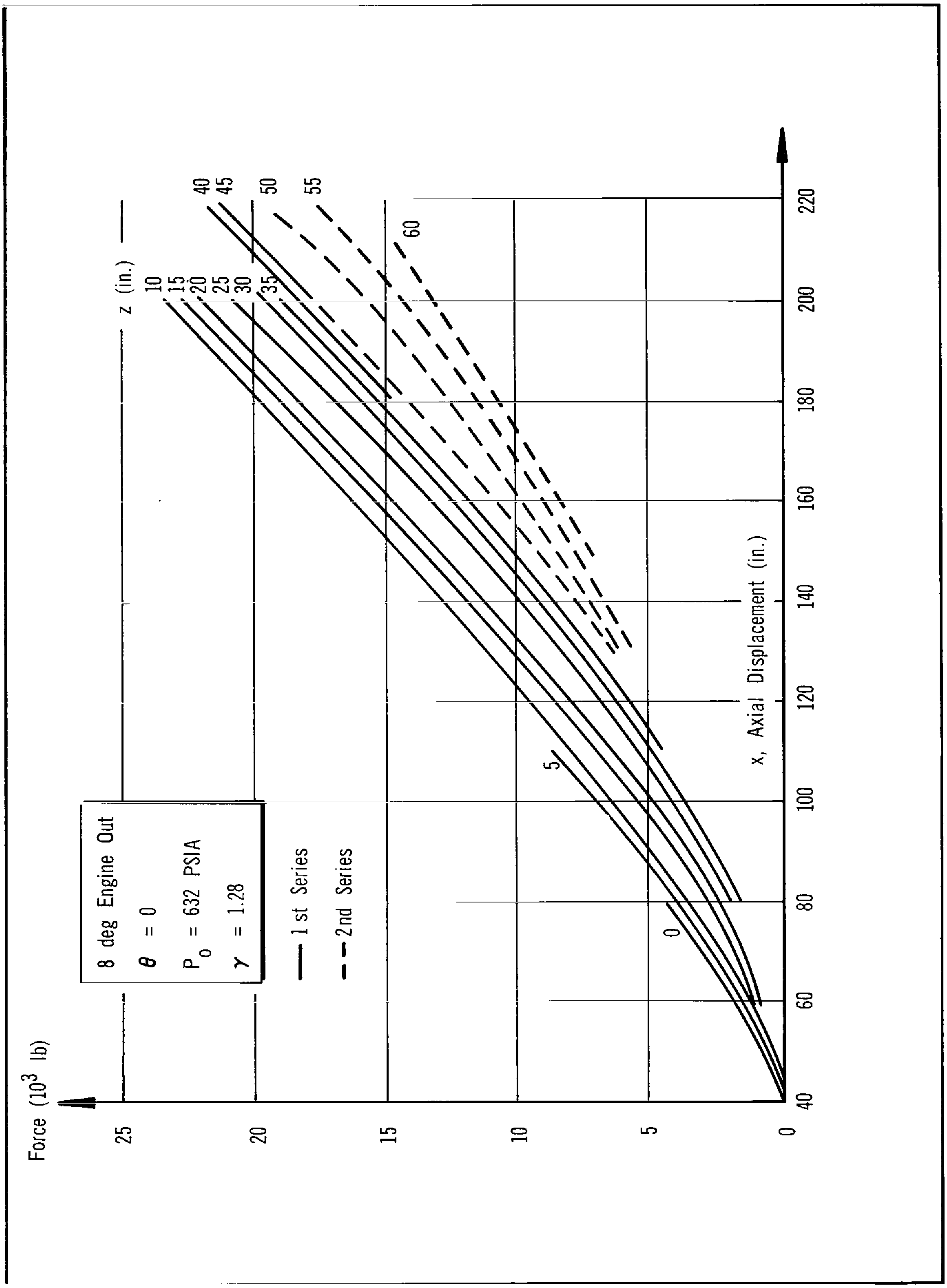

FIGURE 17. DRAG FORCE ON THE INTERSTAGE - ENGINE OUT FAILURE
(γ = 1.28, P_o = 632 psia)

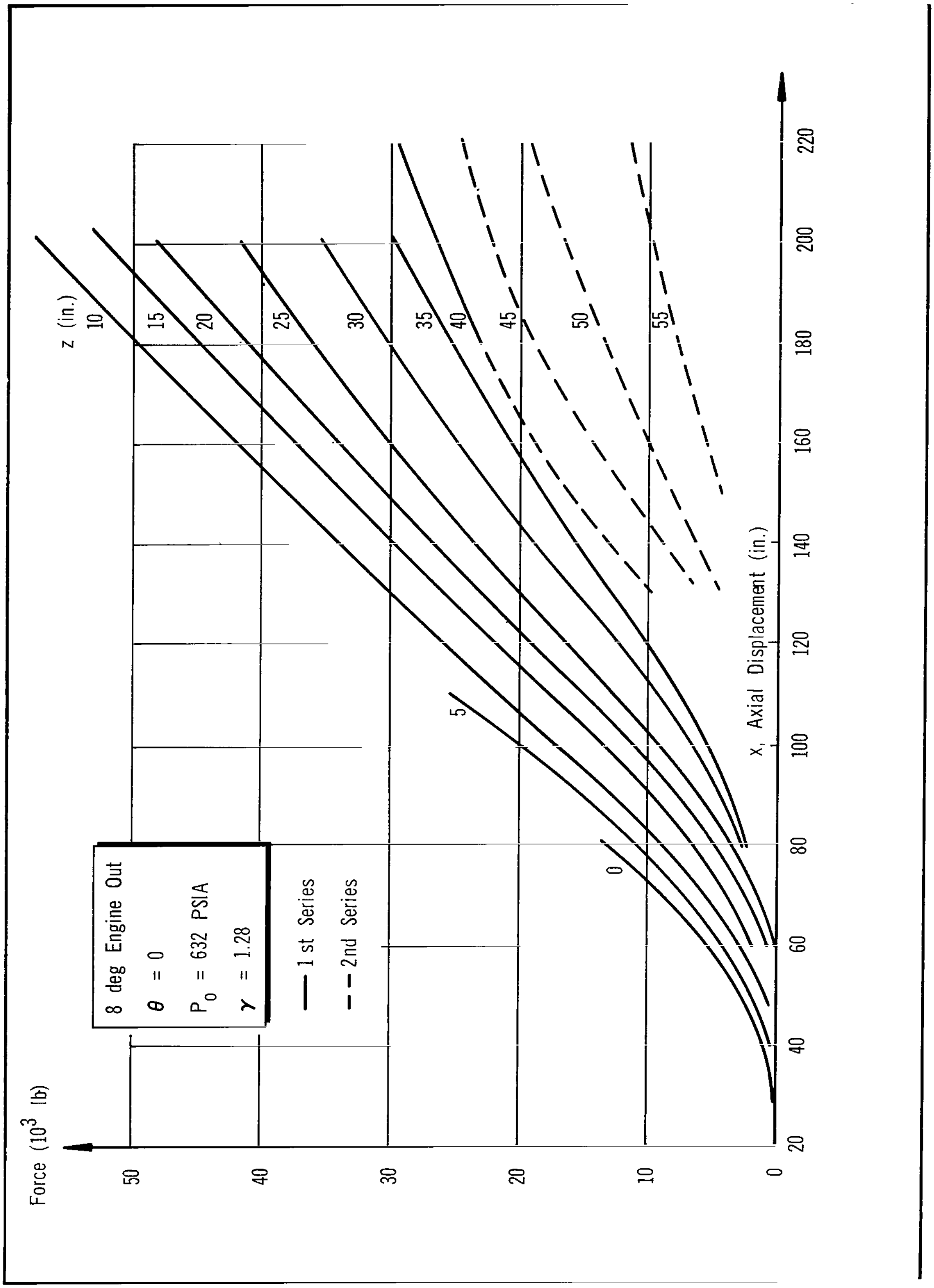

FIGURE 18. NORMAL FORCE ON THE INTERSTAGE - ENGINE OUT FAILURE (γ = 1.28, P_o = 632 psia)

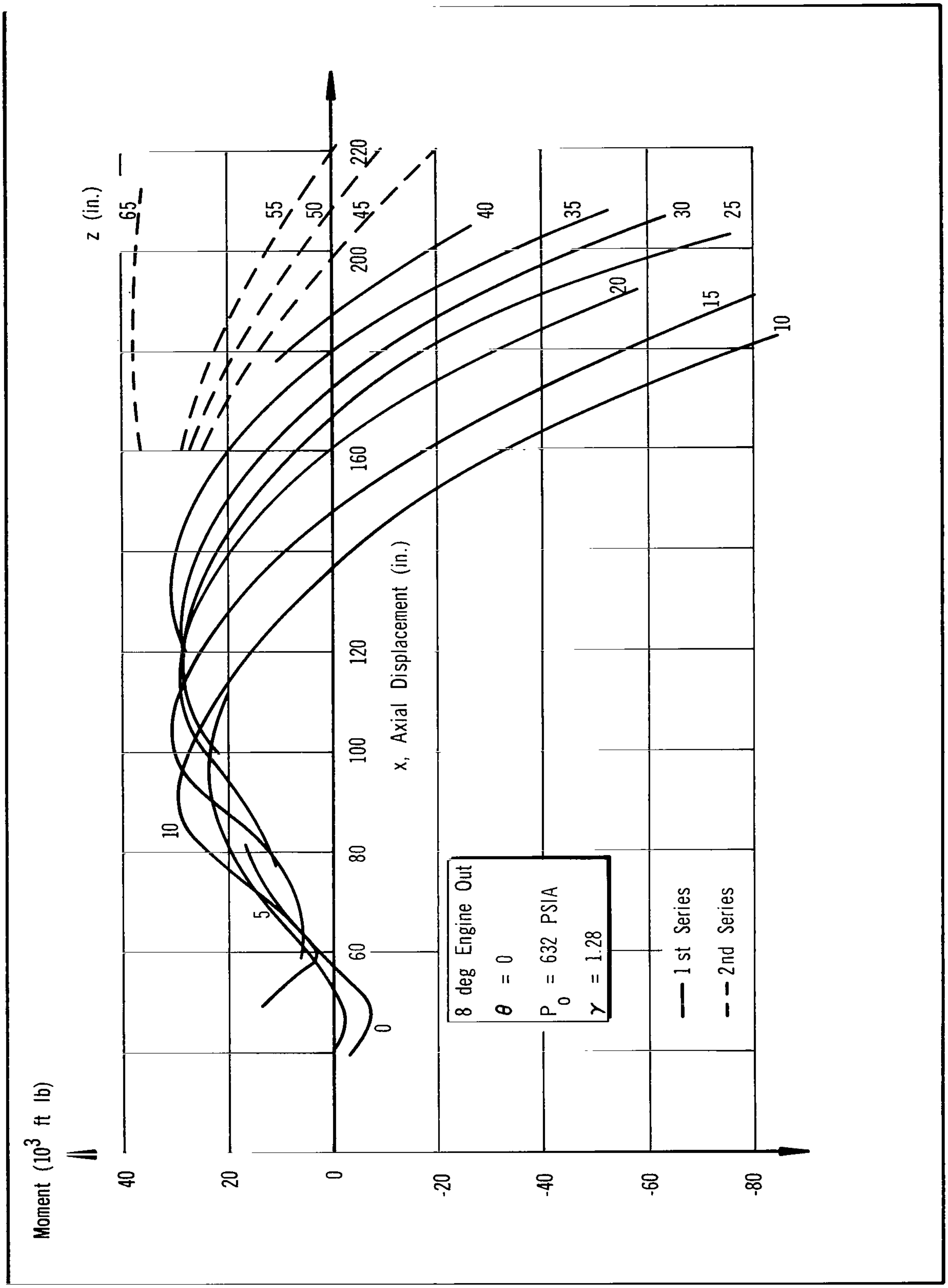

FIGURE 19. MOMENT ON THE INTERSTAGE - ENGINE OUT FAILURE
($\gamma = 1.28$, $P_o = 632$ psia)

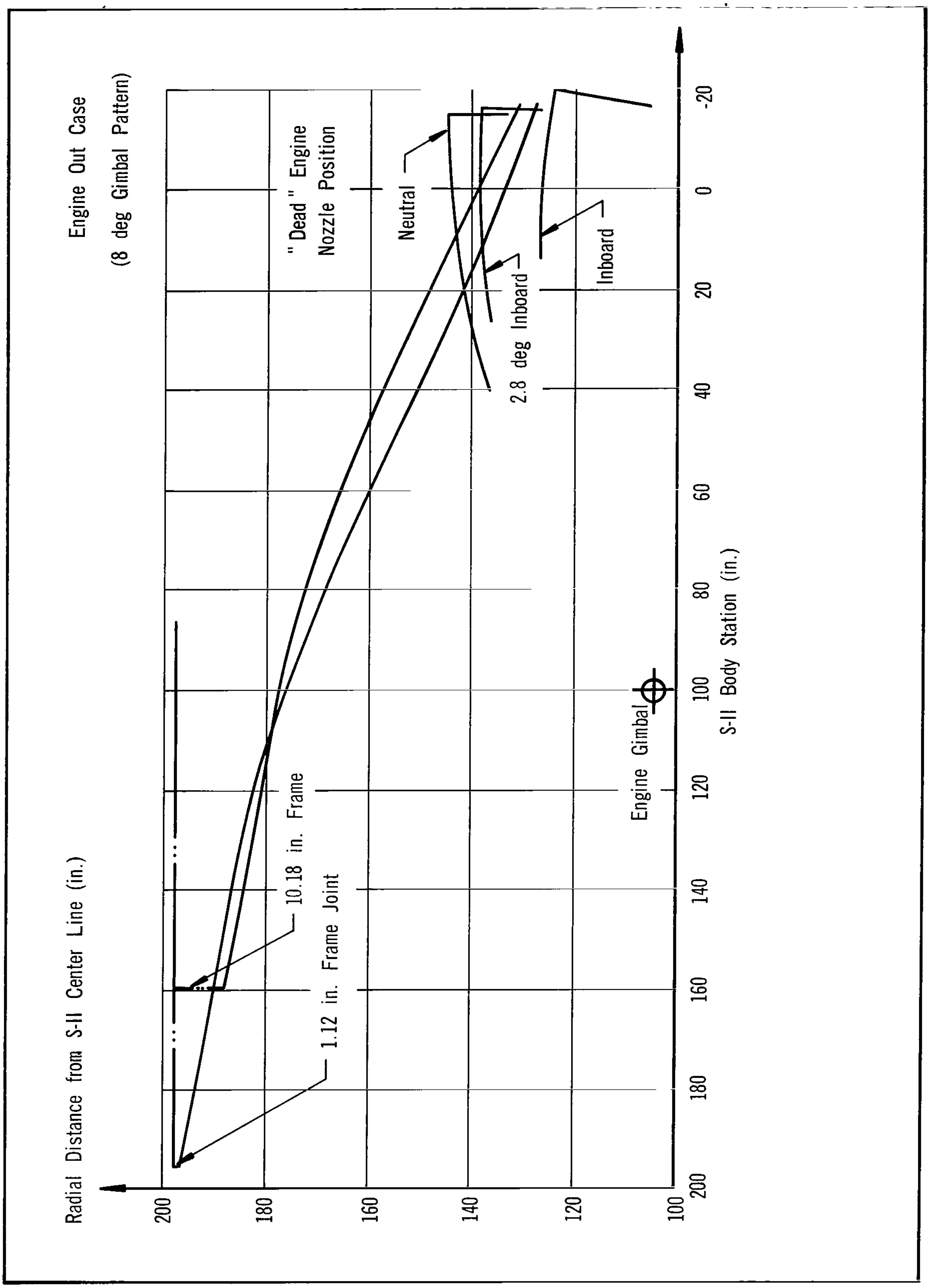

FIGURE 20. MOTION OF S-II/S-IC INTERSTAGE RELATIVE TO J-2 ENGINE NOZZLE

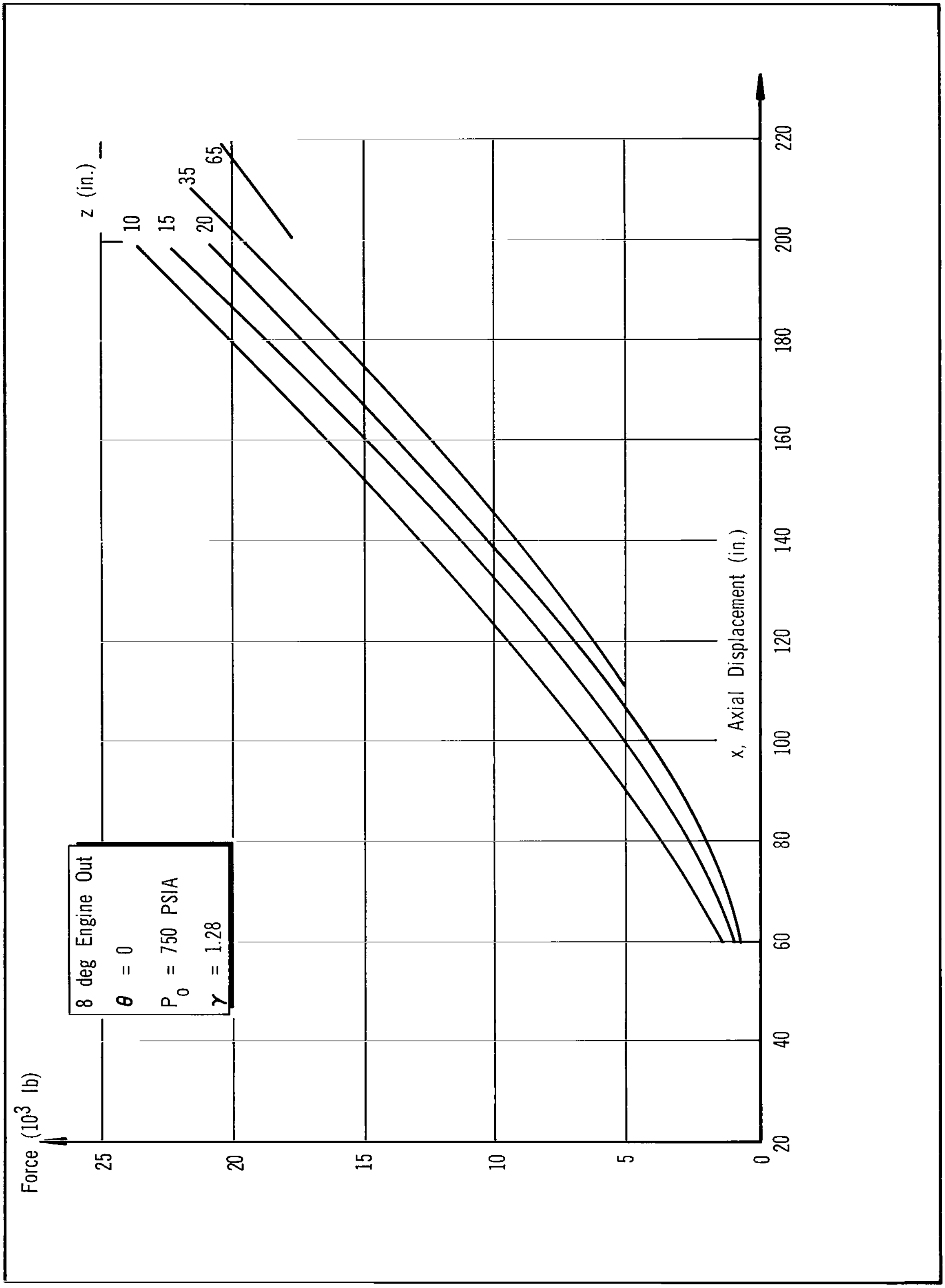

FIGURE 21. DRAG FORCE ON THE INTERSTAGE - ENGINE OUT FAILURE
($\gamma = 1.28$, $P_o = 750$ psia)

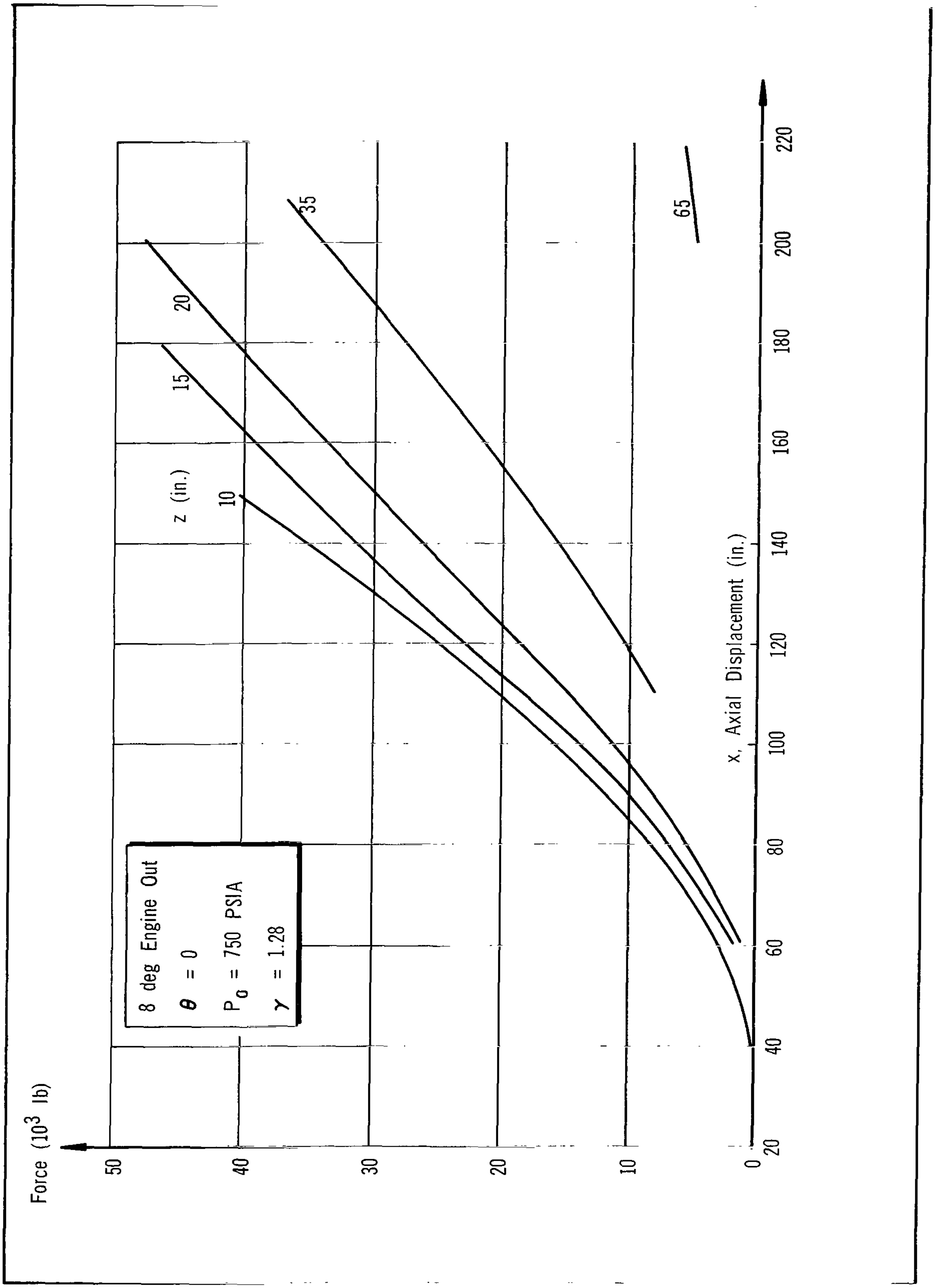

FIGURE 22. NORMAL FORCE ON THE INTERSTAGE - ENGINE OUT FAILURE (γ = 1.28, P_o' = 750 psia)

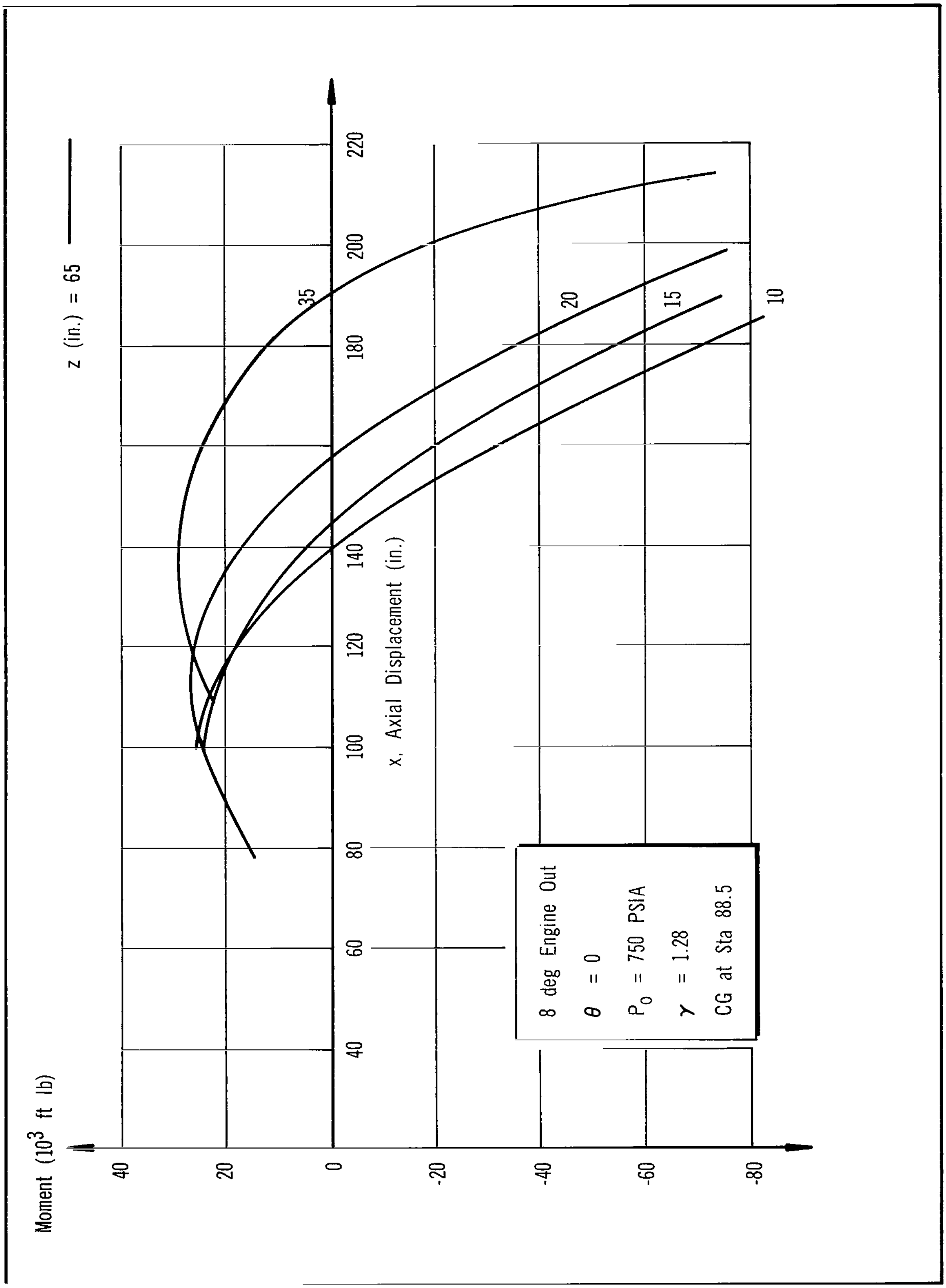

FIGURE 23. MOMENT ON THE INTERSTAGE - ENGINE OUT FAILURE
($\gamma = 1.28$, $P_o = 750$ psia)

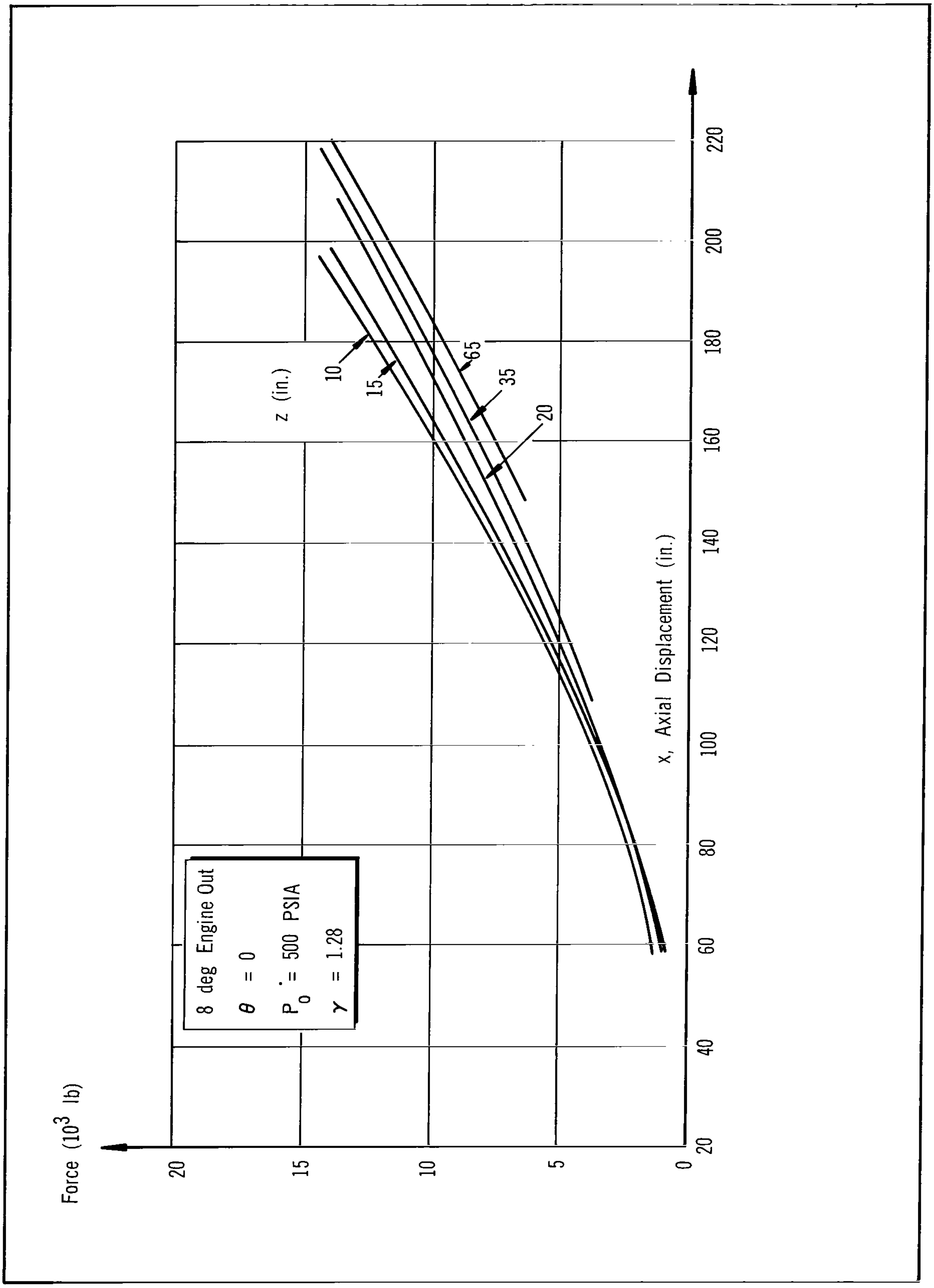

FIGURE 24. DRAG FORCE ON THE INTERSTAGE - ENGINE OUT FAILURE
(γ = 1.28, P_o = 500 psia)

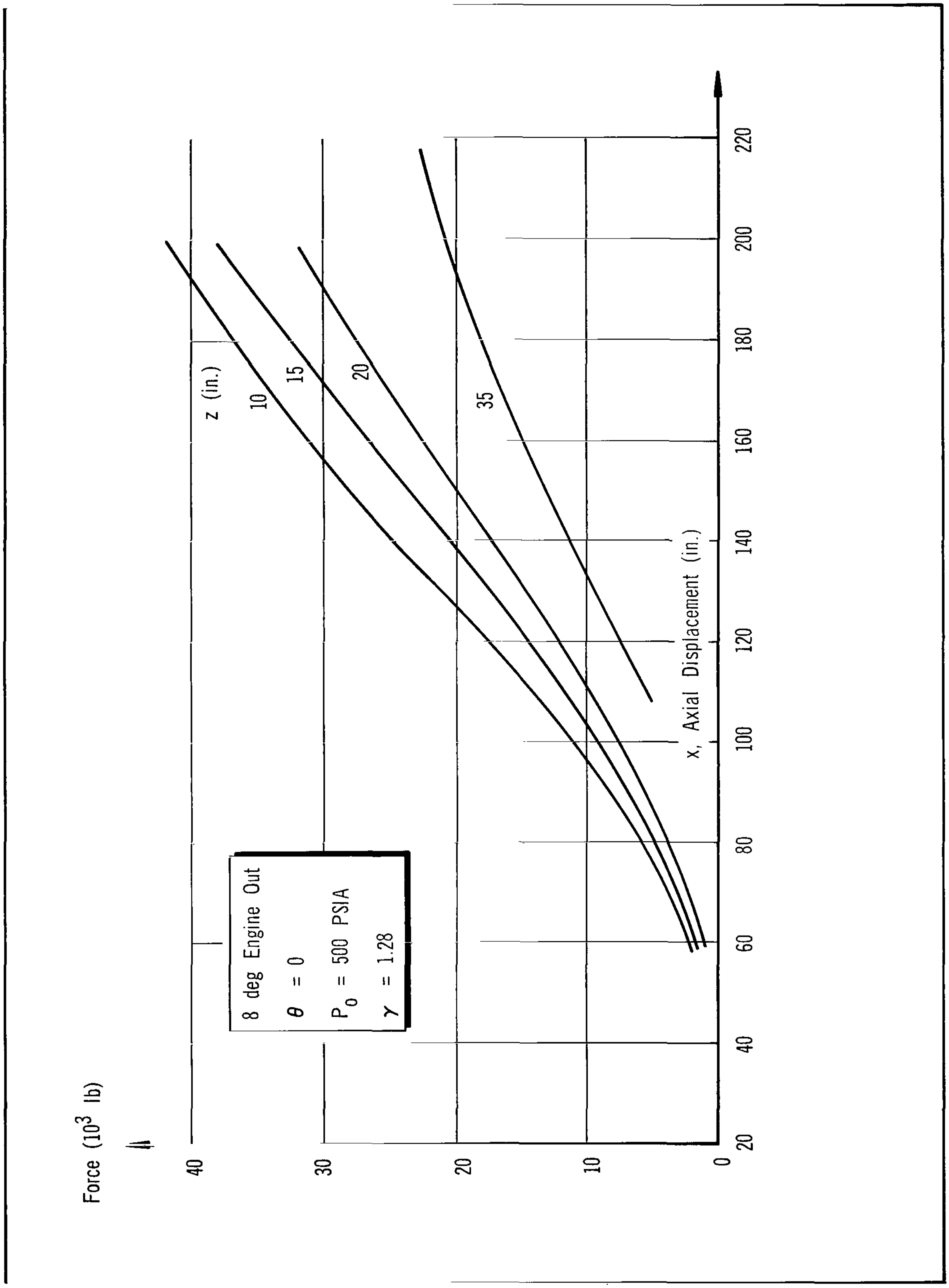

FIGURE 25. NORMAL FORCE ON THE INTERSTAGE - ENGINE OUT FAILURE
(γ = 1.28, P_o = 500 psia)

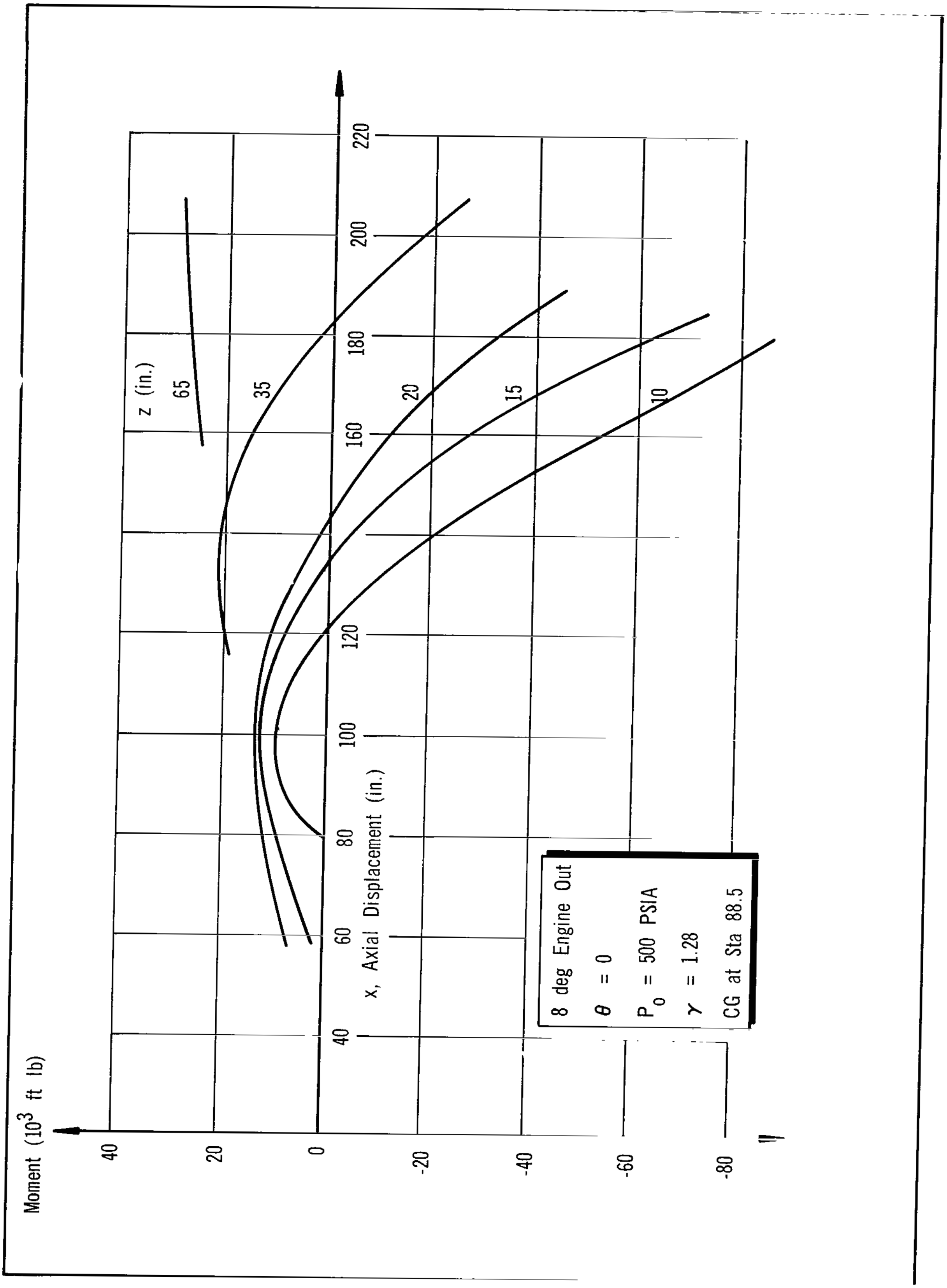

FIGURE 26. MOMENT ON THE INTERSTAGE - ENGINE OUT FAILURE
($\gamma = 1.28$, $P_o = 500$ psia)

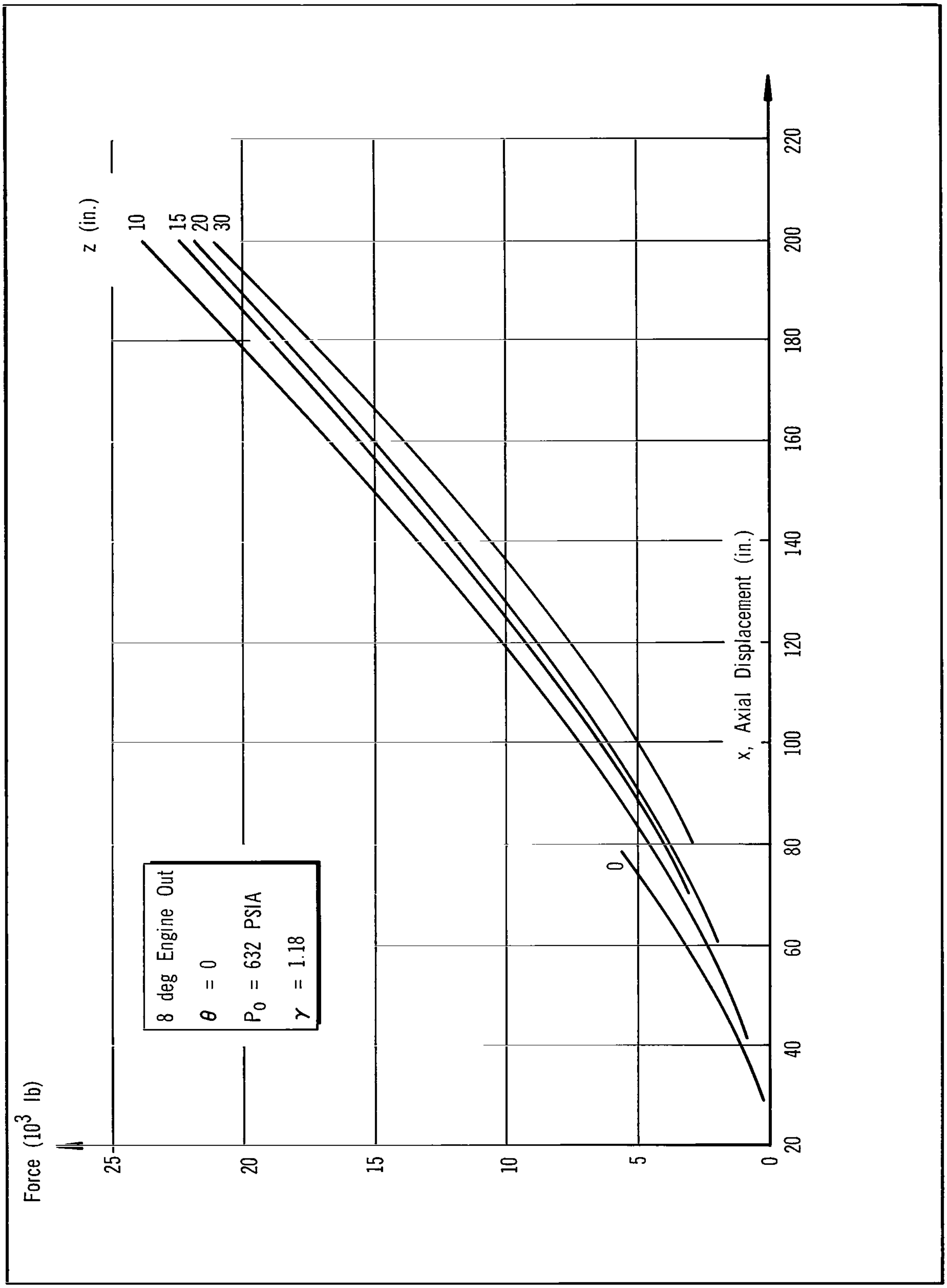

FIGURE 27. DRAG FORCE ON THE INTERSTAGE - ENGINE OUT FAILURE
(γ = 1.18, P_o = 632 psia)

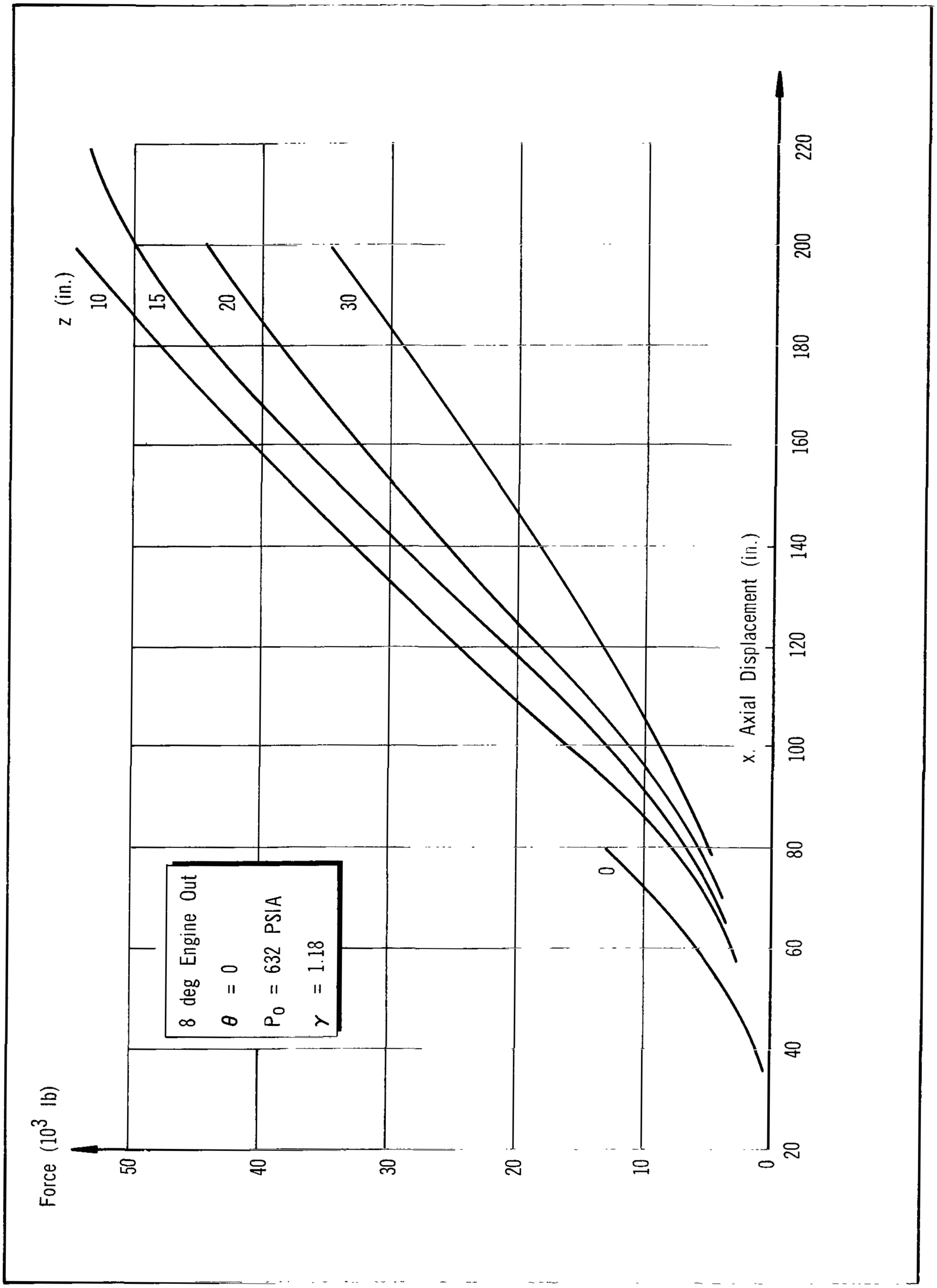

FIGURE 28. NORMAL FORCE ON THE INTERSTAGE - ENGINE OUT FAILURE ($\gamma = 1.18$, $P_o = 632$ psia)

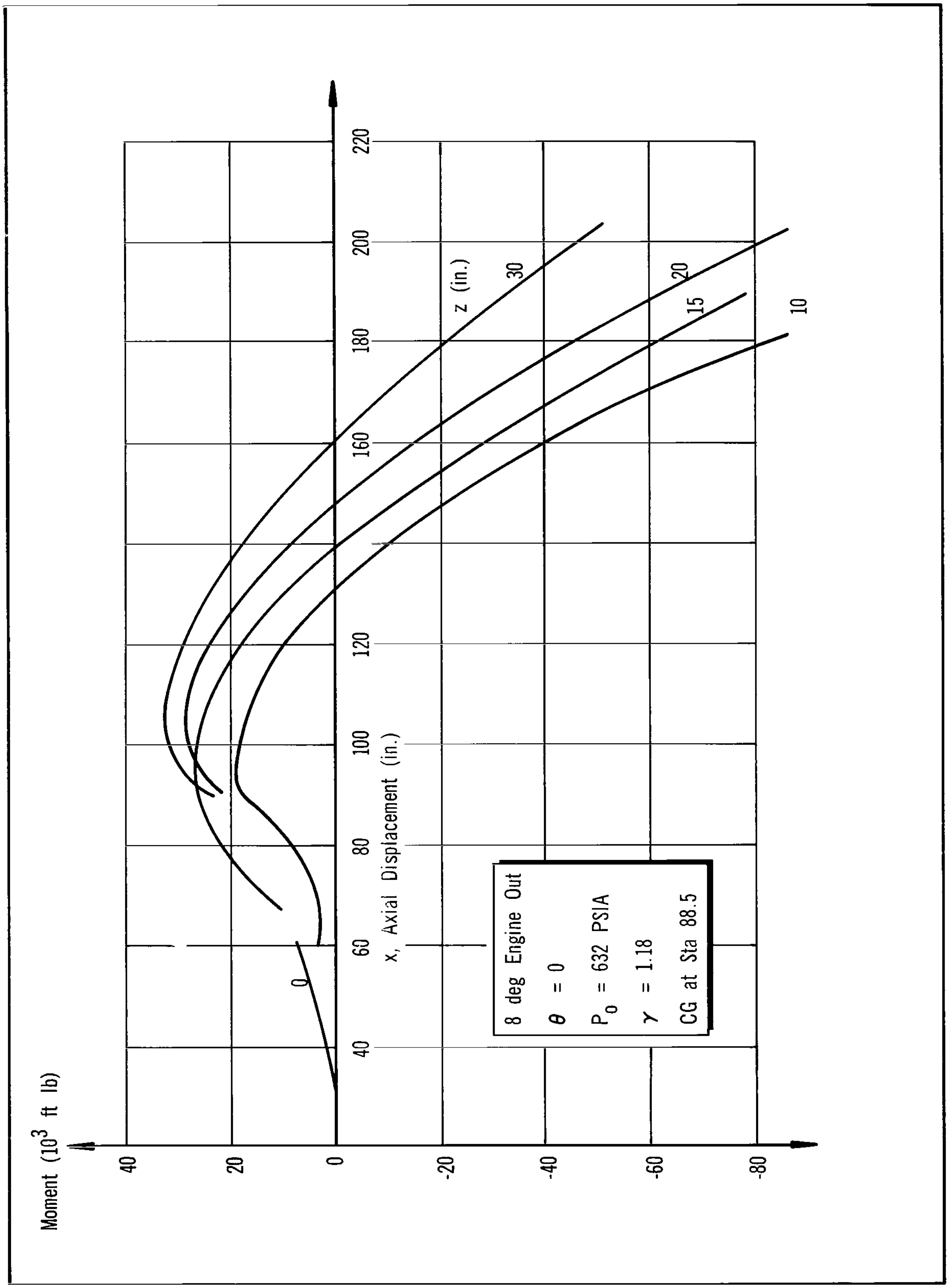

FIGURE 29. MOMENT ON THE INTERSTAGE - ENGINE OUT FAILURE
($\gamma = 1.18$, $P_o = 632$ psia)

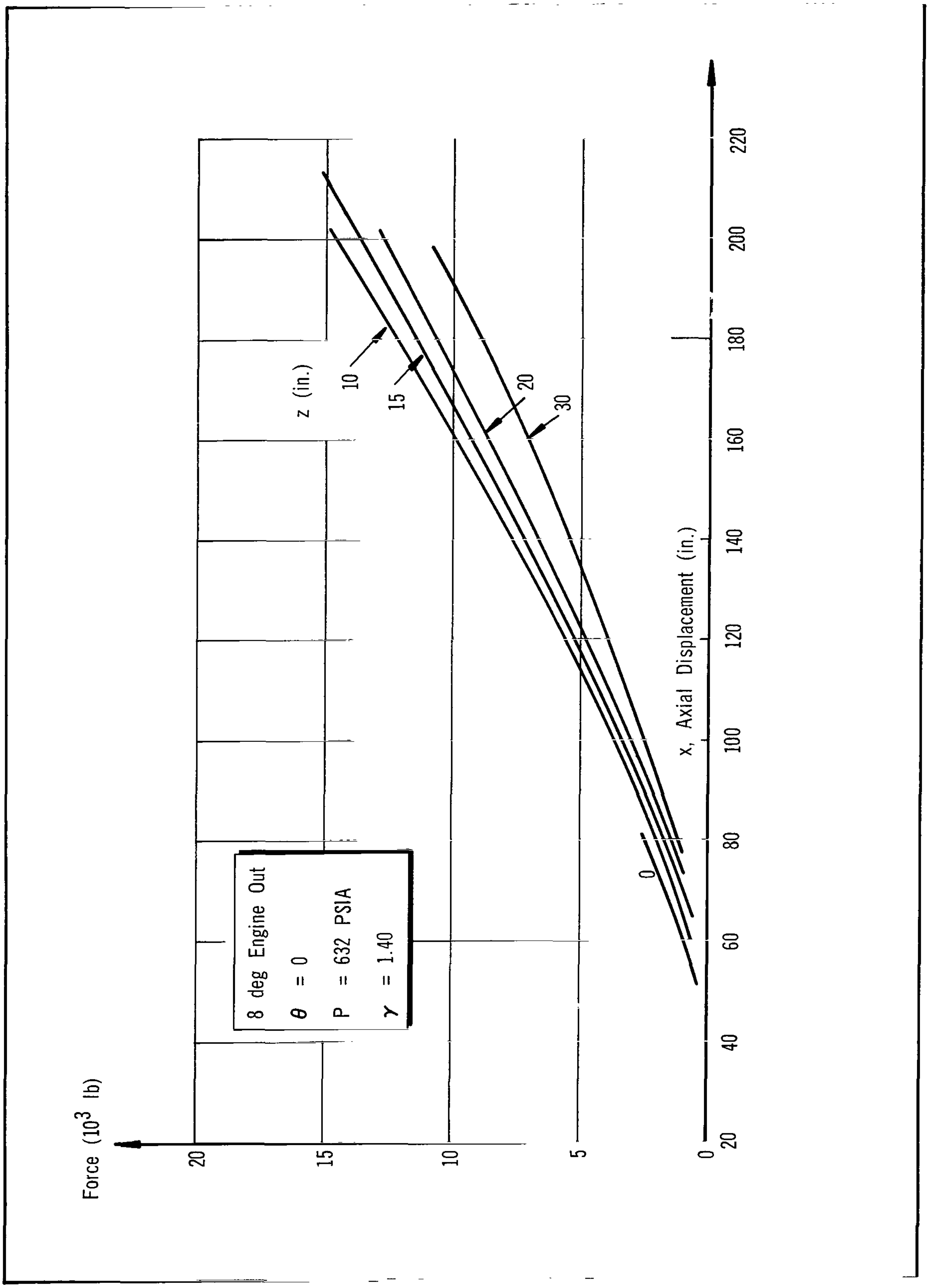

FIGURE 30. DRAG FORCE ON THE INTERSTAGE - ENGINE OUT FAILURE ($\gamma = 1.40$, $P_o = 632$ psia)

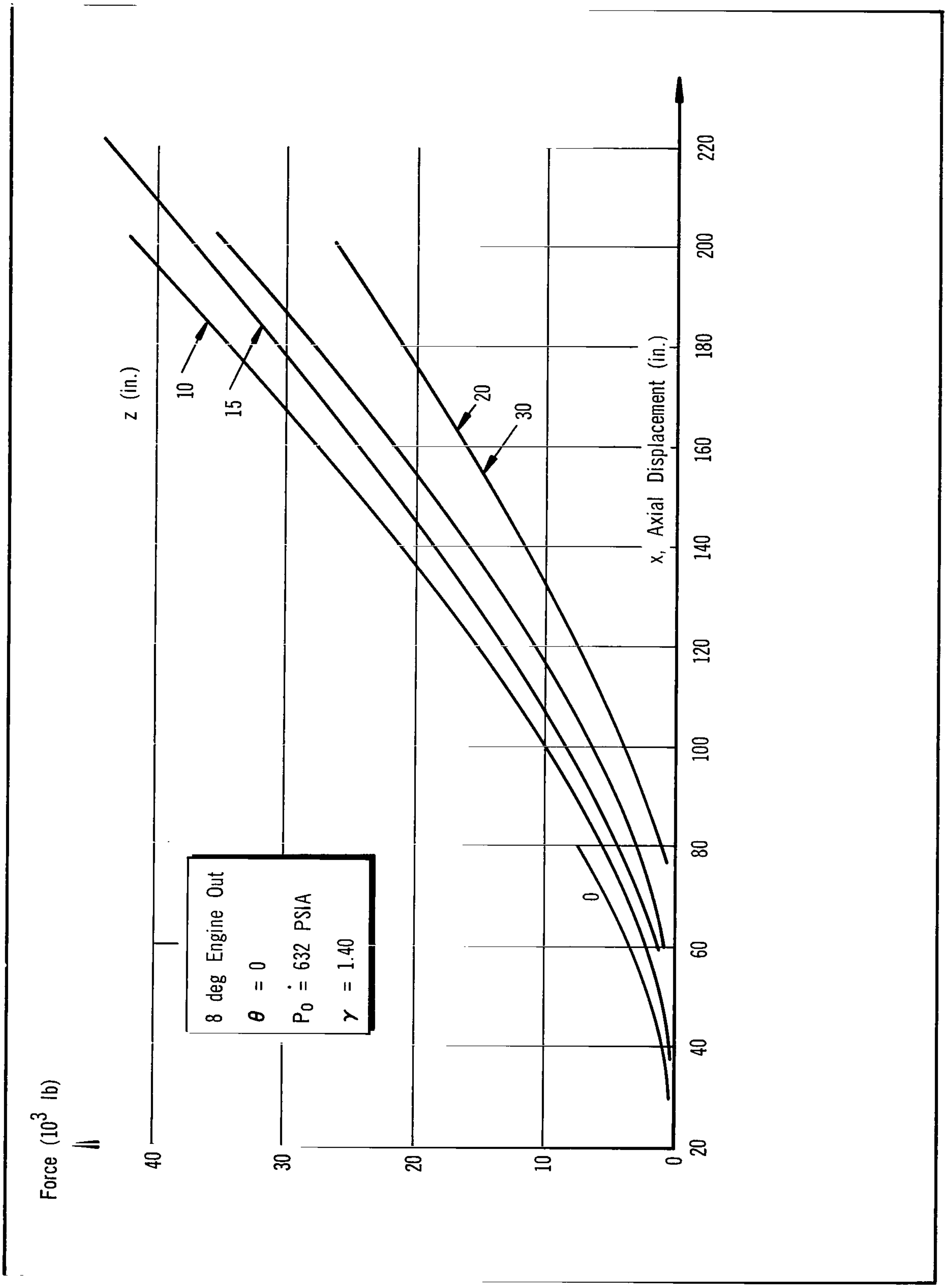

FIGURE 31. NORMAL FORCE ON THE INTERSTAGE - ENGINE OUT FAILURE (γ = 1.40, P_o = 632 psia)

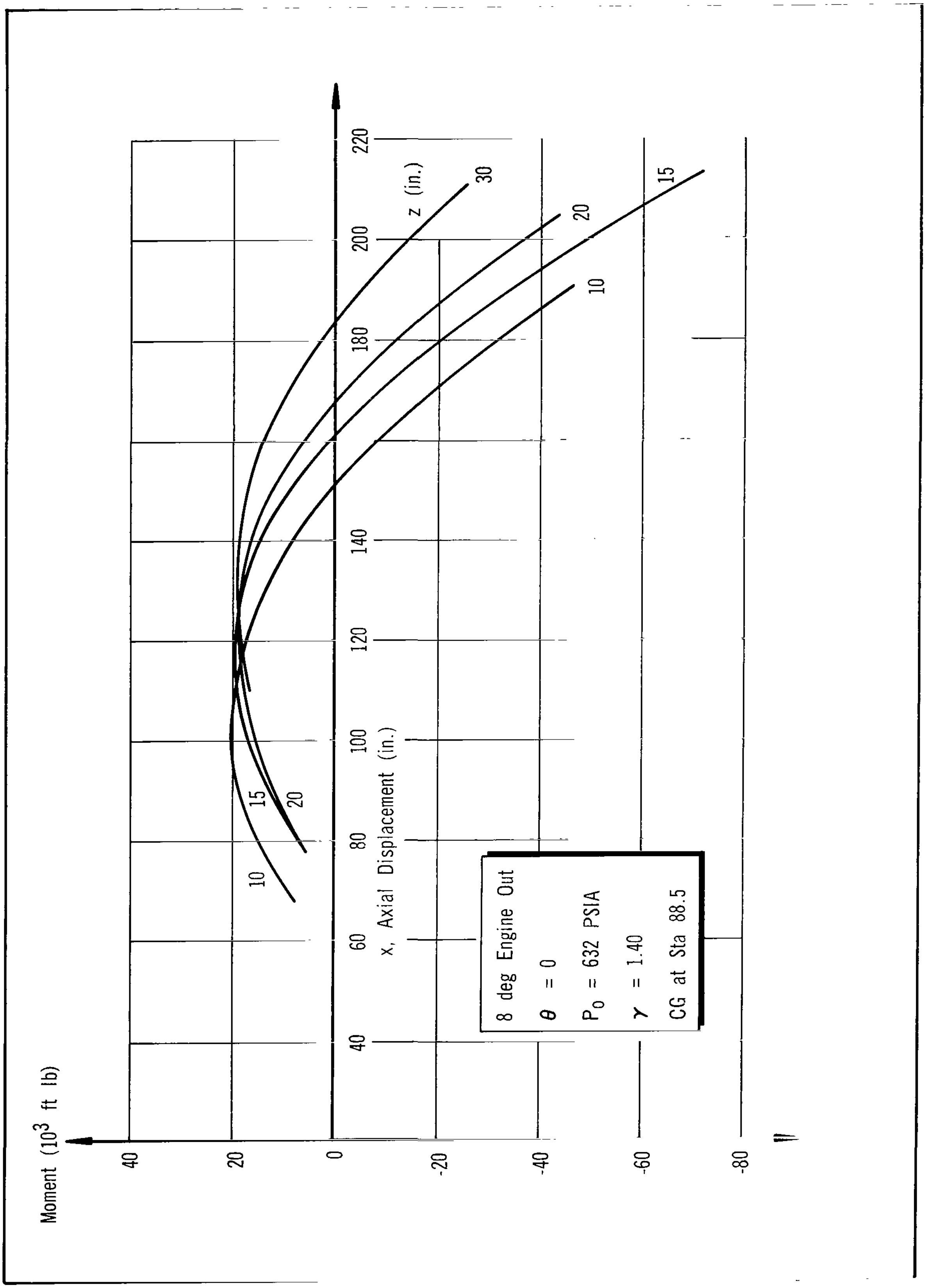

FIGURE 32. MOMENT ON THE INTERSTAGE - ENGINE OUT FAILURE (γ = 1.40, P_o = 632 psia)

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