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AUGUST 1969

A DESIGN STUDY FOR A THERMIONIC REACTOR POWER SYSTEM FOR A NUCLEAR ELECTRIC PROPELLED UNMANNED SPACECRAFT

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COVERING THE PERIOD 5 MAY 1969 TO 5 AUGUST 1969

AUGUST 20, 1969

PREPARED UNDER CONTRACT JPL 952381
FOR
THERMIONIC REACTOR SYSTEMS PROJECT

PROPULSION RESEARCH AND ADVANCED CONCEPTS SECTION
JET PROPULSION LABORATORY
4800 OAK GROVE DRIVE
PASADENA, CALIFORNIA, 91103

1 of 137

NUCLEAR SYSTEMS PROGRAMS
ISOTOPE POWER SYSTEMS OPERATION

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THIS WORK WAS PERFORMED FOR THE JET PROPULSION LABORATORY, CALIFORNIA INSTITUTE OF TECHNOLOGY AS SPONSORED BY THE NATIONAL AERONAUTICS AND SPACE ADMINISTRATION UNDER CONTRACT NAS7-100

ISOTOPE POWER SYSTEMS OPERATION
GENERAL ELECTRIC
SPACE DIVISION

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This report contains information prepared by the General Electric Company under JPL subcontract. Its content is not necessarily endorsed by the Jet Propulsion Laboratory, California Institute of Technology, or the National Aeronautics and Space Administration.

1. INTRODUCTION

A design study program of thermionic reactor power systems for nuclear electric propelled, unmanned spacecraft was initiated by the General Electric Company on February 4, 1969 for the Jet Propulsion Laboratory under Contract Number JPL 952381. The purpose of this program is to provide designs of selected thermionic reactor power systems integrated with nuclear electric unmanned spacecrafts over the range of 70 to 500 kWe unconditioned power. The key design objective is a weight of 10,000 pounds, including reactor, shielding, structure, radiators, power conditioning, and thruster subsystems at a 300 kW(e) unconditioned power level. Spacecraft propulsion will be provided by mercury electron bombardment ion thruster engines.

The program is divided into five principal tasks:

a. Task 1 - System Requirements and Evaluation - The purpose of this task is to establish program guidelines, program functional design requirements and system evaluation criteria.

b. Task 2 - Spacecraft Design - The purpose of this task is to prepare basic spacecraft designs for a Jupiter orbiter mission. Preliminary design layouts of the major spacecraft components and structural analyses of the supporting structure will also be included in this task.

c. Task 3 - Power Plant Design - The purpose of this task is to design and optimize the thermionic reactor power plants for each of three candidate reactor concepts (Gulf-General Atomics, General Electric, and Fairchild-Hiller).

d. Task 4 - System Analysis Development - The purpose of this task is to develop the necessary analytical procedures and computer codes required to conduct power plant design and optimization calculations and to perform parametric studies.

e. Task 5 - Mission Engineering - The purpose of this task is to prepare preliminary definitions of pre-launch, launch and mission operations, and to assess the impact of aerospace nuclear safety requirements upon power plant design.

The design study is performed in two consecutive phases:

a. Phase I - Design of unmanned spacecraft configurations, including power plants, for each of the three candidate thermionic reactor concepts. Key ground rules include:

- 300 kWe unconditioned power
- NaK-78 coolant
- 1350°F reactor outlet temperature
- Copper-stainless steel conduction fin radiators
- 200°F maximum electronic component temperature limits
- 10,000 pounds power plant weight (design objective)
- 10,000 to 15,000 full power hours.

b. Phase II - Investigation of the effect of key parameters on power plant design:

1. Power level: 70 to 500 kWe
2. Coolant: substitution of lithium for NaK-78
3. Radiator type: the use of beryllium/stainless steel or vapor fin radiators
4. Extended life: 20,000 full power hours

Program effort is progressing well, and is currently on schedule. A key Phase I milestone has been completed with the selection of a conical radiator configuration, mounted in the upright position on the Titan IIIC/7 launch vehicle. The power conditioning subsystem definition has been completed for the bonded, wet cell flashlite reactor concept. The level of completion of the spacecraft weight optimization computer code exceeds 70 percent. Other Phase I results have been previously reported (Reference 1).

2. TECHNICAL DISCUSSION

2. TECHNICAL DISCUSSION

The presentation of the following material follows the program task structure as outlined in Section 1. The results of the study for this reporting period are summarized in Section 3, Conclusions, and Section 4, Recommendations.

2.1 SYSTEM REQUIREMENTS

Key program guidelines and fundamental design requirements have been defined and previously reported (Reference 1). This report alters one work element to be performed as follows:

- The effect on system weight of varying NaK-78 reactor outlet temperatures will investigate the temperature range of 1100 to 1600°F.

2.2 SPACECRAFT DESIGN - LAUNCH ORIENTATION AND SPACECRAFT STRUCTURAL REQUIREMENTS

The conical and triform radiator configuration, Point-of-Departure (or baseline) designs previously established (Reference 1) have been evaluated in terms of the additional structure required to survive the launch environments imposed by the Titan IIIC/7 launch vehicle. This evaluation was accomplished as a function of the spacecraft orientation on the launch vehicle, upright or inverted, and the utilization of the launch vehicle shroud as launch support structure. The preferred system is identified as the conical radiator, configured in the upright position on the launch vehicle. This configuration provides the maximum Initial Mass in Earth Orbit (IMEO), and therefore requires the minimum structural addition necessary to survive launch, relative to a spacecraft nominally optimized for thermal performance.

The results of this analysis will be employed in the weight optimization computer code being developed for this study (Paragraph 2.6). Particular structural requirements for the three spacecraft to be defined will be separately evaluated.

2.2.1 LAUNCH ENVIRONMENT

2.2.1.1 Titan IIIC Launch Environment

Discussions with Martin Denver, Titan III Structural Dynamics Loads group, indicated that the Stage I burnout event, including the transient oscillation "Pogo" condition just prior to burnout, results in the most severe upper vehicle-spacecraft loadings. The Stage II burnout event was, in their experience, a less severe loading condition. However, they have no existing payloads of comparable size and flexibility. Both the conical radiator and the triform radiator spacecraft will be required to survive the Stage I burnout loads. However, in order to permit early separation of the supporting truss that is required for the triform radiator spacecraft, it will be required to survive the less severe Stage II burnout loads without a separate supporting truss. This approach permits the truss to be jetisoned at the time of shroud separation, normally accomplished just prior to Stage II burnout, at 280 seconds after launch. The truss weight penalty, in terms of IMEO, is then proportional to that of the shroud. The quasi-steady state load environments are therefore defined as:

• Cylindrical-conical vehicle (Stage I burnout)	3g lateral 6g longitudinal
• Triform vehicle - unsupported (Stage II burnout)	0.67g lateral 4g longitudinal
- with launch support truss (Stage I burnout)	3g lateral 6g longitudinal

When informed that the spacecraft first lateral mode would be below the desired 6 cps, Martin Denver indicated that this may present some problems; but the launch vehicle could be designed around them. This problem was also present on the MOL program, where the Titan IIIM was designed to accommodate 3 cps. The principal problem is that a low spacecraft natural frequency would couple the spacecraft directly with the booster during launch and ascent. This dynamic coupling manifests itself in two ways: first, it creates problems with the autopilot stability, which would be most critical during Stage "0" flight; and second, although the gravity loads given would not be increased, the dynamic effects would be experienced more often during flight rather than in the normal case where the maximum response is seen as a transient at the time of Stage I burnout.

2.2.1.2 Shroud Attachment

The use of a "snubbing" technique to transmit 5000 pound loads laterally from the payload structure to the shroud is possible for a 60 by 10 ft diameter shroud configuration. This load would be taken 60 feet from the interface adapter, and is dumped into the shroud to limit the deflection of payload structure. A major consideration in this approach is that the launch probability (WTR) is reduced when based on the worst quarter winds or annual winds. The airload on the shroud is the major portion of the load encountered during boost phase. This airload can be altered by placarding, in which careful assessment of the environment, especially wind velocity up to 40,000 feet is made. This approach limits the days in which launching can be achieved.

2.2.1.3 <u>Wind Placarding</u>

Martin has performed a wind placarding study for 90 and 105 foot payload fairing configurations. The basic ground rules for each configuration were as follows:

<u>90 foot fairing on a Titan III/C</u>

- 850,000 pound ultimate P_{EQ} transtage
- 3500 pound payload
- Mach 1.4
- q_{α} = 4000 PSF-degree
- q = 800 PSF

<u>105 foot fairing on a Titan III/C</u>

- 1,080,000 pound ultimate P_{EQ} adapter skirt
- 3500 pound payload plus Agena
- Mach 1.4
- q_{α} = 4000 PSF-degree
- q = 800 PSF

Results of Martin study show that for each configuration, the percent of maximum design wind velocity that may be flown is approximately 53 percent. From this, the probability of launch from WTR is 72 percent for the worst quarter winds or 87 percent for annual winds.

2.2.2 CONFIGURATION SELECTION

Two basic vehicle configurations were investigated. One vehicle is a three section cylinder-conical assembly, 990.6 inches long. Details of the design and the loading diagram are shown in Figure 2-1. The seoond vehicle is a three section triform, 1155 inches long. Details of this configuration and a loading diagram are shown in Figure 2-2.

Five load/boundary conditions were analyzed for the cylinder-conical vehicle. The conditions are summarized below and are schematically shown in Figure 2-3. The combined dynamic launch environment is 3 g's lateral superimposed on 6 g's axial.

C1 - Upright unsupported, propellant loading divided between upper section, and base

C2 - Upright with two supports to the shroud, propellant loading divided between upper section and base

C3 - Inverted with two supports to the shroud, propellant loading in base

C4 - Inverted unsupported, propellant loading in base

C5 - Upright with maximum allowable reaction to shroud (5,000 pounds), propellant loading divided between uupper section and base.

Four load/boundary conditions were analyzed using the triform vehicle. The conditions are stated below and illustrated in Figure 2-4. The combined dynamic launch environment employed is 0.67 g's lateral superimposed on 4 g's axial. This load environment occurs after second stage burnout. An auxiliary truss used to support the triform will accommodate the launch loads of 3 g's lateral and 6 g's axial. An analysis was also performed to determine the support truss weight and payload effects.

T1 - Upright unsupported, reactor loading divided between upper section and base

T2 - Upright with two supports to the shroud, propellant loading divided between upper section and base

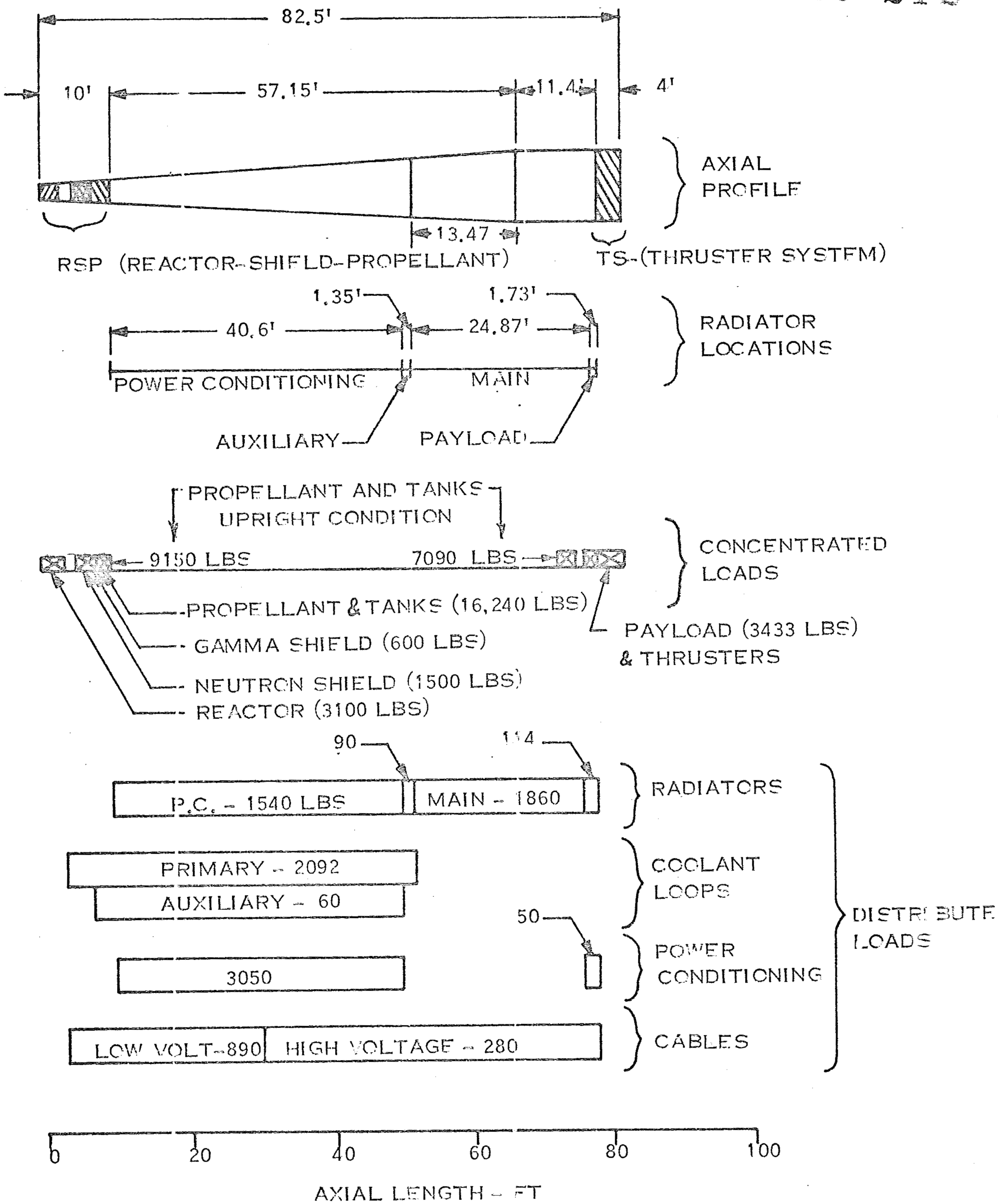


Figure 2-1. Reference Design-Cylinder-Conical Radiator Spacecraft Weight Distributions

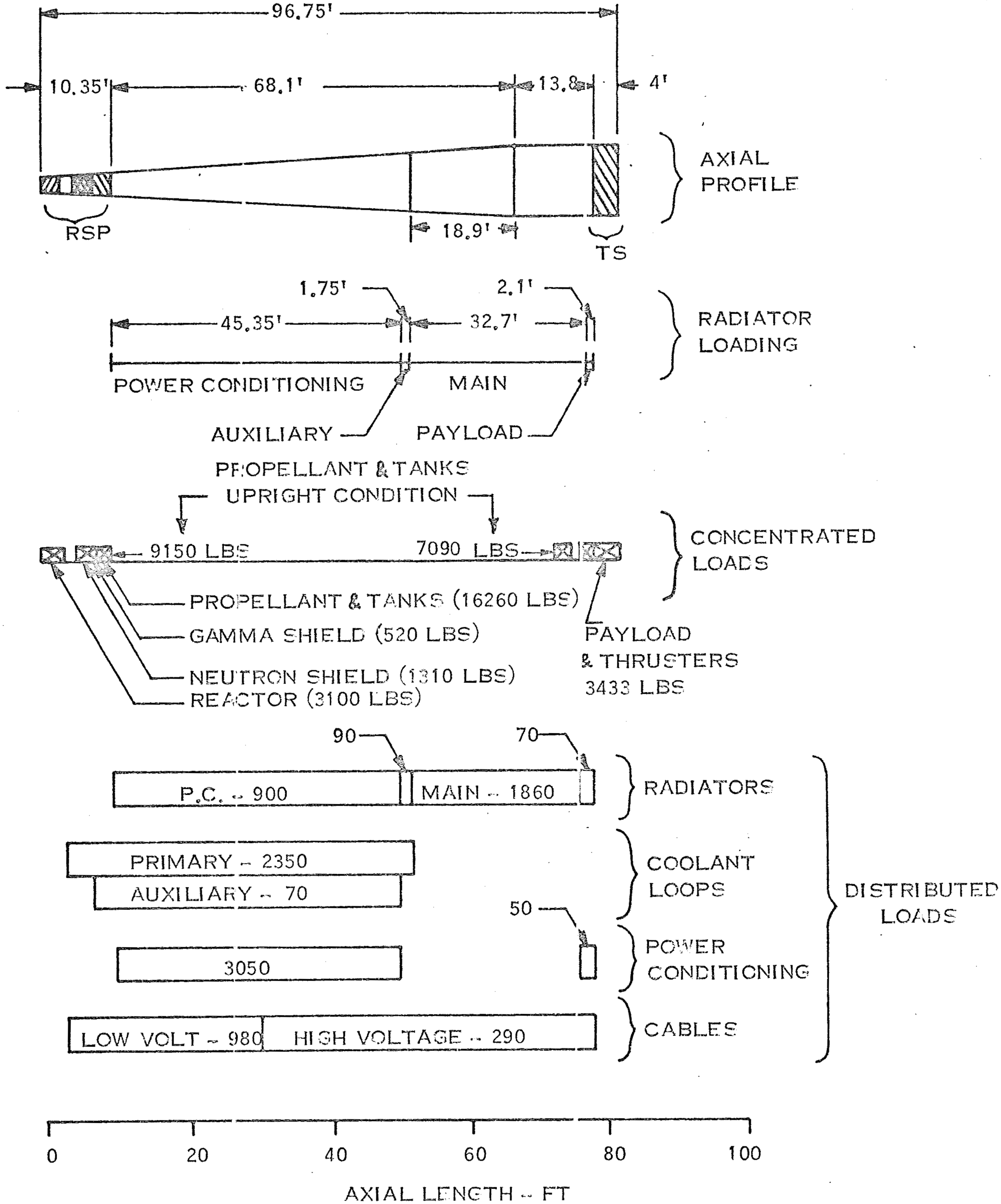


Figure 2-2. Reference Design-Triform Radiator Spacecraft Weight Distributions

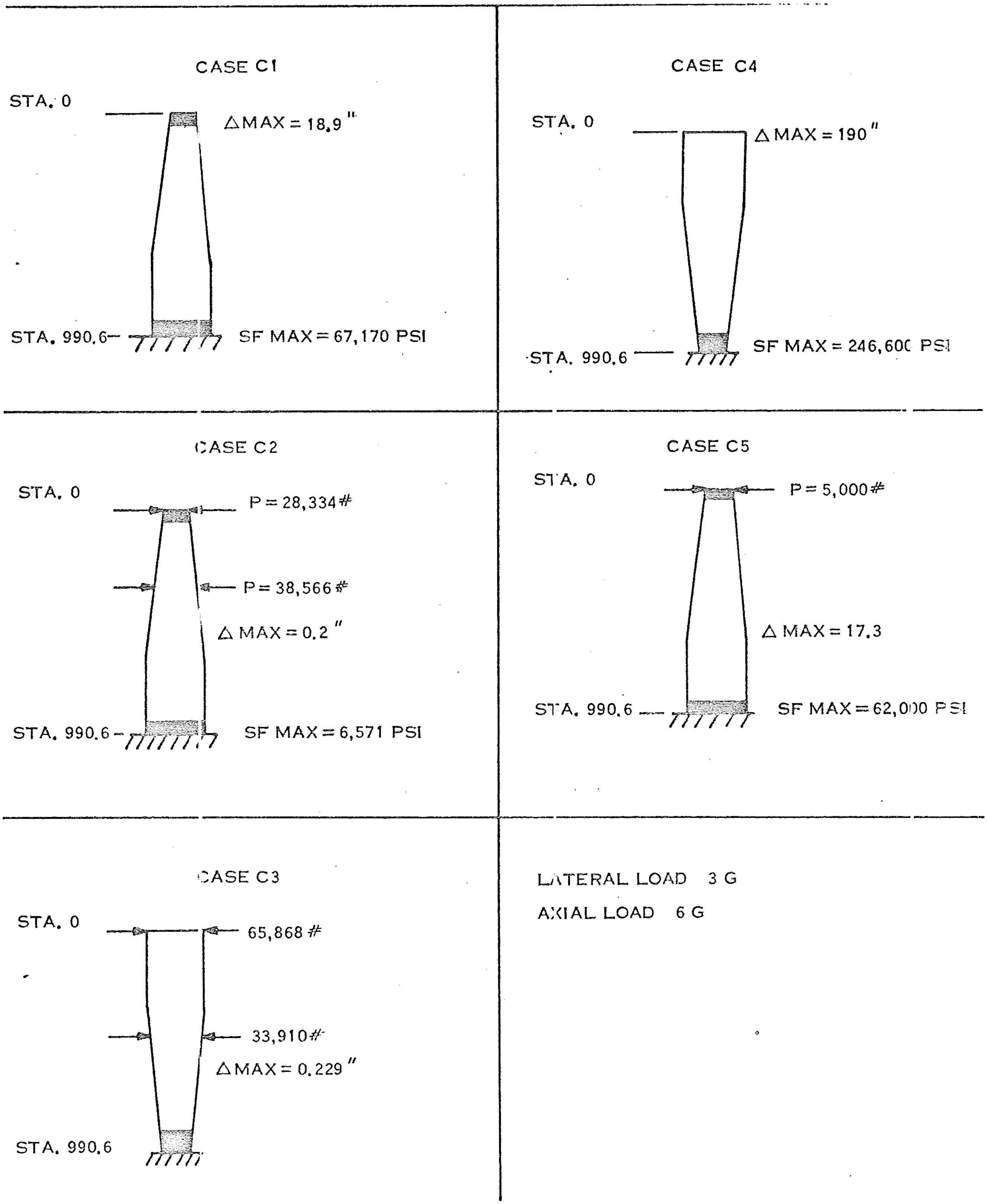
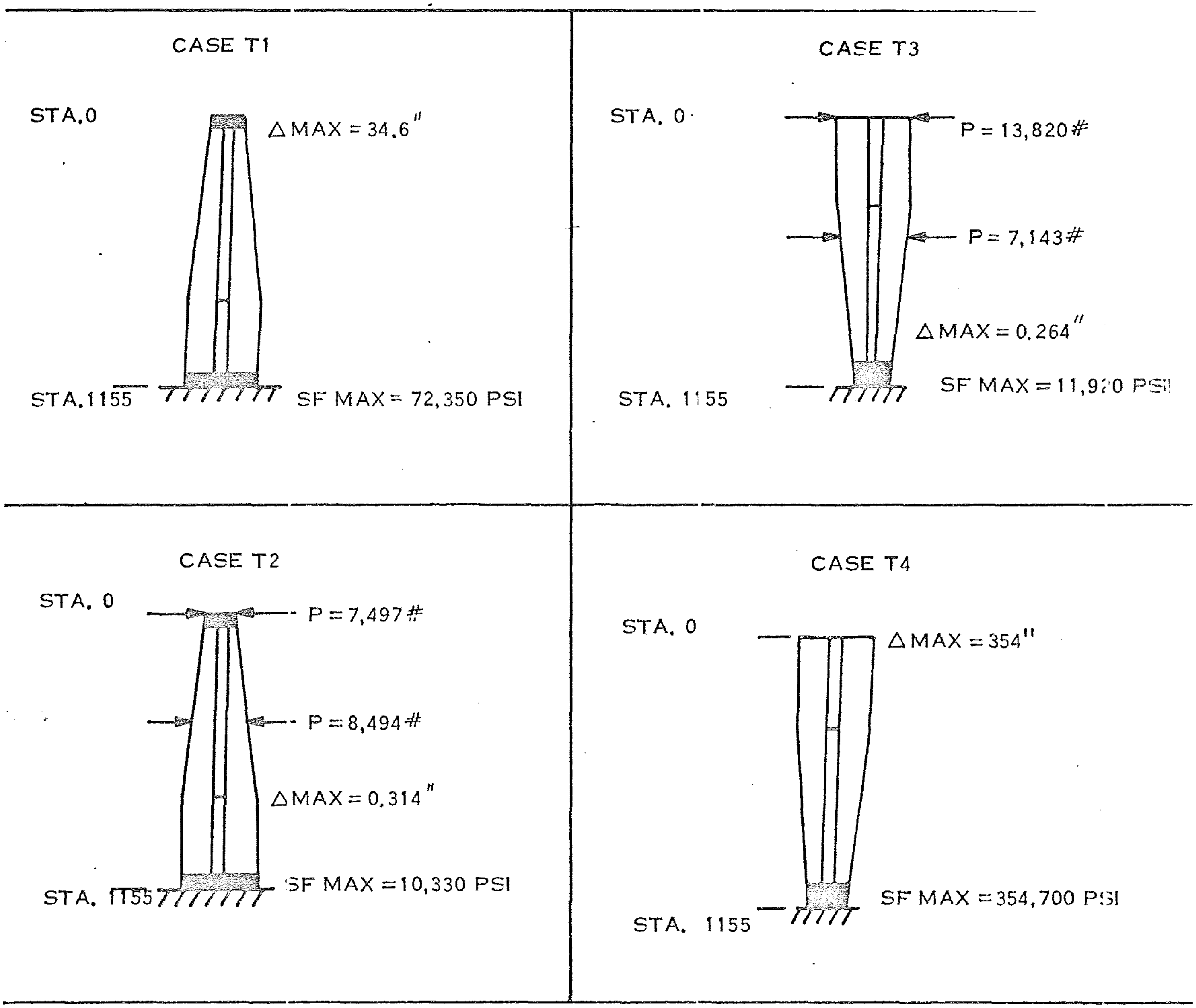


Figure 2-3. Load/Boundary Conditions for Cylindrical-Conical Radiator Spacecraft Design



LATERAL LOAD 0.67 G
AXIAL LOAD 4.0 G

Figure 2-4. Load/Boundary Conditions for Triform Radiator Spacecraft Design

T3 - Inverted with two supports to the shroud, propellant loading in base

T4 - Inverted unsupported, with propellant loading in base.

The propellant and tank weights shown on Figures 2-1 and 2-2 reflect an early selection of 15,000 pounds of propellant plus an assumed eight percent tank fraction (because of the unusual shield configuration of the tank, a truncated cone structure) for a total weight of 16,240 pounds. Current values are 14,500 pounds of mercury propellant, with a probable four percent allowance for structure, feed lines, valves, etc. However, this small difference does not effect the results of this structural analysis.

The data of Figures 2-3 and 2-4 also present maximum stresses, deflections and loads imposed on the shroud, where applicable, for the various cases investigated. The definition of these parameters are discussed below.

2.2.3 STRUCTURAL ANALYSIS SUMMARY

The results of the structural analysis are summarized on Table 2-1 for the five conical and four triform configurations investigated. Maximum stresses, maximum deflections, maximum axial loads (where applicable), fundamental frequency and the total weight of the spacecraft structure required at launch are presented. The weight numbers shown are the total structure required to survive launch including those spacecraft components such as radiator which have been assumed to serve as structure. The definition of this effective spacecraft structure is presented in Paragraph 2.2.4.

The preferred configuration is the conical radiator spacecraft mounted in the upright configuration on the launch vehicle, Case C1, Figure 2-3. It meets all launch requirements with only minor modifications to the baseline spacecraft presented on Figure 2-1. These are the addition of longerons to stiffen the aluminum radiator section, some additional tube wall thickness in the copper/stainless steel radiator section, and circumferential rings throughout to prevent compressive buckling instability. The net weight increase for this additional structure is 1299 pounds, as defined in Paragraph 2.2.4. This configuration is selected because it provides the following advantages:

- Maximum spacecraft IMEO
- Acceptable stresses and deflections

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TABLE 2-1. SUMMARY OF STRUCTURAL ANALYSIS

Load/Boundary Conditions	Case	Max. Stress (psi)	Max. Deflection (in)	Max. Lateral Load (lbs)	Total Space-Craft Structure Wt. at Launch (lbs)	Fundamental Frequency (cps)	Remarks
Conical - 3.0 g's Lateral, 6.0 g's Axial	C1	67,170	18.9	---	4944	1.17	This is the preferred configuration which meets the environmental requirements at the minimum structural weight with few design modifications.
	C2	6,571	0.2	38,566	4944 plus fairing increment	1.36	A major redesign and additional weight would be required in the launch vehicle (LV) shroud to react the lateral loads.
	C3	7,691	0.229	65,868	Same as 2	1.10	Same as 2. Also the LV shroud loads are higher and a larger adapter would be required for attachment to the launch vehicle.
	C4	246,600	190.0	---	4944 plus conical LV adapter	0.56	The stresses and deflections are excessive. Additional structure is required to bring the levels into an acceptable range. A larger adapter is also required.
	C5	62,000	17.3	5,000	4944 plus fairing increment	1.34	This configuration also meets the environmental requirements. A special adapter is necessary to limit the LV shroud lateral load to a 5000 lb. maximum.
Triform - 0.67 g's Lateral, 4.0 g's Axial	T1	72,350	34.6	---	8689	0.49	The configuration, the best identified for the triform, will fly with an auxiliary truss during the launch environment. The total weight is greater than the cone.
	T2	10,330	0.314	8,494	8689 plus fairing increment	0.92	Same as 1. Also the LV shroud would require major redesign and additional weight to react the high lateral loads.
	T3	11,920	0.264	13,820	Same as 2	0.99	Same as 2. Also the LV shroud loads are higher and a larger spacecraft-launch vehicle adapter would be required.
	T4	354,700	354.0	---	8689 plus LV Adapter	0.19	The stresses and deflections are excessive. Additional structure is required to bring the levels into an acceptable range. A larger LV adapter is also required.

- No required redesign of the standard Titan III shroud to provide lateral support
- Potential to eliminate the standard shroud thereby providing an estimated 1000 pound increase in IMEO.

The upright conical, supported by the shroud (Case C2) is eliminated because, although no estimates were made, the weight increase associated with redesigning the shroud is certain to be greater than the integral structure concept selected, reducing the useful IMEO. The inverted conical, supported by the shroud is rejected for the same reasons. The shroud weight penalty would be even greater than for Case C2 since the inverted launch orientation does not efficiently utilize the available conical radiator as structure. Case C4, the unsupported, inverted conical arrangement also does not efficiently utilize the available radiator as structure, and weight of additional structure necessary to provide acceptable stresses and deflections would probably exceed the weight of the Cu/SS radiator, about 2000 pounds. The case where one tie point is provided to the shroud might be acceptable since no shroud redesign is required because the axial load has been limited to 5000 psi, the maximum load that can be reacted to the shroud without major redesign. A spring system would be needed to limit the input loads to the 5000 pound level. Addition to this support has the effect of decreasing the overall deflection by 1.6 inches and the maximum stress by 5170 psi while increasing the frequency by 0.19 cps. These changes from the unsupported Case C1 condition are too small to warrant the additional weight and complexity of the required spring system.

Configuration T1 (Figure 2-4) is the most attractive of the triform radiator configurations. The triform configurations required an auxiliary truss system to prevent the inherent compressive buckling instability of the section. During Stage I launch environment, the auxiliary truss would require a great number of attachments to the triform structure in order to prevent buckling. Releasing these attachments can cause design problems during separation. The estimated minimum weight increase associated with the multiple attachment supporting truss is 4928 pounds.

As discussed in Paragraph 2.1.1, the supporting truss is assumed to be jetisoned, along with the shroud during the Titan IIIC/7 Stage II burn (280 seconds after launch). The basic triform radiator structure is designed to survive Stage 2 burnout loads (0.67 g lateral and 4 g axial) and therefore the lower, subsequent transtage loads. The IMEO penalty associated with the 4928 pound truss is therefore only about 1250 pounds. This, coupled with additional stiffening of the aluminum and Cu/SS radiator section of the triform results in a minimum net structural weight penalty of about 2580 pounds, ideally about twice the net structural weight penalty for the conical structure. However, many structural attachments between the truss and the triform radiator structure are required to prevent buckling failure prior to truss (and shroud) separation. Although not assessed, the weight penalty associated with stiffening the basic triform radiator structure, in order to minimize the number of truss-to-radiator attachment points, represents a permanent weight increase, a reduction in the IMEO which must be carried throughout the Jupiter orbiter mission. The primary reasons for rejecting the triform radiator configuration are:

- Lower IMEO, and/or
- Complex truss attachment and separation
- No potential to eliminate the standard Titan III shroud.

As in the case of the conical configuration, inversion of the triform configurations resulted higher stresses, higher shroud reactions and lower fundamental frequencies than the corresponding upright conditions. In addition, a larger adapter attachment between the spacecraft and the launch vehicle would be required. For these reasons the inverted conditions are considered undesirable.

The addition of shroud supports to the unsupported spacecraft reduced the deflections considerably, but tended to produce fairing loads which could not be accepted without a major redesign of the shroud. Hence, the unsupported (by the shroud) or the single, limited capability, fairing tie for the upright configuration is the most desirable condition for both the conical and triform configurations.

Although several of the configuration/boundary condition cases investigated meet the thermal structural and envelope requirements, none meet the current Titan launch vehicle autopilot minimum frequency requirement of three Hertz. To meet this frequency requirement, the spacecraft stiffness would have to be increased by a factor of 6.6 or the shroud by a factor of eleven with provisions for one support into the existing spacecraft. Probably the simplest, and certainly the minimum weight approach is to modify the autopilot to accept payload frequencies in the range of one cps.

2.2.4 STRUCTURE DEFINITION

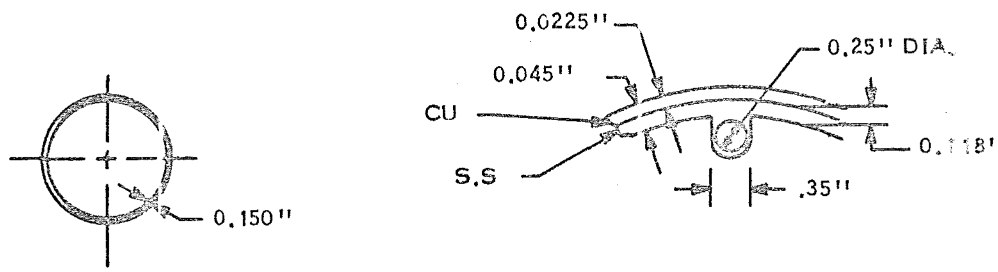
The portion of the baseline thermionic spacecraft designs which can be readily utilized as structure are the radiators, the aluminum passive power conditioning radiator and the active, copper-stainless steel tube and fin primary radiators. The weight and geometry of these components is summarized on Figures 2-5 and 2-6 for the conical and triform radiator spacecraft, respectively, under the following assumptions:

- The spacecraft is at ambient temperature when launched
- The aluminum radiator structure is characterized by the 2024-T3 alloy which has an ultimate tensile strength of 65,000 psi, and a compressive yield strength of 40,000 psi
- The stainless steel radiator structure is characterized by the 301 alloy, half-hard, which has an ultimate tensile strength of 150,000 psi, and a compressive yield strength of 58,000 psi
- The copper structure in the Cu/SS radiator is expressed as an equivalent thickness of stainless steel for stress analysis purposes.

Although the main radiator terminates at the start of the payload equipment bay, at least an equivalent structure will be required to transmit the accumulated loads from this point to the launch vehicle (upright launch) or the launch vehicle shroud (inverted launch). Therefore, the cross sectional characteristics, and the associated weight per unit length of the main radiator are assumed to extend aft through both the payload bay and the ion engine thruster bay, located immediately behind (Figures 2-1 and 2-2). The weight of the payload and special thruster power conditioning radiators which remain to be precisely defined is assumed to be included in the structural weight of these two spacecraft bays.

A preliminary assessment of the baseline structure indicated that it would require some stiffening and/or additional support structure. This requirement would be most severe for the triform, and less for the cylindrical conical radiator. Therefore, the following changes were made in the baseline structure prior to initiation of detailed analyses:

Spacecraft Component	Available Structure Weight, lbs. (Baseline Design)	Assigned Structure Weight, lbs. (As Analyzed)	Required[a] Additional Structure Weight, lbs.	Total[a] Structure Weight, lbs.
Low Voltage Power Conditioning Radiator	1540	1644	150[c]	1694
Auxiliary Radiator	90	206		
Primary Radiator	1860	2266	18[d]	3250
Payload Bay	(b)	300		
Thruster Bay	(b)	360		
Total	3490	4876		4944[e]
Net Structural Weight Penalty IMEO = 4944 - 3490 - 155[e] = 1299				



ALUMINUM RADIATOR

COPPER/SS RADIATOR (180 TUBES - NOMINAL)

STRUCTURE GEOMETRY

(a) Preferred (minimum weight) configuration; upright conical

(b) Except for local radiators, concentrated loads of payload and thruster subsystem components assumed as no structural value.

(c) Circumferential rings.

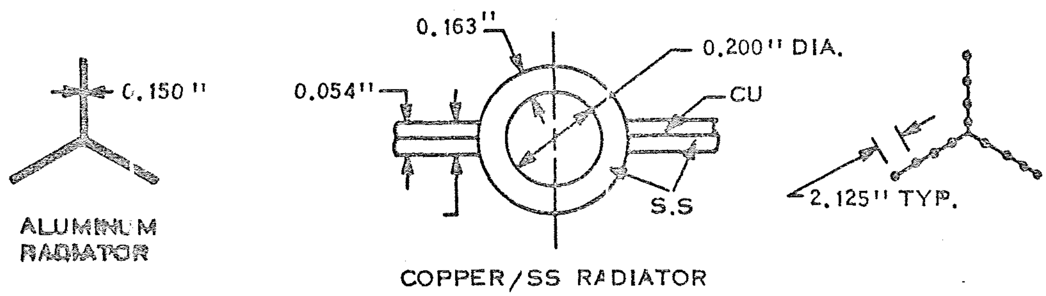
(d) Circumferential rings (300 lbs), less 282 lbs weight reduction from tapered bumper on radiator tubes.

(e) Includes estimated 155 lbs required for radiators in payload and thruster subsystem bay.

Figure 2-5. Structural System Weight Summary, Thermionic Spacecraft Cylindrical-Conical Radiator

Spacecraft Component	Available Structure Weight, lbs. (Baseline Design)	Assigned Structure Weight, lbs. (As Analyzed)	Required[a] Additional Structure Weight, lbs.	Total[a] Structure Weight lbs.
Low Voltage Power Conditioning Radiator	900	900	336[c]	1236
Auxiliary Radiator	90	112	none[c]	112
Primary Radiator	1860	2234	none[c]	2234
Payload Bay	(b)	139	none[c]	139
Thruster Bay	(b)	264	none[c]	264
Launch Support Truss	-	-	4928[d]	4928
Total	2850	3649		8913[e]

Net Structural Weight Penalty at Launch = 8913 2850 -155[e] = 5908 lbs.
Net Structural Weight Penalty, IMEO = 3649 + 336 - 155[e] - 1250[d] = 2580 lbs.



STRUCTURE GEOMETRY

(a) Best (minimum weight) triform configuration; upright and unsupported.

(b) Except for local radiators, concentrated loads of payload and thruster subsystem components assumed as no structural value.

(c) Required additional structure to survive Titan IIIC/7 stage 2 burnout loads.

(d) Required to survive initial launch loads; jettisoned with shroud just prior to stage 2 burnout, IMEO penalty is 1250 lbs.

(e) Includes estimated 155 lbs required for radiators in payload and thruster subsystem bays.

Figure 2-6. Structural System Weight Summary, Thermionic Spacecraft Triform Radiator

a. Ninety aluminum "hat shaped" 0.04 square inch (effective) longerons were added to the aluminum power conditioning radiator in the conical configuration, i.e., one-half of the number of tubes in the copper-stainless steel radiator.

b. The area and therefore moment of inertia of the copper-stainless steel radiator was increased by the addition of 0.1 inch of stainless steel to the armor of each of the 180 tubes. Although this additional structure could be placed elsewhere, the selected location provides additional, although unnecessary, meteoroid protection.

c. The thickness of the stainless steel tubes in the triform radiator was similarly increased.

The effect of these changes on the spacecraft structural weight are also shown on Figures 2-5 and 2-6. The results of the analyses performed support these preliminary weight adjustments in that still additional structure is required to survive launch, primarily in the form of stiffening rings to reduce the buckling loads on the conical radiator and, in the form of separate truss work for the triform radiator. These additional weights are also illustrated on Figures 2-5 and 2-6. The details of this analyses, presented below, also indicate that about one half of the 0.1 inch added to the bumper thickness on the conical radiator can be removed, so long as this is accomplished by tapering this thickness from 0.1 inch at the base of the spacecraft, to zero at the top of the Cu/SS radiator. This effect is included in the total spacecraft structural weight at launch presented on Figures 2-5 and 2-6.

2.2.5 STRUCTURAL ANALYSIS

2.2.5.1 Computer Programmed Analysis

Two computer programs were written to determine the physical properties of the cylinder-conical and the triform spacecraft configurations. The physical shapes and member sizes used in computation were as per the thermionic designs shown in Figures 2-5 and 2-6.

Program "Cone" determined the properties of the cylinder-conical configuration including the area, inertia, torsional and shear constants. An integration method was used to determine the varying properties along specified vehicle stations. The second program "Triform" determined similar properties for the triform configuration.

Due to the number of conditions to be analyzed, combined with the complexity of the loads and redundant boundary restraints, the analysis was completed using the General Electric "MASS" computer program. Both spacecrafts were modeled for the program using 27 nodal points, each with six degrees of freedom. The physical properties, as determined by the "Cone" and "Triform" computer, were used at the respective nodal points.

The concentrated and distributed loads shown in Figures 2-1 and 2-2 were applied to the spacecraft models. A quasi-static steady state load magnification of 3 g's lateral superimposed on 6 g's axial were used for the cylinder-conical model analysis. Load magnification factors of 0.67 g's lateral and 4 g's axial were used for the triform model analysis. The loads used for the triform model are equivalent to the second stage burnout environment. The triform is basically unstable relative to compressive buckling, and the analysis assumed an auxiliary support truss would be used to accommodate the launch load environment prior to Stage 2 burnout. Since the loads distribution is proportional to the magnification factor, their magnitude can be modified for other load environments, if required.

Nine load/boundary conditions, as shown in Figures 2-3 and 2-4, were analyzed. Selected results in terms of stresses, displacements, loads and physical properties were plotted versus station number and are included in Appendix A.

2.2.5.2 Compressive Buckling Analysis

A compressive buckling analysis was performed to determine the stability of the preferred cylinder-conical and triform configurations, Cases C1 and T1. The cylinder-conical configuration was found to be stable, requiring only stiffening rings about the circumference at average intervals of 24 inches. The triform configuration was found to be critical in buckling. Considerable stiffening in the form of attached members to reduce the panel widths or an auxiliary truss is required. The analysis performed on these sections is presented below.

2.2.5.2.1 Instability Analysis of Cone-Cylinder Configuration C1 - The configuration of cone and longerons in question will remain stable up to a certain length after which circular frames or supporting rings must be used to maintain stability. The maximum axial load per longeron is 11,754 pounds; the effective area of longeron with skin is 0.252 square inches and the inertia of longeron with skin is 0.0043 inches. Therefore, the maximum axial stress per longeron is 46,642 psi.

The column equation (Reference 2) used to find the required spacing of circular frames is:

$$F_C = \frac{C \pi^2 E}{(L/P)^2}$$

which yields a required spacing of 15 inches, assuming an end restraint factor of 2.0 for C. Hence, supporting rings are required.

The moment and EI versus station curves presented in Appendix A show that the moment varies more rapidly than the inertia thus allowing an increase in frame spacing, and a tapering of the tube wall thickness along the length of the radiator. An average spacing of 24 inches will be used and, therefore, 30 frames will be required.

The frame size is calculated using the following frame stiffness equation (Reference 2):

$$(EI) = MD^2/16000\ (L)$$

The required frame inertia at the base of the spacecraft is found to be 0.071 inch^4. This inertia requirement is met by a tube frame 1.5 inch square with a 0.049 inch wall. The required frame size at the top of the spacecraft is a 0.625 inch square tube frame with a 0.028 inch wall, which meets an inertia requirement of 0.004 inch^4. Therefore, the average frame area required is 0.165 square inches, and the additional weight per frame is 15 pounds. The total weight for the thirty frames required is 450 pounds. Because smaller, wider spaced frames may be used in the aluminum section, it is estimated that the weight allocation will be 150 pounds in the aluminum radiator and 300 pounds in the Cu/SS radiator.

2.2.5.2.2 Buckling and Lateral Instability Analysis of Triform T1 Configuration - As previously noted, the triform structure is required to be self-supporting at, and after, Stage 2 burnout, where the worst loading conditions are 0.67 g lateral and 4 g axial. Prior to this condition, the 3 g lateral and 6 g axial loads are taken up by a separate truss (See Paragraph 2.2.5.3), which is jettisoned along with the shroud just prior to Stage 2 burnout. The following analysis is directed toward the structural requirements necessary for the triform configuration to survive the Stage 2 burnout loads.

a. Aluminum Section - The maximum column loading at the base of this section is a moment of 7.732×10^6 inch-pounds and a load of 9.162×10^4 pounds. Combining the moment and axial load to find the total column load on one flange (one third) of the triform:

$$P_{axial} = \frac{Mo}{48} + \frac{P}{3}$$

$$P_{axial} = 1.9158 \times 10^5 \text{ pounds}$$

where the factor 48 is the distance in inches from the center to the outer edge of one flange of the triform. The axial stress, based on the local area, is therefore 31,930 psi.

Now, one flange of the 0.15 inch thick aluminum section triform has a b/t of 300 (48/0.15), for which the buckling stress is much lower than the axial stress of 31,930 psi. Therefore, the flanges of the triform will have to be stiffened to prevent them from buckling.

For an efficient section (Reference 2), referring to Figure 2-7, values of bw/bs equal to 0.3 and tw/ts equal to 1.0 are required. Assuming a working stress of σ_c at 25,000 psi, and using the equation (Reference 2)

$$\sigma_c = \frac{Ks\,\pi^2 E}{12(1-\nu^2)} \left(\frac{ts}{bs}\right)^2$$

with a Ks value of 4.0, corresponding to the selected values of bw/bs and tw/ts, nine stiffeners of 1.7 by 0.16 inches are required for each aluminum panel. This increases the total panel area by 38 percent and reduces the axial stress to 22,875 psi. This compares favorably with the assumed stress of 25,000 psi and the stiffened, aluminum radiator section will be stable.

b. Cu/SS Section - Following the same procedure, at the nominal five foot wide base of the active radiator, for the geometry of Figure 2-6, a moment of 15.82×10^6 inch-pounds, and a load of 1.48×10^5 pounds, the axial stress is found to be 41,000 psi. The allowable stress for the existing structure (Ks of 5 (Reference 2)) is found to be 86,862 psi.

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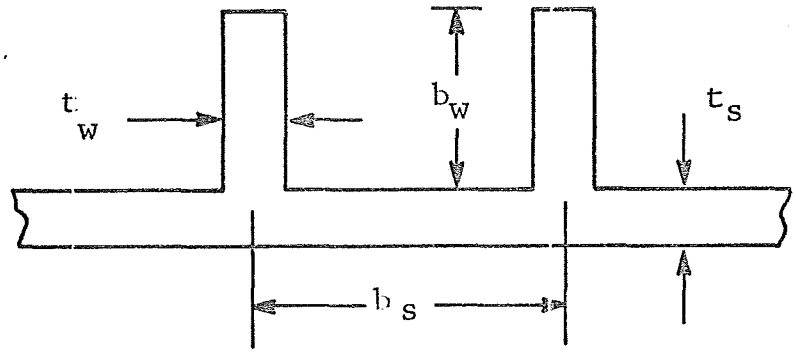

Figure 2-7. Stiffened Panel Geometry

Since the compression yield for 301 SS - half hard is only 58,000 psi, it becomes the allowable. However, the actual stress of 41,000 psi is lower than the compressive yield stress, and therefore, the steel section is stable without reinforcement.

Under the reduced load conditions of 0.67 g lateral and 4 g axial, the weights of the triform, Case I, after second stage burnout are:

- Weight of copper/stainless steel section - 2749 pounds
- Weight of aluminum section - 900 pounds
- Weight of aluminum section, with reinforcement - 1236 pounds.

Hence, after jetisoning the supporting truss required at launch to reduce the load levels, the triform structure will weigh 3985 pounds.

2.2.5.3 Launch Support Truss - Triform

The truss required to withstand the more severe Stage 1 burnout launch loads will now be examined. These conditions are 3 g lateral and 6 g axial.

a. Aluminum Section - Ratioing the moment and axial loads at the base of the aluminum section up to the increased launch loads, the moment increases to 34.62×10^6 inch-pounds and the load increases to 13.74×10^4 pounds. The load on one flange of the triform

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becomes 7.67×10^5 pounds. The maximum Fcy for the reinforced aluminum triform is 25,000 psi. Therefore, the load which can be carried by the reinforced area is:

$$F_{AL} = 25,000\ (8.375) = 209,375 \text{ pounds}$$

Hence, the load which must be carried by the truss is:

$$F_{truss} = P_{axial} - F_{AL} = 557,625 \text{ pounds}$$

Now, assuming the truss to be made of 17-7 PH stainless steel, 1/2 hard, with an allowable F_{cy} of 147,000 psi, the truss area required is 3.8 square inches.

b. Steel Section - Proceeding in the same manner, the area increase at the base of the spacecraft is found to be 5.2 square inches.

The total weight of the truss required for the triform, under launch load conditions, is computed at 4928 pounds. This represents a minimum weight, which requires many truss-to-triform attachment points along the length of the spacecraft (at least every two feet). Weight required for connection of the truss work to the spacecraft has not been calculated, but it could be as much as 10 to 15 percent of the truss weight.

2.2.6 STRUCTURAL DYNAMIC ANALYSIS

The two baseline thermionic spacecraft configurations (conical and triform) discussed in the first section of this report were analyzed. The nine cases discussed there and three additional cases are analyzed and results tabulated herein.

A basic minimum vibrational characteristic of 3 cps on frequency is desirable to insure against interactions between the current launch vehicle control system (e.g., Titan IIIM) and the spacecraft.

The structural dynamic analysis was carried out by applying methods of linear algebra, along with the dynamical equations of motion, to lumped-parameter-models of the two basic thermionic spacecraft configurations. Each configuration corresponds to varying boundary conditions as shown in Figures 2-3 and 2-4.

2.2.6.1 Lumped Mass Model

Both the conical and the triform configurations use the same basic lumped mass model. Figure 2-8 shows the basic dynamic model used. The cantilevered system is shown fixed free with none, 1 and 2 ties into the shroud. The mass to be lumped at the various mass points consists of various concentrated loads such as propellant, reactor, thrusters, and various distributed loads such as structure and radiators. The mass was lumped as shown in Table 2-2. Point zero is taken as the base of the structure. In this configuration, 9150 pounds of propellant and tank were located at the apex near the reactor to provide additional shielding for the reactor. The remaining 7090 pounds of mercury and tank was located at the base. In the inverted launch position, with the reactor (and apex) located at the interface, the 9150 pounds of propellant and tank was also moved to the apex to increase the natural frequency, and reduce the load.

Figures 2-9 and 2-10 show the area and inertia distributions used in determining the spacecraft stiffness. The material used in the structure is stainless steel, copper, and aluminum. Length between mass points was 100 inches and 115 inches for the conical and triform configurations, respectively.

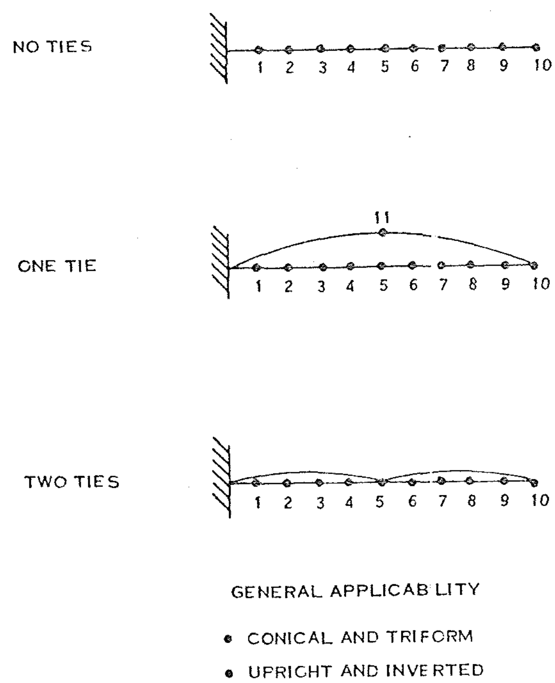

Figure 2-8. Thermionic Powered Spacecraft Dynamic Models

TABLE 2-2. MASS DATA FOR LUMPED MASS MODEL

Upright Conical Configuration

Mass Point	Weight (lb)	Bending Inertia $(lb\text{-}in^2) \times 10^{-4}$
0	7776	142
1	5033	264
2	1635	264
3	1610	232
4	1577	101
5	1825	81
6	1780	65
7	1790	52
8	1740	40
9	12588	31
10	3215	9.5
	Shroud	
0	2000	4550
11	4000	9040
10	2000	4550

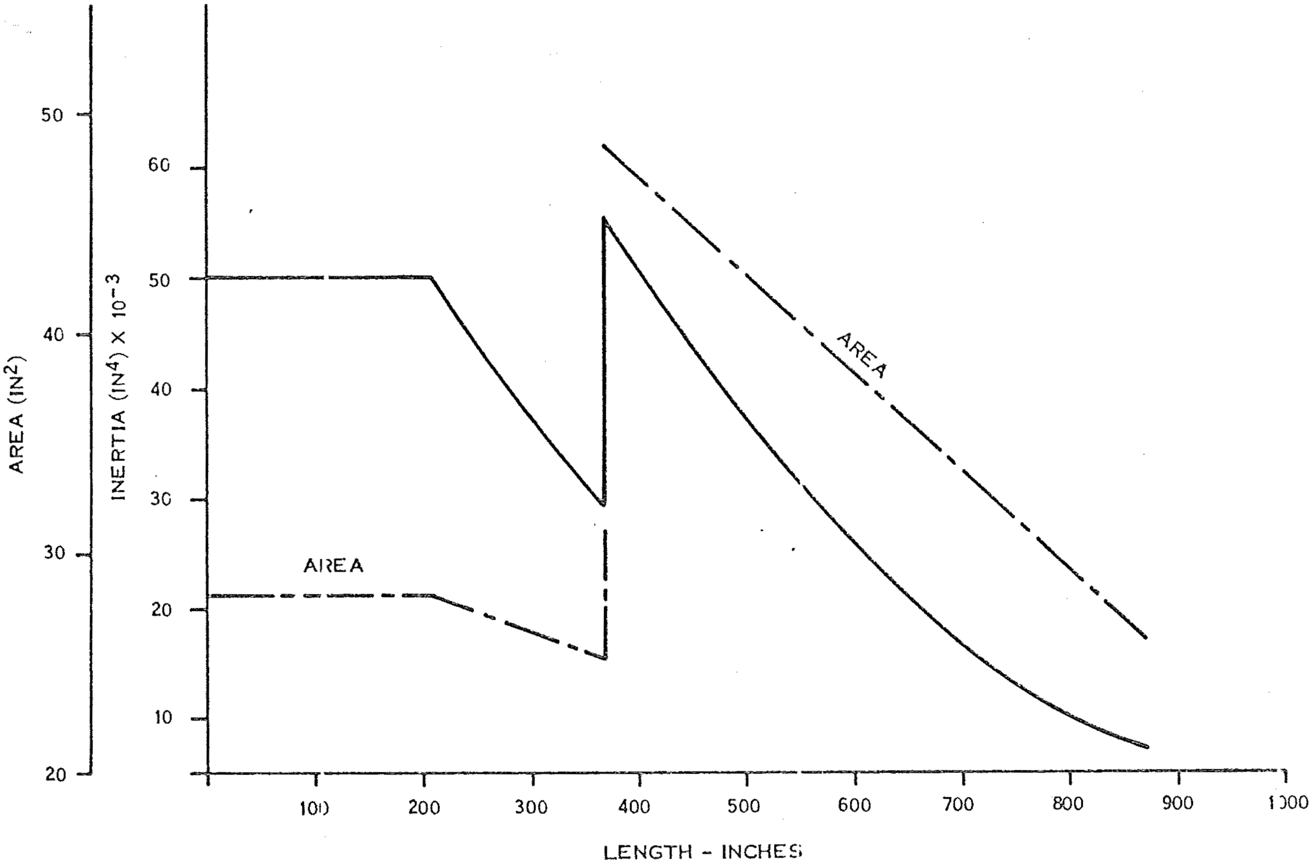

Figure 2-9. Conical Configuration Dynamic Properties

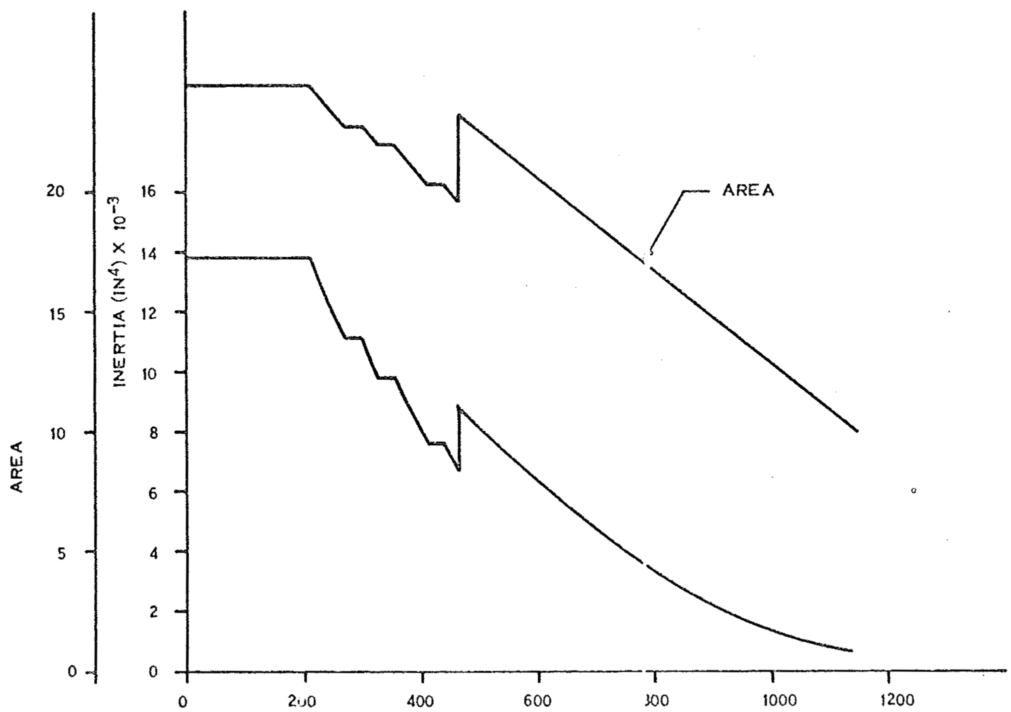

Figure 2-10. Triform Configuration Dynamic Properties

2.2.6.2 Normal Mode Shapes and Natural Frequencies

The two configurations were examined in the upright and inverted positions for cases of zero, 1 and 2 ties into the shroud. The frequencies of vibration as well as the mode shapes of these cases were obtained. Frequency results are presented in Table 2-3 for the lowest three modes. The triform configuration is immediately seen to be very bad dynamically in view of the current 3 cps requirement for prevention of interaction with the launch vehicle control system. The conical configuration is also poor, but better than the triform configuration. For both configurations, the upright position is far better than the inverted. Mode shapes corresponding to the lowest natural frequency of each of the 12 cases are shown in Figure 2-11 through 2-14. These figures show the effect of the ties to the shroud. While two ties are not appreciably better than one tie from a frequency viewpoint, it does significantly reduce displacements.

As a check on the validity of the model used in the analysis, a single degree of freedom frequency computation was made on the upright conical configuration with no shroud ties. This computation yielded 1.27 Hz, which compares with 1.17 Hz obtained from the 10 point 30 degree of freedom model. Thus, a high level of confidence may be associated with the lowest frequency results.

2.2.6.3 Shroud Stiffness Requirements

An approximate calculation was made to determine how much stiffness must be added to the shroud to bring the lowest frequency of the spacecraft shroud system up to 3 cps (MOL Program Requirements). The computation was made for the case of two shroud ties, and for the upright conical configuration.

Since the system is basically a beam, the lowest frequency may be considered proportional to the square root stiffness to mass ratio

$$f_1 \sim \sqrt{EI/\mu}$$

where μ denotes mass per unit length, and the EI here is to be considered an equivalent stiffness. The shroud is essentially an additional spring in parallel with the spacecraft and hence its stiffness is additive. The frequency obtained for this case was 1.36 cps. The additional shroud mass required, in comparison to the unstiffened system weight is found from the equation

$$\frac{f_{desired}}{f_{current}} = \sqrt{\frac{(EI)_{s/c} + (EI)_{new\ shrd}}{(EI)_{s/c} + (EI)_{shrd}}} = \frac{3}{1.36}$$

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TABLE 2-3. THERMIONIC SPACECRAFT STRUCTURAL DYNAMICS

Configuration	Frequencies (cps) 1st	2nd	3rd
Upright Cone			
No Ties	1.17	6.47	11.86
1 Tie	1.32	6.55	11.31
2 Ties	1.36	6.79	11.32
Inverted Cone			
No Ties	.56	4.73	8.77
1 Tie	1.06	4.92	7.22
2 Ties	1.10	5.45	8.43
Upright Triform			
No Ties	.49	2.98	5.82
1 Tie	.89	3.51	6.41
2 Ties	.92	5.10	6.93
Inverted Triform			
No Ties	.19	2.02	4.85
1 Tie	.98	2.51	5.04
2 Ties	.99	3.71	5.36

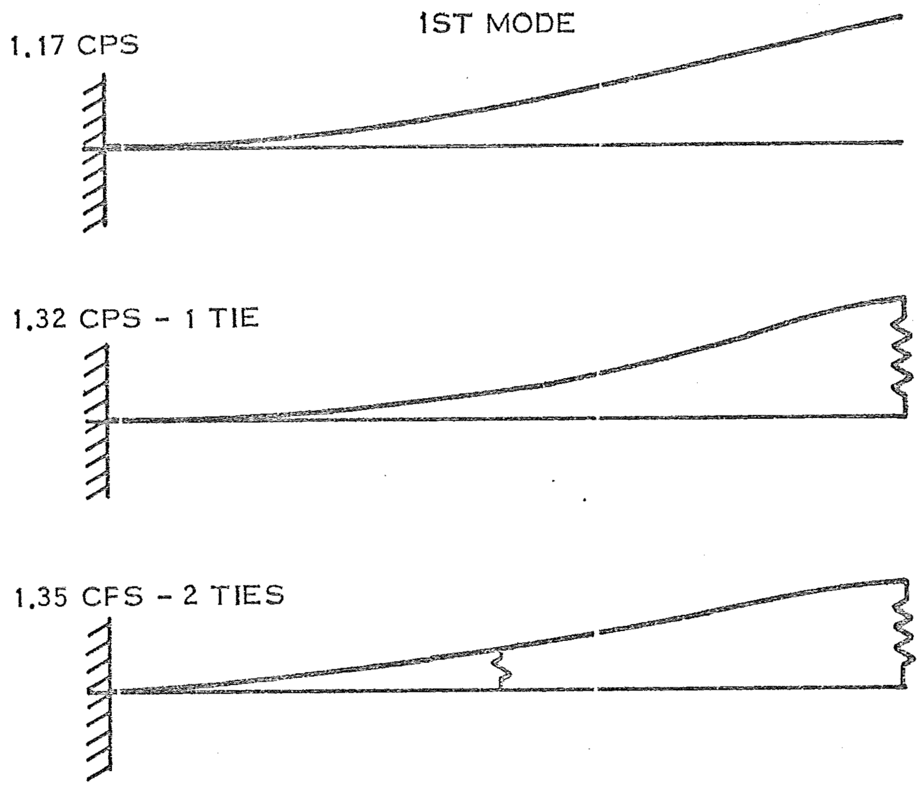

Figure 2-11. Thermionic Spacecraft-Upright Conical Mode Shapes

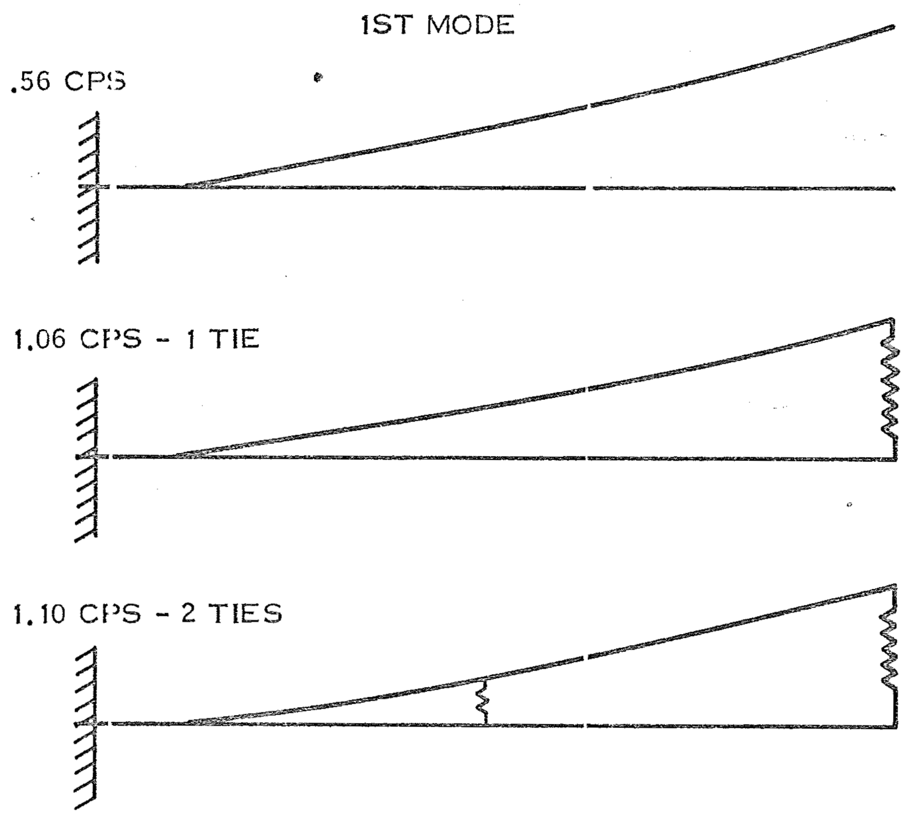

Figure 2-12. Thermionic Spacecraft-Inverted Conical Mode Shapes

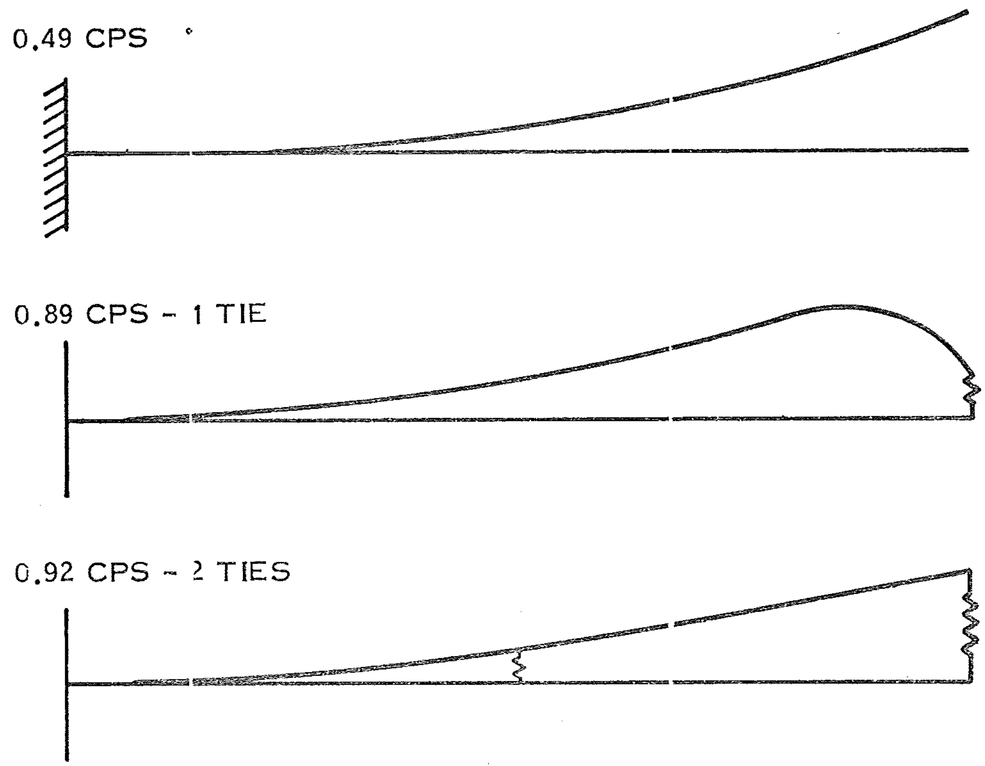

Figure 2-13. Thermionic Spacecraft-Upright Triform Mode Shapes

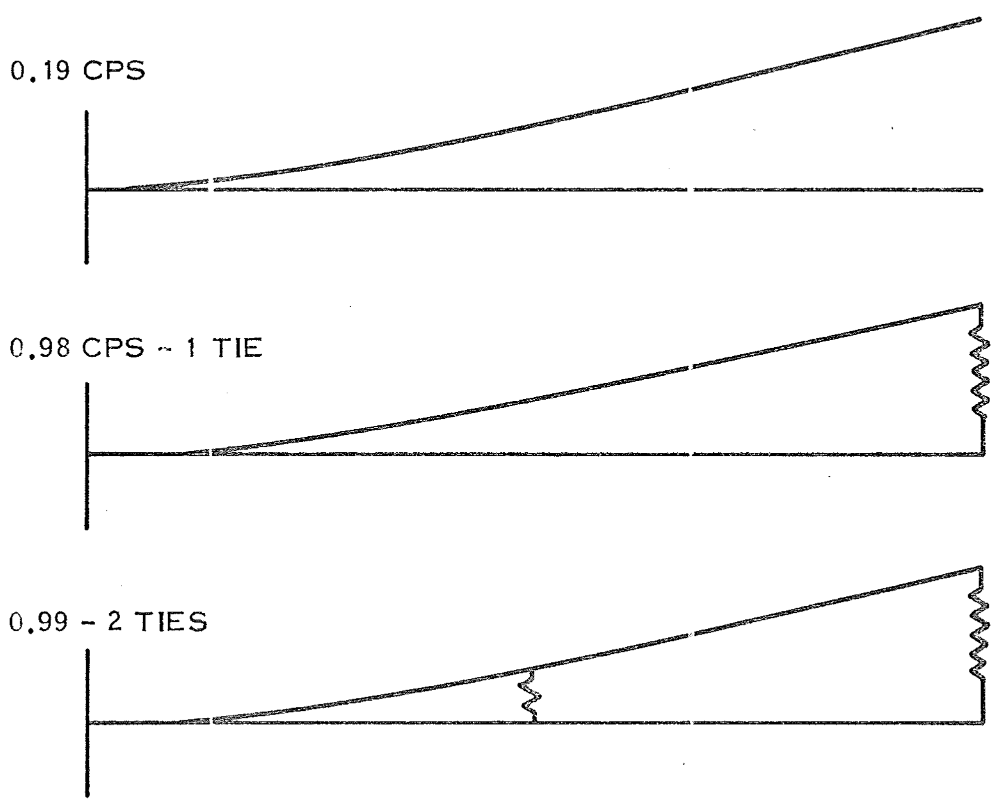

Figure 2-14. Thermionic Spacecraft-Inverted Triform Mode Shapes

or

$$\frac{(EI)_{new\ shrd}}{(EI)_{shrd}} = 11.03$$

as the amount of stiffening required in the shroud to give a minimum natural frequency of 3 cps.

Using this factor of 11 and considering the shroud as a simple monocoque cone whose original weight is about 8000 pounds, the required shroud would weigh around 88,000 pounds. Using the 0.24 pounds of payload per pound of shroud weight penalty (Reference 1) for a 700 nautical mile circular orbit, this shroud increase results in a 21,120 pound payload reduction. Therefore, redesign of the autopilot to accommodate payload frequencies in the one cps range will be required.

2.2.7 APPLICATION OF RESULTS TO SPACECRAFT USING PANCAKE OR EXTERNALLY FUELED REACTORS

The major impact on the arrangement of spacecraft using the pancake, or externally fueled diode reactors, is that their higher voltages are expected to permit the location of the aluminum power conditioning radiator at the base of the spacecraft, immediately above the thrusters. The primary and auxiliary radiators would be located immediately aft of the shield. This arrangement is desirable in that it eliminates the need to locate the low temperature power conditioning radiator between adjacent higher temperature components. However, this change in arrangement will have a minor effect on the overall load distribution, relative to that employed in this analysis. Therefore, the stress distributions are expected to be similar. The net structural weight penalty, and natural frequencies are therefore not expected to change appreciably. However, the distribution of the launch support structure will change.

The aluminum power conditioning radiator, if located at the base, will be required to survive higher stresses, and more stiffening structure will be required. Conversely, the relocated Cu/SS primary radiator will see lower stresses, and less stiffening will be required. The total structural requirements should be about the same as identified for the conical spacecraft evaluated.

2.3 ELECTRIC SYSTEM DESIGN

The purpose of this portion of the study is to design an electrical power system and its components for use in an electrically propelled spacecraft in which the source of electrical power is a thermionic reactor. This section reports the work completed for the flashlite reactor concept.

The depth of the study is sufficient to permit reasonable estimates of the weight, size, and efficiency of equipment which will provide the necessary functions, and to allow comparisons of these characteristics of the flashlite reactor electric system with those of the externally fueled and pancake reactor electric systems.

2.3.1 REQUIREMENTS, DATA, AND ASSUMPTIONS

The primary functions of the electrical system are to convert the electrical power developed by the thermionic reactor to forms suitable for use by the various electrical loads, to distribute the electrical power, and to provide the necessary system and component protection and control.

2.3.1.1 Load Characteristics

A tabulation of the identified spacecraft loads and their electrical requirements is given in Table 2-4. Details of the electrical requirements of the thrusters, which constitute the principal load (Reference 3), are given in Table 2-5. The main portion of the thruster loads is the thruster screens, which require about 7.2 kW each (2.3 amperes at 3100 volts $\pm$ 1% average). A total of 37 thrusters is included, of which 31 are active and 6 are spares.

The ion engines, which represent the principal electrical load of the entire system, are known to arc frequently. When arcs occur, it is necessary to shut down briefly the arcing engine to allow the arc to extinguish, then restart it. Since the engines are a large percentage of the total load, it was necessary to investigate whether the arcing and consequent shutdowns significantly diminish the average load represented by the engines. Analysis shows that even at the extreme arcing rate of 20 per hour the reduction in average load is only about 3.5 percent. Since arcing frequency tends to diminish with time, the reduction in average load by thruster arcing may be neglected.

Details of the analysis are given in Appendix B.

TABLE 2-4. GE THERMIONIC REACTOR SYSTEM ELECTRICAL LOADS

At Reactor		Electrical Requirements			
		100% Reactor Power		10% Reactor Power	
Item	Function	V	KW	V	KW
Primary Loop Coolant Pump	Cools Reactor	100%	10	75%	10
Secondary Loop Coolant Pump	Cools Power Loop	100%	10	75%	10
Shield Pump	Cools Shield	100%	0.12	75%	0.12
Auxiliary Pump	Cools Pumps, etc.	100%	0.1	75%	0.1
Propellant Pump	Pumps Mercury Prop. to Thrusters	100%	0.1	75%	0
Reactor Controls	Controls Reactivity of Reactor	28±5%	2.0	28±5%	2.0
Cesium Heaters	Maintains Temp. of Cesium Vapor	100%	0.5	75%	0.5
At Thrusters					
Thrusters	Propulsion	See Table 2-5	240	-	0
Science Payload Communications		28±5%	1.0	28±5%	1.0
Guidance and Control	Thrust Vector Control Ion Engines	28±5%	Est. 0.5	-	0
Telemetry	Reports on Conditions of Systems	ASSUME INCL. IN COMMUNICATIONS			
Powerplant System Controls	Protection, Switching & Control of Elec. Sys.	28±5%	Est. 0.5	-	0.5

TABLE 2-5. THRUSTER POWER REQUIREMENTS

Supply Number		Type	Output(1)	NOMINAL RATING					MAX. RATING			Control Range, A
				Volts	Amps	Watts	Reg. %	Peak Ripple	Volts	Amps	Amps Limit(2)	
1	Screen	DC	V	3100	2.32	7200	1.0(V)	5	3200	2.32	2.60	2.0 - 2.4
2	Accelerator	DC	F	2000	.02	40	1.0(V)	5 @ 0.2 A	2100	0.20(3)	0.21	---
3	Discharge	DC	V	35	8.3	290	1.0(V)	2	150 @ 50 mA	9 @ 37V	10	7.5 - 9.0
4	Mag. - Man.	DC	F	15	.7	11	1.0(I)	5	20	1.0	1.0	---
5	Cath. Htr.(4)	AC	F	10	4.0	40	5.0	5	11	4.4	4.1	---
6	Cath. Keeper	DC	F	10	0.5	5	1.0(I)	5	150 @ 50 mA	1.0 @ 20V	1.0	---
7	Main Vapor.	AC	V	0.6	1.0	1	Loop	5	8(5)	2.0	2.2	0.5 - 1.5
8	Cath. Vapor	AC	V	0.3	0.5	1	Loop	5	8(5)	1.0	1.1	0.2 - 0.8
9	Neut. Cath. Htr.	AC	F	10	2.0	20	5.0	5	11	2.2	2.2	---
10	Neut. Vapor	AC	V	0.3	0.5	1	Loop	5	8(5)	1.0	1.1	0.2 - 0.8
11	Neut. Keeper	DC	F	10	0.5	5	1.0(I)	5	150 @ 50 mA	1.0 @ 20 V	1.0	---

(1) V = Variable, F = Fixed

(2) Current limit or overload trip level.

(3) Current at this level for less than 5 min. at low repetition rate.

(4) Needed only during startup or until discharge reaches 3A.

(5) Startup only.

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2.3.1.2 Flashlite Reactor Characteristics

Details of reactor electrical characteristics as well as the method recommended for reactor control are presented in References 4 and 5. The reactor power generator is composed of Thermionic Fuel Elements (TFE's) made up of series stacked cells in a configuration resembling batteries in a flashlite. These TFE's are series connected in pairs with the center connection grounded, as shown in Figure 2-15. Each TFE pair requires an individual power converter so that the electrical operation of each TFE can be adjusted for optimum conditions. The outputs of the several converters are subsequently combined in parallel to provide common electrical outputs to the loads.

Reactor characteristics and mission use, of course, influence system design. The electrical system must be designed to provide power to the loads under the following conditions during the flight:

a. Full power operation (300 kW) at beginning of mission (BOM). Under BOM condition all of the Thermionic Fuel Elements (TFE's) are operative; therefore, to achieve full power output, each TFE operates at its lowest power output.

b. Full power operation (300 kW) at end of mission (EOM). At EOM, 10 percent of the fuel elements are assumed to have failed; hence, the remaining TFE's must operate under conditions which will result in the higher power output necessary to achieve full 300 kW reactor power.

c. Ten percent power operation (30 kW) during coast. During this period, the thrusters are inoperative and the only loads connected are hotel loads and payload.

d. TFE failure operation. This mode requires the operation of any converter for maximum power output with a TFE of a pair inoperative.

The reactor electrical characteristics corresponding to these several operating conditions are presented in Table 2-6.

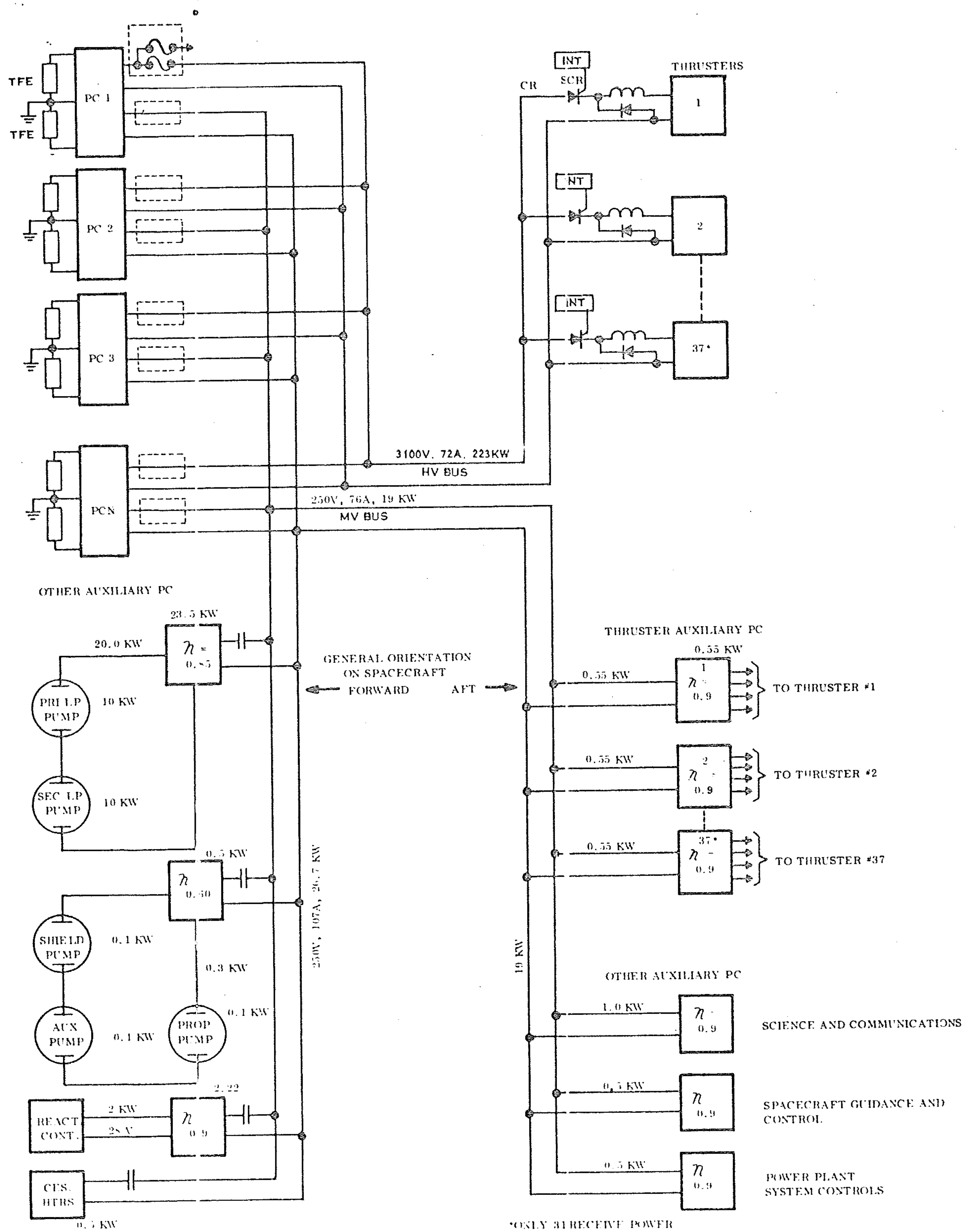

Figure 2-15. Flashlite Reactor Powered Spacecraft Electric Network

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TABLE 2-6. FLASHLITE REACTOR ELECTRICAL CHARACTERISTICS

	BOM	EOM	Coast
Electric Power, (kWe)	300	300	30
Voltage Output, (Volts)	16.8	15.7	12.5
Current, (Amperes)	17.900	19,100	2400
TFE Pairs	108	97*	108
Current/TFE Pair	165.7	196.9	23.8
Emitter Temp., Max., ^{O}K	1950	1950	1600

* 10% TFE Pair Loss at EOM.

2.3.1.3 Electric System Requirements

The primary function of the electrical system is that it transforms the low voltage DC output of the reactor to a higher voltage for transmission and use in the electrical loads. Transmission cable weight and the corresponding power losses associated with power transmission require that power be transmitted at a high of a voltage as practical. The rational for voltage level selection is given in Paragraphs 2.3.4 and 2.3.5.

Secondarily, the electrical power conditioning system is to provide control of the amount of power that is extracted from each TFE pair to insure proper electrical and thermal balance within the reactor. The flashlite reactor is divided into six zones for analysis purposes, with different temperature characteristics. Consequently, for the TFE's in these zones, the electrical output characteristics are different. Further, the TFE's throughout the reactor may be electrically different due to construction variations.

On the basis of these requirements and the data of Table 2-6, the power conversion equipment is designed to accommodate input voltages during normal full power operation from a low of 14 volts to a high of 17 volts, and during the coast phase, accommodate an input of 12 volts. Furthermore, since one half of a TFE pair may fail, provisions are included for allowing the conversion equipment to operate from the remaining TFE. For power conditioner design purposes, this is assumed to be one-half voltage condition at EOM under full power.

2.3.2 SUMMARY OF RESULTS

The electrical system proposed for the flashlite thermionic reactor is described in Paragraph 2.3.3.

The weight of the equipment for the electrical system, including transmission, distribution and interconnecting cables, but not radiators (which are assumed to be the primary structural mounting member for the electrical equipment), is estimated to be 4864 pounds. Total electrical power losses for the system are estimated to be 52,870 watts, for an overall efficiency of 82.3 percent. A breakdown of the principal components of weight is given in Tables 2-7 and 2-8. The electrical power losses are given in Table 2-9. Refer to Paragraph 2.3.6 for a discussion of the selected components and main power converter configuration.

Transmission cables from the reactor to the power conditioning equipment weigh 917 pounds. In order to withstand the 1600°F heat of the reactor and 800°F shield and to minimize weight and power losses, the transmission cables are assumed to be sodium contained in stainless steel tubes, 0.7 inches in diameter. Wiring for high voltage power distribution from the primary power conditioners is composed of aluminum, which weighs approximately 7 pounds total. Interconnection wiring, primarily for medium voltage power distribution between the power conditioning area and reactor and engine areas, weighs 13 pounds.

TABLE 2-7. ELECTRIC SYSTEM WEIGHT SUMMARY
FLASHLITE REACTOR SYSTEM

Component	Weight, lbs.
Main converters	2690
Auxiliary P.C.	507
Auxiliary thruster P.C.	272
Power Distribution Cables	935
Screen supply interrupters	460
Total	4864

The results of the electric power system definition for the spacecraft based on the flashlite reactor have major application to spacecraft based on the pancake and externally fueled reactors. Key points of similarity and differences identified, relative to the results presented here for the flashlite reactor electric system, are:

- Efficiencies of the main power conditioning units will be improved because of the higher voltages
- Low voltage bus bar weights will be reduced
- The main power conditioning units may be located aft, with a minimum of separate units for the 31 operating thrusters
- The weight and power loss associated with the screen isolation may be eliminated by providing isolation at the main high voltage power conditioning unit transformer
- Weights and efficiencies of power conditioning components other than the main high voltage units are not expected to change appreciably.

2.3.3 ELECTRICAL POWER SYSTEM DESCRIPTION

The basic electrical power system proposed for the spacecraft utilizing the flashlite thermionic reactor is shown in Figure 2-15. In this system each TFE pair is provided with a power conversion module and each module provides a medium and high output voltage level of 250 volts and 3100 volts, respectively. The outputs of each module are filtered and all modules are connected in parallel to create the distribution power busses.

The high voltage output bus provides power to all of the screen electrodes of the ion engine thrusters. The 3100-volt level is established by the voltage requirements of the screens.

The 250-volt output provides power to the remaining spacecraft loads including the several power supplies required for each thruster (Table 2-5) as well as for the hotel loads and payloads. The 250-volt level is selected as a convenient, relatively high voltage for auxiliary power distribution. The 3100-volt level used for the screens also could have been used to distribute the auxiliary power, but because of handling and component selection problems, 3100 volts is considered an inconvenient voltage level for uses other than those for which it is necessary.

TABLE 2-8. FLASHLITE REACTOR SYSTEM MAIN CONVERTER WEIGHT BREAKDOWN

Component	Weight, lbs.
Bypass rectifiers	1.0
Input filter	
Choke	3.0
Capacitor	1.0
Inverter	
Power transformer	4.0
Transistors	1.0
Current transformer	0.25
Contactor	2.0
Base drive circuits	0.5
HV output	
Rectifiers	0.05
Filter inductor	1.5
Filter capacitor	1.5
MV output	
Rectifiers (SCR)	0.2
Filter inductor	0.5
Filter capacitor	0.5
Control circuits	0.5
Total electric parts, (single TFE pair)	17.50
Total electric parts (108 TFE pairs)	1890
Wire, brackets, hardware, heat paths	800
Total Main Converter System Weight	2690

TABLE 2-9. SUMMARY OF ELECTRIC LOSSES FOR FLASHLITE REACTOR/SPACECRAFT ELECTRIC SYSTEM

Component	Losses, watts(e)	
Main Power Conditioner		
Transistor Conduction Loss (0.55x165)	91	
Transistor Switching Loss	25	
Transistor Base drive Loss (3v x 165/10)	49	
Transformer (3%)	85	
Input filter (1%)	28	
Output rectifiers (HV)	3	
Output filter (HV)	12	
Output rectifiers (MV)	4	
Output filter (MV)	2	
Control circuits	10	
Total losses, single TFE pair unit	309	
Total main power conditioning losses, 108 units		33,400
Screen supply interrupter		1,250
EM Pump Power Conditioning		3,700
Thruster auxiliary P.C.*		*
Payload Power Conditioning		100
Reactor, power plant and spacecraft controls		322
Total power conditioning losses, watts		38,772
Total transmission cable losses, watts		14,100
Total power losses, watts		52,872
Overall efficiency @ 300 kWe reactor output:	82.3 percent	

* Losses are included in thruster efficiency calculation

Power to the hotel loads and to the auxiliary thruster power supplies and the payloads is distributed by means of two 250-volt busses, one group of loads near the reactor and one at the thrusters/payload area. The bus near the front of the spacecraft supplies power to the following loads:

- Primary loop coolant pump
- Secondary loop coolant pump
- Shield coolant pump
- Auxiliary pump (supplies coolant to cool the other pumps)
- Propellant pump
- Reactor controls
- Cesium heaters.

The bus at the thruster payload supplies power to the following loads:

- Thruster auxiliary power conditioners
- Payload power conditioners
    - Science
    - Command and telemetry
- Guidance and control power conditioners
- Powerplant system control power conditioners

2.3.4 MAIN POWER CONVERTER DESIGN

2.3.4.1 Design Approach

Details of the basic TFE power converter modules selected for the flashlite reactor system are shown schematically on Figure 2-16.

Either of two philosophies may be used in sizing the conversion equipment. One is to design the converters for the TFE pairs operating in the several zones of the reactor. This approach results

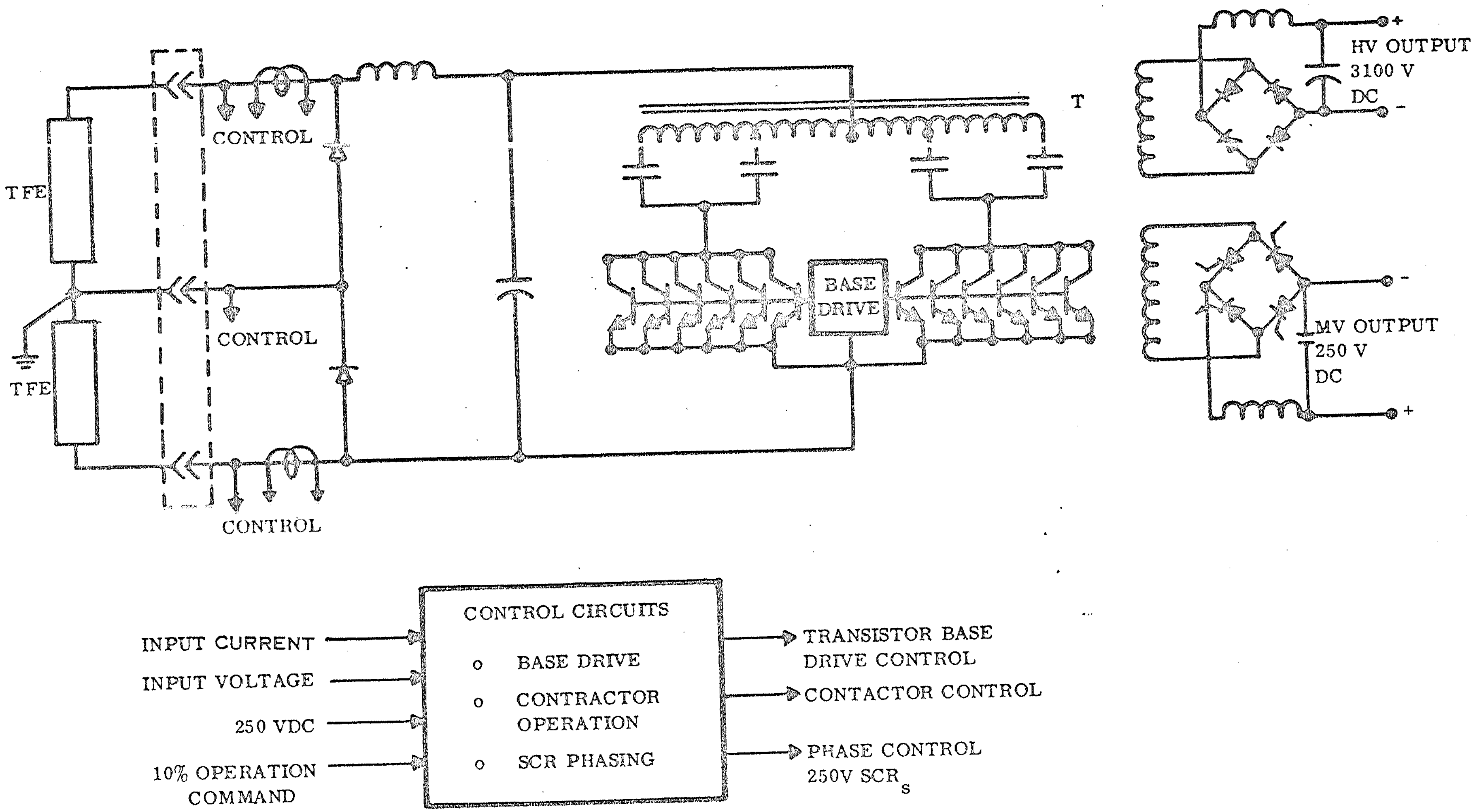

Figure 2-16. Basic Converter Module, Schematic

in minimum weight equipment, but may require several different designs. For estimating purposes, equipment weights for this approach can be calculated using average TFE characteristics, recognizing that some equipment may be smaller and lighter than average and some may be larger and heavier. Because of the difference in the electrical characteristics in the six reactor zones, a maximum of six different designs would be required.

The alternate philosophy is to have a single design of power conversion equipment and apply this design to all TFE pair modules. This approach requires that the conversion equipment design be capable of operating with all extremes of TFE characteristics. It must be capable of handling the largest current and the highest voltage of all individual TFE pairs. Considering all TFE modules then, power conversion would be overdesigned since the maximum current and maximum voltage do not result coincidentally in any single TFE pair.

Although the latter approach is the preferable one from the standpoint of design commonality, the first approach will be used for equipment sizing for this study, since it results in the optimum design for a weight limited spacecraft. The power conversion equipment will be sized for average TFE current and average TFE voltage. It should be remembered, however, that some converters may be larger and some smaller than average.

From the TFE data for the 300 kWe operating points shown on Table 2-6, it is clear that the TFE pair average current is largest at end of mission, 197 amperes, and average voltage is highest at beginning of mission, 16.8 volts. The end of mission current increase when compared with beginning of mission is primarily due to the allowed loss of 10% of the TFE's, not reactor characteristic change.

Over the life of the reactor, while delivering full power and excluding failure of one-half of a TFE pair, the average output voltage will range from 15.7 volts to 16.8 volts. In considering the total voltage range for which to design the primary power converters however, it is necessary to consider also the voltage range required by the reactor current regulating control scheme. For this purpose, acknowledging that the primary user of power are the relatively constant ion bombardment engines, assume the spacecraft load can change instantaneously by 10 percent full load, 30 kW. The control system described for the flashlite reactor requires that in the steady state, TFE current be proportional to reactor thermal power so that emitter temperature is controlled following electrical load changes (Reference 4). Transiently, in the first few milliseconds after an electrical load change, diode temperatures remain constant and diode voltage and current follow the isothermal characteristic curves, as shown for example, on Figure 2-17. For large load changes, the corresponding thermionic diode voltage change would be large, but for relatively small load change of concern here, the corresponding instantaneous voltage change is quite small - perhaps 0.8 volts which is approximately 5 percent at the operating levels. Assuming that the control system limits the total excursion to this value as a maximum, then the total input voltage range for which the conversion equipment should be designed is from about 14 volts to about 18 volts, with additional provisions for operation at the failed half input voltage and the coast voltage corresponding to 10 percent power. For this range of input voltages the output voltage should be held constant. Electrical input characteristics for the primary power conditioners design then are as follows:

Input Voltage

Full power: 14 to 18 VDC
Coast power: 11 VDC (min)
Half TFE failure: 7 to 9 VDC

Input current

Full power: 196.9 amperes (max.)
Coast power: 23.8 amperes

Equivalent Input Power Rating

$(18)(196.9) = 3.55$ kW

2.3.4.2 Inverter Design

The basic conversion function is performed by a parallel inverter which is the preferred circuit to minimize inverter losses. The inverter is capable of operating in three different modes:

- Thrust operation, 100 percent power
- One-half voltage operation, corresponding to the failure of one TFE of the pair
- Coast mode, 10 percent power.

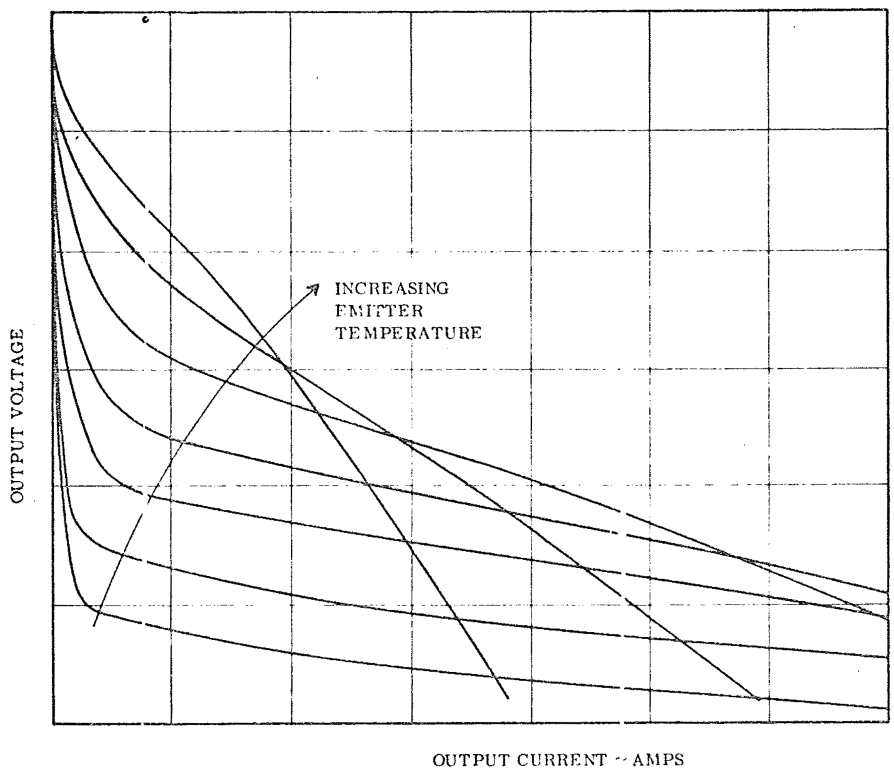

Figure 2-17. Typical Thermionic Reactor I-V Characteristics

Since the voltage level for normal operation and the failed condition are sufficiently different, provisions are included to select taps on the primary of the main power transformer for these two conditions. During normal operation which is either at 100 percent load during thrust or 10 percent load during coast, the switching devices are connected by means of a contactor to the extreme taps on the power transformer. In the event of failure of one of the TFE's of the pair supplying this converter, the switching devices are transferred to the lower voltage tap. For the 10 percent power operation during coast, the TFE output voltage is sufficiently similar to the regular 100 percent power operation that no transformer tap change is necessary. In this application where transistors have been selected for the switching devices, the base drive circuits must be designed to recognize the collector current change and reduce the base current accordingly, in order to reduce the unnecessary losses in the base drive circuits during the coast mode.

An alternative to switching the main transistor groups between taps by means of a contactor is to provide a second set of switching devices permanently connected to the one-half voltage transformer taps. Since, however, these switching devices are required to handle the same current levels as the primary transistor groups, the total number of transistors doubles, and base drive circuits must be duplicated. It is lighter and less complex to have one set of transistors and select the proper transformer taps by contactor switching.

2.3.4.2.1 Switching Device Selection - One of the first decisions to be made in considering power conversion equipment for the flashlite reactors system is the type of power switching device to use in the inverter, whether transistor or SCR. Because the voltage output level of the TFE pairs is relatively low, it is important that the losses of the switching devices be low in the interest of efficiency. These losses are primarily composed of conduction losses plus losses during the switch transition times. Typically, transistors are superior when compared to SCR's from the standpoint of conduction losses (saturation voltage drop of 0.8 volts or less for transistors compared to 1 to 1.5 volts for SCR's). Also, transistors have much shorter switching times than SCR's and therefore can be used at higher frequencies to reduce transformer weight. Thus, the choice is transistors. Furthermore, the selection is confined to silicon transistors. Germanium transistors, the other possibility, are eliminated from consideration because of low operating temperature tolerance. Hence, the design is based on the use of silicon transistors. To meet the necessary current handling capability, six silicon transistors are switched in parallel to generate alternating current for transformation and subsequent rectification.

Two factors influence the selection of the specific power switching transistors; both affect efficiency. These factors are the switching speed and the conduction drop as a result of the collector-to-emitter saturation voltage.

An examination of the characteristics of several types of silicon power transistors currently available indicates that they can be divided typically into two general categories. The first, typified by the RCA 2N3263 (25 amp, 150V), Delco 2N2580 (10 amp., 400 volt), and Westinghouse 1776-1460 (60 amp, 140 volt), exhibit a saturation voltage drop of about 0.75 volts and switching speeds of about 0.5 and 1.0 microseconds (neglecting storage time, which can be compensated for by special circuit techniques).

The second category is defined by a relative newcomer, a Westinghouse low-saturation voltage drop transistor - 0.2 volt at 76 amperes. This device has a switching speed of about 5 microseconds.

Some of the characteristics of these devices are given in Table 2-10.

TABLE 2-10. CHARACTERISTICS OF CANDIDATE POWER TRANSISTORS

Type	Max. V_{CEO} (volts)	Max. I_C (amp)	Oper. I_C (amp)	Oper. I_b (amp)	$V_{CE(SAT)}$ (volts)	$V_{BE(SAT)}$ (volts)	tr (usec)	tf* (usec)	Max. T_J °C	Q_{JC} °C/W
RCA 2N3263	150	25	15	1.2	0.75 max.@15A	1.6 max.	0.5 max.	0.5 max.	200	1.0
Delco 2N2580	400	10	5	1.0	0.7 typ.	1.5 max.	0.7 typ.	0.6 typ.	150	0.7
W 1776-1440	140	40	20	2.0	0.75 est.	1.5 est.	0.5 max.	0.4 typ.	200	0.67
W 1776-1660	140	60	30	3.0	0.75 est.	1.5 est.	0.5 est.	0.45 max.	200	0.67
W Low Sat V	Up to 120	75A	40	4.0	0.2 max.	1.5 est.	5	5	200	0.7 est.

Other high power transistors in addition to those shown in Table 2-10 were considered, such as Solitron 2N4865 and SDT8921 which are 100 ampere units, but which exhibit relatively high saturation voltages at the higher operating currents.

The choice is then between high-speed transistors, which typically have a saturation voltage of 0.75 volts and switching speeds of less than 1 microsecond, and the slower, low-saturation-drop unit with a voltage drop of 0.2 volts and switching speeds of 5 microseconds.

Both saturation voltage and switching times contribute to transistor losses. Analysis shows that these losses on a par unit basis are represented by the following expressions:

$$\frac{P_C}{W} = \left[1-0.002\ T_R f \right] \frac{V_{CE(SAT)}}{E}$$

$$\frac{P_S}{W} = 0.00067 \left[T_R + T_F \right] \left[1- \frac{V_{CE(SAT)}}{E} \right]$$

In these equations:

P_C = Conduction power loss, watts

P_S = Switching loss, watts

W = Power being converted - (input voltage) X (input current)

T_R = Transistor rise time, microseconds

T_F = Transistor fall time, microseconds

f = Switching frequency, kiloHertz

$V_{CE(SAT)}$ = Transistor collector-emitter saturation voltage, volts

E = Supply voltage

These components of transistor losses have been evaluated by means of a computer program for various values of saturation voltage and transistor switching speed as a function of switching frequency. An input voltage of 16 volts and a power level of 3 kW were assumed. Results are shown in Figure 2-18. Total losses versus frequency for the Westinghouse low saturation drop unit and the typical high speed unit are shown in Figure 2-19. These curves show that at switching speeds below 5.8 kHz the low saturation drop transistor is preferred,

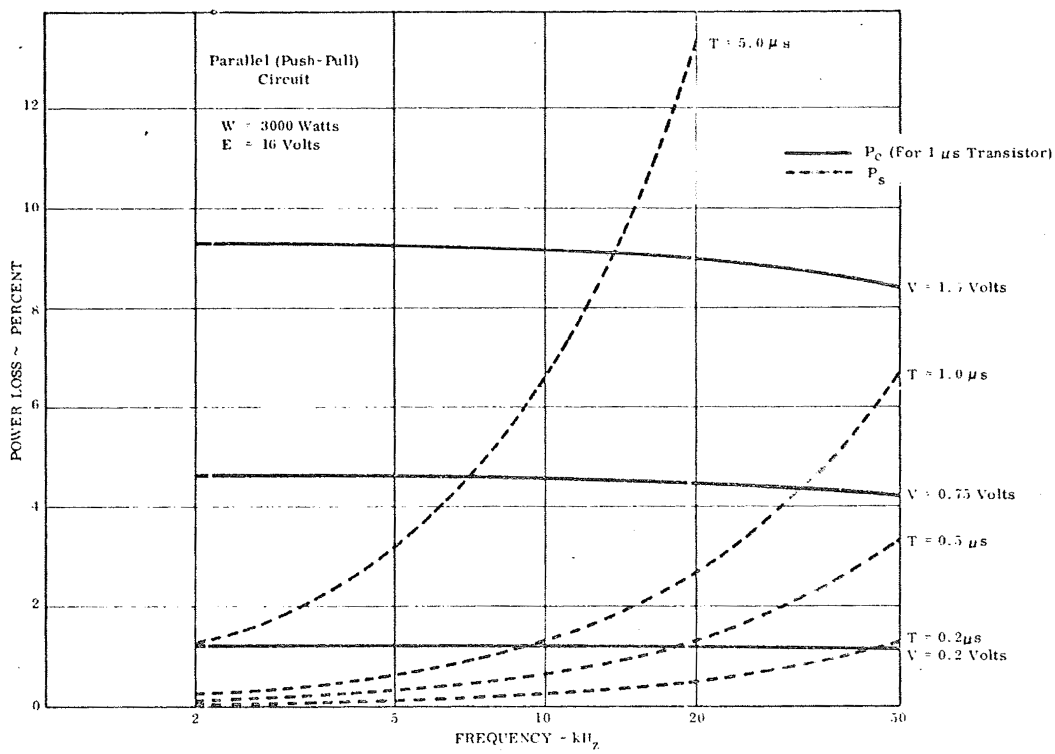

Figure 2-18. Transistor Conduction and Switching Loss

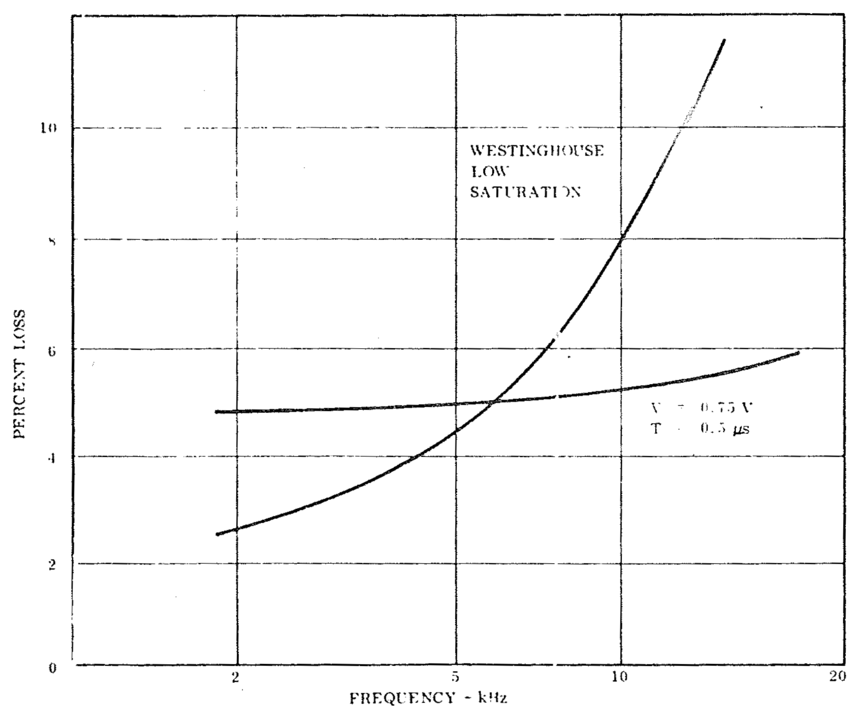

Figure 2-19. Comparison of Transistor Losses with Frequency

because the combined conduction and switching losses are lower than those of the faster transistors. Above 5.8 hKz, switching losses in the 5 microsecond transistor rise rapidly, and the faster transistors are preferred.

2.3.4.2.2 Operating Frequency Selection - The preferred switching frequency must be identified. As far as the transistors are concerned, regardless which type is used, losses rise with frequency, although for high speed transistors the rise is very gradual. Magnetic core losses in the converter power transformer also go up with frequency, but the amount of core material required decreases, and total core loss remains about constant. In general, constant efficiency transformers can be assumed. At very high frequencies, say 20 kHz and above, losses in output rectifiers must be considered. Otherwise, other losses can be considered to be independent of frequency.

The increased transistor losses associated with increasing frequency require additional radiator weight. On the other hand, magnetics weight drops with increasing frequency. A brief analysis shows that for an incremental power system weight of 50 pound/kW, in general the weight gain/penalty is less than 100 pound total over the frequency range of 2 to 20 kHz. Hence, the selection of frequency may be made considering other basis as well as a weight-efficiency tradeoff. An operating frequency of 10 kHz was selected because of previous design experience.

2.3.4.2.3 Transformer Material Selection - One of the factors which relates to both operating frequency and weight is the type of core material used in the power transformers. One of the candidates is a ferrite. This type of material has the advantages of relatively low density (about 5.3 gm/cm^3 compared to about 8.5 gm/cm^3 for electrical steel) and low core loss at high frequencies, but it has the disadvantages of relatively low saturation flux density and low Curie temperature, the temperature of about 180°C at which it loses its magnetic properties. In view of the operating temperature specified for the electronic equipment in this application including the attendent component thermal gradient and the unknown characteristics of ferrite in a nuclear environment, use of this type of material will not be considered. Its evaluation for the thermionic spacecraft application will be left to a more detailed design than this study permits.

2.3.4.2.4 <u>Inverter Characteristic Summary</u> - For purposes of this study, the power conversion equipment design is based on the following selections:

• Switching devices	High speed silicon transistors (Westinghouse 1776 - 1460)
• Operating frequency	10 kHz
• Magnetic core material	Electrical steel such as Hymu-80
• Module size	Full size for one TFE pair 196.9 amperes, maximum 11-17 volt with provisions for half voltage operation
• Reliability provisions	No additional circuit redundancy

2.3.4.3 <u>Component Size Identification</u>

A complicating factor in the use of power transistors in this application is their limited current rating compared to the total current delivered by the source. For example, the rating of the Westinghouse 1776 - 1460 is 60 amperes, whereas the EOM current of a TFE pair is 196.9 amperes. If the transistors are operated at 30 amperes, both to reduce the saturation collector-emitter voltage drop and to provide normal design margin for reliability, six transistors operating in parallel are required per group. The problems associated with operating many transistors in parallel are at least twofold: proper sharing of current and coordination of turn-off characteristics, especially storage time, so that the transistors in a group all turn off together and one transistor does not carry all of the current during the switching interval. Successful operation of parallel transistors in power converters is common. For purposes of this study it will be assumed that, by proper control of device characteristics during manufacture, possibly by device selection, and by special circuit techniques, proper operation of up to 10 power transistors in parallel can be achieved at the desired operating frequency without sacrifice of efficiency and with minimal weight increase for additional circuit components.

It should be noted that the saturation voltage drop of a transistor is a function of the transistor collector current. Hence, to within limits, low saturation drop of even ordinary power transistors can be achieved by operating them at low currents. In part, this is the reason for operating the selected transistors at half rated current.

Clearly, there are practical limits to the reduction in saturation voltage which can be gained by the transistor paralleling technique. For example, if 10 transistors are operating in parallel the addition of one will reduce the current per transistor by about 9 percent and will only reduce the saturation voltage by a similar amount (i.e., $V_{CE(SAT)10} = 0.44V$, $V_{CE(SAT)11} = 0.40V$). A separate study would be required to determine the optimum balance between saturation voltage and number of transistors.

A decision required in connection with power converter design is the basic size of the converter module. Conversion of all the power of one TFE pair can be performed in a single converter with a single power transformer. On the other hand the conversion equipment for a single TFE pair can consist of a number of small modules with their inputs connected in parallel and their outputs in series or parallel. The single converter has the advantage of lowest weight, but has the disadvantage of providing no redundancy. The modular approach has the advantage of a high degree of redundancy but the disadvantage of greater weight.

An estimate of the weight penalty can be made by the following reasoning. The bulk of the weight of a dc-to-dc converter is the power transformer. Let it be assumed that the transformer represents half the total weight, and that the transformer weight varies at the 3/4 power of its electrical rating (Reference 6).

Let W_{10} = weight of transformer for module of 10 modules

W_1 = weight of transformer for single converter

then

$$\frac{W_1}{W_{10}} = \left(\frac{P_1}{P_{10}}\right)^{3/4} = (10)^{3/4} = 5.63$$

Thus, the weight of a single, full size transformer is 5.6 times that of the transformer in a single module of 1/10 the power rating. Since 10 small transformers is the equivalent of a single large one from a power standpoint, ten small transformers would weigh 10/5.6, or nearly 1.8 times more than a single large unit. Since transformer weight is assumed to represent 1/2 total equipment weight and all other weight is considered to be equivalent in the two cases, the total equipment weight of 10 small modules will be 1.4 times that of a single large converter.

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The former analysis does not allow overating of the small modules to take advantage of redundancy. If the modular equipment were to be designed so that loss of a single module could be tolerated without loss of capacity, each of the 10 modules would have to be designed so that 9 could handle the total power output of the TFE pairs. Hence, each would have to be capable of handling 10/9, or 1.11 of its nominal power; in other words, each should be designed for 11 percent excess capacity. Such excess capacity has not been factored into the computations.

2.3.4.4 Redundancy Considerations

Since the flashlite reactor contains 108 TFE pairs, each of which represents a separate power source, it is assumed that no redundancy is required in the conversion equipment. A loss of one power converter channel represents a loss of less than 1 percent in the total power available from the reactor.

If redundancy is desired, however, some of the methods of providing it are as follows:

a. Use single, full capacity converters for each TFE pair and include additional converters which can be switched in in place of failed units. There would be significant difficulties in providing for fault detection and switching. This does not appear to be a practical approach.

b. For each TFE pair, provide a redundant full capacity converter so that if one fails the other can take over. This approach doubles the weight of the conversion equipment and appears prohibitive from the weight standpoint.

c. For each TFE pair, provide N converters in parallel, each with sufficient capacity so that one can fail and the others take over the full load without loss of power. This is similar to the second approach, except that more than a redundant unit would be provided. The penalty for this approach, as noted before, is one of weight: modularized equipment is simply heavier than concentrated equipment of the same rating.

d. Provide circuit redundancy rather than equipment redundancy. That is, instead of providing complete spare modules or converters, design the necessary conversion equipment conservatively and provide redundant circuits to minimize the probability of failure.

For the flashlite reactor system the no redundancy approach is selected for the following reasons:

a. With 108 individual power sources available, failure of any one converter channel represents loss of less than 1 percent of total power.

b. Study ground rules provide reactors designed to provide BOM power at EOM, even if 10 percent of the TFE units are lost due to failure.

c. To provide redundancy by additional converters represents a substantial weight penalty for the conversion equipment.

2.3.5 MAIN CONVERTER MODULE INTEGRATION

2.3.5.1 Reactor Integration

The main converter module detailed on Figure 2-16 is connected to the TFE pair through a limiter or fuse, the function of which is to open the circuit between the TFE pair and the converter in case of internal converter faults. The intention is to prevent physical damage within the converter because of high short circuit currents. It is recognized that operation of the fuse open circuits the TFE pair, and may cause overheating and failure of the TFE pairs. The alternative would be to provide some means of short circuiting the TFE's in the case of disconnection of the converter. In this initial study, short circuiting means are not provided because the condition of open-circuiting by converter failure is considered equivalent to open-circuiting of a TFE because of an internal fault. Consequently, there are no provisions against overheating for either a TFE failure or power conditioning failure. Future study should be performed to determine if a problem could exist.

Diodes across each TFE are included within the converter to provide a path for the current from the surviving TFE, in the event of open cir cuit failure of the other.

An input filter consisting of a capacitor and reactor is included in the converter design to limit the voltage swings at the input to the converter during those portions of the normal operating cycle when the converter transistors are off and the TFE pairs are unloaded.

At a 10 kHz switching frequency for the converters connected to each TFE pair, it can be assumed that the fluctuations in unfiltered TFE current, represented by converter switching with pulse width modulation, are not detrimental to the thermionic diodes. Diodes have long thermal time constants of several seconds at least, so the rapid switching will not affect instantaneous temperatures.

Filtering is not needed from the standpoint of the diodes. However, instantaneous changes in current between some large value and zero will cause large instantaneous changes in diode output voltages as shown in Figure 2-17, which presents typical I-V characteristics of the diodes. During the intervals when current is zero, diode voltage will go to rather high values. Hence, from the standpoint of protection of the converters, input filtering is required. In addition, the filter circuits provide nearly constant current flow in the low voltage leads during the converter switching, and effectively reduce the low voltage cable power loss. (Refer to Appendix C for the filter calculations.)

Solar cell systems should not require input filters because of the relatively constant voltage at low currents. Note the typical solar I-V characteristics at low currents on Figure 2-20 (10) in comparison to TFE characteristics on Figure 2-17.

Current transformers in the converters are included to provide signals representing TFE currents for system control for load sharing, reactor control and for telemetry information.

The two output voltage levels are created by separate secondary windings on the same single power transformers. Each output is furnished with its own fuse or limiter to protect the converter against physical damage in the event of a load fault or a distribution line fault. The alternatives to this type of protection for these faults requires further consideration.

The electric system performs load sharing control as well as voltage regulation.

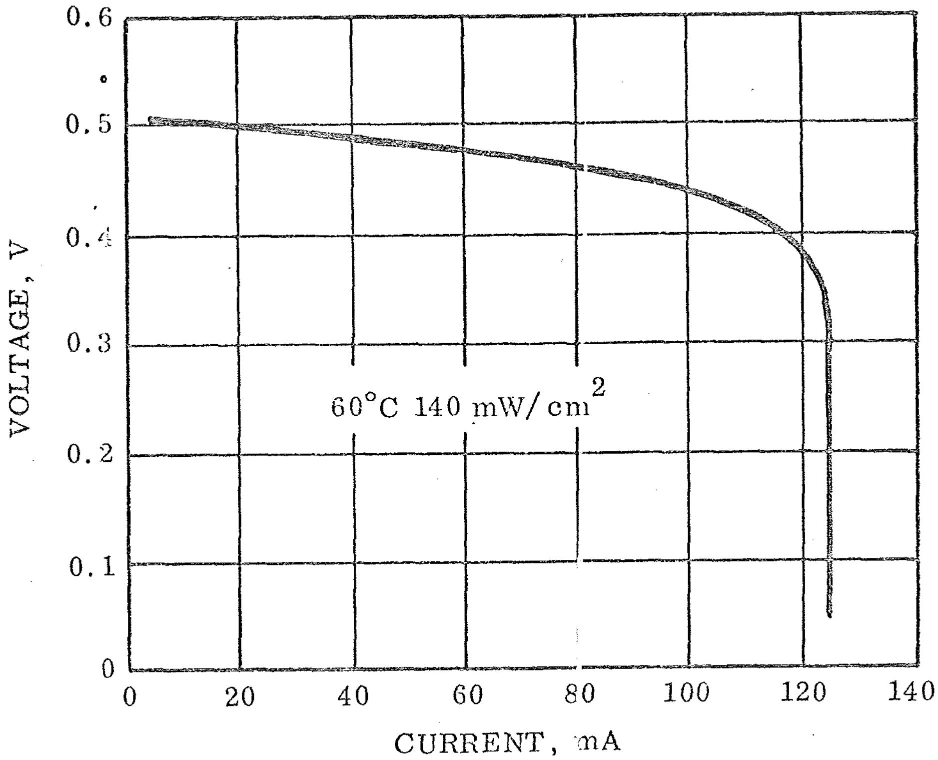

Figure 2-20. Typical I-V Curve of Thin Solar Cell

During each of the three modes of operation; normal, one-half voltage with one TFE failed, and 10 percent power, the load sharing by the TFE's controlled by pulse width modulation cycling of the individual converters. Control of the inverter conduction cycle relative to the non-conduction time is exercised by regulating circuits which sense the input current. Modifying functions to the control is the location of the TFE in the reactor, and whether the system is operating in the coast phase.

During normal and half voltage operation, when the principal load is the thruster screens and the high voltage output is utilized, voltage regulation is exercised by regulating circuits which sense the high voltage at the load bus and control the reactor operation to maintain this voltage constant. The 250-volt output is separately regulated by phase controlling SCR's as the rectifiers in its output circuit.

During the coast period when 10 percent power is required, the thrusters are deenergized, and there is no load on the 3100-volt bus. Reactor control is maintained by switching regulation to the 250-volt bus.

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To provide the necessary control functions three sets of control circuits are included. The first set operates on the main converter transistor base drives and regulates the TFE load sharing, sensed at the individual power converter input bus. The second set of controls regulate the output voltages by adjustment of the reactor and the SCR phase control of the 250-volt bus.

The third set of control circuits operates the contactor, which switches the main transistor groups from the normal to the 1/2-voltage taps. These control circuits sense voltage unbalance in the TFE pairs and operate the contactors if the voltages become unbalanced because of a fault in one of the TFE's.

2.3.5.2 Thruster Integration

The requirement that the power conversion equipment operate from individual TFE pairs so that the operating conditions of the TFE's be controllable individually, requires a system in which the outputs of the individual power converters can be combined electrically at a DC level. Several factors suggest that one of the voltage levels of the combined converter outputs be that required by the ion thruster screens, which in this case is 3100 volts.

- Three-quarters of the total reactor capacity is consumed by the thruster screens. Of the total of 300 kilowatts reactor electrical output, 223 kilowatts is required by the screens of the 31 operating ion engines.

- If another voltage were used for distribution, an additional conversion process would be required to provide screen power. Additional conversion is undesirable because it involves additional weight and additional losses.

- A high distribution voltage such as that required by the screen supplies tends to minimize conductor size and, hence, conductor weight in the distribution lines.

In order that a common screen supply be feasible several factors must be considered. If all screens are fed from a common supply all are interconnected electrically. Hence, it is necessary that such interconnection be compatible with the complete electrical system, including the thruster auxiliary power conditioners. Also, it must be possible to isolate individual thrusters from the common supply in the event that the thrusters fail on momentary arc-over.

To date operating experience has been confined to the operation of single thrusters with their own power supplies. There is no known case of the operation of several thrusters from a common supply. However, tests of three and more devices from a common supply are planned in the near future at General Electric in Evendale, Ohio. Studies have shown no significant problems.

Examination of thruster electrical connections which have been used to date (References 7 through 10) show that one side of the screen supply connects to system ground and the other to the thruster screen. A small resistor is usually inserted in the negative or ground lead to provide a signal representing screen current. The screen current signal in the common supply configuration can be derived satisfactorily either from a resistor in the positive lead or from a current measuring transformer electrically isolated from the screen power supply leads. Individual fault isolation also can be accomplished by including an isolating device such as a static switch, a relay contact, a fuse or some other circuit interrupting device in the leads between the common screen supply bus and the individual thrusters. Transient isolation to decouple the individual thrusters from the common screen supply can be achieved by the use of inductors in the lines between the common screen supply and the individual thrusters. In the event of arcs within the thruster, between the screens and the other electrodes, the inductors would prevent the current from changing abruptly and would absorb the supply voltage until the thrusters could be isolated from the screen supply by means of the individual circuit interrupters.

Thus, it appears feasible to operate all thrusters from the common supply and thus avoid multiple power conversion for the high voltage screen power. This is a major assumption in the design of the electrical system for the flashlite thermionic reactor system, and is the only identified technique to eliminate the additional losses and weight that would be associated with providing thruster isolation via a second power conditioning stage.

The technique identified for thruster isolation is presented in Paragraph 2.3.5.4.

2.3.5.3 <u>Power Distribution Voltage Selection</u>

The medium voltage power distribution is required to supply power to the remaining thruster loads and the payload and hotel loads. These are in two locations with 60 percent of the total load requirement near the reactor and with the remaining 40 percent near the spacecraft thrusters.

With 19 kWe load located approximately 40 feet from the supply, a reasonably high voltage is necessary for distribution to minimize cable weight and power loss. Since many electrical components and insulations are rated to operate to 600 VDC, allowing 50 percent derating, an optimum potential of 250 volts was selected.

A number of options exist in the manner in which the two output voltage levels, 3100 volts and 250 volts, can be created for the multiplicity of interconnected modules.

<u>High Voltage *Outputs Parallel - Medium Voltage **Outputs Parallel</u> - This concept requires high voltage rectifiers on the output of each module and an individual filter on each output.

One or more modules, designated the master module shall be designed for voltage regulation, with provisions for adjustment of the reactor.

If the high voltage outputs are designed to regulate the reactor characteristics, the medium voltage outputs would also be regulated adequately only if the medium voltage circuits had the same electrical relationships as the high voltage circuits. To insure proper voltage regulation of the medium voltage outputs, regulation must be provided by phase control of the medium voltage output rectifiers; hence the need for silicon controlled rectifiers (SCR).

<u>High Voltage Outputs Series - Medium Voltage Outputs Parallel</u> - Each high voltage circuit output would be relatively low voltage (3100/10, or about 30 volts), but each would have to carry full output current. Hence, rectifier power losses would be high, and would lower overall efficiency by approximately 4 percent.

Each module could be separately voltage-regulated, with the overall reference voltage level adjusted so that the high voltage is regulated.

Only a single high-voltage filter would be required. To make this filter small, the individual inverters could be staggered in phase relationship so that ripple frequency would be high.

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* 3100 volt screen supply
** 250 volt hotel/thruster supply

If each high voltage output were separately regulated, provisions must be made to allow the medium voltage outputs to share the load properly.

High Voltage Outputs Series - Medium Voltage Outputs Series - Individual low voltage outputs would be very low and corresponding rectifier losses would be high. This is not a satisfactory approach for that reason.

Separate Converters for High Voltage and Medium Voltage Outputs - High voltage outputs in series or parallel, medium voltage outputs in parallel, each separately regulated. The disadvantage of this approach is that two sets of converters are required, and weight would be high.

Of these alternates, the high voltage parallel/medium voltage parallel method, appears superior. It is heavier than the high voltage series/medium voltage parallel method, by virtue of requiring individual filters in the high-voltage circuit, but it is about 4 percent more efficient. For weight and efficiency calculations, this is the system which will be assumed. To achieve voltage regulation of the medium voltage circuits, phase-controlled output rectifiers are assumed, although it is recognized that the additional regulating loop thus created may be difficult to construct due to the affect upon system stability. These can be examined in a more detailed study.

2.3.5.4 Screen Circuit Control

Each individual thruster screen is fed from the common high voltage bus at the thrusters through a series network consisting of a high speed electronic switch (SCR) and a series reactor (L). The SCR interrupts the circuit between the power supply and the thruster screens in the event of arcs within the thrusters, as detected by a sudden drop in voltage at the screens, the appearance of voltage across the series reactor, L, or some other signal. Following circuit interruption by the SCR, energy stored in L continues to supply power to the arc for a period of up to two milliseconds. The SCR remains off for a period of 0.2 second to allow time for the arc to clear and thruster conditions to return to normal. After 0.2 second, the SCR is again switched on, reestablishing screen voltage and hopefully restoring full thruster operation. If, for example, the arc restrikes two more times within a short period of time, say, 5 seconds, the screen supply to that thruster and the inputs to the auxiliary power supplies for that thruster are permanently disconnected. This thruster is considered completely disabled and one of

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the six spare thrusters is placed automatically into operation to replace it.

During the spacecraft coast period when the thrusters are not required to operate, power to the thrusters is disconnected by the static switches in the screen supplies and by the contractors in the input circuits to the auxiliary thruster power supplies.

A simplified schematic diagram of the static switch used as the screen circuit interrupter is shown in Figure 2-21A. A number of SCR's are connected in series to withstand the high voltage of the screen supply, and there are resistor networks across the SCR's and the resistor-capacitor networks to provide for proper steady state and transient voltage division.

One advantage of using individual interrupters for each thruster is that each thruster screen supply conceivably can be fed by its own individual transmission line. This provides a measure of redundancy in the transmission system. The same system of individual transmission lines, however, can also be employed with a single switch, if located with the power conversion equipment near the reactor. Another consideration is that the interruption of the high voltage screen supply must be coordinated with the interruption of other supplies, including the vaporizer, arc, screen, and accelerator supplies.

As already described, the electrical system configured for the flashlite thermionic reactor is based on the use of a common screen supply, with individual static circuit interrupters provided for each thruster. The interrupters operate immediately upon the development of a fault and series inductors provide the energy necessary to clear the fault, as well as providing momentary, transient circuit isolation during faults. In order to minimize system weight, it is assumed that electromechanical switches for permanent circuit interruption are not required.

One additional alternative could be considered. The above discussion suggests that each thruster screen circuit be provided with its own circuit interruption device. This suggests that any thruster that arcs, internally, can be isolated individually for the required period in order to allow the arc to clear, without interrupting power to the other thrusters. Another mode of operation is to interrupt power to all thrusters when an arc occurs in any one of them and then to re-apply power after the prescribed delay period, allowing the arc to clear. This somewhat reduces the total average thrust, but also reduces the amount of switching equipment required from 37 pieces to one piece. Of course, some means still must be provided to permit isolating individual thrusters should total thruster failure occur. These alternatives are illustrated in Figure 2-21. This latter arrangement, in effect, treats the entire thruster as a single thruster except that it allows isolation of the individual pieces when necessary.

Further study may be required to firmly decide between these two approaches. The system which provides separate isolation for each thruster is preferred at this time, because of its greater reliability.

2.3.6 MAIN CONVERTER MECHANICAL DESIGN

2.3.6.1 Geometry

Components of the main power conditioner are to be mounted using a baseplate integral to the radiator. Figure 2-22A shows the components configured within a one square foot area. The suggested layout was designed to accept power at one side and have the outputs on the opposite side, thus simplifying the component construction, testing and integration.

2.3.6.2 Component Size

The following components have been selected for use in the main power conditioners. Weights for each device is shown in Table 2-8.

- Input Filter

  Inductor: 2.0" x 4.0" dia.
  5 h, 7 turns, 5 cm length 10 cm diameter
  Awg #4, copper wire

  Capacitor: 1.3 x 2.5 x 3.0" H
  4 - GE-KSR Tantalum Foil,
  200 μ f, 100v, type 29F3265

- Bypass Rectifiers: 2.5" x 1.2" dia.
  200A, 200v
  Type GE-IN3264

- Inverter
  Transformer: 5.0" x 3.0" x 4.0" H
  Electrical steel, Hymu-80
  Input: 14-18VDC, 196.9A maximum
  Outputs: 3100VDC, 2.3A
  250 VDC, 1.7A
  Tapped Primary

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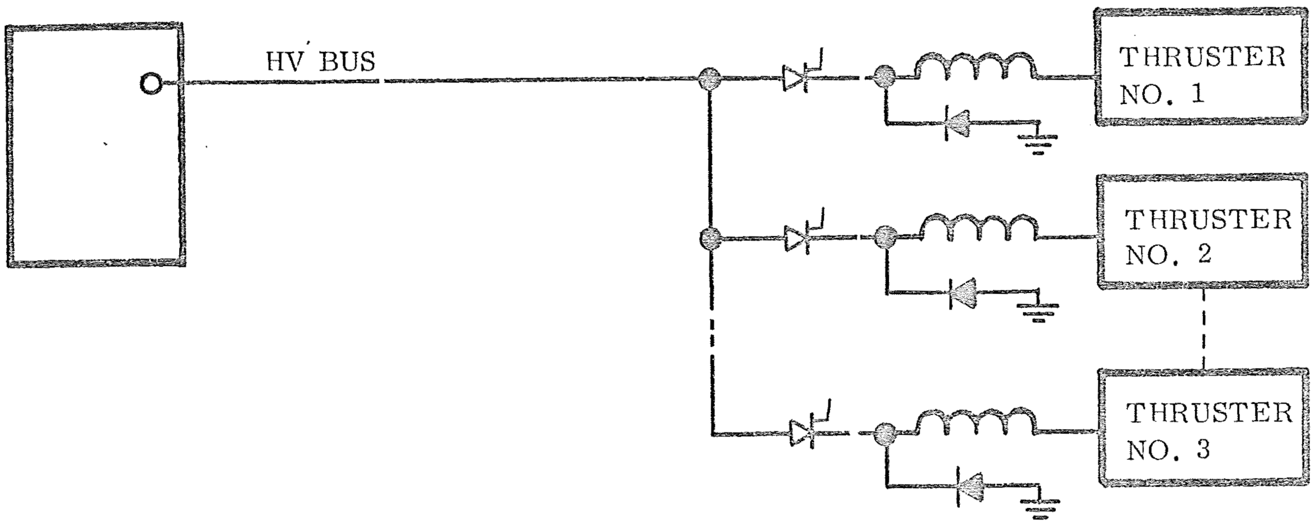
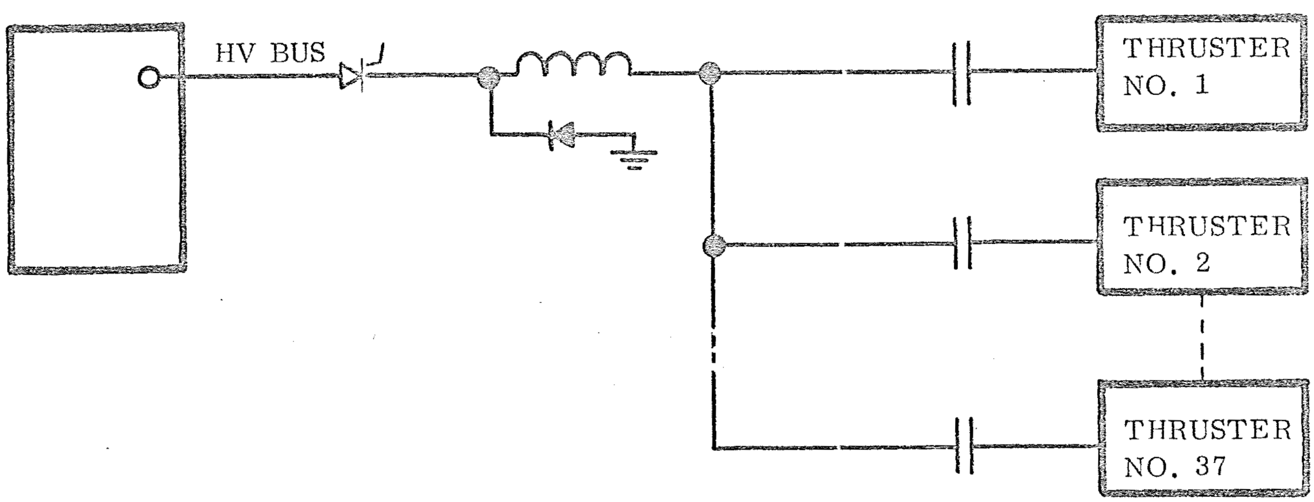


A. INDIVIDUAL SCREEN CIRCUIT INTERRUPTION



B. COMMON SCREEN CIRCUIT INTERRUPTION

Figure 2-21. Alternative Screen Supply Interruption Techniques

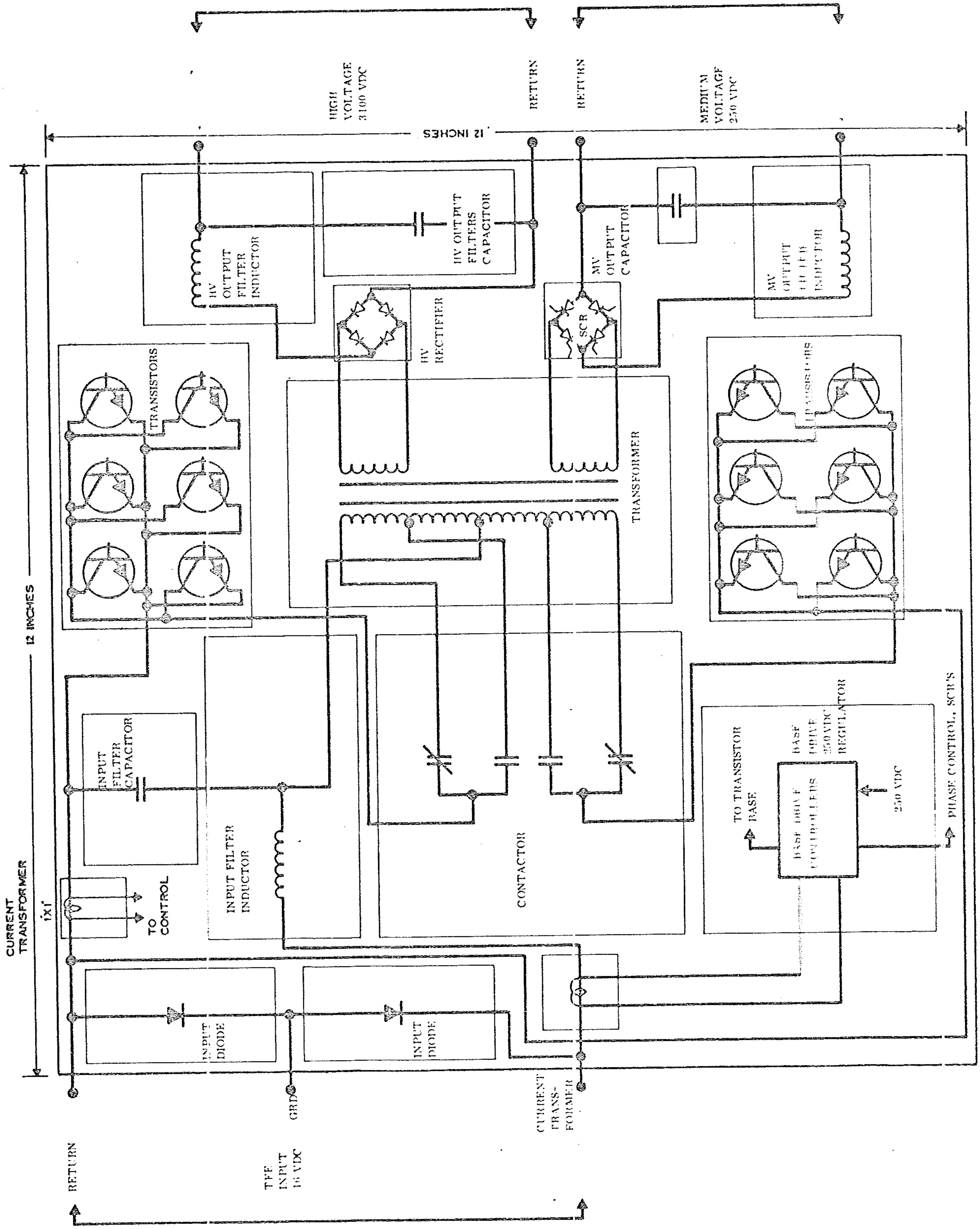


Figure 2-22. Component Geometry Main Power Converters

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- Inverter (Continued)
  Transistors, mounted on two panels bonded to radiator,
  as illustrated on Figure 2-23.
  Six transistors/heat sink transistor
  Transistor type: Westinghouse 1776-1460
  0.5" x 0.9" dia.
  60A, 140v

- HV Output Rectifiers: Bonded block, 1.0" x 1.0" x 0.5"H
  12 diodes/block, 3 diodes/branch
  Diodes: 3A 800v
  Type: GE-A15N
  0.15 x 0.2 dia.
  Axial lead

- HV Filter: Inductor: 2.25" x 1.8" x 1.8" H
  8 cu inches

  Capacitor: 3.8" x 1.6" dia.
  Axial

- MV Output Rectifiers: 0.4" x 0.3" x 0.6" H
  3-Silicon controlled rectifiers
  Stacked flat pack
  SCR: Similar to Type GE-C106
  0.4" x 0.3" x 0.2" H

- MV Filter:
  Inductor: 2.0" x 1.5" x 1.0" H

  Capacitor: 1.0" x 3.6" dia.
  Tubular tantalum foil

- Contactor: 4.0" x 4.0" x 3.0" H
  250A, 120VDC, DPDT, latching

- Control Circuits: 3.0" x 3.0" x 1.5" H
  (Base drive, SCR 5 control boards
  phasing) 2 power transistors, similar to 1776-1460

- Current transformer: 1.0" x 1.0" x 1.0" H
  2 toroids and power supply

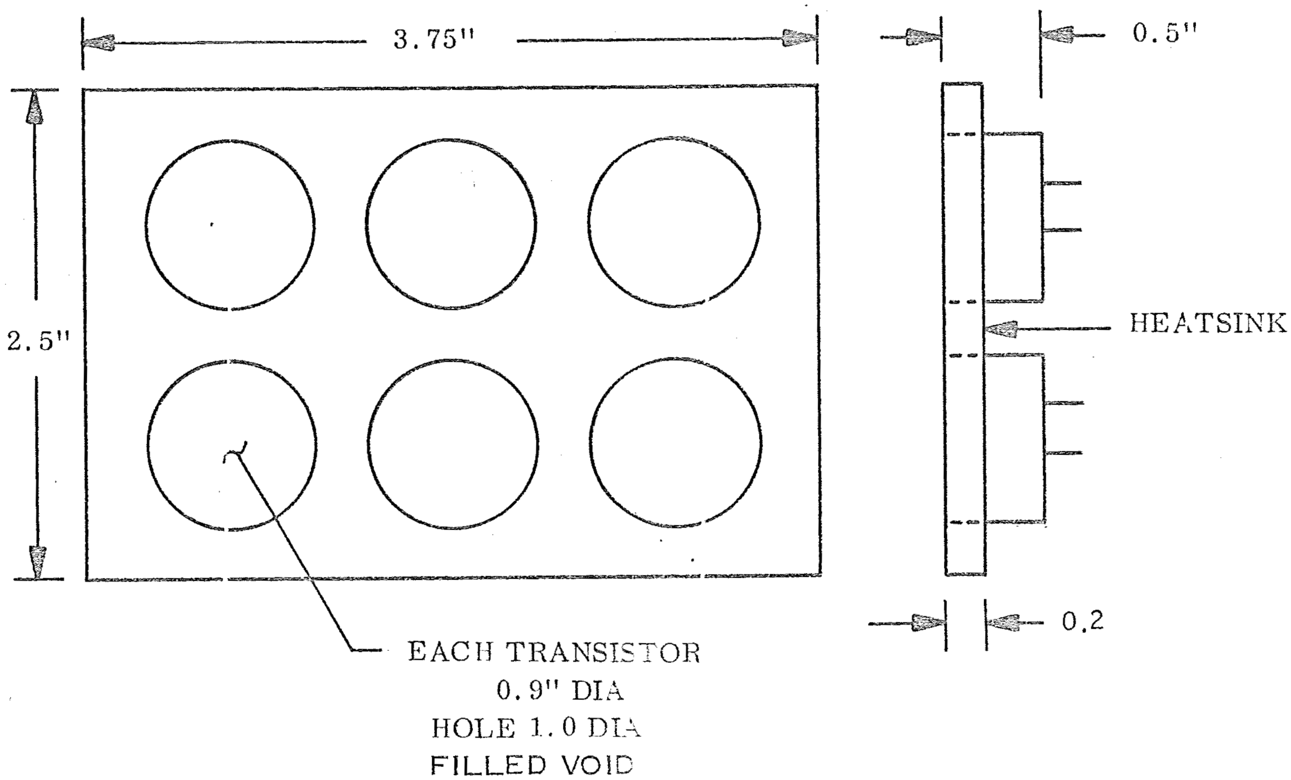

Figure 2-23. Transistor Mounting Detail

2.3.7 AUXILIARY POWER CONDITIONING

2.3.7.1 EM Pump Power Conditioning

DC conduction electromagnetic pumps were selected for use with the thermionic reactor system. These pumps require very high current at very low voltage, specifically for the primary pump, 5000 amperes at 0.5 volt. Special additional power conditioning equipment, therefore, is necessary. Using conventional power conversion schemes for very low voltage, efficiencies of less than 50 percent are encountered. With DC-AC-DC conversion, the voltage drop in the output rectifiers approximates or exceeds the output voltage required and hence the efficiency is poor.

In order to obtain the extremely low DC output voltage required at the pumps, standard low-voltage conversion to a higher output voltage is performed and several pumps are connected in series. Now, with the rectifiers dropping 0.7 VDC and the output typically 10 VDC, an efficiency of approximately 85 percent is realizable.

Two power conditioners are used in the system. One feeds the main coolant loop primary and secondary pumps which require 10 kW each. The other feeds the auxiliary pump, the shield pump, and the propellant pump, requiring an estimated 0.1 kW each. Each of the primary and secondary pumps are assumed to be divided into 10 parallel fluid ducts, requiring 0.5 volt for each duct, all connected in series. The conditioner would have an efficiency of 85 percent as previously mentioned.

The remaining pumps are single duct machines, which when connected in series require a power conditioner to supply 0.3 kW at 1.5 volts DC. Efficiency for this supply would be approximately 60 percent, but for this relatively low power level the loss would be about 200 watts.

Power conditioner circuits are a conventional parallel-commutated SCR inverter with a counter-tapped transformer combined with a low voltage rectifier as shown in Figure 2-24. A standard 8 pound/kWe output has been applied for weight estimation for the main EM pumps. The characteristics of the power conditioning for the EM pumps are summarized on Table 2-11.

An important consideration in making the decision to use DC conduction pumps was the reactive power weight penalty occasioned by using AC induction pumps. For example, the five pumps in the system using three-phase AC power, required an estimated 1150 pounds for power conditioning. This would be approximately equivalent to 30 pound/KVA, which with a 0.55 power factor would be about 50 pound/kW. Largely, the weight would be increased due to the capacitors necessary for correction of the power factor and to the cabling cross-sectional area increase necessary to handle the reactive volt-amperes.

2.3.7.2 Other Auxiliary Power Conditioning

Auxiliary power conditioning is also required for the following operations:

- Reactor Control
- Power Plant Control
- Special Ion Engine Units
- Spacecraft Guidance and Control
- Payload Power Conditioning.

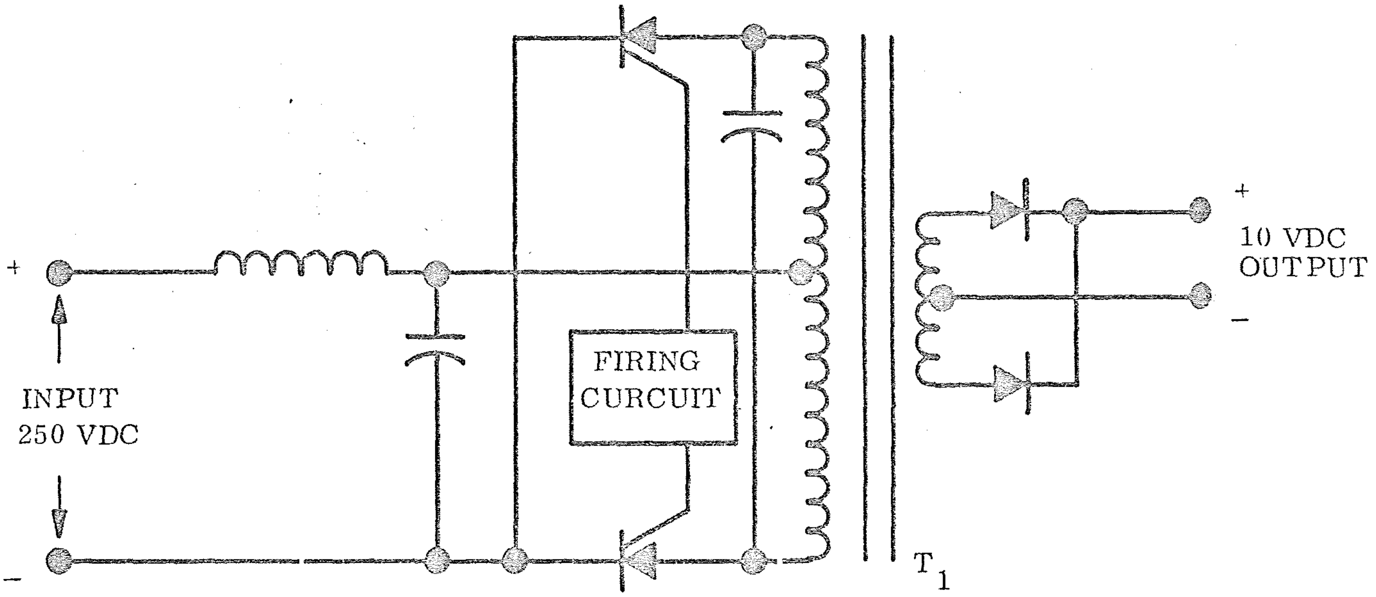

Figure 2-24. DC-EM Pump Power Conditioning
Parallel-Commutated SCR Converter

The weight of the power conditioning for all these units, except the special ion engine units, is greater than the 8 pounds/kWe output employed for the main EM pumps because of the smaller size of these special purpose units, as shown in Table 2-11. The weights presented in Table 2-11 for the special ion thruster units are those provided by JPL. The efficiency of these auxiliary units is 90 percent. No losses are shown for the special ion engine units, since this power loss is already factored into the ion thruster efficiency used to calculate the beam power.

2.3.8 ELECTRIC CABLE DESIGN

Three sets of power distribution cables are required for the flash-lite reactor electrical system. Low voltage cables conduct power from each TFE pair to the corresponding power conditioning module, and medium voltage cables distribute power from the medium voltage bus in the power conditioning bay to the hotel loads near the reactor and near the thrusters. Screen supply power is distributed via the high voltage cables from the power conditioners to the thrusters.

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TABLE 2-11. AUXILIARY POWER CONDITIONING CHARACTERISTICS

Component Application	Power Input kWe	Efficiency Percent	Weight Pounds	Power Losses watts(e)
Main EM Pumps	23.5	85	160	3500
Auxiliary EM Pumps	0.50	60	10	200
Reactor Control	2.22	90	15	222
Power Plant Control	0.50	90	10	50
Spacecraft Control	0.50	90	10	50
Special Ion Engine Units	17.0	90	272	---
Payload Units (included in 2200 lb payload weight)	1.0	90	30	100
TOTALS			507	4122

In selecting the materials and cross-section area of the various cables, a weight optimization was performed. An optimization was made between cable weight and the corresponding inverse electrical losses reflected in the compensating power plant weight.

The following expression relating the cable weight to the power loss was developed:

$$\text{Total Weight} = \frac{\rho\,(I)^2(L)^2(D)}{P} + 0.05\,P$$

where ρ = resistivity of the cable material
I = current
L - length
D = density of the cable material
P = power loss

The first term is the cable contribution and the second term is the power plant weight using a plant specific weight of 50 pounds/kWe.

The relation for minimum power loss is obtained by taking the first derivative of the weight equation and solving for minimum power with the derivative equal to zero, which yields:

$$P_{MIN} = \left(\frac{K}{0.05} \right)^{\frac{1}{2}}$$

where

$$K = (I)^2 (L)^2 D$$

Optimized weight for the cable is then obtained by use of the first term of the weight equation. The corresponding optimized cable cross-section can be calculated once the power loss is known.

Characteristics of the following materials were examined for application to the three cable set requirements:

- Copper (Cu)
- Aluminum (Al)
- Sodium (Na)
- Beryllium (Be)
- Nickel Clad Silver (NiAg)
- Nickel Clad Copper (NiCu)
- Sodium Potassium (NaK)

Figure 2-25 shows a comparison of the leading candidate materials for the low, medium and high voltage cables. On the basis of weight/power loss optimization and mechanical integrity at the operating temperatures, the materials summarized on Table 2-12 were selected for the cable sets. The detailed optimization results for each of the three cable systems are presented on Figures 2-26 through 2-28.

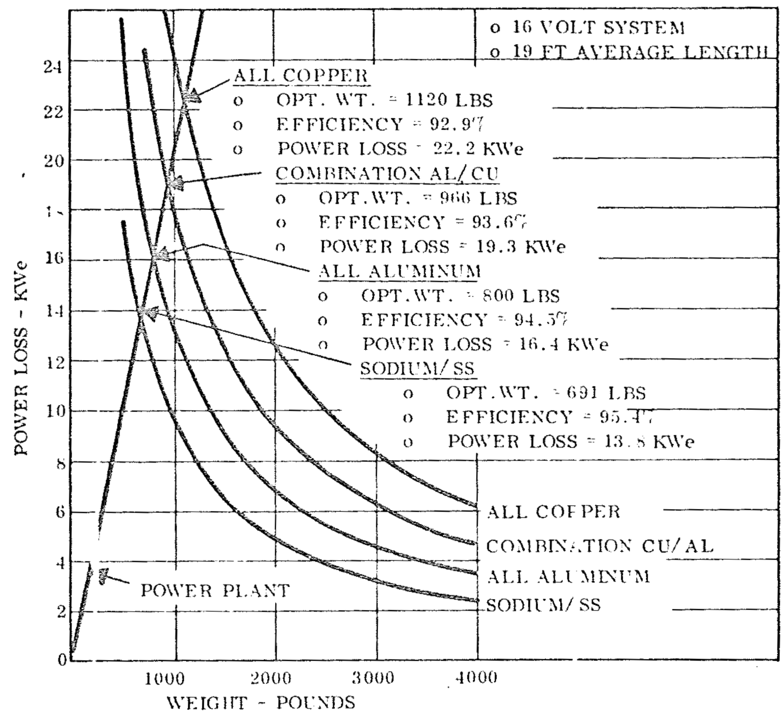

Figure 2-25. Low Voltage Cable System Weights

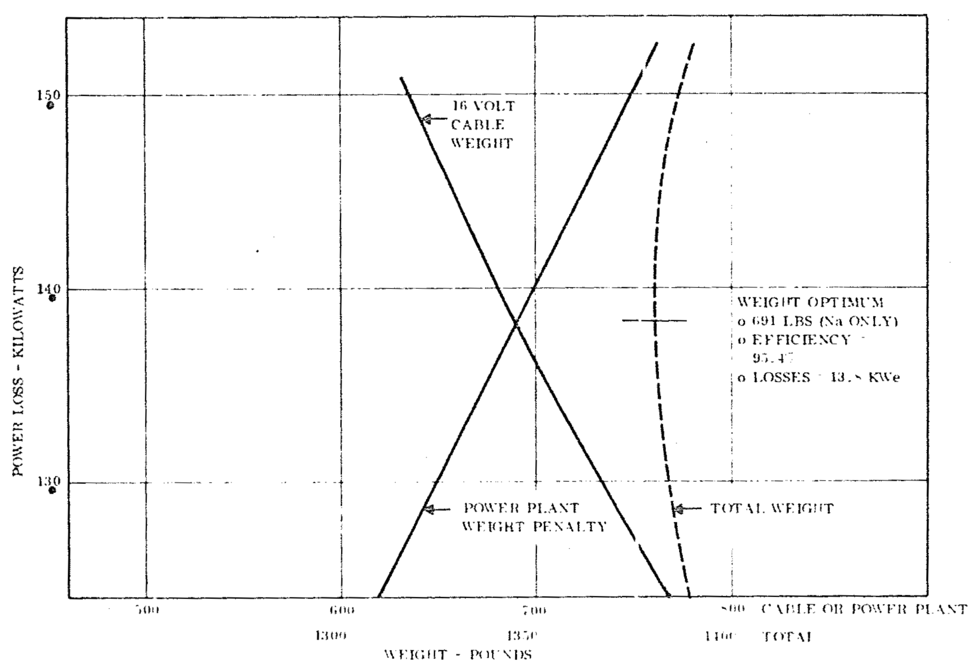

Figure 2-26. Detailed Weight Optimization, Sodium-Stainless Steel Low Voltage Cable System

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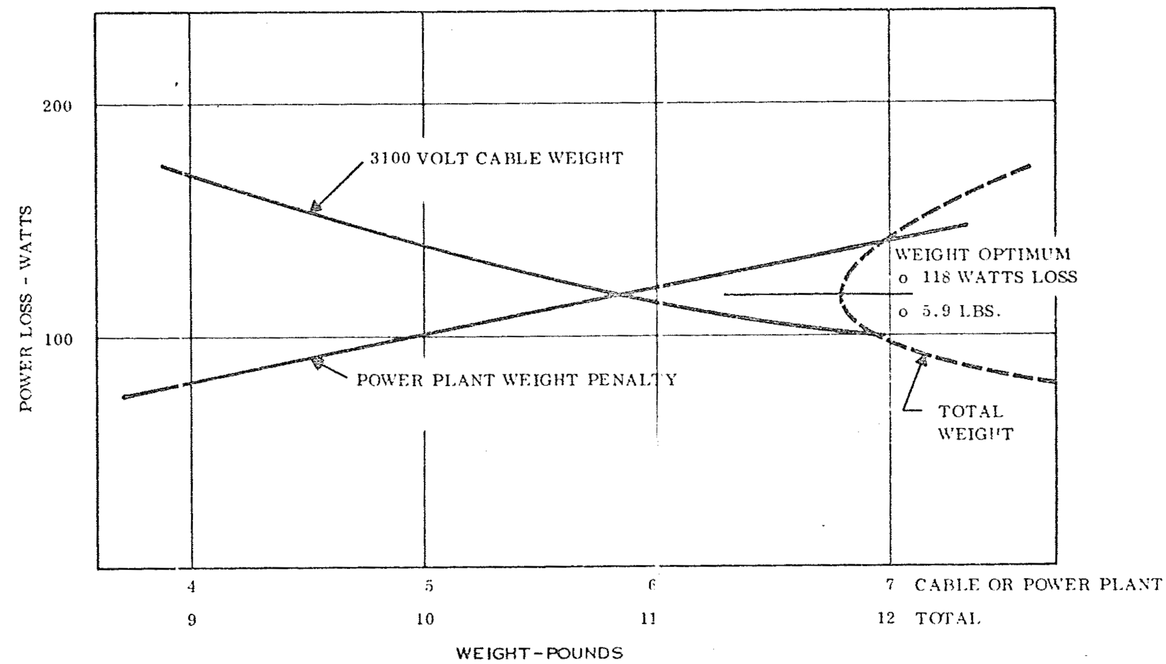

Figure 2-27. Detailed Weight Optimization, Aluminum High Voltage Cable System

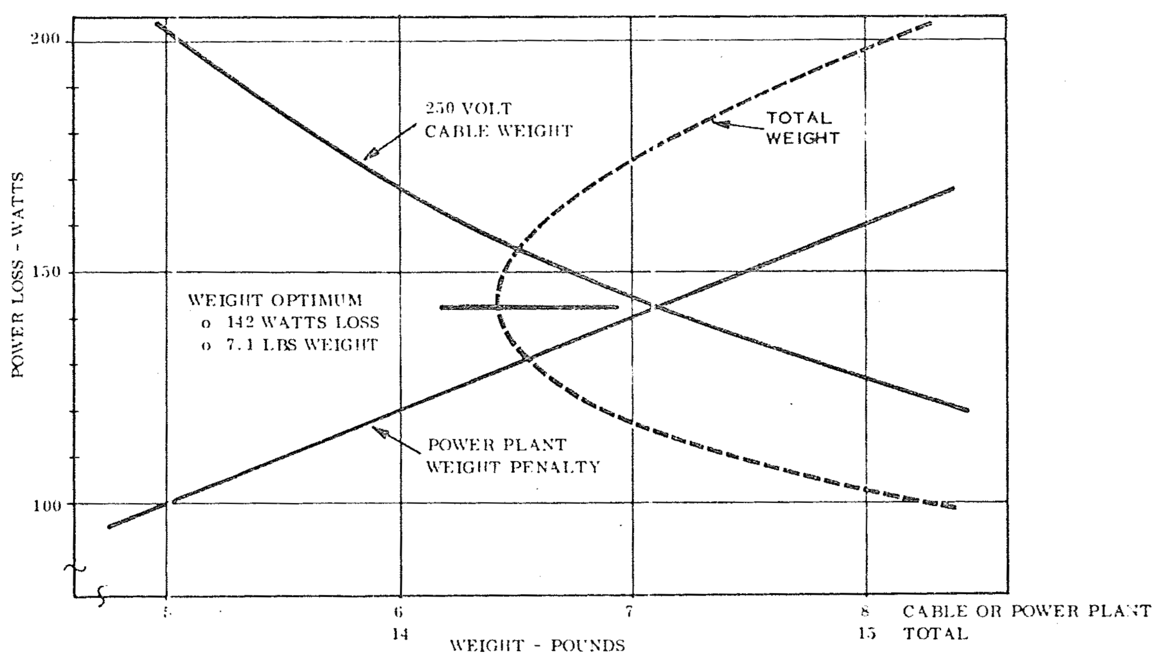

Figure 2-28. Detailed Weight Optimization, Aluminum Medium Voltage Cable System

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TABLE 2-12. SELECTED POWER CABLE CHARACTERISTICS

Cable	Material	Weight	Power Loss
Low Voltage	Sodium/Stainless Steel	917	13,800 W
Medium Voltage			
Reactor Area	Aluminum	7	140 W
Thruster Area	Aluminum	5	50 W
High Voltage	Aluminum	6	120 W

The total weight for all three cable systems is 935 lb., and the total power loss is approximately 14,100 watts, or 4.67 percent overall.

2.3.8.1 Low Voltage Cables

Considering the low generation voltage (16 volts), the quantity of power to be transmitted, and the transportation distance, selection of the low voltage cable material is more critical than that for the other cables. An optimized system using all copper would weight 1120 pounds, and would incur a power loss of 22.2 kWe for an efficiency of 93 percent. An all aluminum cable, which would not be satisfactory due to its low strength in the high temperature reactor equipment bay, would weigh 800 pounds, and would be 94.5 percent efficient. A cable which does have acceptable strength characteristics and could be used as an alternate is a cable composed of copper in the reactor/shield area combined with an aluminum cable for the lower temperature areas. The combination cable would weight 970 pounds, with an efficiency of 93.6 percent.

The selected cable configuration is sodium enclosed in a 10-mil thick stainless steel tube, weighing 690 pounds, for the sodium and 227 pounds for the tube. The cable is connected to the TFE flexwire at the reactor and is wrap-routed outside the shield and then routed along the outside of the power conditioning bay, penetrating into the bay at the appropriate power conditioner. The current carrying capability of the stainless steel was neglected in the optimization. Considering the stainless steel, the losses would be lowered, which means the cable diameter could be reduced, resulting in a cable weight reduction. On the other hand, keeping the cable size constant, the efficiency could be improved. Consequently, the total weight of 917 pounds and efficiency of 95.4 percent would be a worst case.

2.3.8.2 Medium/High Voltage Cables

Weight optimization shows that aluminum is the preferred, readily available material for the medium and high voltage cable material, but other materials such as copper may be used. The conductor material selection is not critical when the cable weights and power losses are compared with other system contributors. Nor are cross-sectional areas of the conductors critical, within limits.

2.3.9 COMPARISON WITH SOLAR ELECTRIC PROPULSION POWER CONDITIONING SYSTEM WEIGHTS

Table 2-13 summarizes the power conditioning system weights for a solar electric propulsion system under the following assumptions:

a. The component weights are those identified for the flashlite reactor power conditioning

b. The overall power level is 300 kWe

c. The power conditioning is accomplished by 108 separate units, each nominally rated at about 3 kWe

d. The solar electric power conditioning is assumed to be 90 percent efficient for purposes of computing radiator weight.

Comparison of these results, which indicate a potential for 6.6 pounds/kWe overall, with the data of Tables 2-7 and 2-8 shows that the major causes for the higher weights for the overall power conditioning for the flashlite thermionic reactor spacecraft are in the requirements for the following components:

a. Input Filter

b. Auxiliary Plant Power Conditioning

c. Screen Supply Interrupter

Of these major areas, both the input filters and the auxiliary plant power conditioning for items such as EM pumps, reactor and power

TABLE 2-13. SOLAR ELECTRIC POWER CONDITIONING SUBSYSTEM WEIGHTS*

COMPONENT	WEIGHT LBS.
Inverter	
Power transformer	4.0
Transistors	1.0
Base drive circuits	0.5
Output rectifiers	0.05
Control circuits	0.5
Hardware	1.5
Total for one 3 kWe (nominal) module	7.55
Total for 108 - 3 kWe (nominal) modules	815
High voltage filter	30
Thruster Auxiliary P.C.	278
Subtotal 3.7 lb/kWe @ 300 kWe	1113
Radiator (90% efficient P.C.)	880
Total 6.6 lb/kWe @ 300 kWe	1993

* Based on flashlite reactor power conditioning component weights.

plant control, will be required for all three reactor concepts. However, the total input filter weight will be less for the pancake and externally fueled reactor based power plants because fewer units (perhaps only one), and not 108, will be required.

It is noted that the radiator weight quoted for the solar electric PC system is estimated to be about 30 percent lighter than that currently estimated for the flashlite reactor. This occurs because it is assumed that the solar electric system would not be subject to area restrictions imposed by launch vehicle shroud length, which result in thicker, heavier fins, relative to a minimum weight radiator. This effect may also apply to the power plants based on the pancake and flashlite reactors, and is one of the variables being optimized in the design of all three thermionic spacecraft.

2.4 PRE-STARTUP TEMPERATURES IN EARTH ORBIT

An investigation of radiator panel temperatures was conducted for a typical fin-tube geometry in a 750 nm sun oriented, ecliptic orbit. It is shown that the panel temperature falls below 12°F during the shade portion of the orbit, thereby freezing the NaK coolant. In the absence of an insulated preheat or an auxiliary energy source the radiator coolant will be fluid only on the sun side and for a small portion of the shade side. Therefore, system startup would have to occur on the sun side of the orbit.

2.4.1 ANALYSIS

A critical aspect of spacecraft heat rejection system design is the behavior of the radiator under startup conditions. Fundamental to the problem of startup is the necessity for the radiator to respond to increasing power loads. This requirement demands that the radiator coolant be in a fluid condition when startup is initiated.

A study was performed to estimate if the coolant in the Thermionic Spacecraft radiator system would freeze during the launch and orbit stabilization period. Since the launch time, trajectory and other specifics are unknown at this time, the object was to select a typical situation and assess the severity of the radiator startup problem. The assumptions used in this investigation include:

a. Conduction fin offset - tube geometry, stainless steel armor, stainless steel/copper fins (See Figure 2-29)

b. Incident heat flux varies with position as in a 750 nm ecliptic orbit

c. NaK (78 wt % K) radiator coolant - freezing temperature of 12°F

d. Radiator emissivity and solar absorptivity = 0.9

e. NaK is pumped into loop just prior to startup, therefore, its latent heat of fusion does not contribute to radiator heat capacity

f. The radiator is cylindrical and is slowly rotating.

The analysis was performed with the THTD heat transfer code.

2.4.2 RESULTS

The results obtained from the analysis are shown in Figures 2-30 and 2-31. Examination of Figure 2-30 shows that for a wide range of radiator temperatures at the beginning of the sun portion of the orbit, the temperature of the radiator will reach approximately 120 to 140°F by the time it starts the shade portion. However, this situation results in a radiator temperature of -15°F by the time the vehicle again receives solar flux. In order for the radiator to remain above 12°F during the entire orbit, it must begin the swing behind the earth at about 310°F. The assumption that the NaK is not in the radiator is not required. Its effect is to reduce the temperatures during heatup by about 10°F, and increase the temperatures during cool down by the same amount, relative to the data of Figures 2-30 and 2-31.

Whether or not the radiator will require pre-heating, insulation or an auxiliary power supply will depend on the startup power profile of the remainder of the system. A distinct possibility is present for system startup during the sun portion of the orbit, or using an orbit where a greater part of the time is spent in the solar flux. Alternately, an orbit with a beta angle other than 90 degrees may be selected. The radiator average temperature as a function of beta angle (angle between the sun ray and the orbit plane) for an isothermal cylindrical shape at an altitude of 750 nm is shown on Figures 2-32 and 2-33. The cylinder considered was oriented with its roll axis parallel to the earth's surface, and perpendicular to the earth's surface. The ends of the cylinder were assumed to be blocked from seeing the external sink. The external conditions used were nominal, in terms of solar, albedo, earth and day of year.

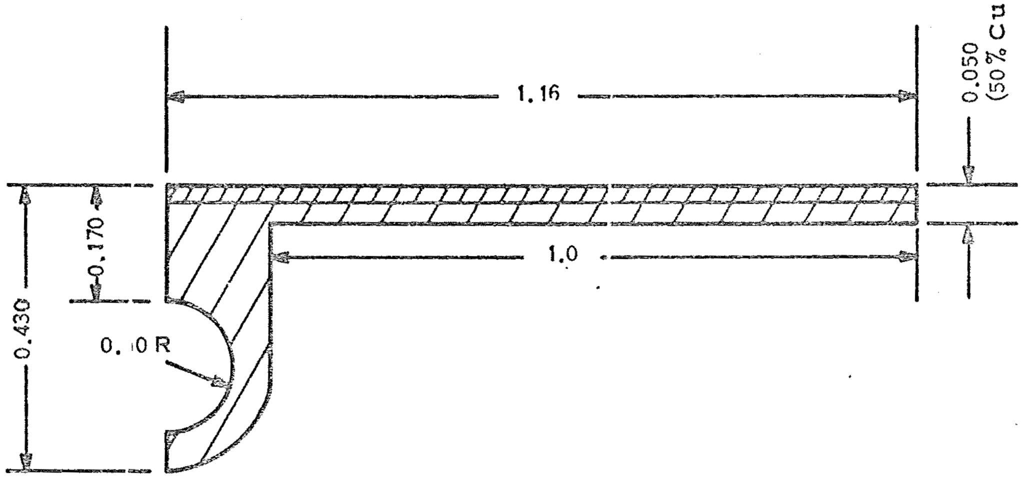

Figure 2-29. Model for Thermionic Spacecraft Radiator Startup Study

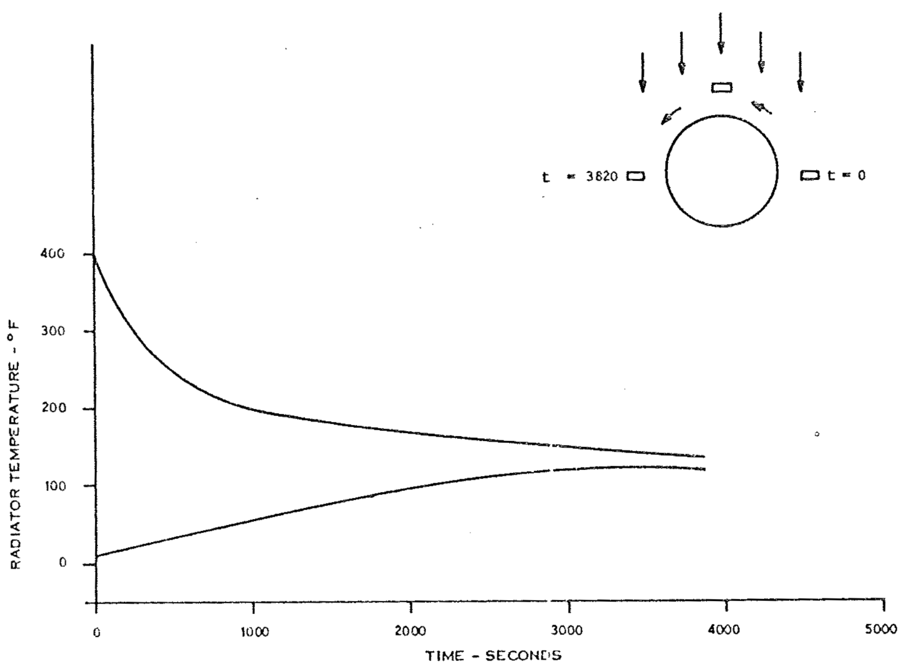

Figure 2-30. Radiator Temperature on Sun Side

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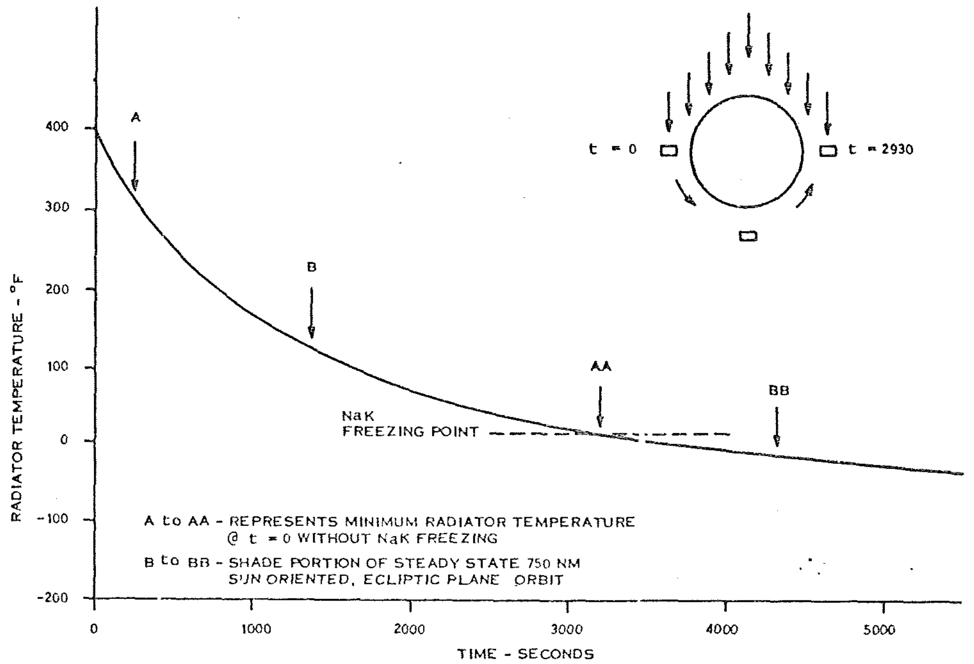

Figure 2-31. Radiator Temperature on Shade Side

The curves labeled orbit average in Figures 2-32 and 2-33 show the temperature for the whole body averaged over the orbit. Maximum instantaneous is the highest temperature during the orbit and minimum instantaneous is the lowest. For the case with the roll axis parallel to the earth's surface, the minimum temperature is -144°F and for the perpendicular case the minimum temperature is -175°F, when the beta angle is approximately less than 60°. The amount of shade time during which the sink is this minimum value can be found by referring to the curve in Figure 2-34 which gives the amount of shade time as a function of beta.

Clearly, proper selection of the earth departure orbit will eliminate the need for special startup heating or insulation for NaK-78 cooled power plants.

2.5 POWER PLANT WEIGHTS

This paragraph presents the current spacecraft weights identified or estimated for that spacecraft based on the bonded, wet cell flash-lite reactor, as summarized on Table 2-14 and discussed below. The weight data presented include both new component estimates, as well as those carried over from the earlier part of the study (Reference 1).

TABLE 2-14. SUMMARY - CURRENT THERMIONIC SPACECRAFT WEIGHTS FOR BONDED WET CELL FLASHLIGHT REACTOR.

Subsystem/Component	Weights, Pounds			
Propulsion System				17853
Power Plant Subsystem			11442	
Reactor[a]		2960		
Permanent Shield		1500		
Radiators		2072		
Primary	1940			
Auxiliary	90			
Shield	42			
Coolant Loops		2015		
Reactor[b]	355**			
Heat Exchanger	175			
Primary[b]	1385**			
Auxiliary[b]	60**			
Shield[b]	40**			
Power Cables		929		
Low Voltage	917			
Hotel Load	12			
Power Conditioning Electronics (Hotel Load)		195		
EM Pumps[c]	170			
Reactor Control	15			
Power Plant Control	10			
Power Conditioning Radiators (Hotel Load)		191		
EM Pumps[c][d]	176			
Reactor Control[d]	10			
Power Plant Control[e]	5			
Structure (Launch Support)		1530		
Radiators[f]	695			
P/C/G Bay[g]	265			
Thruster Bay[h]	340			
Launch Vehicle Adapter[i]	230			
Power System Control		50		
Thruster Subsystem			6411	
Ion Engine Subsystem		1233		
Ion Engines (37 units, incl. 6 spares)	585			
TVC Unit	548			
Miscellaneous	100			
Power Conditioning Electronics		3592		
HV Power Supply	2960			
Special Ion Engine Units	272			
Thruster Isolation (TFE Reactors Only)	460			
Power Conditioning Radiators		1600		
HV Power Supply	1470			
Special Ion Engine Units[j]	75			
Thruster Isolation (TFE Reactors Only)[j]	55			
High Voltage Power Cables		6		
Propellant System				15020
Propellant (Hg)			14500	
Tanks, Feed Lines, Pumps, Valves, etc.			520	
Shield Configuration (4%)	33	330		
Other (Located in Ion Engine Bay, 3%)		190		
Spacecraft Guidance and Control System[l]				50**
Computer and Electronics			38	
Power Conditioning			10	
Radiator			2	
Payload System				2200
Science Allocation			2105	
Communications			60	
Radiator (0.8 kW(t) rejected)			35	
Total Spacecraft at Launch				35123
** Estimated				

NOTES:

(a) Includes coolant, cesium reservoir and temperature control, reactor control system (not including PC) and reactor output power leads.

(b) Includes ducting, accumulators, EM pumps, insulation, and coolant.

(c) Includes power conditioning for the EM pumps used for the reactor loop, for the primary loop if two-loop, auxiliary loop and shield loop.

(d) These radiators integral with Thruster Subsystem Power Conditioning Radiators.

(e) This radiator located on payload equipment bay.

(f) All radiators including structure support for LVPC for thruster subsystem.

(g) Payload, power plant control and guidance bay, less contribution from local PC radiators.

(h) Structure required to carry loads past ion engine units, their special PC and TVC system, less contribution from local PC radiators.

(i) Approximately 2-foot long, 9.7 foot diameter section to house 9-foot diameter by 20-inch deep antenna, and transfer launch loads to LV (disposable).

(j) Located on Thruster Equipment Bay.

(k) Required cooling subsystem included in auxiliary loop and related PC included in hotel load.

(l) Estimated at 50 pounds total, including Computer and Electronics, Power Conditioning and Radiator.

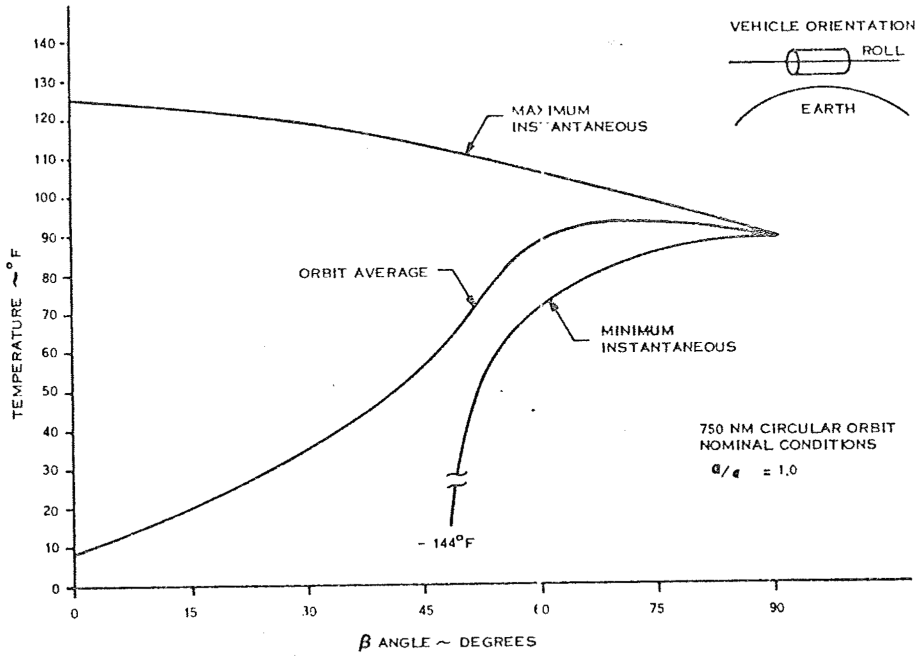

Figure 2-32. Radiator Average Temperature vs Beta Angle

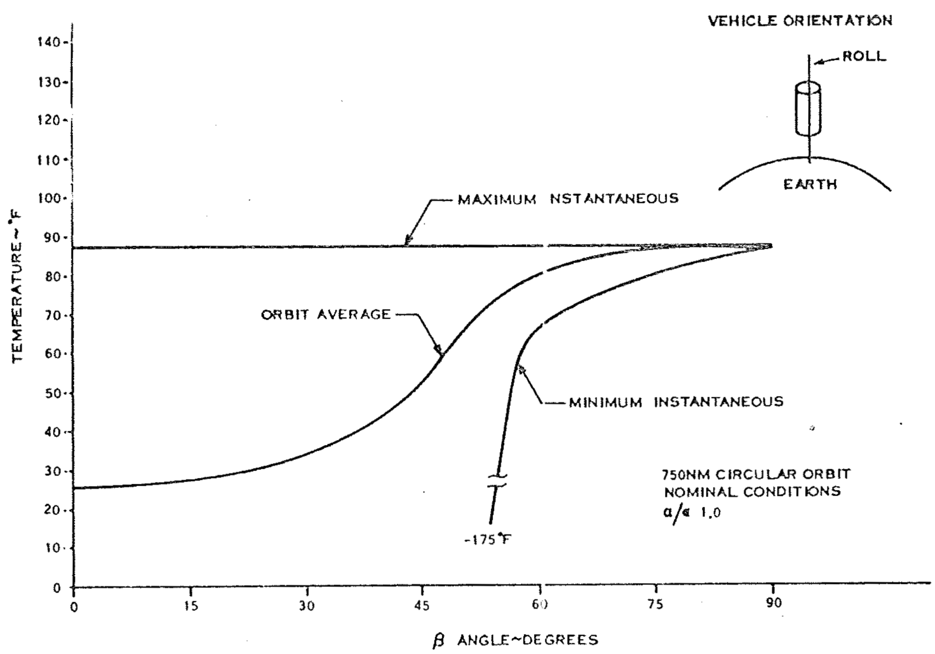

Figure 2-33. Radiator Average Temperature vs Beta Angle

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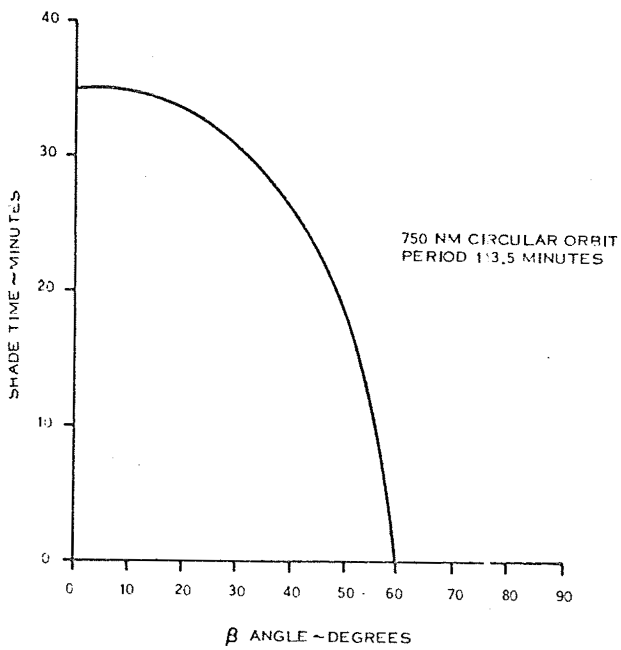

Figure 2-34. Shade Time vs Beta Angle

The format employed in the weight presentation reflects that currently evolving within NASA (Reference 12). Although subject to minor revision, it contains the following major elements:

- Propulsion System
  - Power Plant Subsystem
  - Thruster Subsystem
- Propellant Subsystem
- Spacecraft Guidance and Control Subsystem
- Payload Subsystem.

The remainder of this paragraph discusses the sources of the weight estimates quoted. Weight changes, where identified, are relative to previously reported values (Reference 1).

2.5.1 PROPULSION SYSTEM WEIGHTS

The propulsion system is defined as the power plant subsystem and the thruster subsystem. It is noted that all power conditioning components associated with the ion engine thruster requirements, as well as their related waste heat radiators, are included within the thruster subsystem.

2.5.1.1 Propulsion Subsystem

Reactor weights are those provided by GE-Nuclear Thermionics Power Operation for this study. The maximum variation about the 2960 pounds quoted is +80 pounds, depending on the reactor coolant temperature rise and pressure loss.

Shield weights are those initially estimated for the lithium hydrid neutron shield (Reference 1). No permanent gamma shielding is identified at this time, pending further evaluation by ORNL, under the requirement for a two-series-loop primary coolant system.

Radiator weights are quoted for the two-series-loop system (Reference 1), plus original estimates for auxiliary and shield cooling, based on Cu/SS materials for 0.95 non-puncture probability. All power conditioning radiators are separately identified below.

Coolant Loop weight changes reflect the most recent estimates for the two-series-loop system, assuming three-inch diameter ducts in the primary loop. All ducts have 60 mil stainless steel walls.

Power Cable weights are dominated by the LV cables, either aluminum, or sodium encased in stainless steel. The HV leads are assumed to be part of the thruster subsystem, which particularly requires the bulk of the thermionic reactor electric power output in this application.

Power Conditioning Electronic weights quoted for the hotel load are those identified for the flashlite reactor in Paragraph 2.3.

Power Conditioning Radiator weights quoted for the hotel load power conditioning radiators are based on an earlier estimate of about 44 pounds/kW thermal rejected heat (Reference 1). This weight estimate can be reduced by as much as 33 percent, while still meeting launch environment structural requirements. However, a decrease in fin efficiency, and therefore an overall increase in the spacecraft length and weight may be required. This particular weight/area trade-off is scheduled for further evaluation, particularly as related to the LV-to-HV power conditioning radiator weight,

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2.5.4 PAYLOAD SUBSYSTEM

Reported total weights remain essentially unchanged from those previously reported. Considerable effort is yet required to define the efficient utilization of the net 2105 lbs. of science payload.

2.5.5 SUMMARY - SPACECRAFT WEIGHT

The currently estimated spacecraft propulsion system weight in earth orbit for the flashlite thermionic reactor is 17,900 pounds, which translates to

a. 59.4 pounds/kWe (GUP)

b. 74.3 pounds/kWe (NPP)

2.6 COMPUTER PROGRAM

A computer program is being written to assist in the design and optimization of the thermionic reactor power systems and spacecraft for specified sets of conditions. The design logic of the program has been slightly altered from that previously reported with the modified logic as shown in Figure 2-35. The changes include:

a. An iteration of the shield characteristics based on the computed separation distance of the power conditioning modules from the shield.

b. Iteration of the main radiation loop piping characteristics prior to the iteration of the low voltage cable characteristics rather than vice versa as previously proposed.

The system conditions and configuration options which will be designated and held constant for a particular case are as follows:

System Conditions

- Gross reactor output power
- Mission and reactor operating times
- Integrated radiation dose limits
- Maximum vehicle diameter

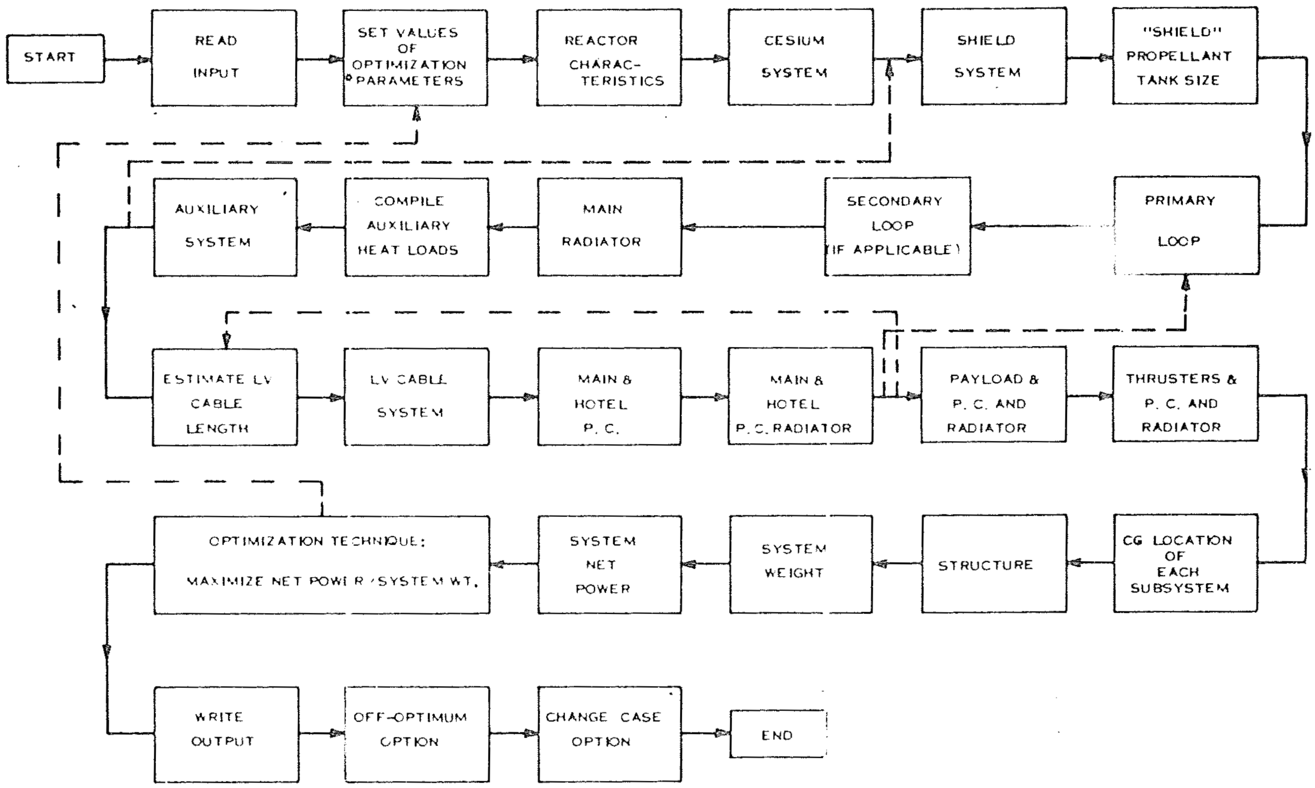

Figure 2-35. Simplified Logic Diagram for Computer Program

- Propellant Weight
- Payload weight
- Thruster weight
- Payload power requirements
- Control systems power requirement
- Cesium reservoir power requirement
- Average sink temperature
- Maximum reactor temperature
- Maximum power conditioning temperature
- Maximum shield temperature

- Maximum coolant temperature in each subsystem
- Maximum PC radiator (passive) temperature
- Number of redundant pumps per subsystem

System Configuration Options

- Reactor type
- Relative radiator location - main radiator forward or PC radiator forward
- Shield concept - normal shield or gamma shielding provided by propellant
- PC cooling mode - active or passive
- Shield cooling mode - active or passive
- Materials selection for each subsystem including coolant compositions
- Main heat rejection loop configuration - single loop or double loop in series

For a given set of the above system conditions and configuration options, the program will determine the characteristics of the power plant and the spacecraft which will provide the maximum power per unit weight to the thruster engines. In arriving at this optimum system, the program will vary and select the optimum combination of the following system conditions:

- Vehicle length
- Shield half angle
- Reactor loop pipe diameter
- Reactor loop temperature rise
- Main radiator loop pipe diameter*
- Main radiator loop flow rate*
- Cross section area of cable

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- Heat exchanger effectiveness*
- Main radiator area ratio
- Shield loop pipe diameter**
- Shield loop coolant temperature drop**
- Shield radiator area ratio**
- PC loop pipe diameter**
- PC loop coolant temperature**
- PC radiator area ratio**
- Auxiliary loop pipe diameter
- Auxiliary loop coolant temperature drop
- Auxiliary radiator area ratio
- Passive PC radiator fin thickness.

Investigation of the effect of the input system conditions and configuration on the optimized system characteristics are achieved by successive operation of the program with the input values of interest.

The complete program, including the design sequence, some of the individual component models and the integration of the optimization technique, must be particularized for the reactor type to be investigated. However, models such as those for the shield and radiator components presently included in the program are general. Programming has been completed up to the power conditioning systems, and models have been defined for all components. Some of these, such as the shield model, are preliminary, and will be refined when specific designs of these components are generated.

Details of the computer code will not be reported until after its completion.

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* These items apply only to a dual loop configuration.
** These items apply only if the pertinent subsystem is actively cooled.

3. CONCLUSIONS

4. RECOMMENDATIONS

5. NEW TECHNOLOGY

6. REFERENCES

3. CONCLUSIONS

1. The conical (or conical-cylindrical) readiator configuration, integrated with the spacecraft and launched in the upright (apex:top) position on the Titan IIIC/7 launch vehicle (Reference 1), will maximize the IMEO and minimize the earth orbit departure weight. This configuration requires the minimum parasitic launch support structure of about 4.3 pounds/kWe of Gross Unconditioned Power (GUP) for the 300 kWe GUP in-core thermionic reactor power plants. Since 240 kWe conditioned power are required for the thruster subsystem, this weight penalty may also be expressed as about 5.4 pounds/kWe of Net Plant Power (NPP).*

2. The triform and other such radiator geometries, such as flat plate and curciform, are not attractive. Their minimum parasitic structural requirements are at least twice those of the conical radiator, because the radiator system available for use as structure is not efficiently oriented, relative to the conical radiator.

3. The inverted (apex:down) orientation on the launch vehicle is not attractive. Although initially attractive, because it permitted the concentration of major spacecraft weights, the reactor, shield and propellant at the lowest point during launch, relative to the launch vehicle, the structure available in required power plant components is not efficiently utilized. Resulting deflections and loads would require greater parasitic structures for both the conical and triform radiator spacecraft, relative to their orientation in the upright position (apex:up) on the launch vehicle.

4. Spacecraft of the type evolving in this study will have lowest natural frequencies of the order of one cycle per second. Redesign of the autopilot for the Titan IIIC/7 launch vehicle

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* Beam Power = NPP η_{MPC} η_{TS}

where η_{MPC} = Efficiency of Main Power Conditioning

η_{TS} = Efficiency of Thruster Subsystem, including PC to provide the many special thruster loads.

will be required to permit launching. This appraoch was utilized in the MOL program, and it is the best technique to maximize IMEO.

5. No electric system redundancy is required. The failure of any one of the 108 main converter units will result in a power loss of less than one percent. Additionally, the reactor is designed to provide a ten percent power margin.

6. All thermionic reactor main power conditioning units will require filtering of the reactor input power in order to prevent damage to the solid state switching units, due to voltage spikes which will inherently occur during switching. For the flashlite reactor with 108 main converter units, the filter units represent a weight penalty of about 1.4 pounds/kWe (GUP).

7. The flashlite reactor electric system for this application requires that all the thruster units operate in parallel off of a single high voltage bus. This requirement dictates that electric isolation be provided for each thruster, preventing the dumping of all thruster beam power into a single thruster unit in the event of thruster arcing. The weight penalty for the thruster isolation system for all 37 thruster units is about 1.5 pounds/kWe (GUP).

8. No special thermal insulation will be required to permit power plant startup in the 750 nautical mile earth departure orbit, when NaK-78 is employed as the primary radiator fluid. In the worst case, an ecliptic orbit, startup procedures would be initiated on the dark side, where the NaK-78 does freeze, with actual startup occurring on the sun side, where it is molten.

4. RECOMMENDATIONS

1. The spacecraft should employ the conical or conical-cylindrical radiator configuration and be launched in the upright (apex:up) position on the Titan IIIC/7 launch vehicle.

2. Consideration should be given to eliminating the standard launch shroud by employing the radiator as the primary shroud and designing a special shroud to cover the reactor during launch.

3. The aspects of the Titan IIIC/7 autopilot re-design to accept a minimum payload frequency of the order of one cycle per second should be investigated.

5. NEW TECHNOLOGY

No new technology items have been identified.

6. REFERENCES

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11. L. E. Weaver, H. G. Gronroos, J. G. Guppy, J. P. Davis, "A Control System Study for an In-Case Thermionic Reactor".

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12. N. K. Simon, Jet Propulsion Laboratory, Personal Communications.

APPENDICES

APPENDIX A

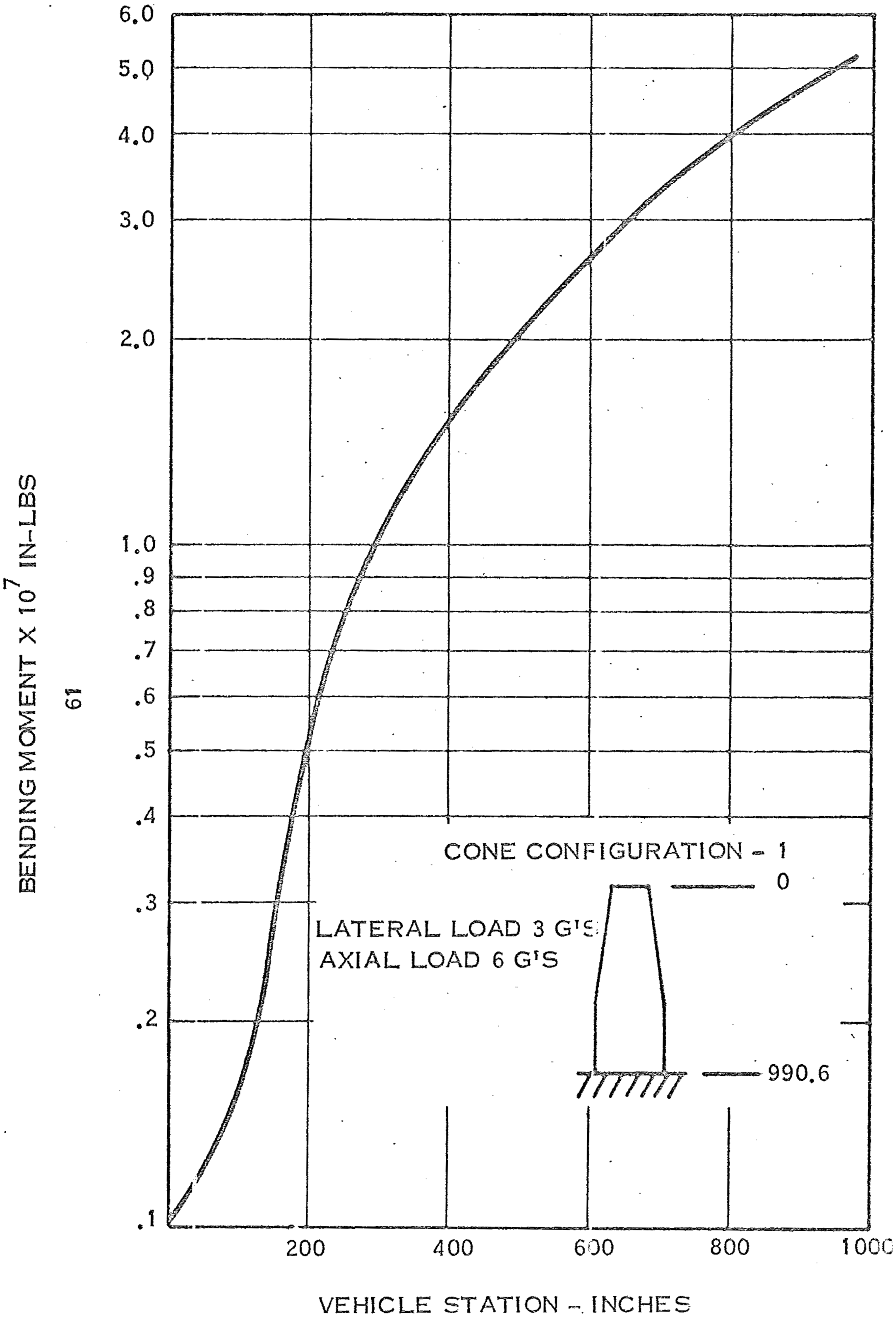
Stress Analyses Data

This appendix presents selected results from the computerized stress analysis. Data are presented for the following configurations:

- Upright Conical - Unsupported
- Upright Conical - Two Supports
- Inverted Conical - Two Supports
- Upright Triform - Unsupported

The data presented for each configuration are:

- Bending Moment Distribution
- Bending Stress Distribution
- Axial Load Distribution
- Shear Force Distribution
- EI Distribution
- Deflection Distribution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.2
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200
400
600
800
1000
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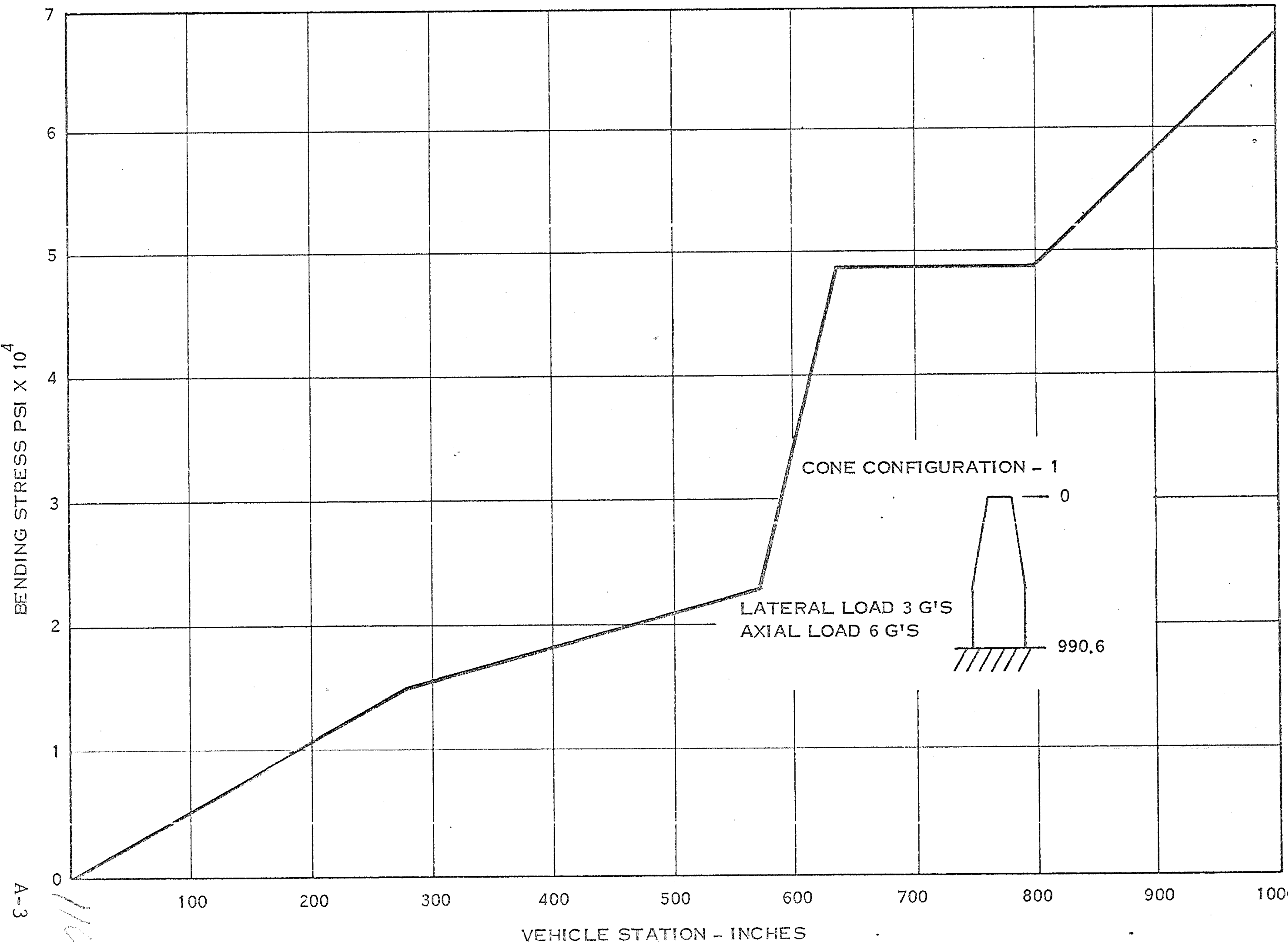
AXIAL LOAD 6 G'S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2
3
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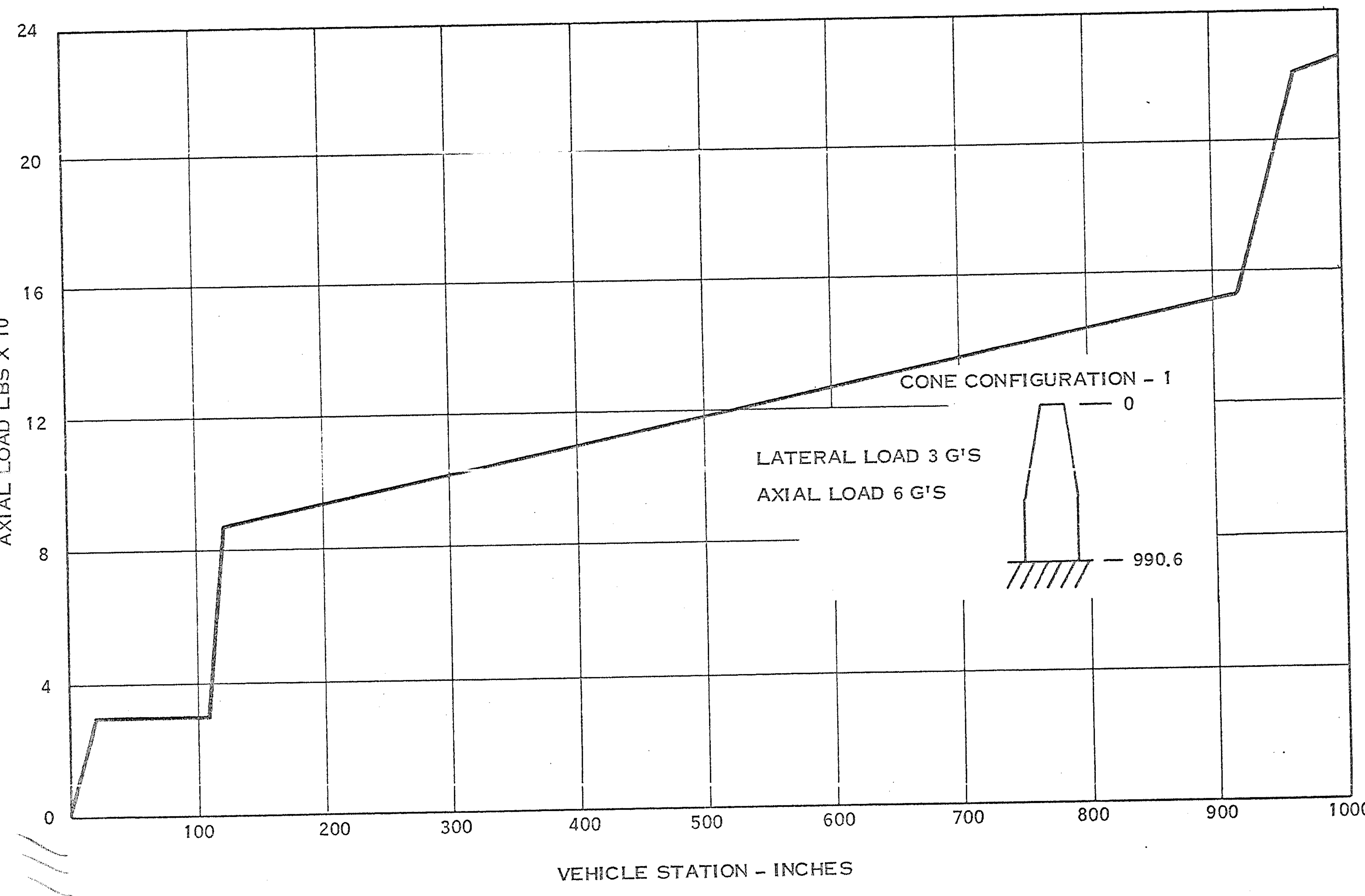
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AXIAL LOAD LBS X 10
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4
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20
24
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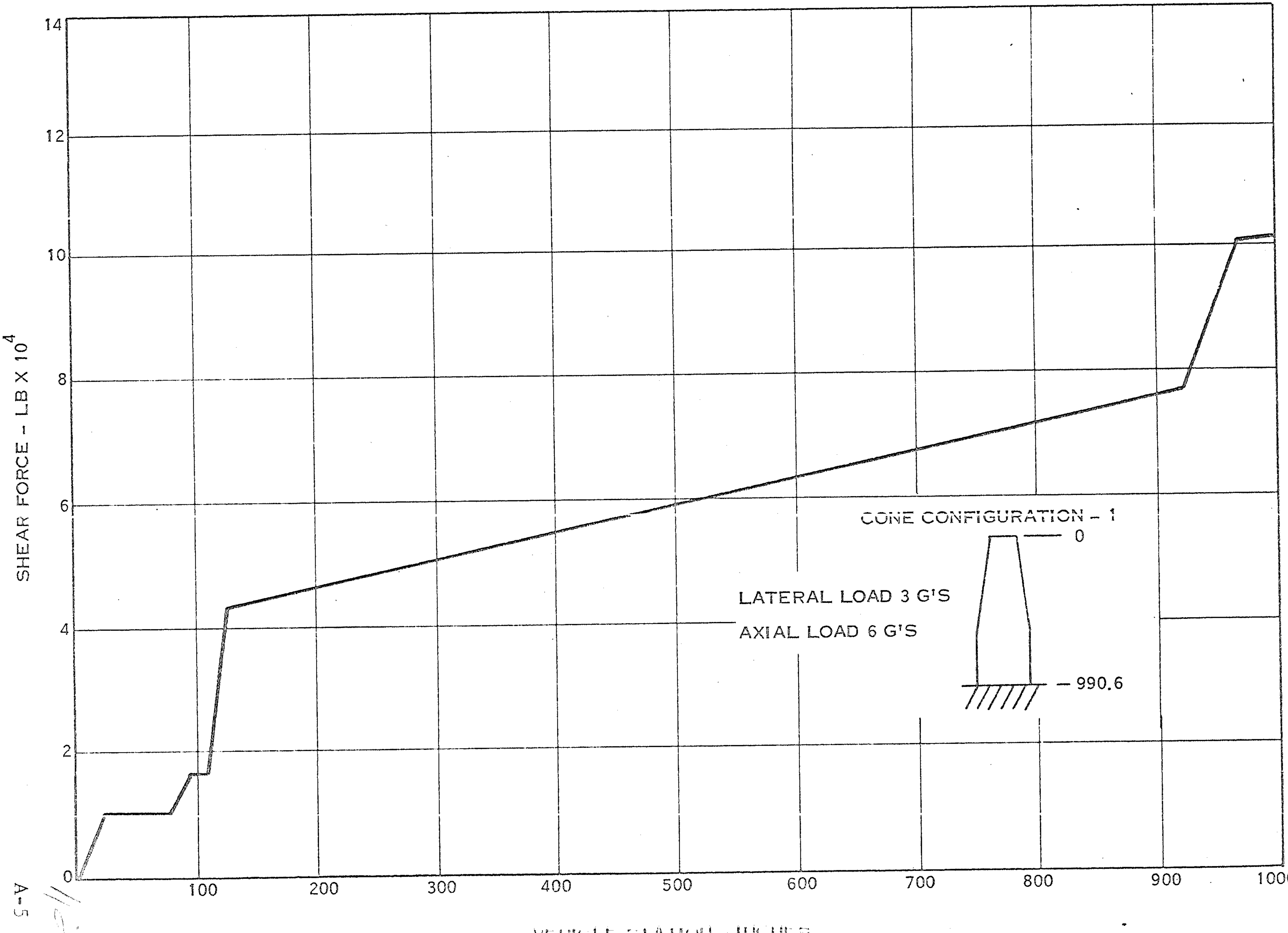
800
900
1000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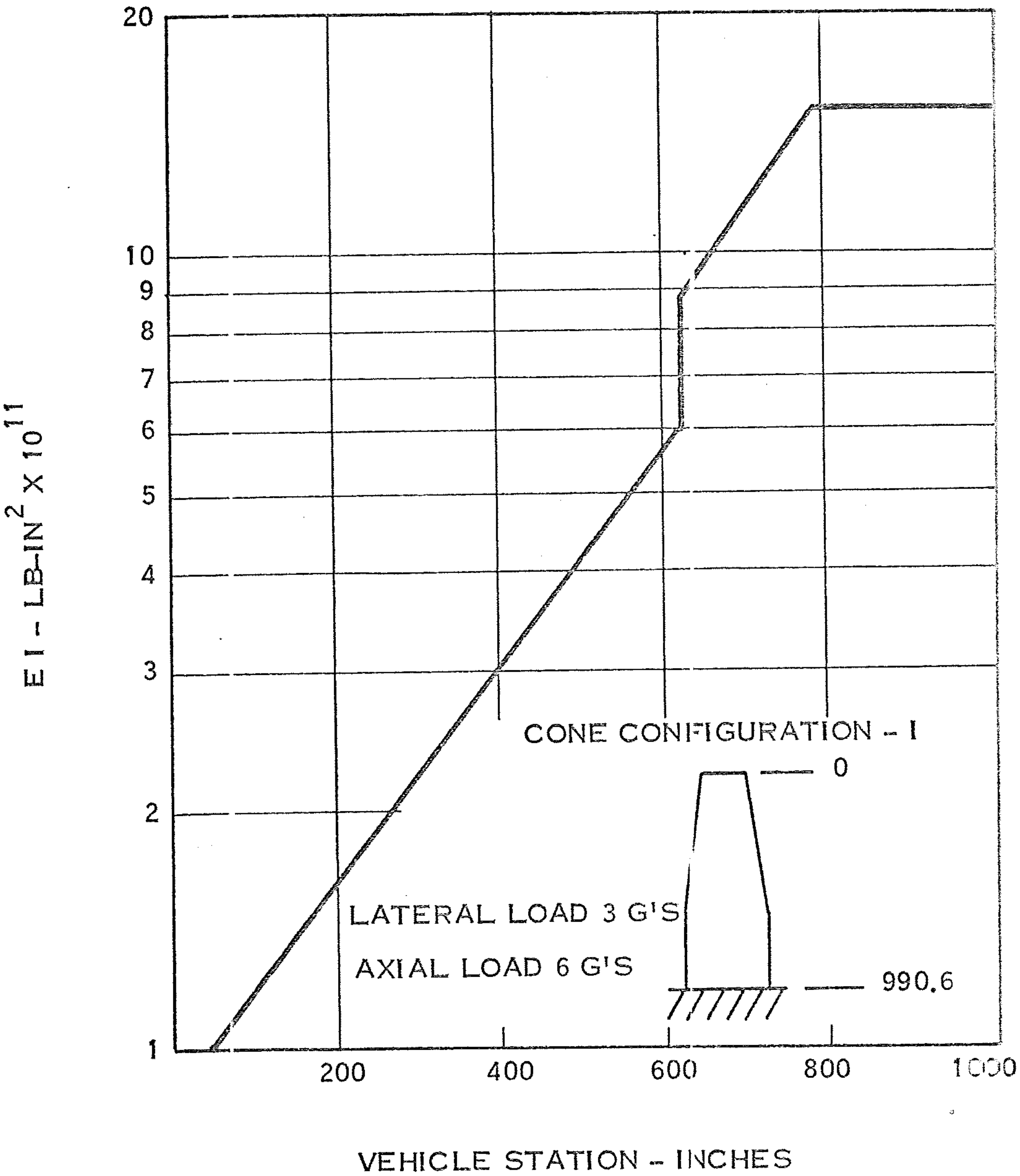
AXIAL LOAD 6 G'S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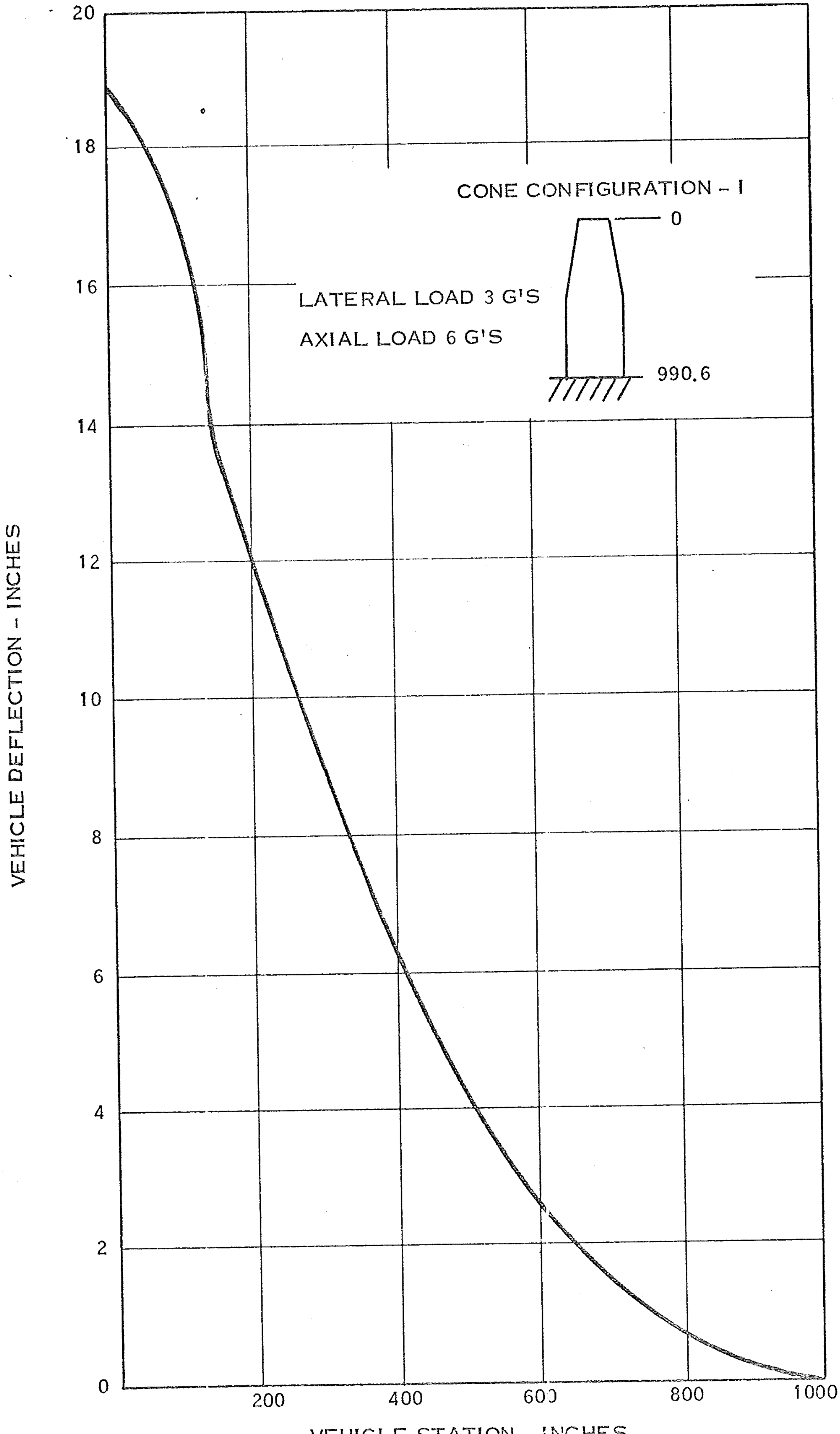
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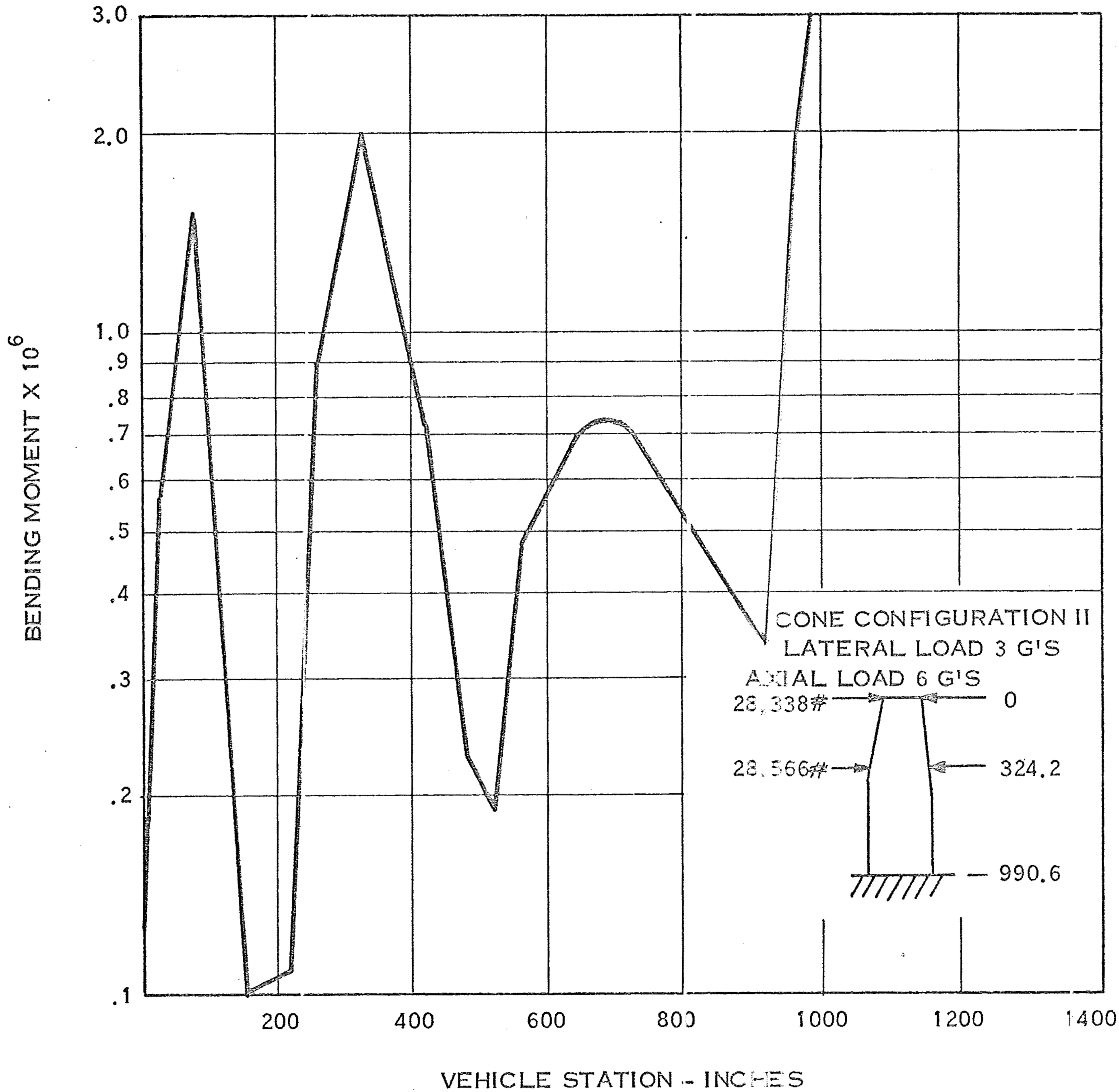
VEHICLE STATION - INCHES

114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600
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1000
1200
1400
VEHICLE STATION - INCHES
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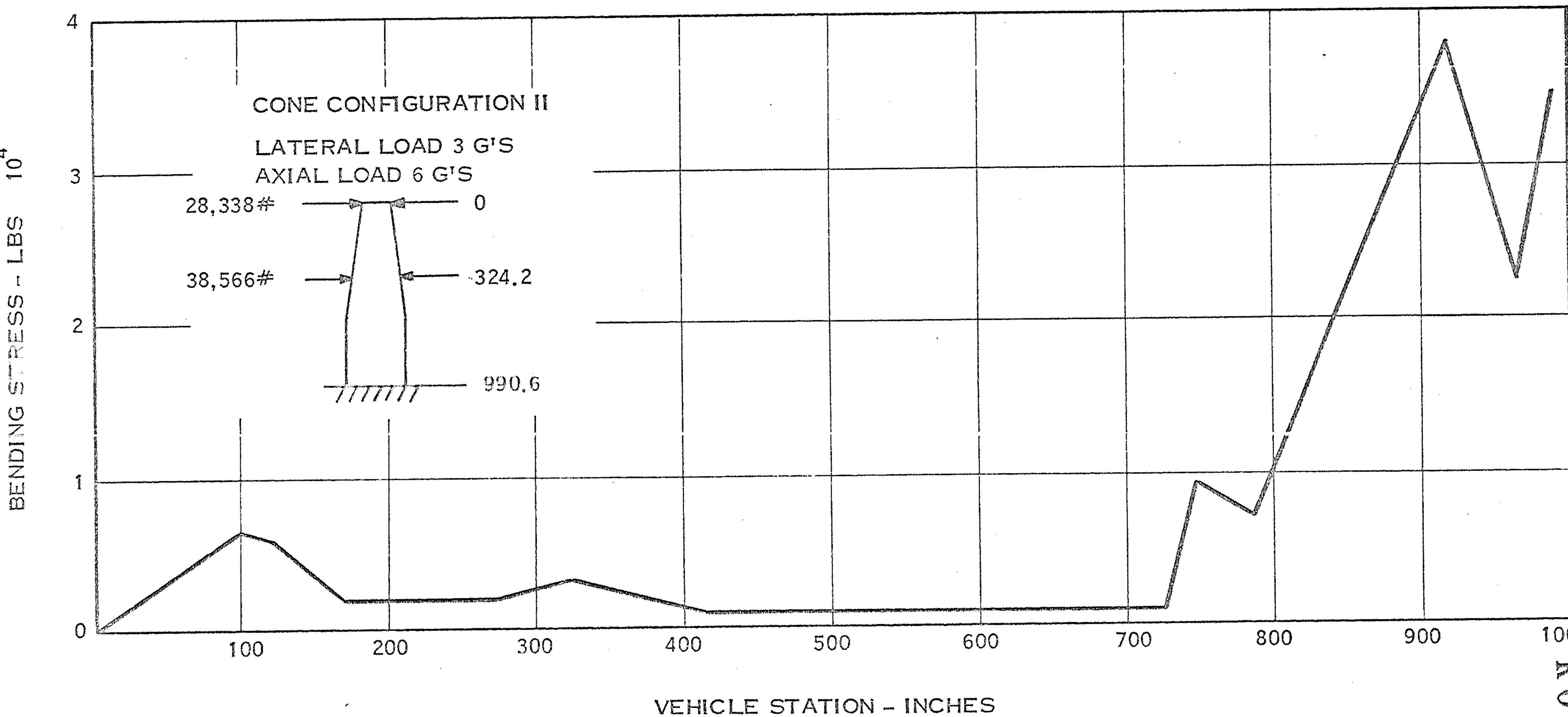
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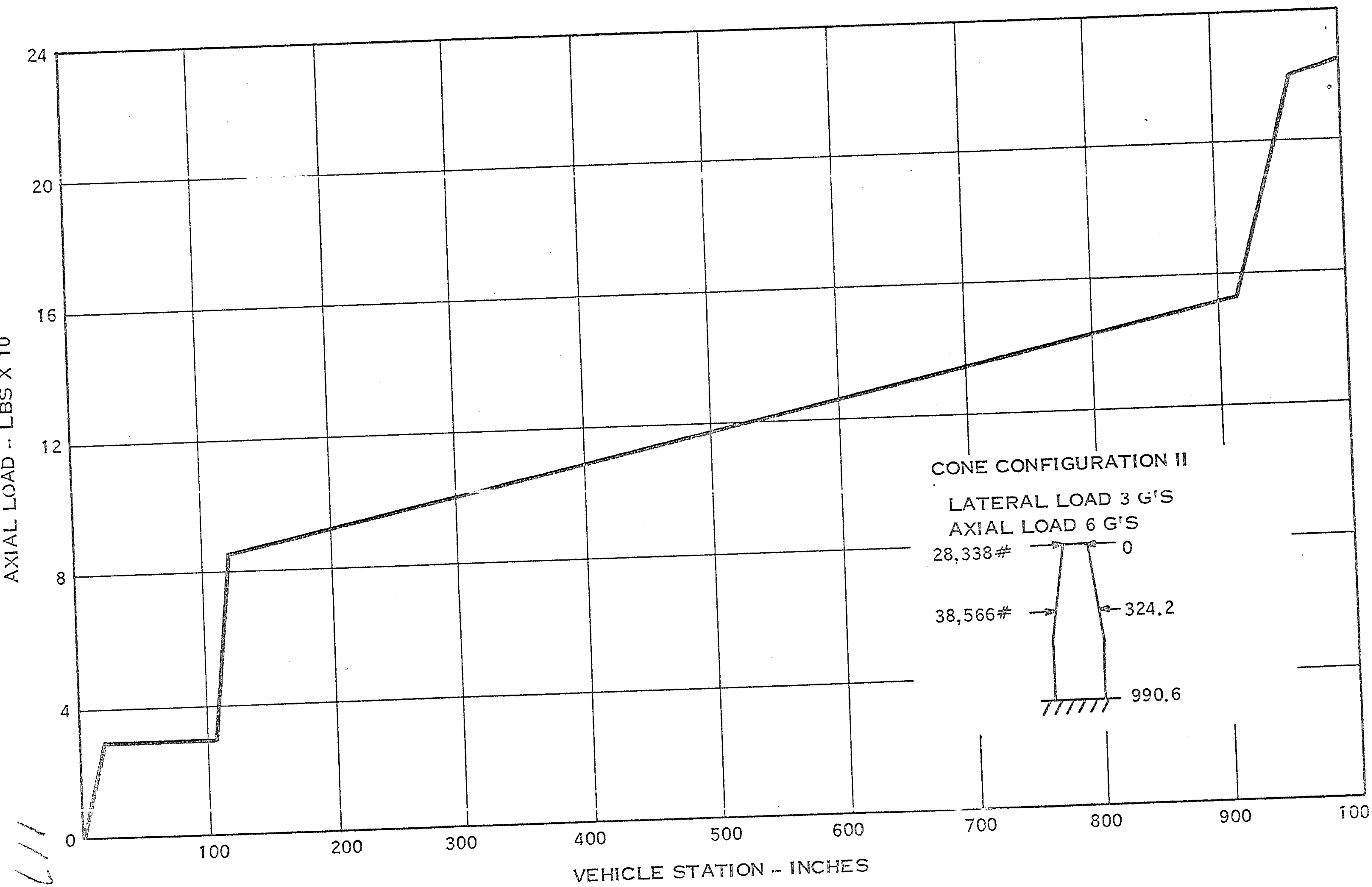
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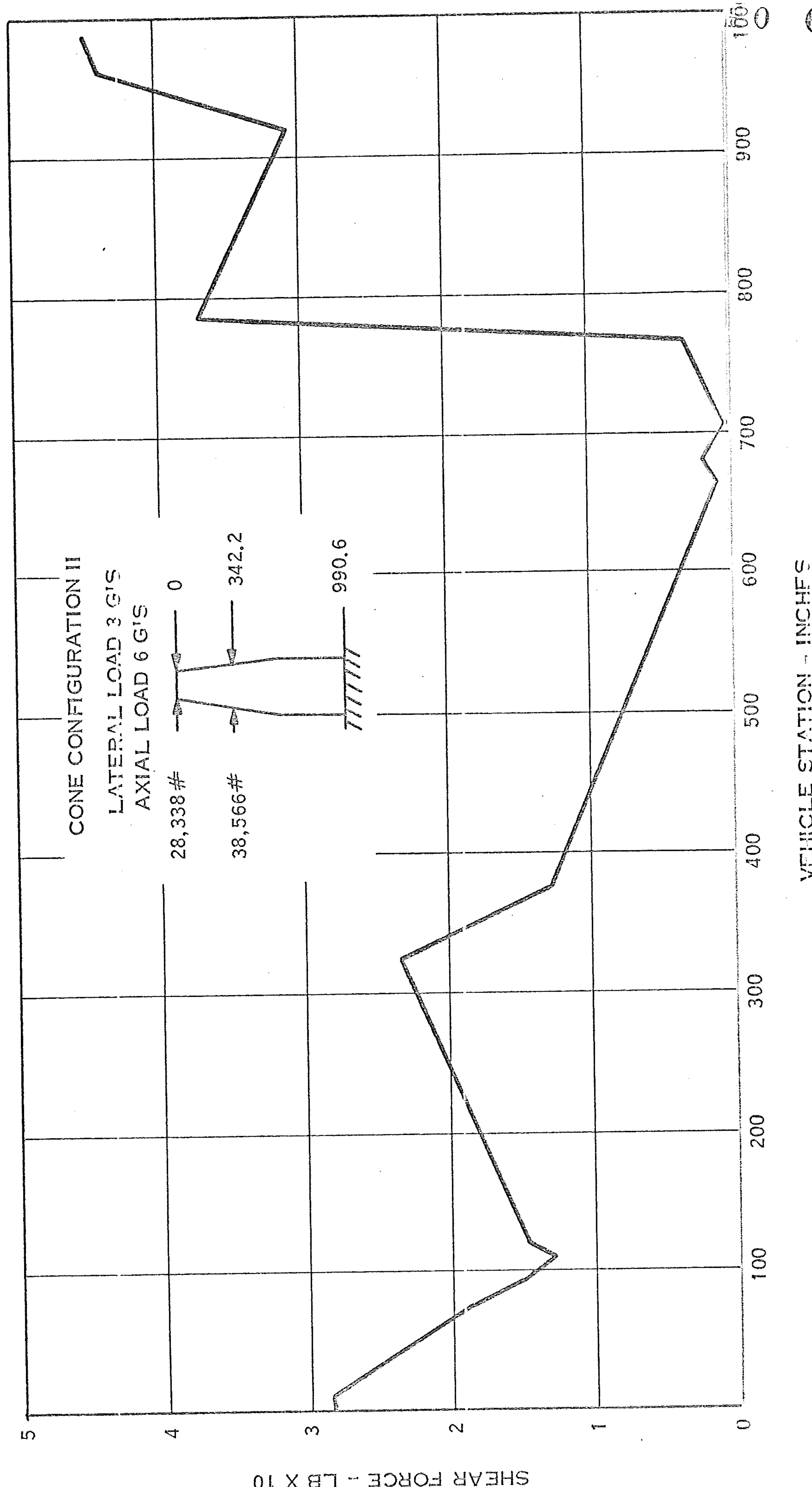
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3
4
5

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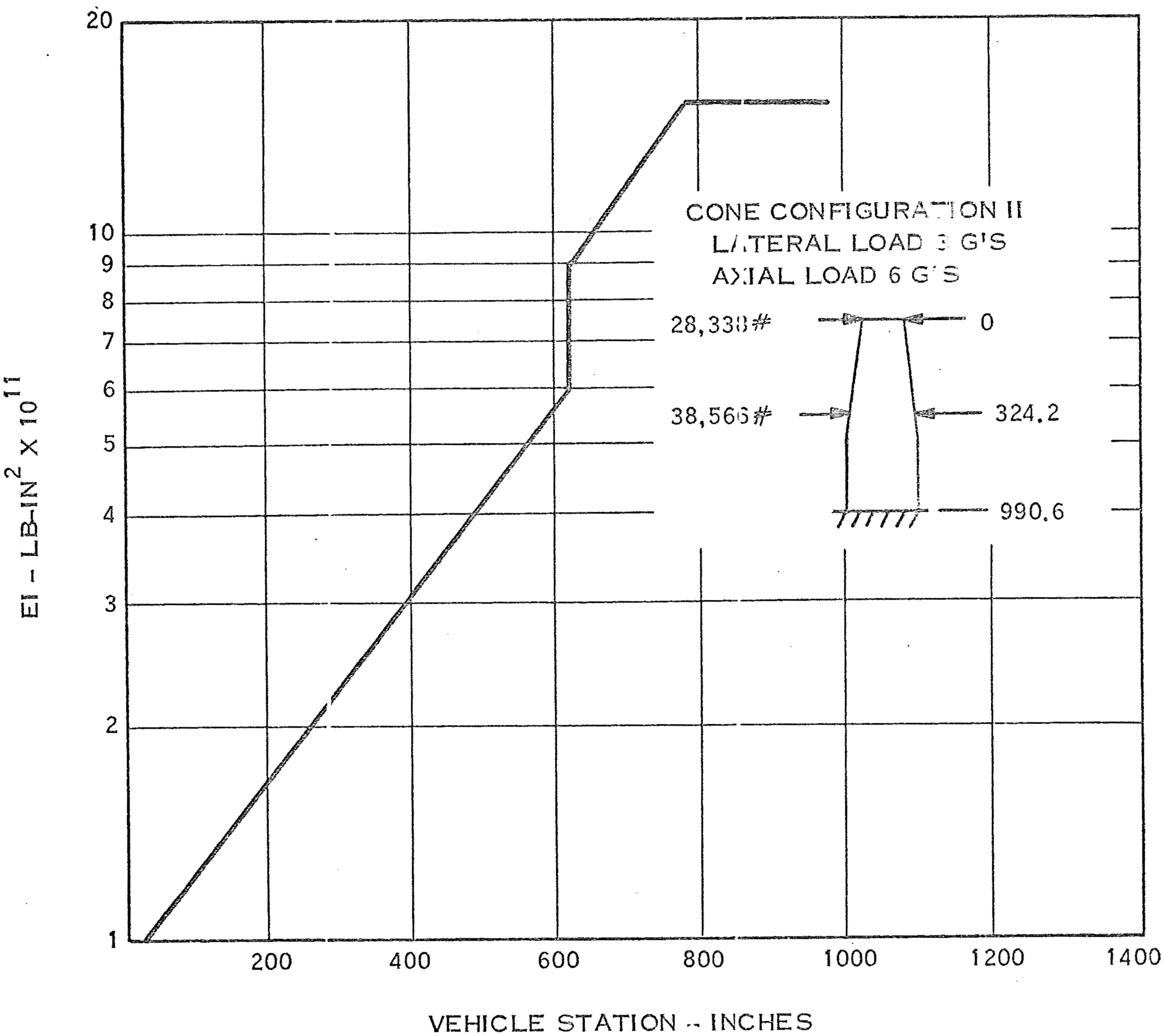

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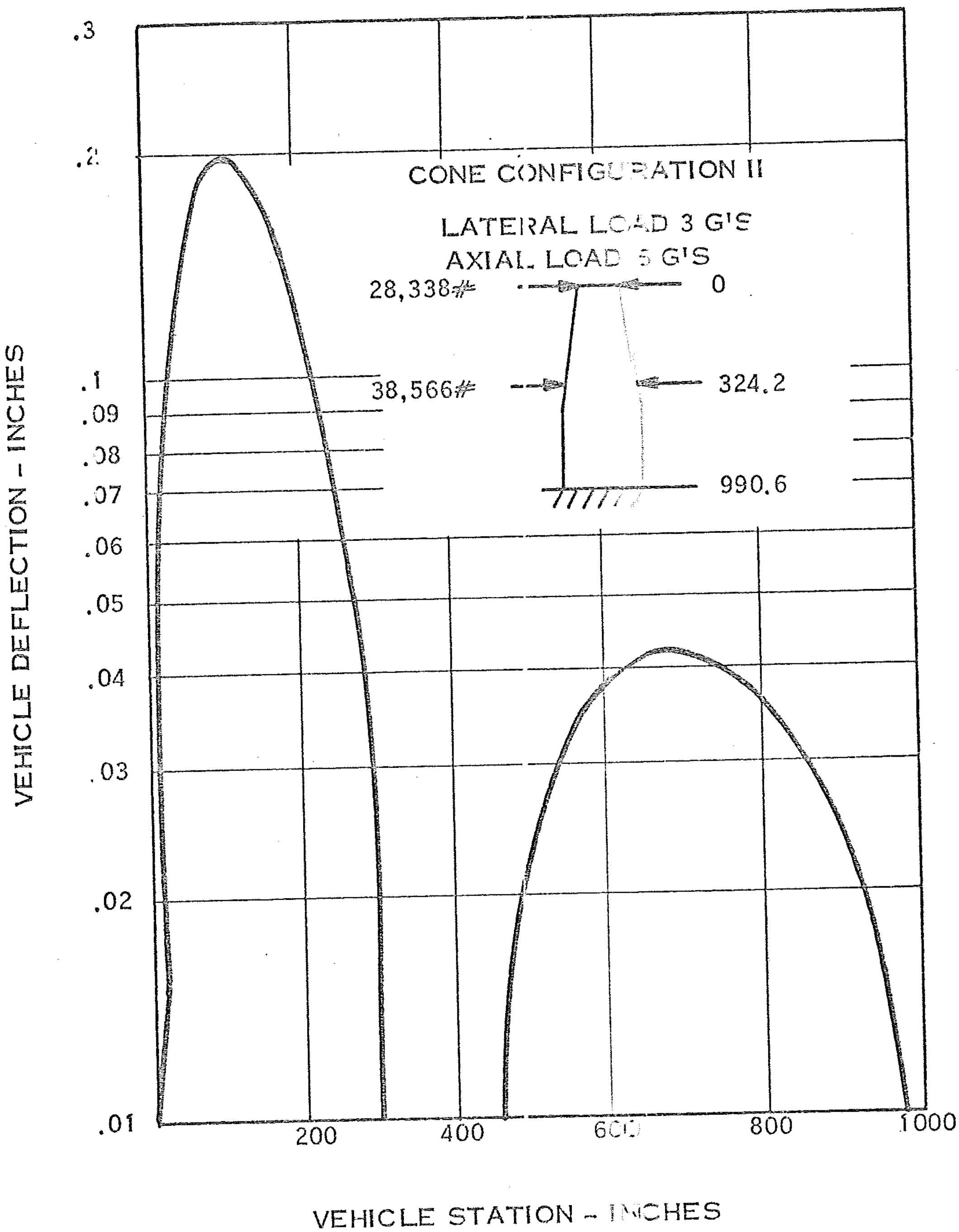
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VEHICLE STATION - INCHES

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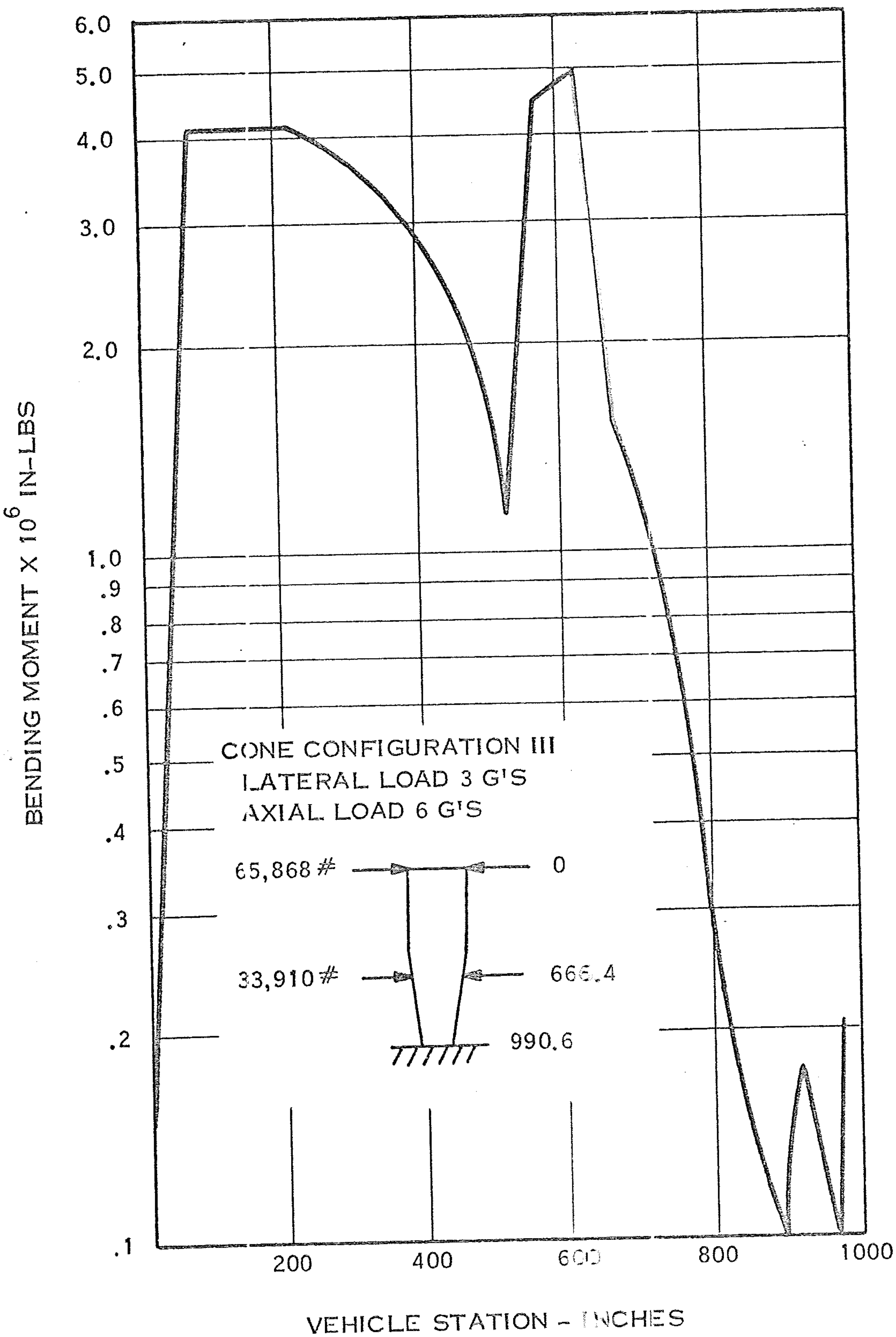
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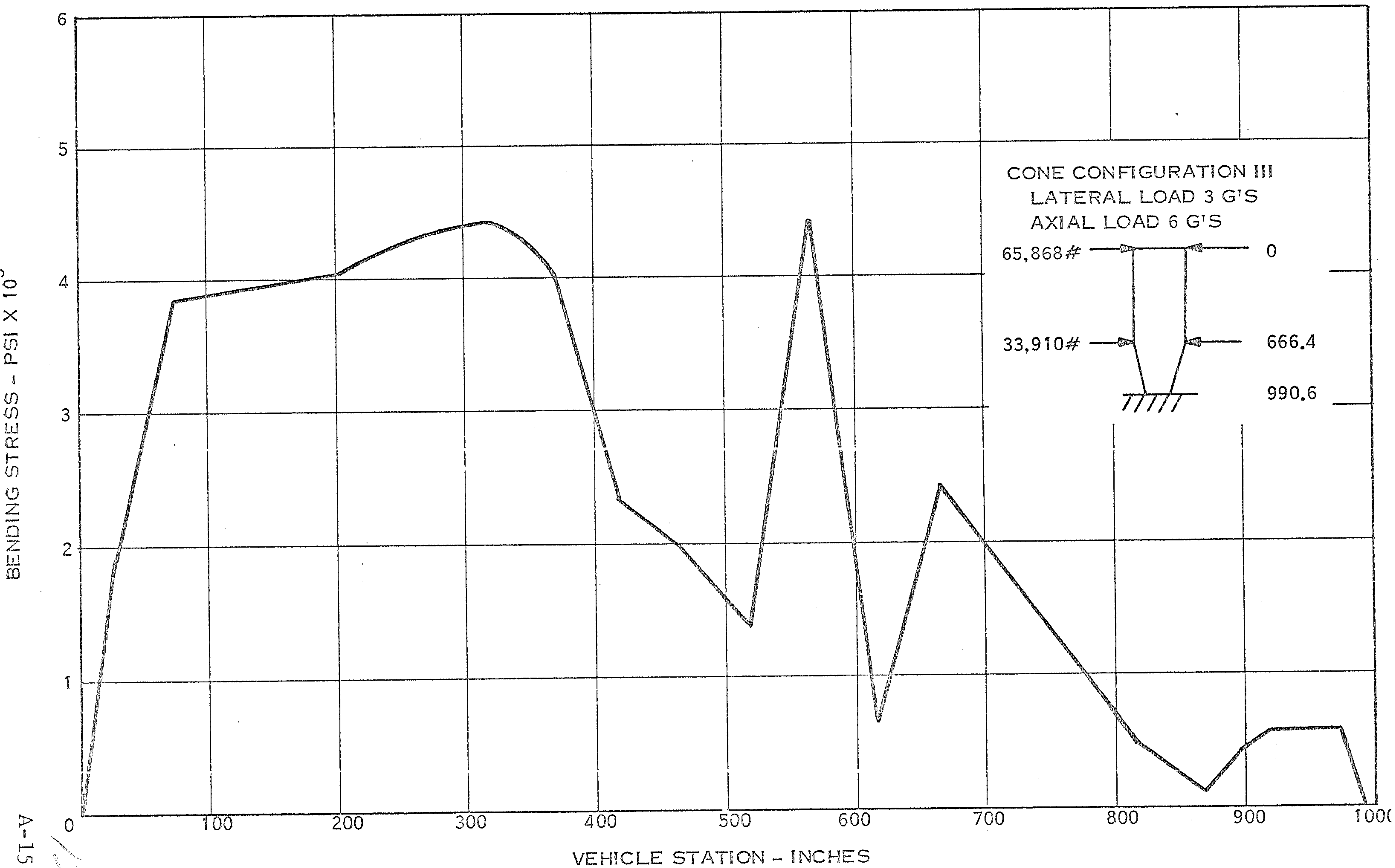
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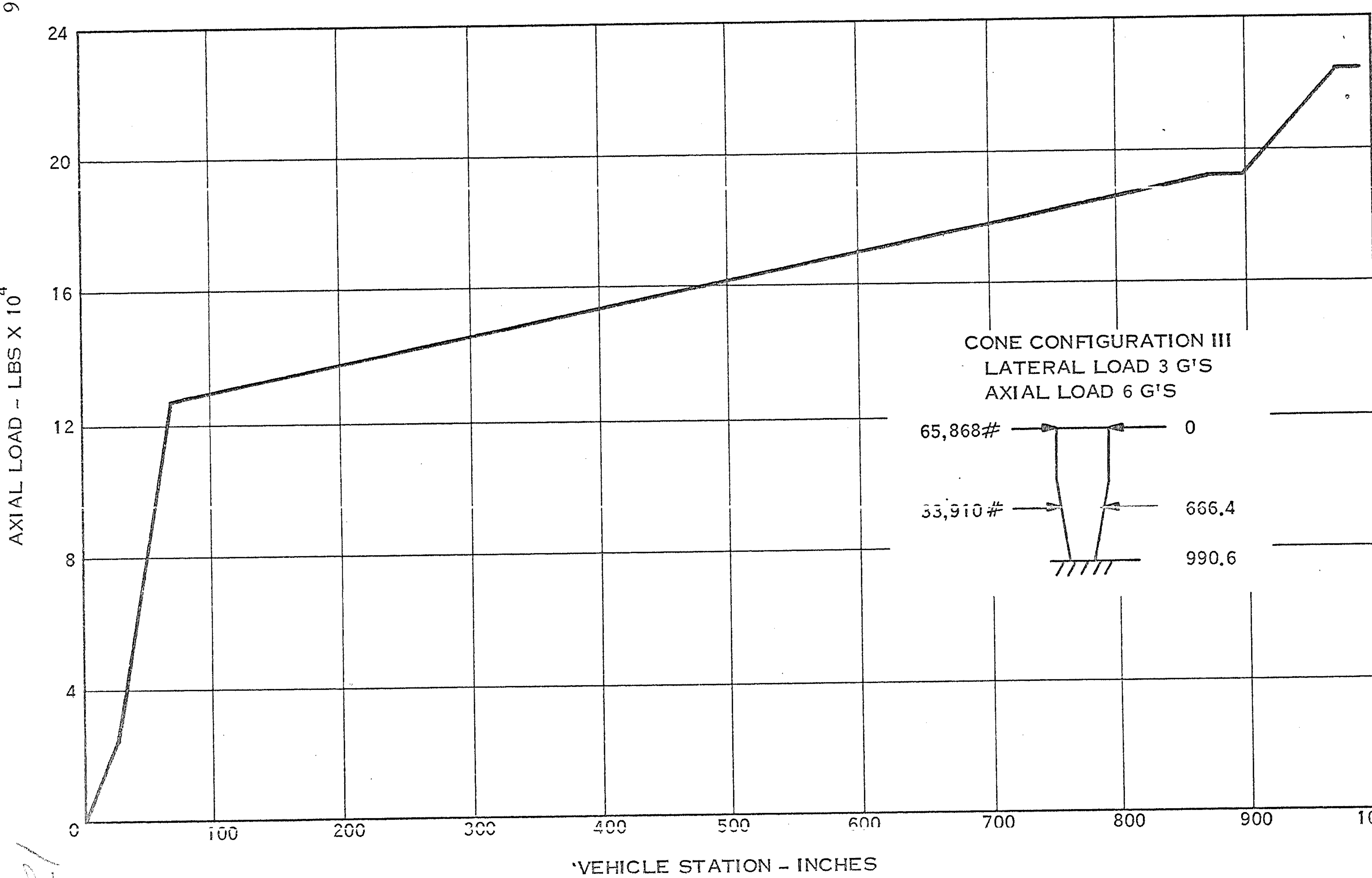
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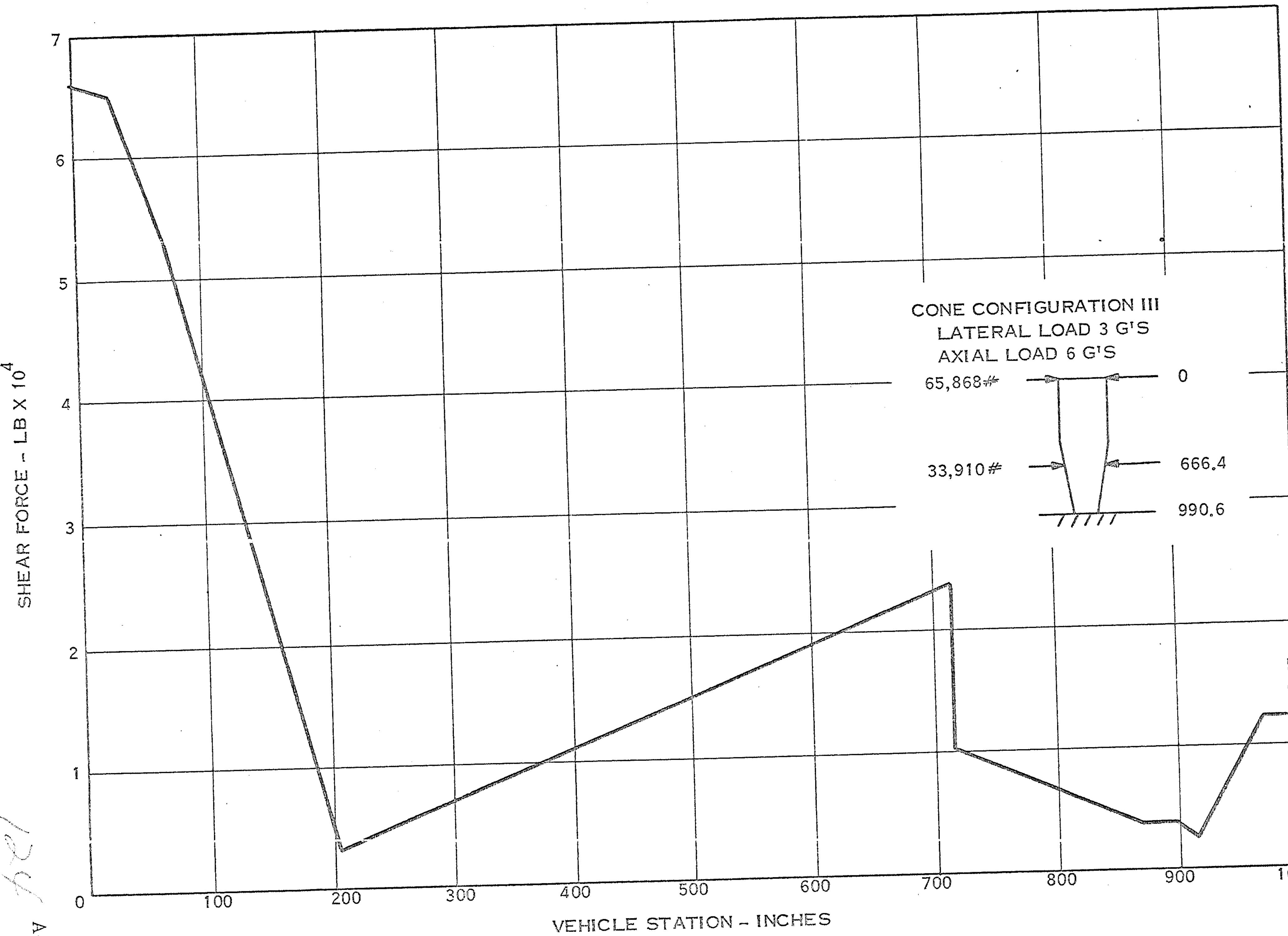
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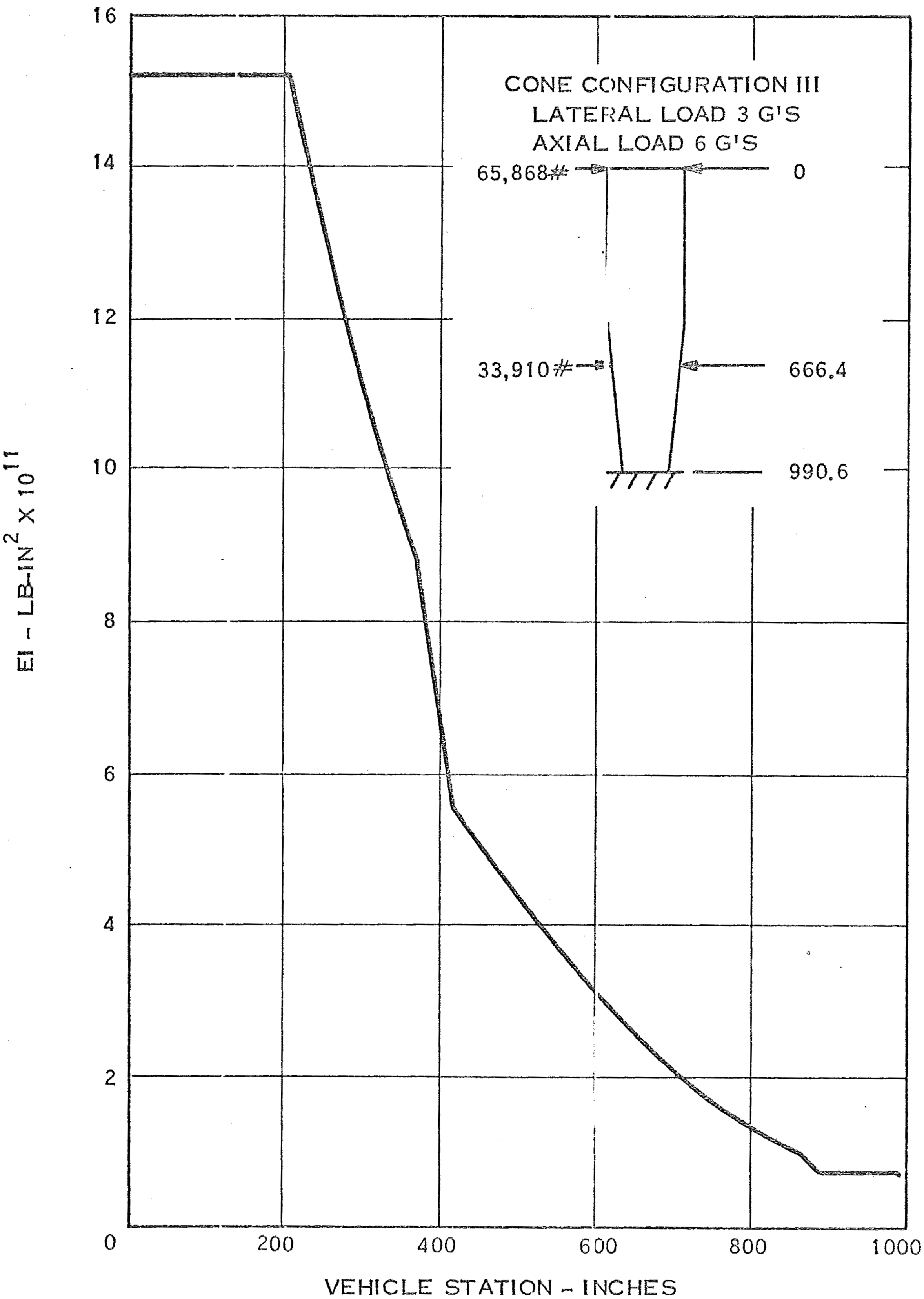
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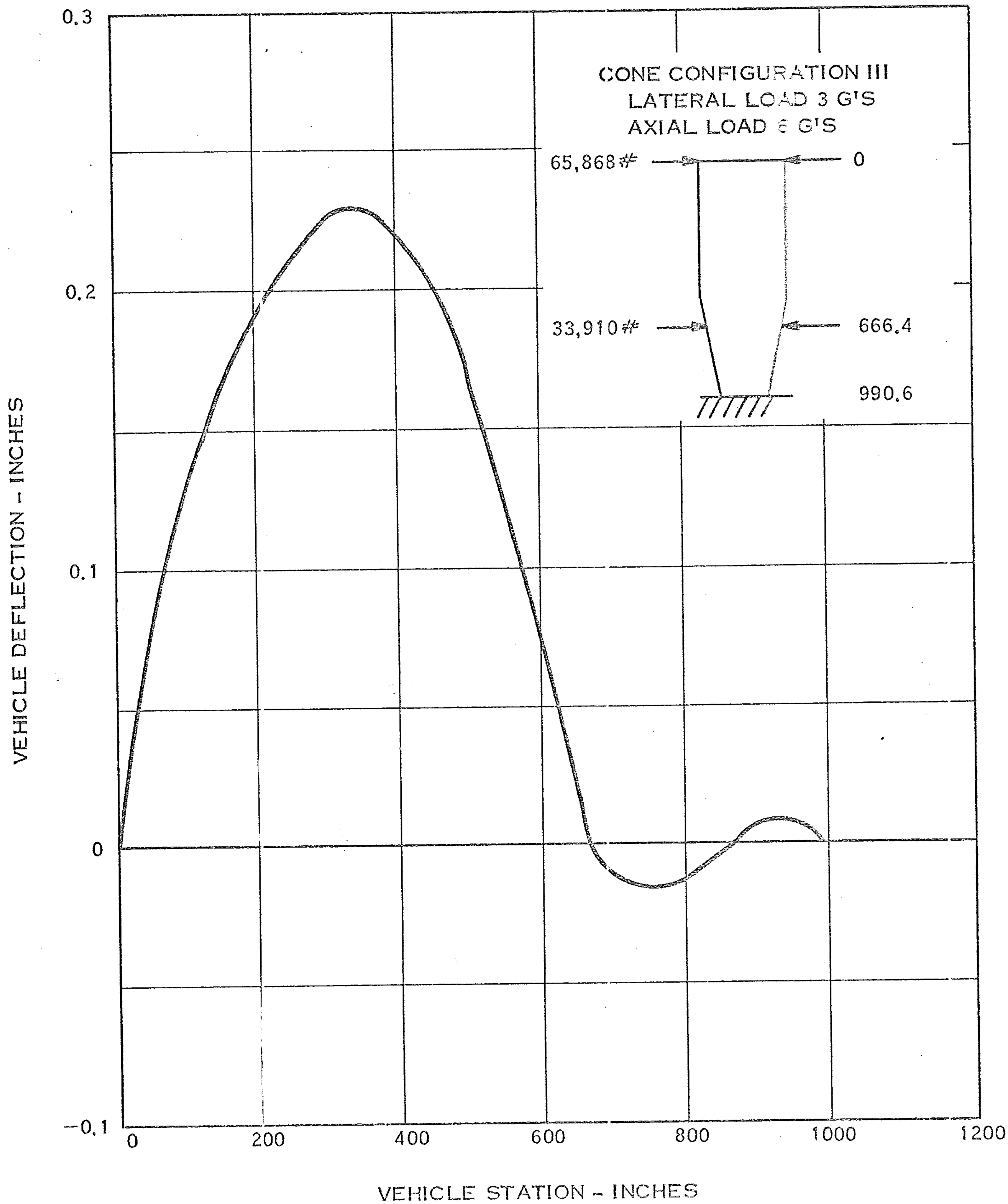
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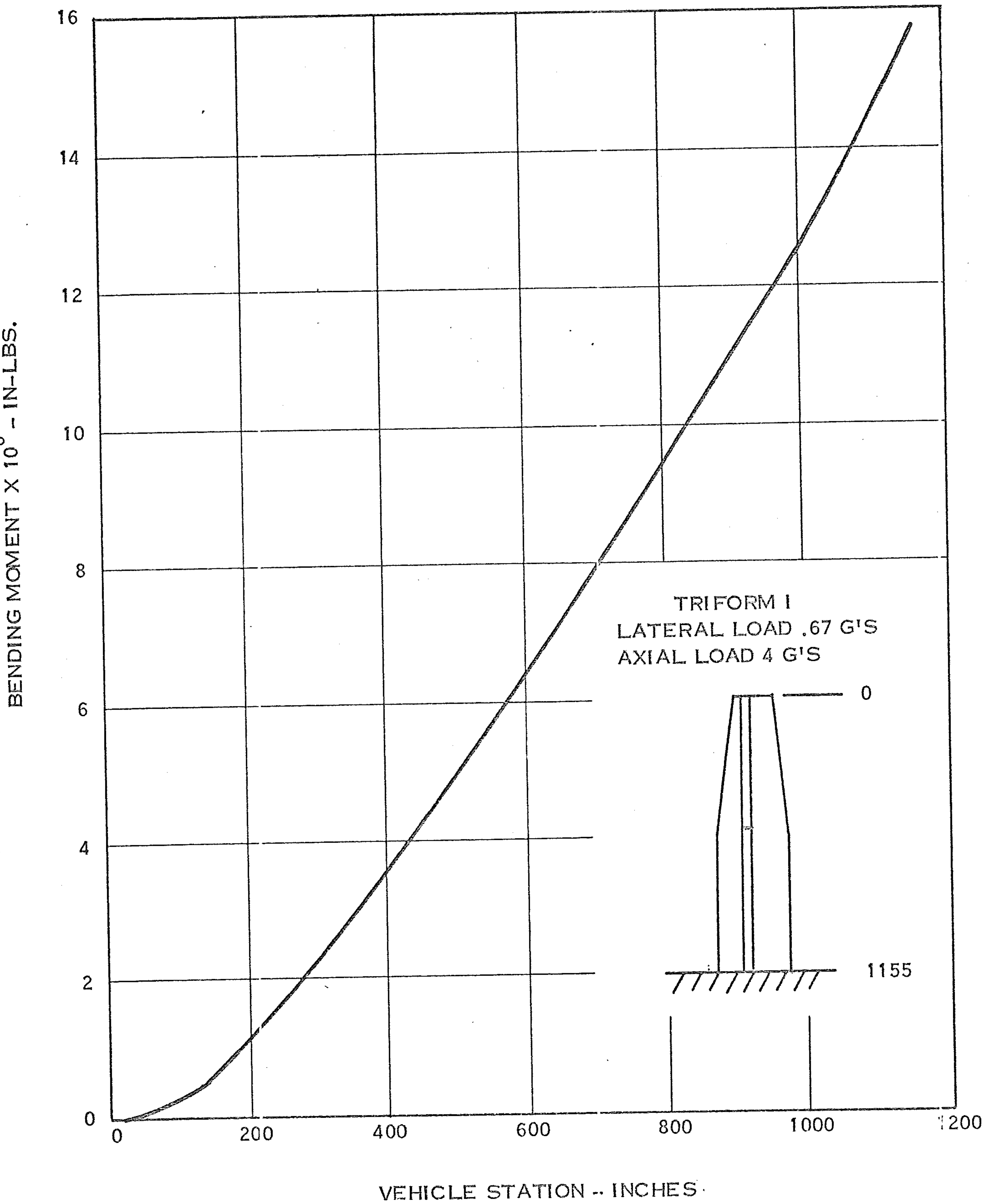
VEHICLE STATION - INCHES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6
4
2
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400
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800
1000
1200
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AXIAL LOAD 4 G'S
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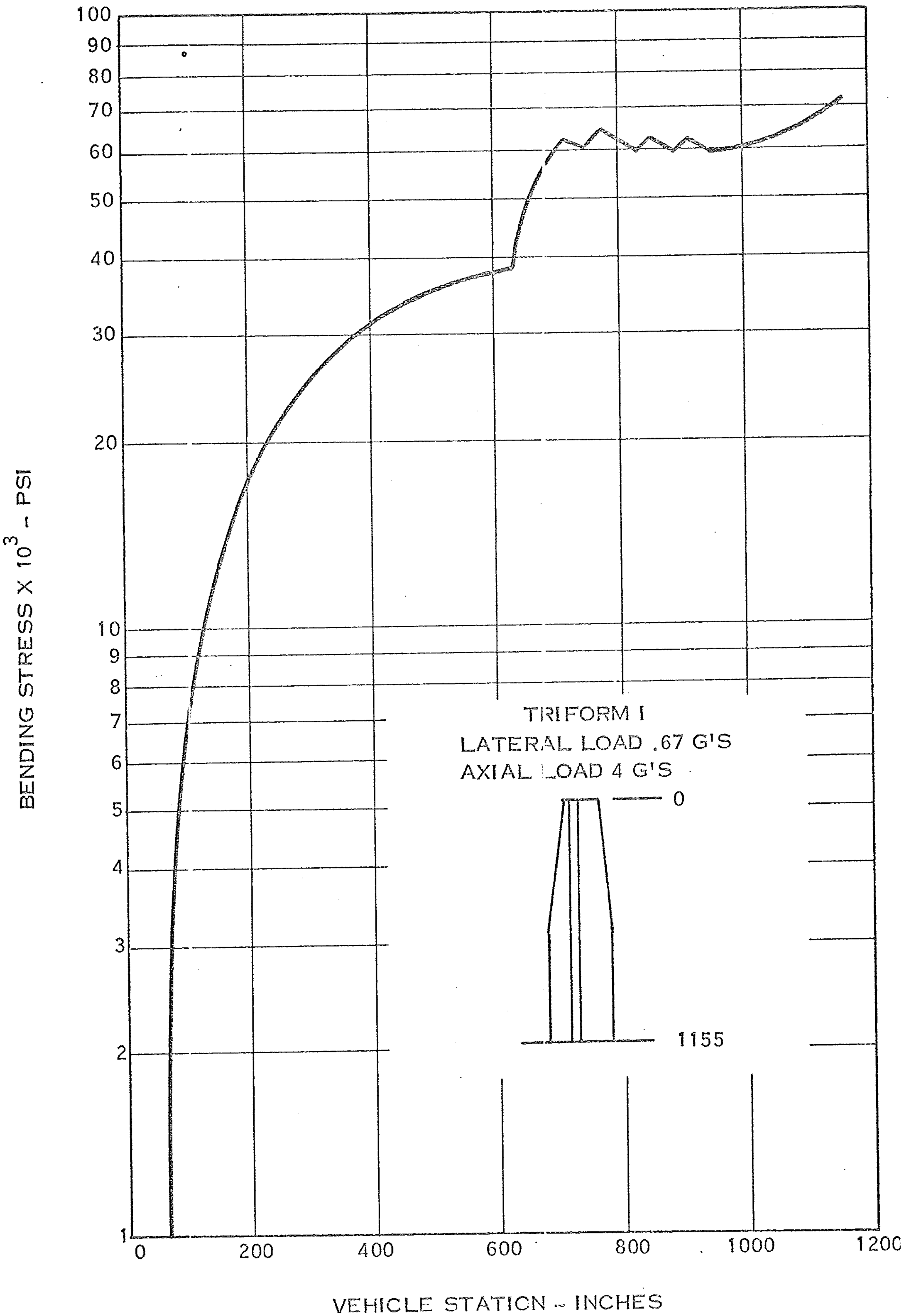
1155

127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800
1000
1200
BENDING STRESS X 10^3 - PSI
VEHICLE STATION ~ INCHES
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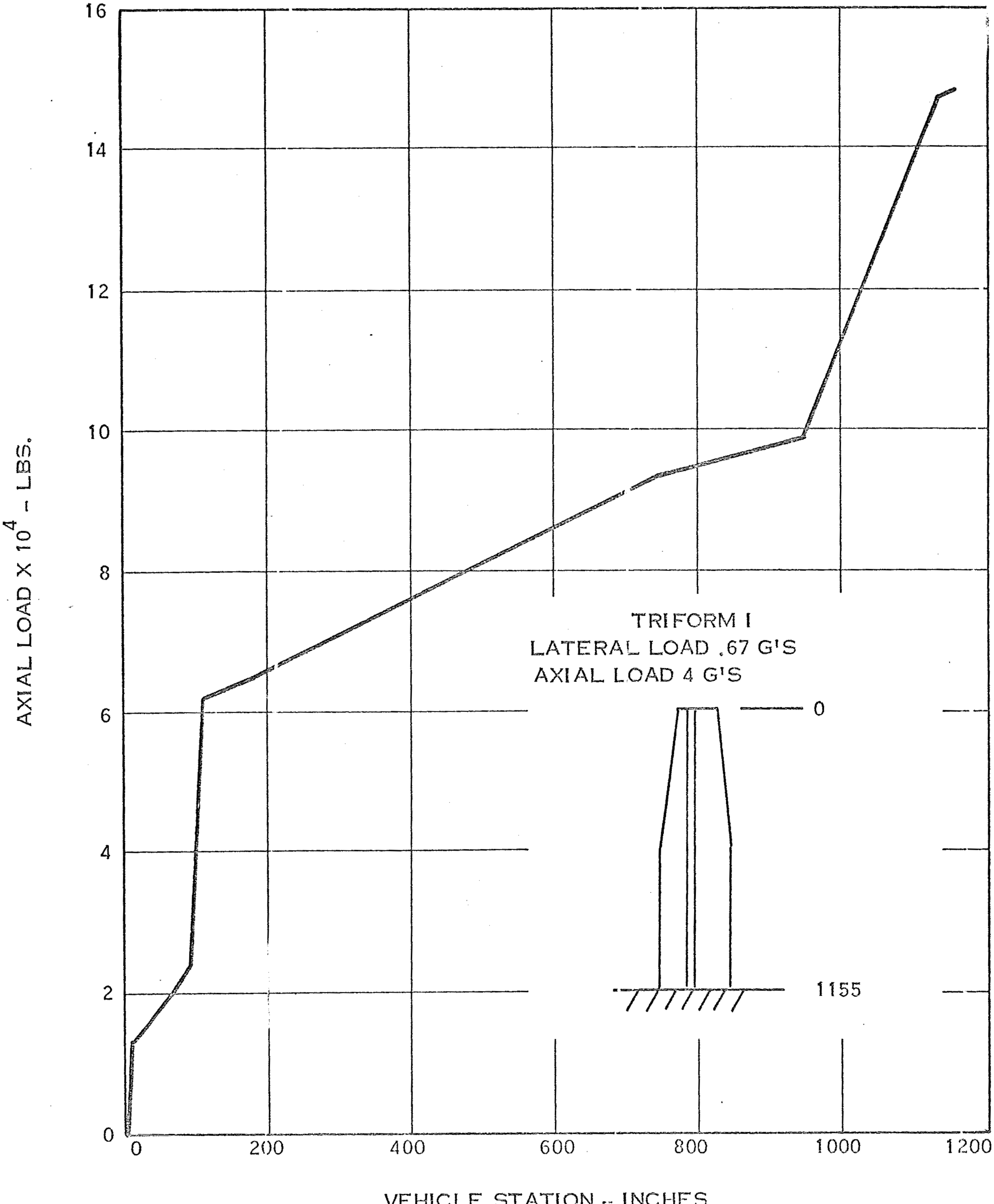
AXIAL LOAD 4 G'S
0
1155

128



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12
10
8
6
4
2
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0
200
400
600
800
1000
1200
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TRIFORM I
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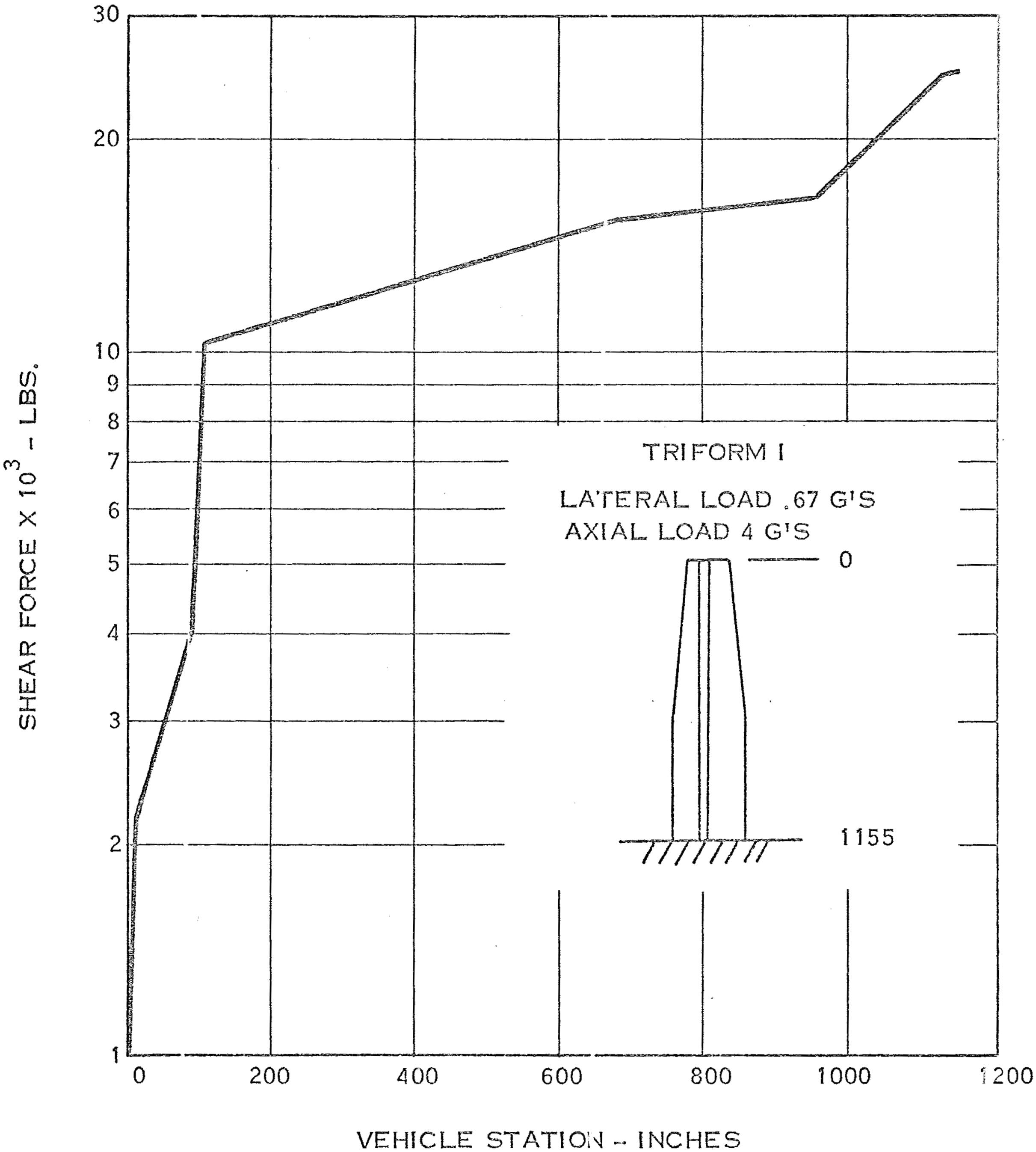
AXIAL LOAD 4 G'S
0
1155

129



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10
9
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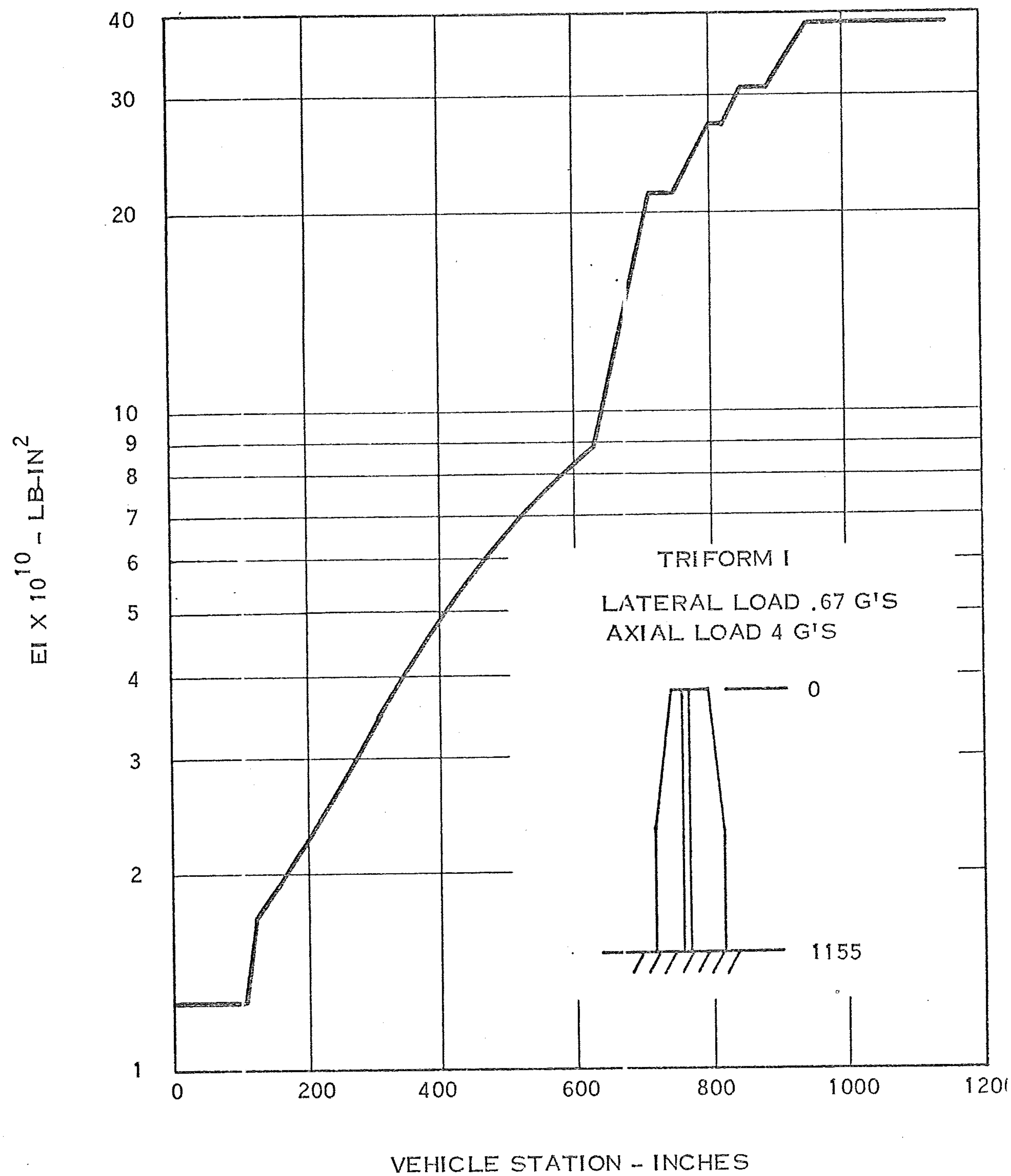
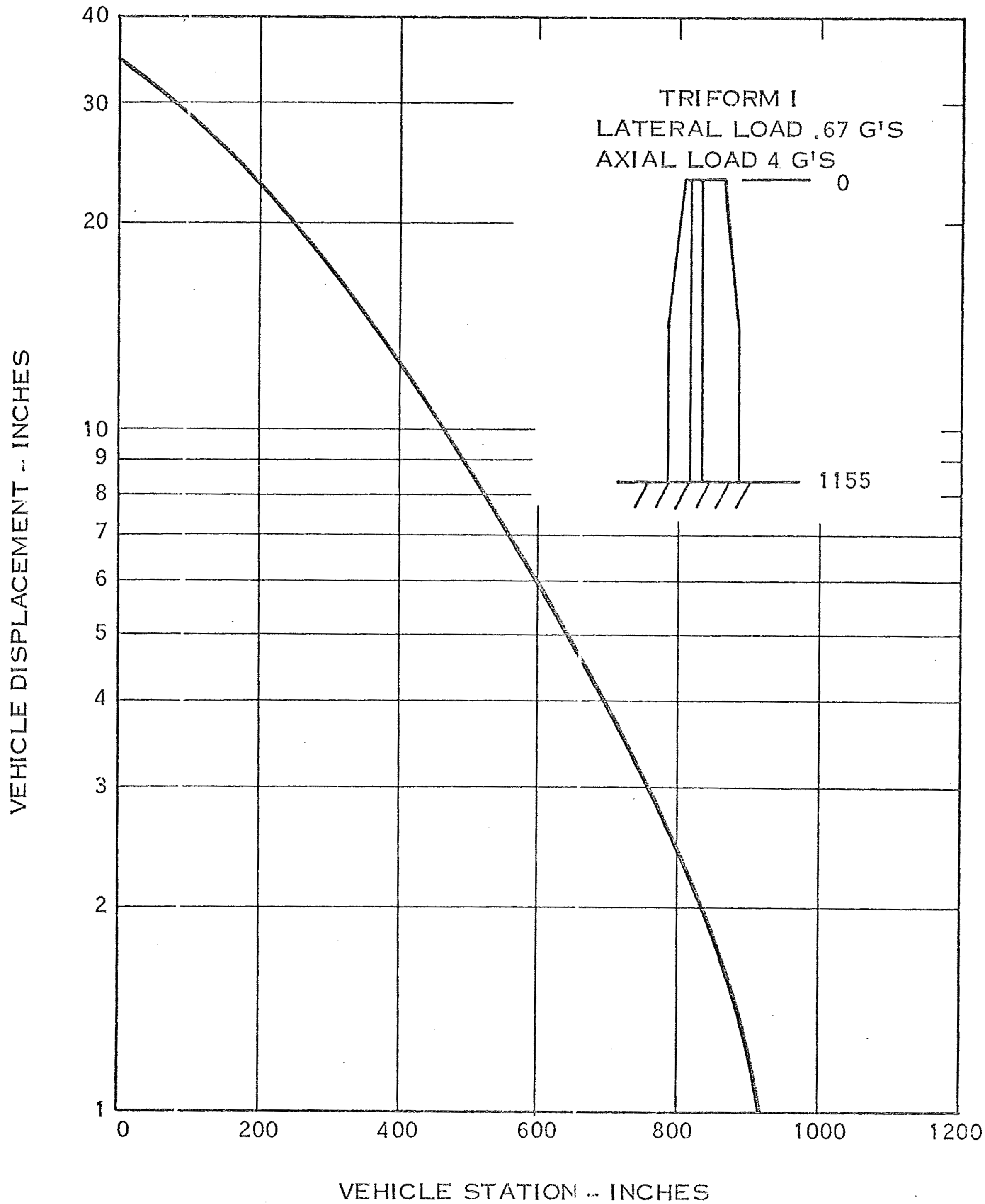
7
6
5
4
3
2
1
0
200
400
600
800
1000
1200
SHEAR FORCE X 10^3 - LBS.
VEHICLE STATION - INCHES
TRIFORM I
LATERAL LOAD .67 G'S
AXIAL LOAD 4 G'S
0
1155


EI X 10^10 - LB-IN^2
1
2
3
4
5
6
7
8
9
10
20
30
40
0
200
400
600
800
1000
1200
VEHICLE STATION - INCHES
TRIFORM I
LATERAL LOAD .67 G'S
AXIAL LOAD 4 G'S
0
1155


TRIFORM I
LATERAL LOAD .67 G'S
AXIAL LOAD 4 G'S
0
1155
VEHICLE DISPLACEMENT - INCHES
40
30
20
10
9
8
7
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5
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3
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1
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200
400
600
800
1000
1200
VEHICLE STATION - INCHES

APPENDIX B

Effects of Thruster Arcing on Average Power Requirements

It is known that arcing frequently occurs within mercury ion engines (thrusters), usually between the screen and the other electrodes. When this happens it is necessary to shut the engine down briefly to allow the arc to extinguish, then restart it. It is necessary to determine how much does this frequent arcing reduce the average power drawn by the thrusters and how is this likely to affect the instantaneous bus voltages.

Assumptions

1. Arcs occur 20 times per hour in each thruster.
2. Thrusters must be shut down for 0.2 sec after arc persists for 2 ms. to allow arc to clear.
3. During the shutdown period, effectively power is removed from the thruster.
4. Each thruster nominally uses 7.75 kW.
5. Thirty-one thrusters out of the total 37 are active at any given time.

Analysis

Average Power

Proportion of total time any one thruster is out is

$$\frac{0.2 \text{ sec/arc} \times 20 \text{ arcs/hr}}{3600 \text{ sec/hr}} = 0.00111$$

Proportion of total time all thrusters are out is

31 x 0.00111 = 0.0344, say 3.5%.

Average power consumed by loads is 96.5% of nominal. Since arcs tend to clear up with thruster age, dropping to perhaps 2/hr., this reduction in average power can be ignored in calculations.

Transient Voltage

One thruster represents (1/31) of total thruster load, which is about 80% of total reactor load. Therefore, one thruster is (1/31) x .8 = 2.6% load change.

It is possible that more than one thruster will arc at the same time, thus causing a greater transient. Probability is as follows.

At any given instant of time, the probability that any one thruster will be out is $\frac{0.2 \times 20}{3600} = .00111$.

The probability that two will be out simultaneously is the combination of 31 things taken 2 at a time times the probability of each happening.

$$P = \frac{31 \times 30}{1 \times 2} \times 0.00111 \times 0.00111$$

$$= 15 \times 31 \times (1.11)^2 \times (10)^{-6}$$

$$= 1.5 \times 3.1 \times (1.11)^2 \times 10^{-6} \times 10^2$$

$$= 4.65 \times 1.23 \times 10^{-4}$$

$$= 5.78 \times 10^{-4} = 0.000578 - \underline{\text{less than } 0.1\%}$$

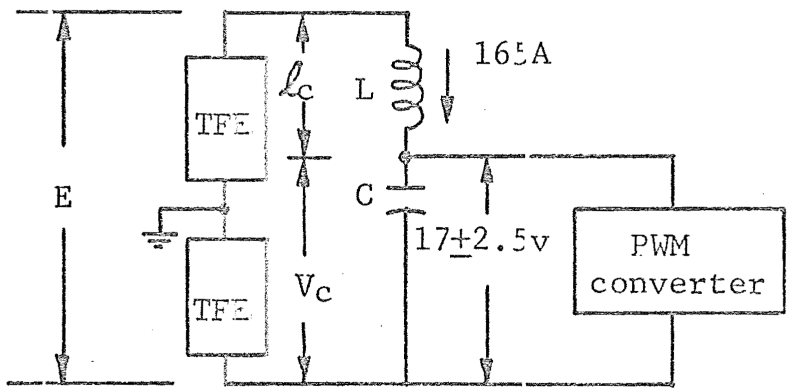
Hence, for calculating transients, assume that only one thruster goes off at any one time.

APPENDIX C

INPUT FILTER DESIGN

The voltage fluctuations to the power converters shall be confined to $\pm$ 2.5v. From the TFE VI curves, Figure 2-17 ; a current excursion of about $\pm$ 50% about a nominal operating point of 165 amperes can be tolerated.

Assume an LC filter and assume that L results in constant current. Assume that the load is a pulse-width modulated (PWM) converter, also assume the operating frequency is 10 kHz. The following circuit applies:

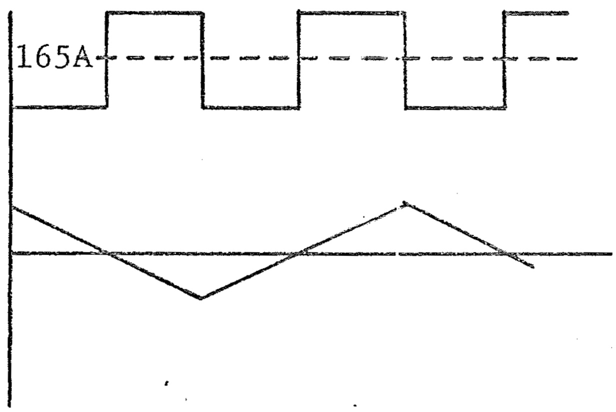


If the current is constant, E must be constant. Therefore, voltage variations across C are reflected in variations across L.

The design will accommodate a voltage range of 12 to 17 volts; therefore at maximum input voltage and minimum input current the duty cycle of the PWM will be approximately 50 percent.

When load is on (converter conducting) current into load is 2 x 165 or 330 amperes, half from L and half from C.

When load is off, current into C is 165 amperes from L



$$\text{Voltage across C} = \frac{\int i dt}{c}$$

$$C = \frac{I \int dt}{c} = \frac{165 \times \frac{10^{-4}}{4}}{e}$$

$$C = \frac{165 \times \frac{10^{-4}}{4}}{5} = \frac{165}{20} \times 10^{-4}$$

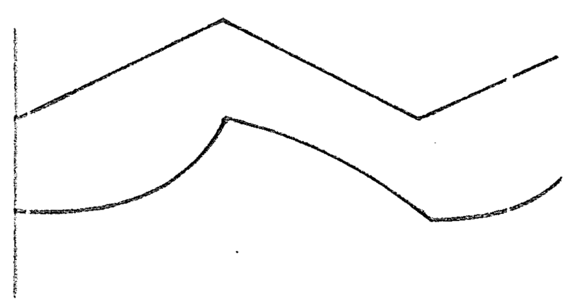
$$C = 800 \ \mu fd$$

Although the calculation for C was based on the assumption that current was constant, some variation will occur to account for the voltage across L.

Computation of actual current waveshape is a complex transient problem; an approximation assumes current variation in L to be $\pm 8\%$ (although 50% can be tolerated) and that the variation is linear.

The relationship between voltage and current is as follows:

$$e = L \frac{di}{dt}$$



$$kt = L \frac{di}{dt}$$

$$Ldi = ktdt$$

$$Li = k\frac{t^2}{2}$$

If the current variation is 0.10 x 165 = 16.5 amperes in 0.25×10^{-4} seconds, the filter size may be approximated as follows:

$$e = L \frac{di}{dt}$$

$$2.5v = L \frac{\frac{12.7}{10^{-4}}}{4} = \frac{50.8}{10^{-4}}$$

$$L = \frac{2.5 \times 10^{-4}}{50.80}$$

$$L = 5 \mu \text{henrys}$$

$$L = 12.6rN^2 \left\{ \ln \frac{8r}{\ell} - \frac{1}{2} + \frac{\ell^2}{32r^2} \left(\ln \frac{8r}{\ell} + \frac{1}{4} \right) \right\} 10^{-9} \text{ henrys}$$

* Hudson's Manual, p. 195, Eqn. 787

Let ℓ = 5 cm, r = 5 cm

N = 7 turns