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HUNTSVILLE, ALABAMA

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August 3, 1960

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SATURN C-2
PHASE I PRELIMINARY DESIGN REPORT (U)

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION

MSFC - Form 523 (September 1960)

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SATURN C-2
PHASE I PRELIMINARY DESIGN REPORT (U)

By

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O refs

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HUNTSVILLE, ALABAMA

ABSTRACT

This report presents the results of the phase I preliminary design study of the SATURN C-2 vehicle. The basic characteristics of the vehicle are:

STAGE	THRUST (POUNDS)	PROPELLANT CAPACITY (POUNDS)	
First	1,500,000	650,000	O_2/RP
Second	800,000	330,000	O_2/H_2
Third	80,000	100,000	O_2/H_2
Fourth	40,000	29,000	O_2/H_2

The vehicle design was investigated for two first stages; ie., a modified cluster tank (modified C-1 booster) and a single tank design. Second stage diameters of 220-in. and 260-in. were studied. Dynamic loads, stage separation dynamics, mass characteristics, various structural arrangements, and second stage propulsion parameters were investigated. Line drawings and mass characteristics for two, three, and four stage versions, and for a nuclear third stage, are given in the appendix. Results of the stage propellant optimization studies are reported.

The recommendations resulting from this design study are:

1. The operational SATURN C-2 vehicle should have the following nominal characteristics:
   a. First stage with a 260-in. diameter, single tank structure, and eight gimballed H-1 engines.
   b. Second stage with 260-in. diameter and four 200K thrust oxygen-hydrogen engines.
   c. Third stage with four 20K*thrust oxygen-hydrogen engines and 220-in. tank diameter.
   d. Fourth stage, when used, a modified CENTAUR stage with the same engine as the third stage.

2. Immediate action should be taken to develop cost and scheduling data for the second stage, such that adequate 1962 FY funds will be available to initiate the second stage development in early fiscal 1962.

3. Detail cost, schedule, and design effort comparison should be conducted on the present first stage, and the single tank first stage, to determine the logical point for phasing the new structure into the program.

4. Control requirements and dynamics of the C-2 vehicle should be investigated in detail, and general requirements and capabilities for nuclear third stages and orbit-launched vehicles be investigated.

* This level has since been reduced to 17.5K primarily to reduce engine development costs.

[redacted]

5. A model description of the SATURN C-2 vehicle, and model specification of the second stage, should be developed on the basis of a thorough preliminary design for the purpose of bid request and negotiation of the second stage development contract.

This report documents the presentation made to Dr. von Braun and the Division Directors on June 3, 1960 and includes most of the supporting data of the study. Phase II of this study is now in progress and includes influences advanced SATURN class vehicle choices may have on the upper stage design parameters.

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I. INTRODUCTION

The SATURN program was formally initiated in August 1958 by Advanced Research Projects Agency (ARPA order 14-59) with the immediate objective of demonstrating the feasibility of obtaining a 1.5 million lb thrust propulsion system based on clusters of modified IRBM or ICBM missile engines. As a result of ABMA proposals, the original order was amended in November 1958 to include the fabrication and flight testing of four SATURN boosters. In December 1958 a SATURN System Study was initiated by ARPA based upon upper stages derived from modified ICBM hardware. The study is reported in Ref. 1. An evaluation committee consisting of Department of Defense and National Aeronautics & Space Administration personnel selected the TITAN booster as second stage and the CENTAUR as third stage.

During the summer of 1959 it became apparent that the restriction of upper stages to modified ICBM hardware seriously limited the payload capability, mission flexibility, and growth potential of the SATURN. ARPA, therefore, requested a new SATURN system study with the only significant restriction being the use of engines then under development. The results of this study are presented in Ref. 2 which recommended the SATURN B with an 880K thrust 220-in. diameter O_2/RP second stage and an 80K thrust third stage with oxygen-hydrogen (O_2/H_2) propellants. This study also showed the potential of a SATURN C and recommended early development of a large O_2/H_2 engine.

In October of 1959 the President announced the transfer of the Development Operations Division of ABMA, and control of the SATURN project, to the National Aeronautics and Space Administration. NASA then formed a committee to determine a long term development plan for the SATURN vehicle system.

The committee was composed of ARPA, Air Force, ABMA, Development Operations Division, and NASA Personnel.

The significant decisions of the committee were that:

a. The operational vehicle should be a SATURN C with all upper stages using the high energy propellants O_2/H_2.
b. The development plan should consider five stages (shown in Table I) as possible building blocks. The stage selection should be such that progress to the full potential of the SATURN C, including an uprated booster, would use all the stages developed.
c. Detail study should be conducted to determine which stages would be used and to define these stages.

TABLE I

STAGE BUILDING BLOCKS FOR THE SATURN C

Stage Designation	Propulsion System	Propellant Type
S-I	8 x 188K	O_2/RP
S-II	4 x 200K	O_2/H_2
S-III	2 x 200K	O_2/H_2
S-IV	4 x 20K	O_2/H_2
S-V	2 x 20K	O_2/H_2

The stages S-II and S-III would be paced by the development of the 200K O_2/H_2 engine. In order to accelerate the SATURN development and provide a good initial payload capability it was decided that the vehicle development would include the following vehicles:

SATURN C-1	S-I, S-IV, S-V
SATURN C-2	S-I, S-II, S-IV, S-V
SATURN C-3	Uprated (new)S-I, S-II, S-III and S-IV or S-V

After discussion of the ground rules with NASA Headquarters, the Future Projects Design Branch initiated studies to define the stages to be used, and to define the C-1 and C-2 vehicles in some detail. The influence of the C-3 configuration on the C-2 configuration is rather qualitative since the C-3 booster thrust level remains defined only as in the range from 1.5 to 3 million pounds. Even this definition is adequate to show that the S-III stage thrust is too low for use as a second stage and too high for use as a third stage of the C-3. No other influence of the C-3 vehicle was considered.

II. SUMMARY OF PRELIMINARY DESIGN INVESTIGATIONS

The design studies were concentrated on the C-2 configuration since the C-3 vehicle remains undefined and the C-1 vehicle was defined as a developmental step toward the C-2 vehicle. The C-1 offers early heavy payload capability; there is at present no critical high priority mission for the C-1. The C-1 will be an operational vehicle of limited lifetime.

The primary mission assigned to the SATURN program is manned lunar circumnavigation. The minimum acceptable payload for this mission is 15,000 lb injected at escape velocity. The NASA Space Task Group stresses this as a minimum payload capability for a three stage SATURN C-2. Performance in this mission is critical and can be compromised only for reliability. This was the controlling factor in selecting the design propellant capacity of the upper stages.

The preliminary investigation considered the following stage configurations:

FIRST STAGE	a. Modified cluster tank structure b. New single tank structure
SECOND STAGE	a. Four 200K thrust O_2/H_2 engines b. Two 200K thrust O_2/H_2 engines c. Stage diameters of 200 in. and 260 in.
THIRD STAGE	a. Four 20K thrust modified LR115 engines b. Six 20K thrust modified LR115 engines c. One 200K thrust O_2/H_2 engine
FOURTH STAGE	a. Modified CENTAUR stage

Table II summarizes the results of the optimization of the three-stage SATURN C-2. The four-stage version was also optimized. A detailed report of the three and four stage C-2 performance optimization study is given in Ref. 3.

Considering the primary mission and the performance results presented in Table II, the following selections for stage propulsion and usable stage propellant capacity were selected.

STAGE	PROPULSION	TANK CAPACITY FOR USABLE PROPELLANTS (POUNDS)
FIRST	Eight H-I engines O_2/RP	650,000
SECOND	Four 200K engines O_2/H_2	330,000
THIRD	Four 20K engines O_2/H_2 (modified LR115)	100,000
FOURTH	Two 20K engines O_2/H_2 (modified LR115)	29,000

TABLE II

OPTIMUM PAYLOAD AND PROPELLANT LOADINGS FOR THREE-STAGE VERSION OF C-2

Configuration	Parameter	Mission			
		100 S. Mi.	96- Minute	Escape	24-Hour Equatorial
Second Stage	Net Payload, lb	54,250	50,390	16,400	9,335
4 x 200K	Consumable Propellants, lb				
Third Stage	Stage I	600,000	600,000	600,000	600,000
4 x 20K	Stage II	321,500	322,000	325,000	327,000
	Stage III	67,650	71,200	102,200	107,467
	Flight Performance Reserves, lb	4,510	4,360	2,740	2,300
Second Stage	Net Payload, lb	53,560	49,650	15,240	8,015
4 x 200K	Consumable Propellants, lb				
Third Stage	Stage I	600,000	600,000	600,000	600,000
6 x 20K	Stage II	297,500	299,000	310,000	315,000
	Stage III	91,200	93,500	117,500	119,909
	Flight Performance Reserves, lb	4,615	4,465	2,780	2,306
Second Stage	Net Payload, lb	55,315	51,280	15,890	8,560
4 x 200K	Consumbale Propellants, lb				
Third Stage	Stage I	600,000	600,000	600,000	600,000
1 x 200K	Stage II	257,500	258,500	275,000	286,000
	Stage III	129,300	132,450	151,900	148,460
	Flight Performance Reserves, lb	4,758	4,602	2,860	2,354
Second Stage	Net Payload, lb	44,650	41,390	12,750	6,820
2 x 200K	Consumable Propellants, lb				
Third Stage	Stage I	700,000	700,000	700,000	700,000
4 x 20K	Stage II	253,500	254,400	263,000	264,000
	Stage III	51,500	54,000	74,780	80,040
	Flight Performance Reserves, lb	3,930	3,800	2,360	1,961

Three vehicles based on these selected parameters were investigated in some detail. The vehicles are shown in Fig. 1 as four-stage vehicles. The center vehicle with 220-in. diameter second stage was rejected due to its very low free-free mode bending frequency and the severe performance restrictions the 220-in. diameter will impose on the vehicle when a nuclear third stage is developed. The vehicle with the single tank booster structure at the left in Fig. 1, and the vehicle with a modification of the present SATURN booster, are recommended for detailed analysis to determine the most desirable configuration for the operational SATURN C-2. A weight summary of these vehicles is given in Table III.

The stage design weights and engine performance parameters of the upper stages are based upon preliminary design data. Therefore, it is suggested that any agencies or design groups planning payloads for the SATURN C-2 vehicle use the nominal payload capabilities in Table IV for preliminary design of the payloads.

TABLE IV

SATURN C-2 NOMINAL PAYLOAD CAPABILITY

MISSION	3-STAGE VEHICLE (POUNDS)	4-STAGE VEHICLE (POUNDS)
100-Statute Mile Orbit	49,000	
96-Minute Orbit	45,000	
Escape	15,000	18,000
24-hr Orbit (equatorial, launched due east from AMR)	8,000	12,000

Preliminary design investigations were made in the following areas and are reported in subsequent chapters:

Engine Expansion Ratio
Engine Oxidizer to Fuel Ratio
Vehicle Frequency
Bending Moments
Stage Separation
Stage Design (structure and propulsion system)
Parametric Studies of the SATURN C-2 for possible use in an orbital refueled space vehicle system.

This design study was independent of any potential contractor design study for the second stage of the SATURN C-1, which is to be used also as the third stage of the SATURN C-2. Since completion of this first phase investigation, the contract for development of this stage has been awarded to the Douglas Aircraft Co.

A joint Douglas - Marshall Space Flight Center conference to establish the design criteria for this stage was held on May 24-25, 1960.

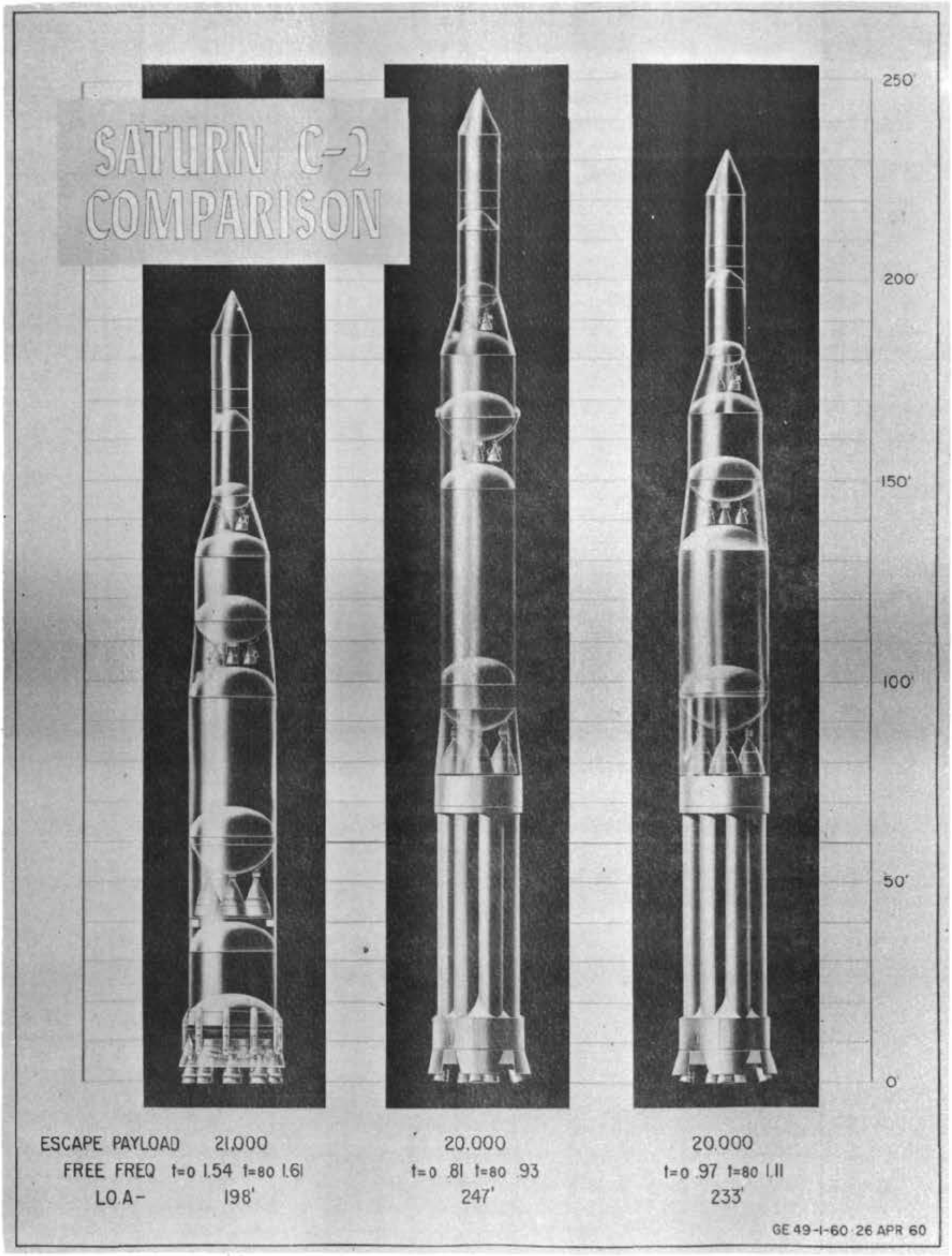
SATURN C-2
COMPARISON
250'
200'
150'
100'
50'
0'
ESCAPE PAYLOAD 21,000
FREE FREQ t=0 1.54 t=80 1.61
LO.A- 198'
20,000
t=0 .81 t=80 .93
247'
20,000
t=0 .97 t=80 1.11
233'
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Figure I

TABLE III [redacted]

SATURN C-2 WEIGHT SUMMARY

4-STAGE ESCAPE MISSION

	W_8 EXPECTED PROPELLANT CONSUMPTION	W_s STAGE DRY WEIGHT	W_n STAGE BURN-OUT WEIGHT	W_0 TAKE-OFF WEIGHT
STAGE I (CLUSTERED)	600,000 650,000 (capacity)	88,500	109,500	1,200,000
STAGE I (SINGLE TANK)	646,500 650,000 (capacity)	49,000	63,000	
STAGE II	320,000 330,000 (capacity)	26,620	31,610	490,500
STAGE III	71,500 100,000 (capacity)	8,430 *	14,510 *	137,870 **
STAGE IV	29,000	5,000 *	6,150 *	51,860

* Jettisonable insulation of 1,660 lb. in Stage II, 590 lb in Stage III, and 300 lb in Stage IV included.

** Weight of 130 lb for a propulsion device used to separate Stage III from Stage II.

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[redacted]

Table III

The resulting stage design is quite similar in geometry and weight to the third stage design developed in this study. There are extensive differences as to details of the stage design; however, these do not affect the vehicle performance or overall dynamics for preliminary design investigations. The Douglas proposal, as modified in the design criteria conference, is discussed in Section V.

III. RECOMMENDATIONS

Based on this preliminary investigation it is recommended that:

1. The operational SATURN C-2 vehicle be defined with the following nominal characteristics:
   a. First stage with 260-in. diameter single tank structure and eight gimballed H-1 engines.
   b. Second stage with 260-in. diameter and four 200K O_2/H_2 engines.
   c. Third stage with 220-in. diameter and four 20K (modified LR115) O_2/H_2 engines.
   d. Fourth stage, when used, a modified CENTAUR stage with the same engine as used in the third stage.

2. Immediate action be taken to develop cost and scheduling data for the second stage, such that adequate 1962 FY funds will be available to initiate the second stage development in early fiscal 1962.

3. Detail cost, schedule, and design effort comparison be conducted on the present first stage, and the single tank first stage, to determine the logical point for phasing the new structure into the program.

4. Control requirements and dynamics of the C-2 vehicle be investigated in detail, and general requirements and capabilities for nuclear third stages and orbit-launched vehicles be investigated.

5. A model description of the SATURN C-2 vehicle, and model specification of the second stage, be developed on the basis of a thorough preliminary design for the purpose of bid request and negotiation of the second stage development contract.

IV. SELECTION OF PROPULSION PARAMETERS

During the course of this study the information generated was applied in the bid specifications for the 200K O_2/H_2 engine. The engine development contract was awarded to Rocketdyne Division of North American Aviation during the drafting of this report. The 200K engine as shown in the second stage was derived from the data available at the beginning of this study, then modified to reflect the operating parameters determined from the study. The engine system and packaging of the proposed engine will result in modification to the suction line layout and gimbal actuator mounting. The basic thrust structure, tank pressures, and expansion area ratio are the same as developed in this study.

The results of the engine expansion area ratio study are summarized in Fig. II. The figure shows the increased weight and specific impulse with increasing expansion ratio which was used to calculate the payload curves. Two parametric payload curves are shown labeled 30 lb per in. of ΔL and 70 lb per in. of ΔL. The curves include the effects of engine weight change, the specific impulse change, and the indicated structural weight change. The weight change in the structure includes the weight of the increasing length of interstage adapter as well as weight increase in each upper stage due to increased vehicle bending moment with increasing length. The curves indicate that the performance varies very little with changing expansion ratio in the neighborhood of the optimum ratio.

At 35:1 area ratio, little gain is possible with increased area ratio. Since the specific engine characteristics are not yet available, 35:1 was selected as a tenative best compromise between performance, separation problems, and engine clearance for gimballing. The study is reported in detail in Ref. 4.

The engine mixture ratio was investigated at values of 5 and 6. The peak of the theoretical specific impulse (shifting equilibrium) vs mixture ratio curve occurs at a mixture ratio of about 4.5 and is quite flat. The experimental data available indicate that about 95% of theoretical values can be obtained. A design point of 5 was selected. Going to a mixture ratio of 6 for the second and third stages shortens the C-2 vehicle by about 7 feet but costs a loss of 10 seconds in specific impulse. This reduces the escape payload by 9%. Pending detailed engine data, an oxygen-hydrogen mixture ratio of 5 is recommended and was the basis of the upper stage designs.

C-2 ENGINE PARAMETERS

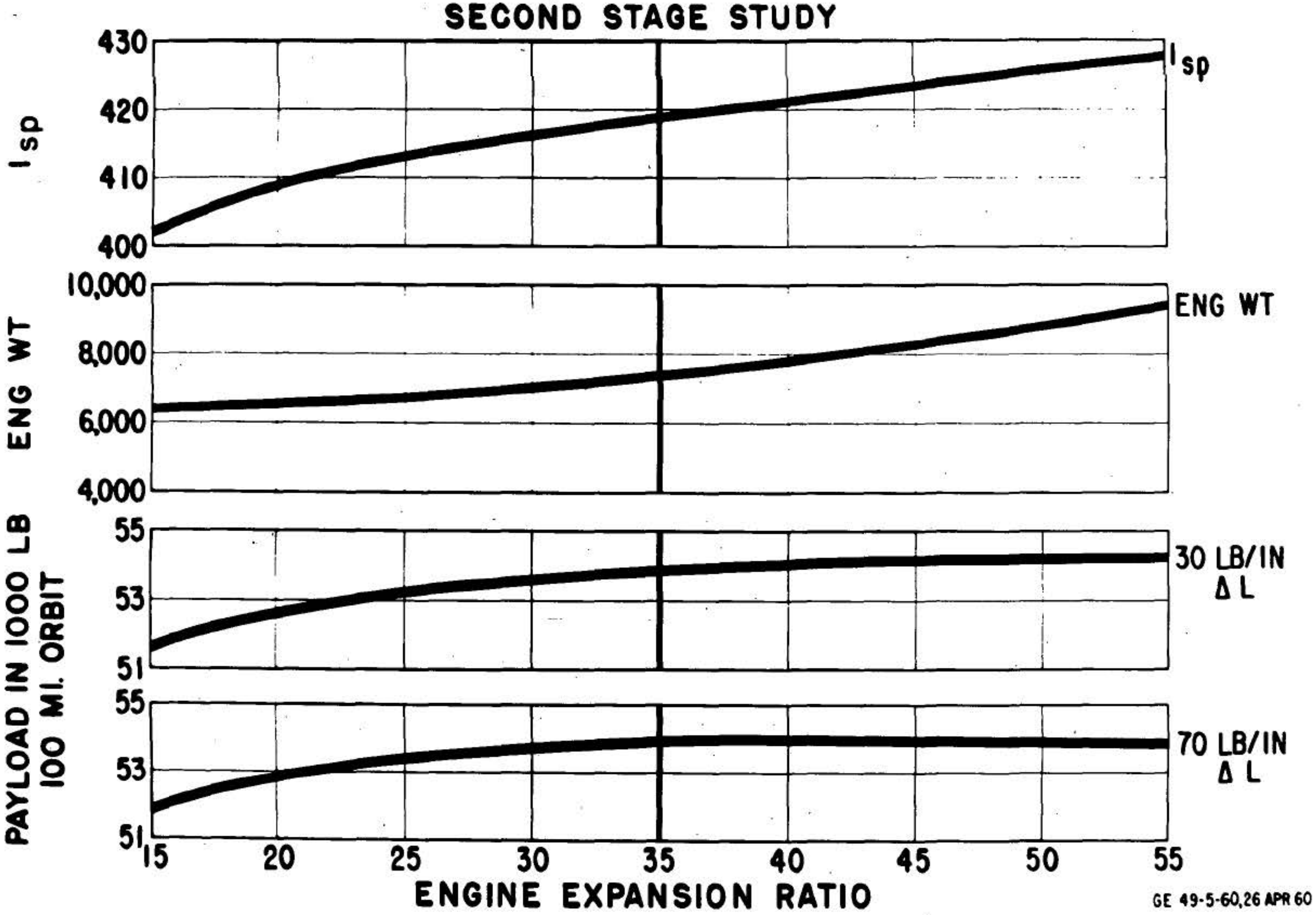

Figure 2

V. DESCRIPTION OF SELECTED SATURN C-2 VEHICLES

In Section II of this report several areas were listed in which preliminary design investigations were made. The first two areas concerning the expansion ratio and the oxidizer fuel ratio were discussed in Section IV. In the first part of this section the vehicle frequency, bending moment, and stage separation are discussed and, in the second part, the vehicle stage design is covered. The remaining area, dealing with orbital refueling, is discussed in Section VI.

A. Vehicle Dynamics, Loads, and Stage Separation.

1. Vehicle Dynamics - As stated in Section II of this report, three vehicles were chosen for more detailed investigation. These three vehicles are shown in Fig. 1. In this figure, for each vehicle, the first mode free-free bending frequency is given for take-off and q_{max} conditions. From discussions with personnel of the Guidance and Control Division, a tentative minimum value for the ratio of first mode free-free bending frequency to control frequency was set at 5. Present control frequency for the clustered booster is between 0.2 and 0.3 cycles per second. Based on these numbers the minimum free-free bending frequency of the C-2 vehicle would be between 1.0 and 1.5 cycles per second. This requirement rules out the second configuration which has the 220-in. diameter second stage. The two remaining configurations meet the bending frequency requirements, however, the configuration with the clustered tank booster is marginal. As indicated in Fig. 1, the bending frequency of the single tank booster configuration is considerably higher. This is due mainly to the decrease in length because of the more compact arrangement afforded by the single tank. There is a small effect on the bending frequency due to stiffness distribution of the two selected configurations. Figure 3 shows the stiffness distribution of the two configurations. Increase in stiffness obtained by using the single tank booster is shown quite clearly by this figure.

2. Vehicle Loads - For the two remaining vehicles discussed under paragraph 1 above, studies were conducted to determine the loads for the point at which dynamic pressure is maximum (q_{max}). Newtonian Theory was used to predict the aerodynamic loading for this condition. The aerodynamic loadings for both the cluster and the single tank four stage configurations at q_{max} conditions are shown in Fig. 4. Loading for the cluster tank booster is for an angle of attack of 11.8 degrees. The corresponding normal force coefficient C_{N_α} and center of pressure C_p are 3.40 per radian at station 1490. The angle of attack of 11.8 degrees is the steady state angle of attack which produces an aerodynamic moment equal to 67% of the moment produced by gimballing the four control engines of the cluster booster 7 degrees. The remaining 33% of the engine moment is used for restoring moment and inertia effects.

From the aerodynamic loading curve and a preliminary weight breakdown, the loads at q_{max} for the cluster booster configuration were

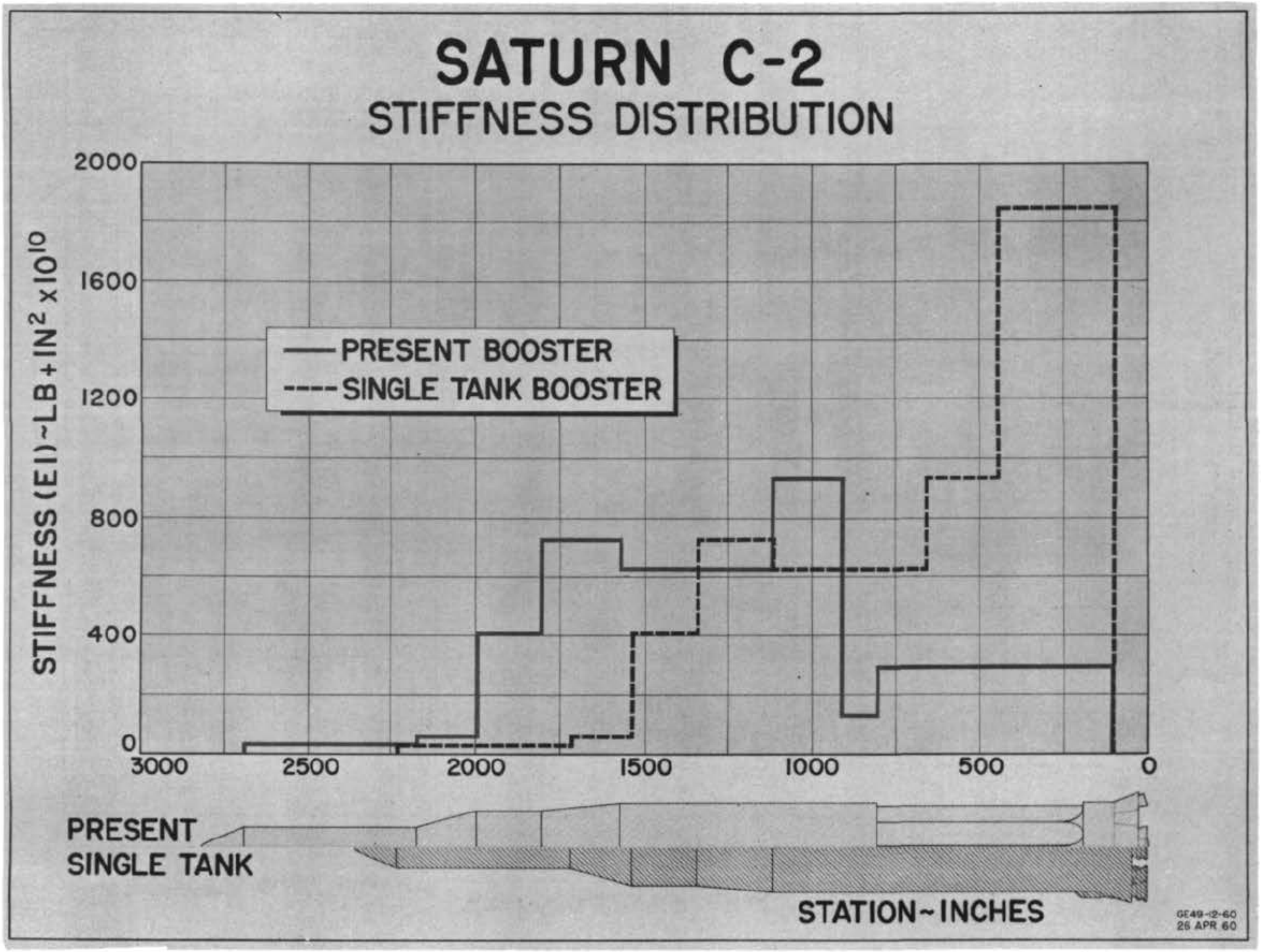
SATURN C-2
STIFFNESS DISTRIBUTION
STIFFNESS (EI)~LB + IN² x10¹⁰
2000
1600
1200
800
400
0
PRESENT BOOSTER
SINGLE TANK BOOSTER
3000
2500
2000
1500
1000
500
0
PRESENT
SINGLE TANK
STATION~INCHES
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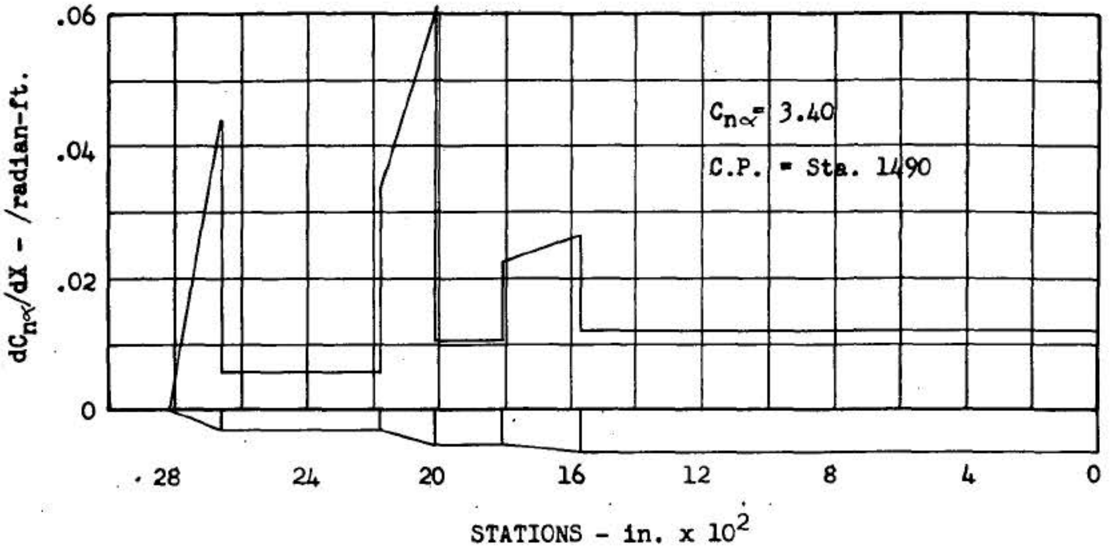
Figure 3

SATURN C-2
AERODYNAMIC LOADING ESTIMATES
(NEWTONIAN THEORY)

CLUSTER TANK



SINGLE TANK

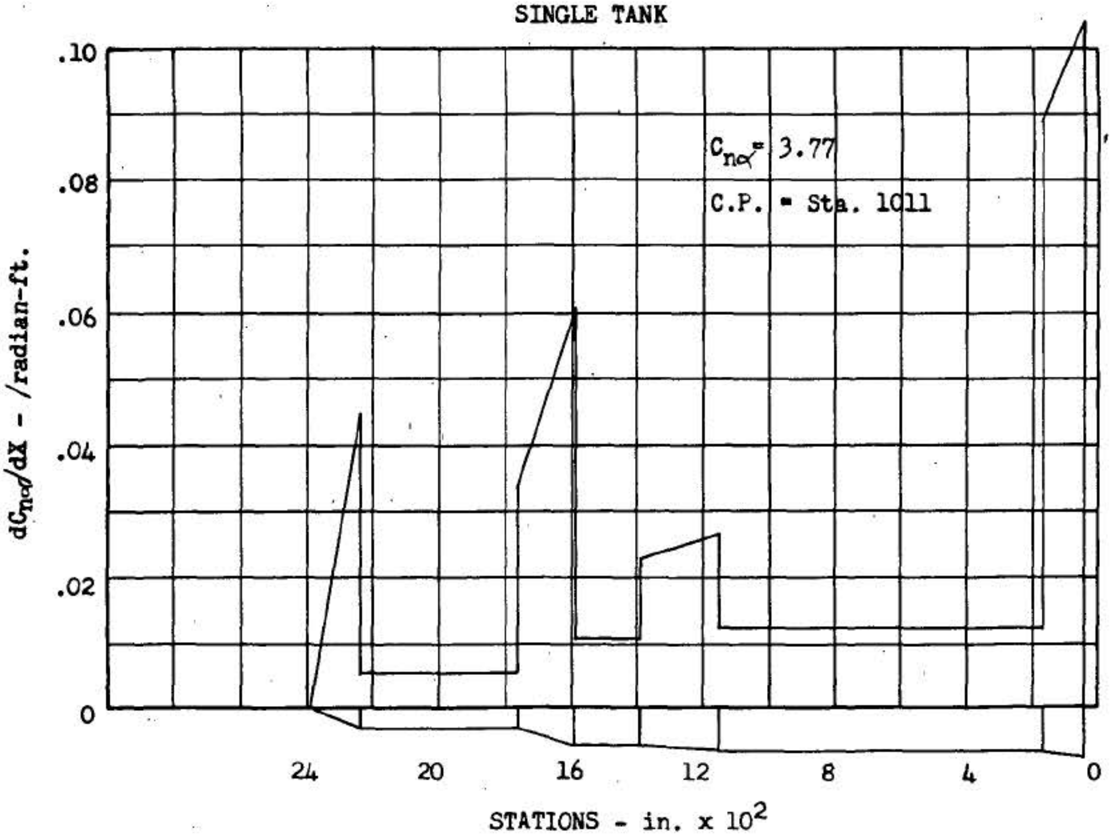


Figure 4

computed. Longitudinal, shear, and bending moment load curves are shown in Figs. 5 through 7 . The center of gravity and moment of inertia used to compute these loads are given in Appendix C .

For the single tank booster configuration, the same angle of attack of 11.8 degrees was used so that a direct comparison of the vehicles could be made. Using the same angle of attack also provided a means of determining the approximate gimbal angle requirement of the single tank booster engines. The resulting load curves for the single tank booster configuration are shown in Figs. 8 through 10. The gimbal angle of the single tank booster comparable to the 7 degrees on the cluster is only 4.3 degrees.

3. Stage Separation - From previous studies the separation of the booster from the upper stages was known to be a problem area and, therefore, was studied in some detail. Results from the initial studies have already been published in Ref. 5 . At initiation of the separation studies, the shape of the thrust buildup curve was not known. In order to proceed with the study, a set of thrust buildup curves were assumed. The assumed curves bracket the curve given by Rocketdyne in their proposal. Additional studies were made using the Rocketdyne curve and the results are given herein.

Aerodynamic loads during separation were computed using the Second Order Shock Expansion Theory. The aerodynamic loading used for the separation studies is shown in Fig. 11. Using this aerodynamic loading and the thrust buildup from the Rocketdyne proposal, the time history of angle of attack after booster separation was computed for several initial angles of attack and delay times. The results from these computations are given in Appendix D . Figure12 presents the limiting conditions of initial angle and delay time. Table V contains the values for the various constants used in computing the separation data. It appears that a delay time of approximately 0.6 seconds could be used with a reasonable margin of safety if the initial angle of attack can be held to within one degree.

TABLE V

SEPARATION STUDY CONSTANTS

Vehicle Weight at Second Stage Ignition	489,580 lb
Moment of Inertia at Ignition	5.883×10^8 lb-ft^2
Center of Pressure	Sta 1919
Center of Gravity	Sta 1434
Gimbal Point	Sta 1046
C_{z_α}	3.29 rad^{-1}
Nominal Dynamic Pressure	360 psf
Maximum Gimbal Angle	$\pm$ 7 deg

FOUR STAGE SATURN C-2 (CLUSTERED BOOSTER)
LONGITUDINAL LOAD DISTRIBUTION

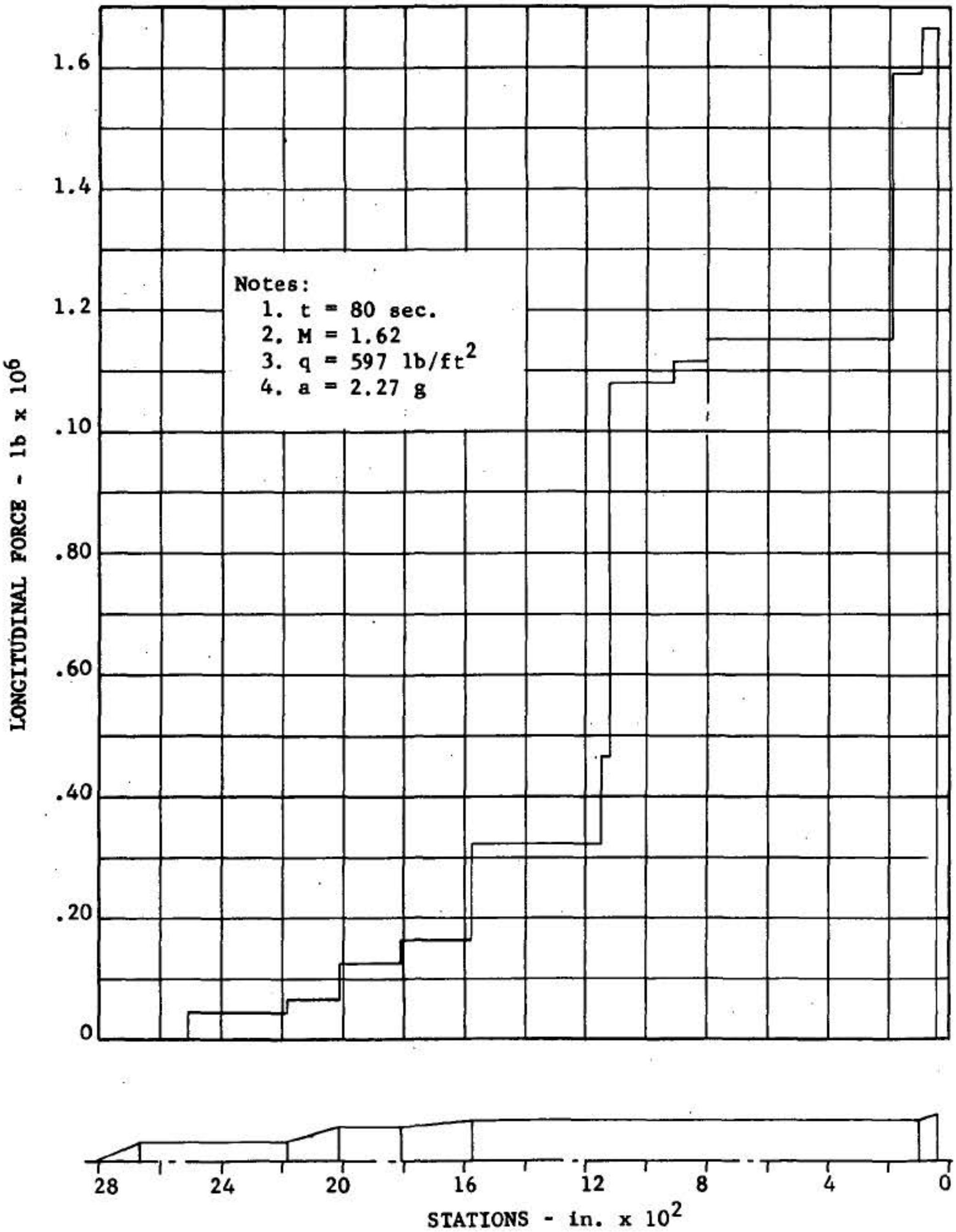


Figure 5

FOUR STAGE SATURN C-2 (CLUSTERED BOOSTER)
SHEAR LOAD DISTRIBUTION

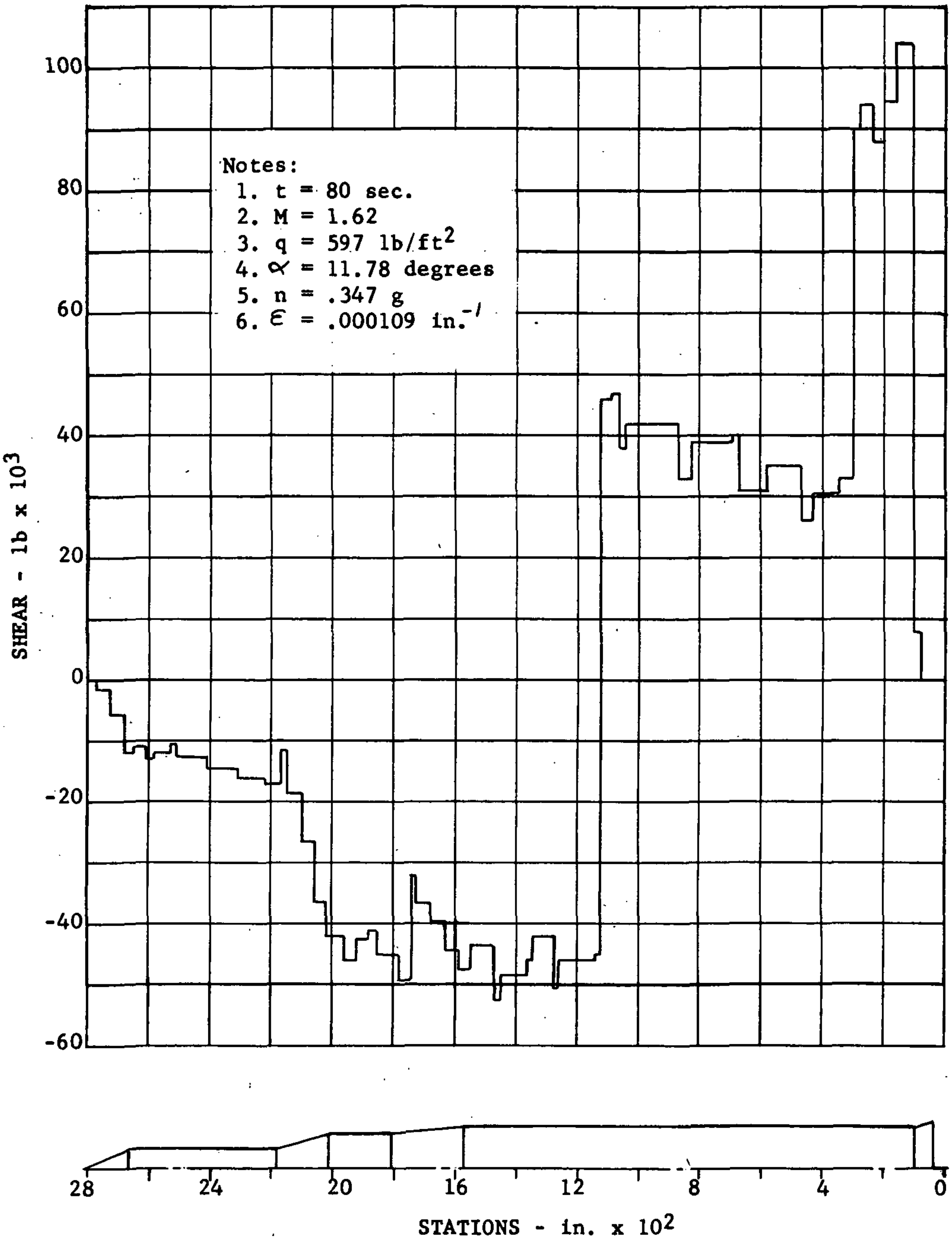
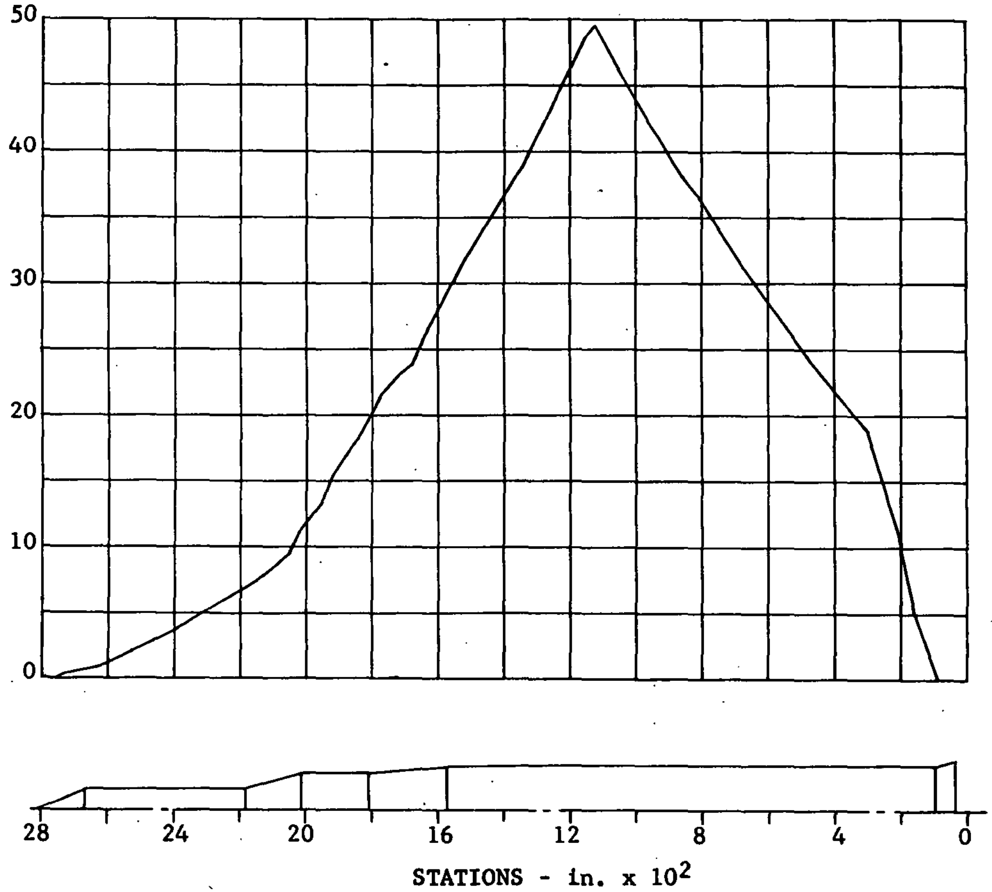


Figure 6

FOUR STAGE SATURN C-2 (CLUSTERED BOOSTER)
BENDING MOMENT LOAD DISTRIBUTION



Notes:
1. t = 80 sec.
2. M = 1.62
3. q = 597 lb/ft^2
4. α = 11.78 degrees
5. n = .347 g
6. ε = .000109 in.$^{-1}$

Figure 7

FOUR STAGE SATURN C-2 (SINGLE TANK BOOSTER)
LONGITUDINAL LOAD DISTRIBUTION

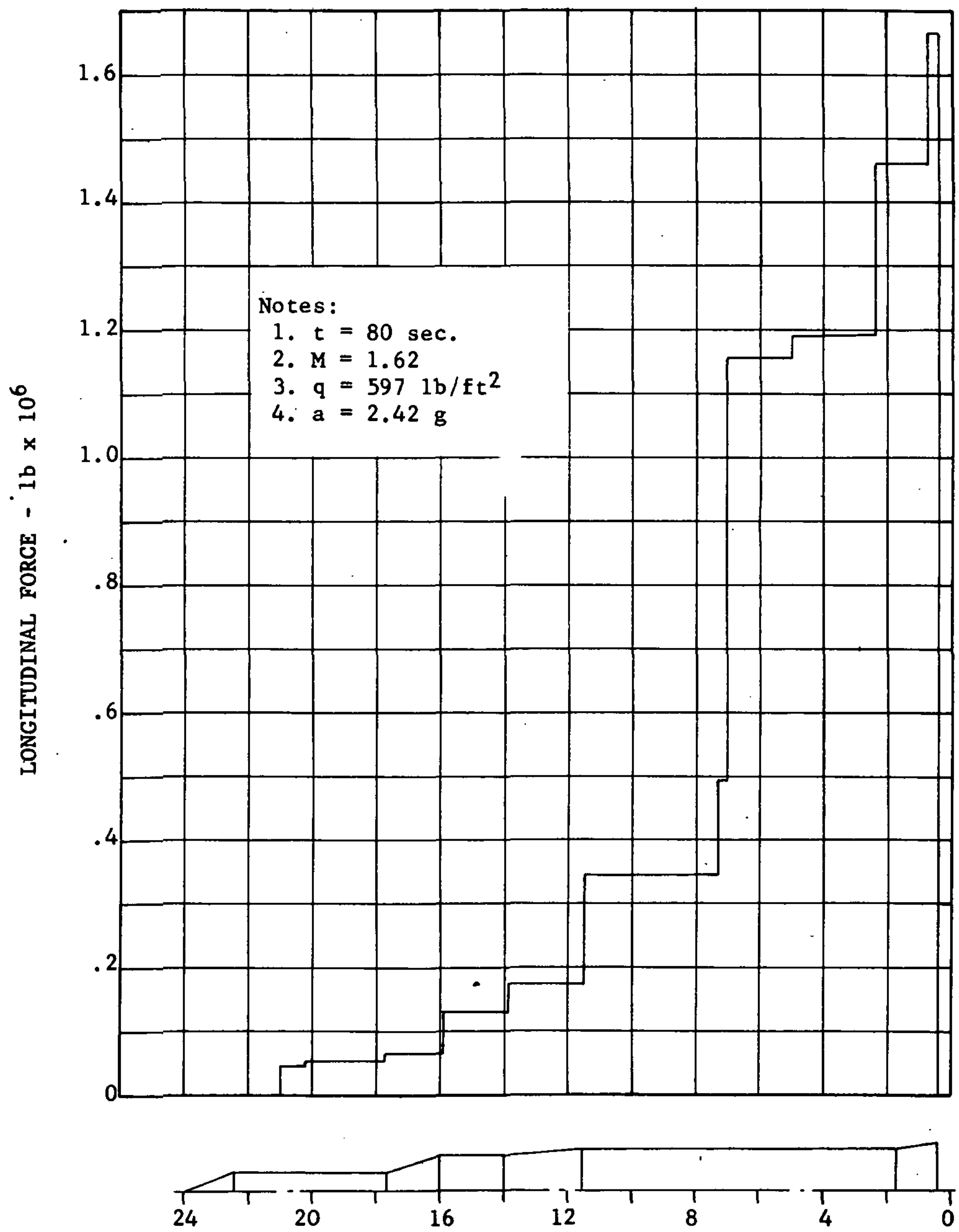


Figure 8

FOUR STAGE SATURN C-2 (SINGLE TANK BOOSTER)
SHEAR LOAD DISTRIBUTION

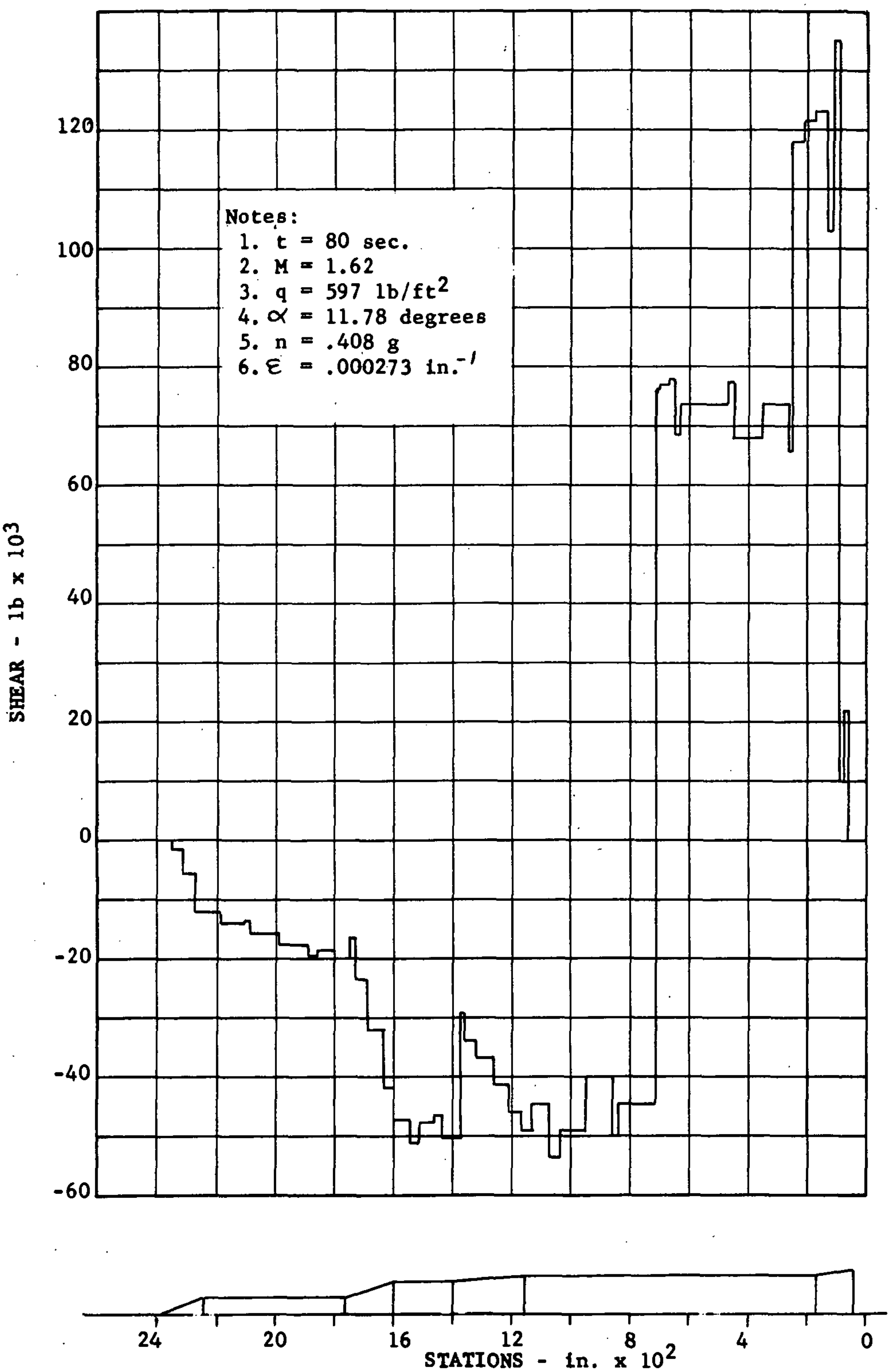
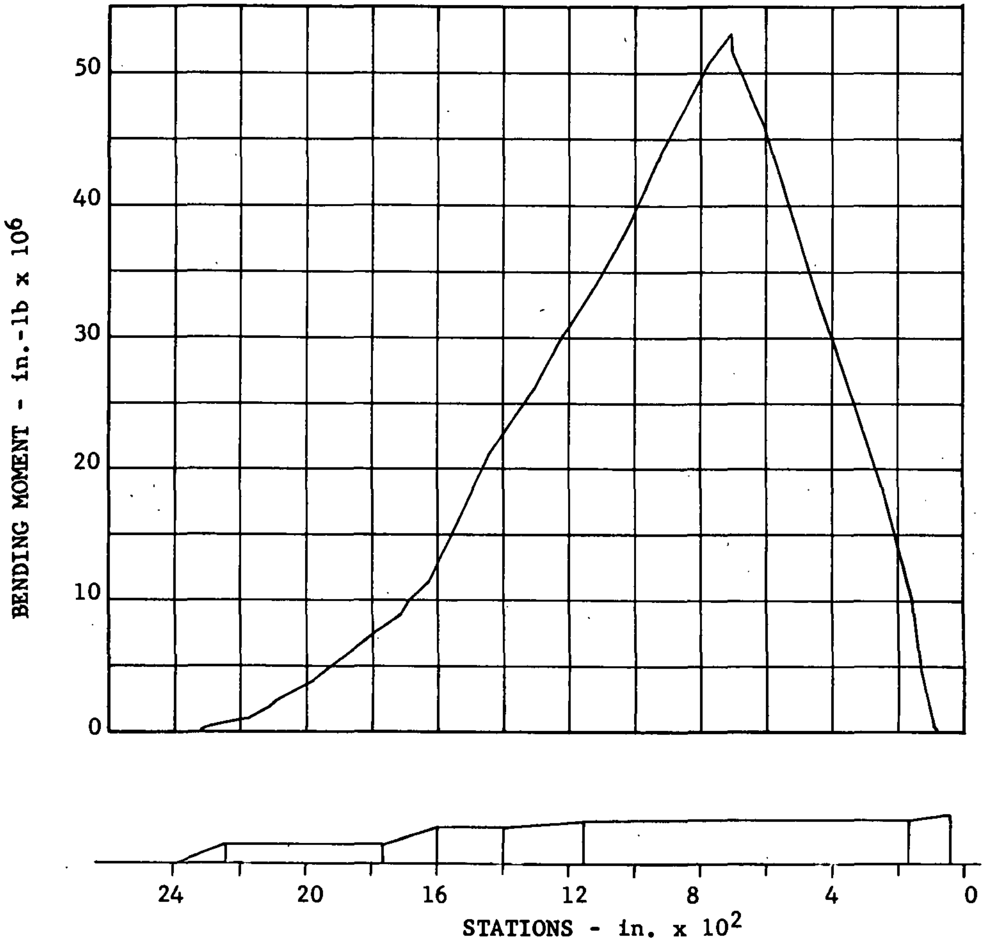


Figure 9

FOUR STAGE SATURN C-2 (SINGLE TANK BOOSTER)
BENDING MOMENT LOAD DISTRIBUTION



Notes:
1. t = 80 sec.
2. M = 1.62
3. q = 597 lb/ft^2
4. α = 11.78 degrees
5. n = .408 g
6. ε = .000273 in.$^{-1}$

Figure 10

AERODYNAMIC LOADING ESTIMATES FOR SEPARATION STUDIES

(MACH NO. 2.82)

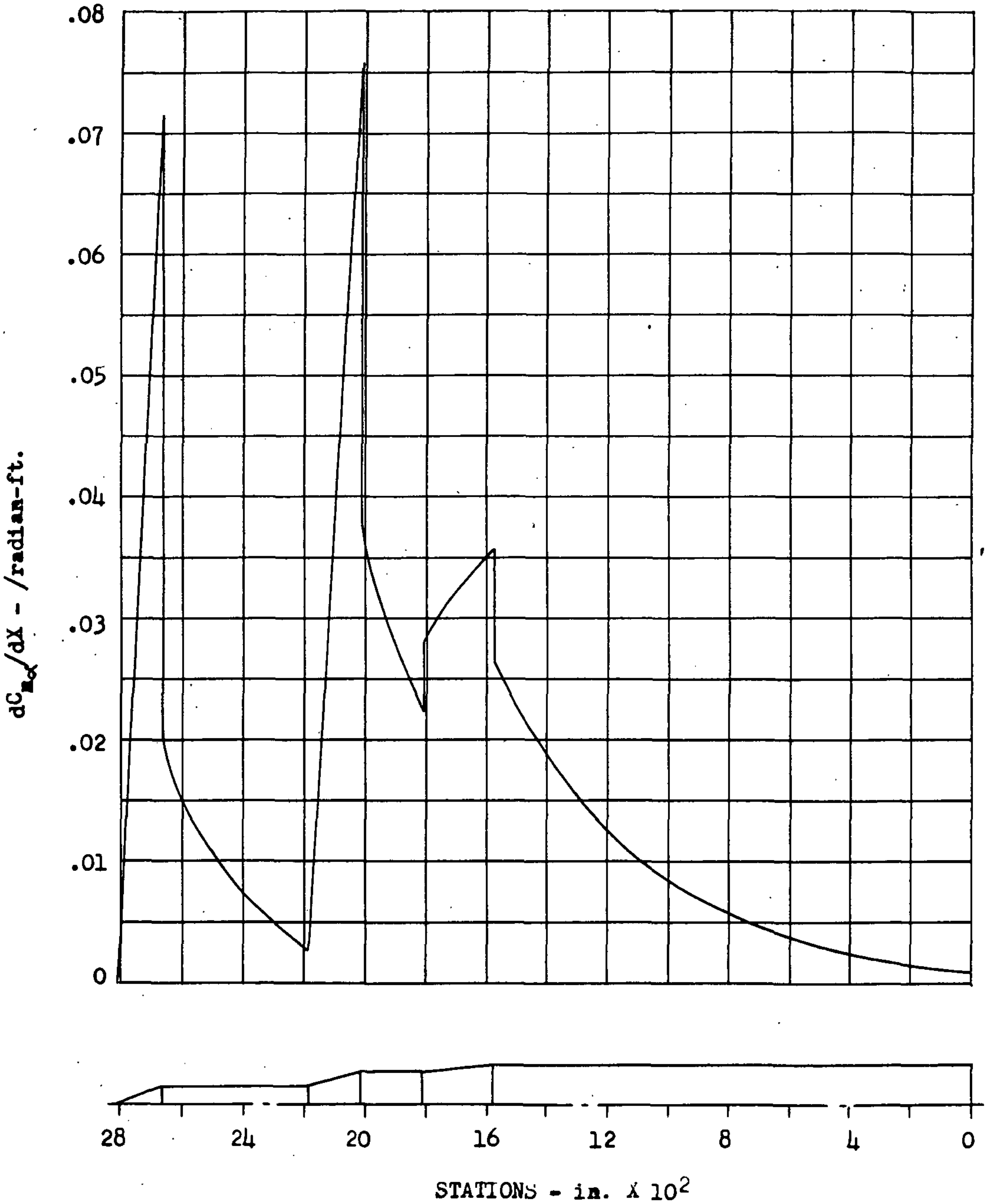


Figure 11

INITIAL ANGLE OF ATTACK AND DELAY TIME LIMITS
FOR SECOND STAGE SEPARATION

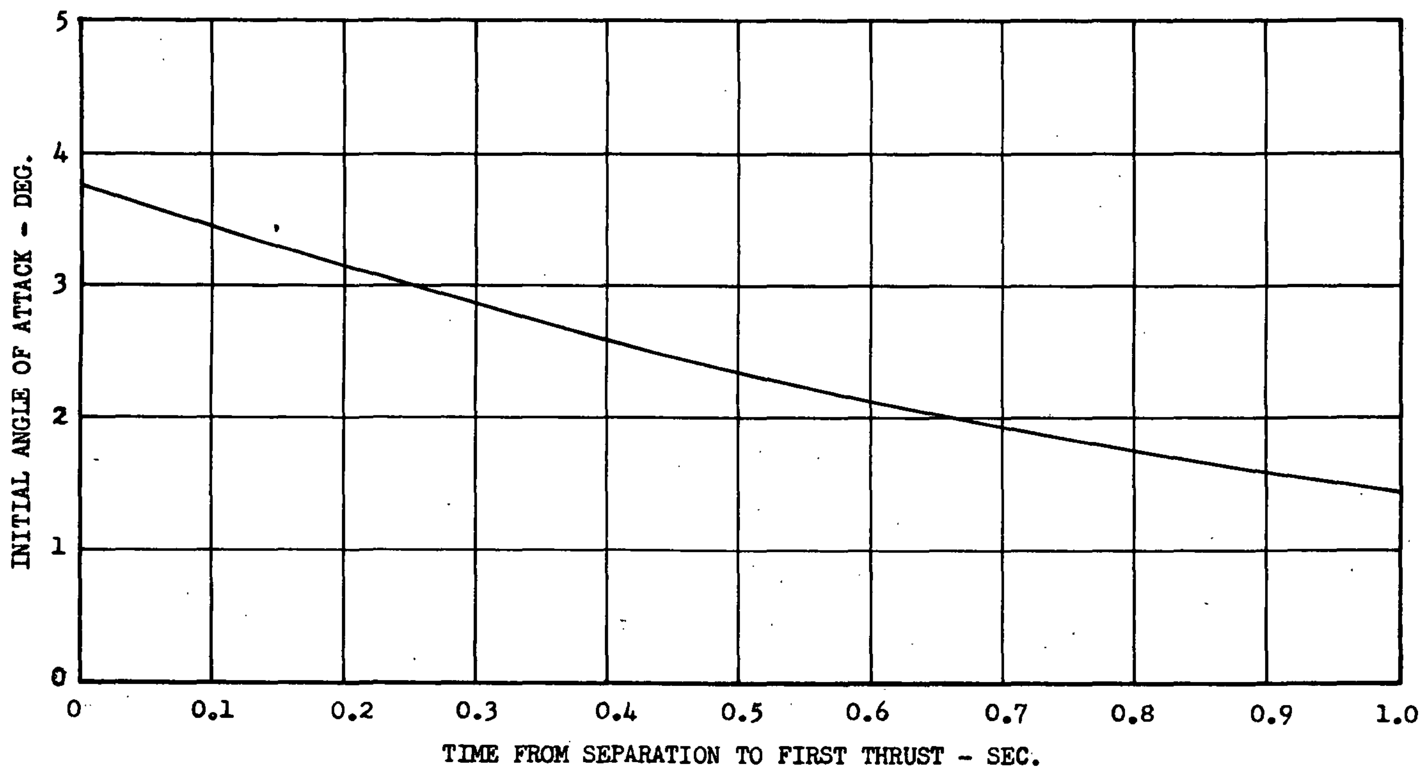


Figure 12

B. Vehicle Stage Design

1. First Stage

a. Clustered Booster - The clustered booster used for the preliminary design studies on the SATURN C-2 vehicle is similar to the present C-1 booster except that it has been shortened approximately 65-in. to reduce the consumable propellant loading to 650,000 lb. Extensive structural redesign to strengthen the forward adapter piece and outer tank attachment structure is necessary to provide for the 260-in. diameter second stage and the loads imposed by the SATURN C-2 configuration. Detailed analysis of these requirements was not conducted. A cutaway isometric sketch of the booster is shown in Fig. 13. A scale drawing of the overall vehicle is shown in Fig. 14 giving gimbal and separation stations.

b. Single Tank Booster - Basic design of the single tank booster was studied earlier, based on eight engines at 250K thrust each, and the results were given in Ref. 6 . This design study was reviewed and modified for the eight H-1 engines. The single tank booster is designed for a consumable propellant capacity of 650,000 lb. The engines are arranged in a circle with a mounting diameter slightly less than 260-in. All eight engines are gimballed and used for control. Material used is stainless steel. This is a conservative design approach; aluminum would give a slightly lighter structure weight except in the tail area where heating is present.

The principle advantages of this design are:

(a) Production simplicity and subsequent reduction in cost
(b) Lower weight which increases vehicle performance by 5%
(c) Compact rigid structure which increases the vehicle first mode bending frequency to 1.6 cycles per second, obviating the need for complex phasing networks and bending mode accelerometers in the guidance system, thus increasing the reliability
(d) More rugged structure for booster recovery re-entry and water impact

A comparative weight summary on the single tank booster and present booster is given in Table VI. Figure 15 shows the major characteristics of the booster. Figure 16 shows the overall vehicle giving gimbal and separation stations.

2. Second Stage

The proposed design for the SATURN C-2 second stage is shown in Fig. 17. The engine shown is not the J-2 Rocketdyne engine but a hypothetical engine derived from the data available at the beginning of the SATURN C-2 study. The consumable propellant capacity for the stage is 330,000 lb. Maximum tank pressures for this stage are 46 psia (tank bottom) in the oxygen tank and 25 psia in the hydrogen tank. Table VII presents a detailed weight breakdown of the stage.

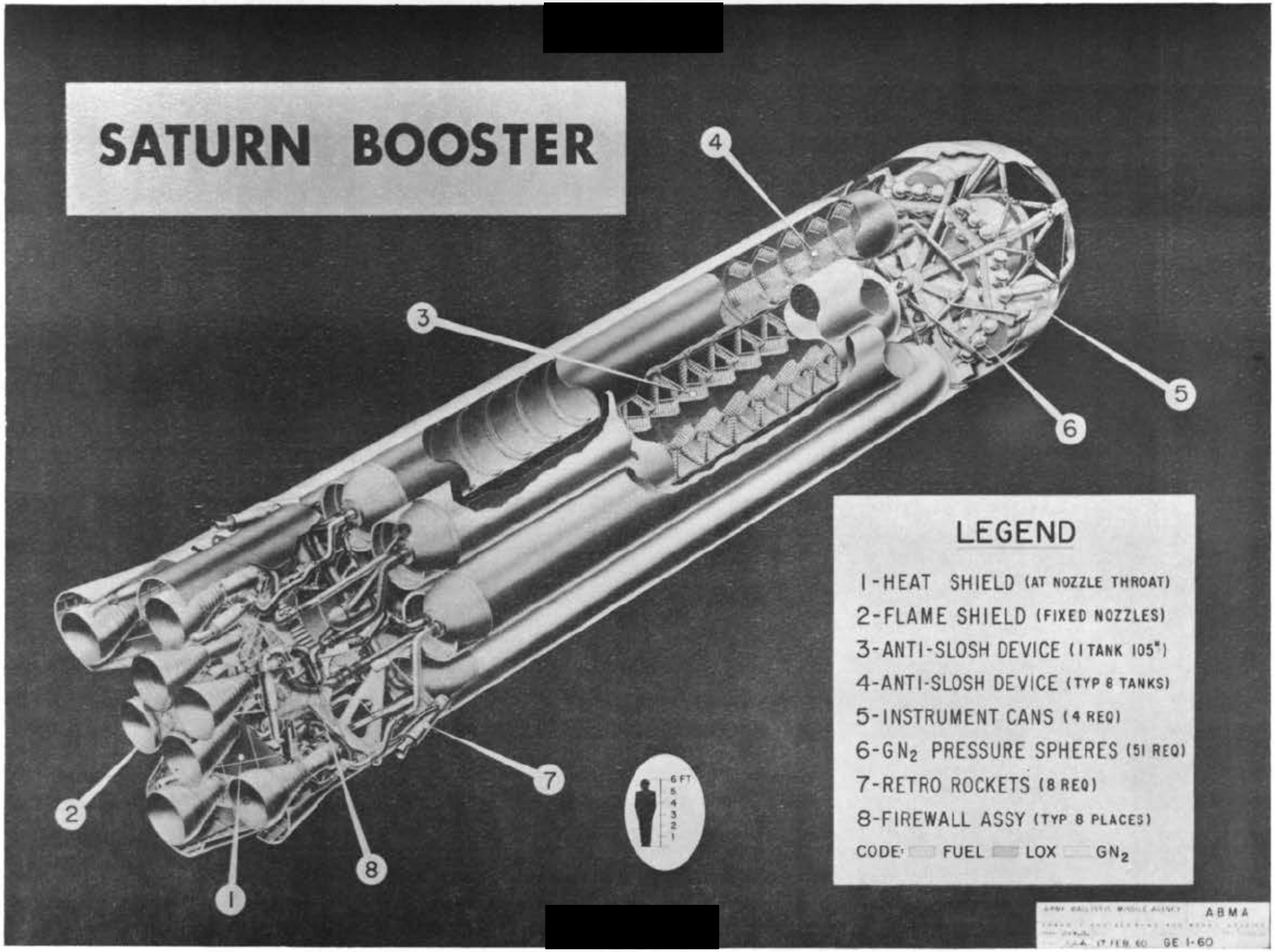
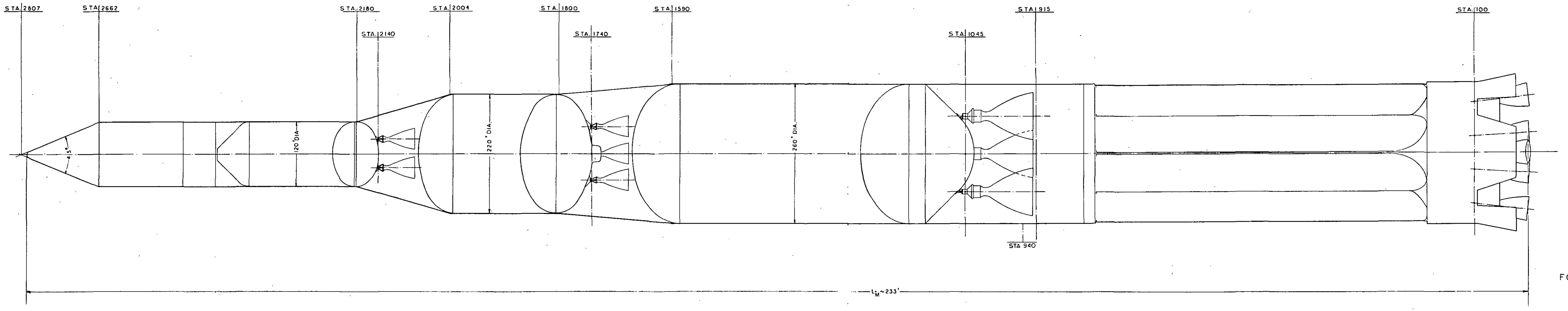

[redacted]
SATURN BOOSTER
LEGEND
1-HEAT SHIELD (AT NOZZLE THROAT)
2-FLAME SHIELD (FIXED NOZZLES)
3-ANTI-SLOSH DEVICE (1 TANK 105")
4-ANTI-SLOSH DEVICE (TYP 8 TANKS)
5-INSTRUMENT CANS (4 REQ)
6-GN_2 PRESSURE SPHERES (51 REQ)
7-RETRO ROCKETS (8 REQ)
8-FIREWALL ASSY (TYP 8 PLACES)
CODE: FUEL LOX GN_2
1
2
3
4
5
6
7
8
6 FT
5
4
3
2
1
ABMA
17 FEB 60
GE 1-60
[redacted]

Figure 13



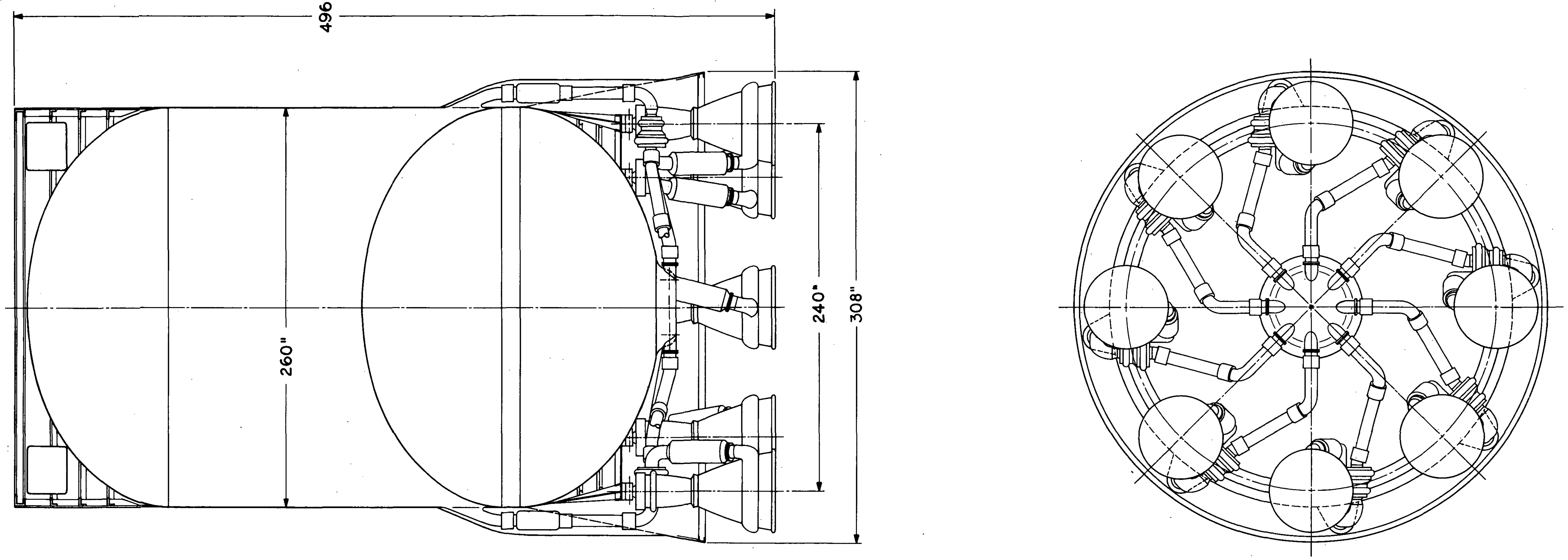
SATURN C-2
FOUR STAGE-VEHICLE
ESCAPE PAYLOAD
FIG. 14

TABLE VI

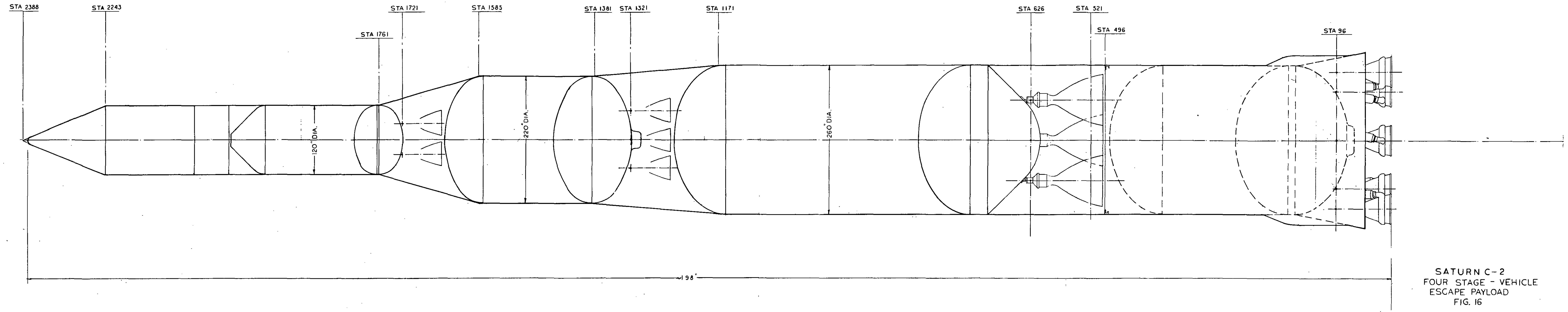
COMPARISON OF THE CLUSTER AND THE SINGLE TANK BOOSTER WEIGHTS

	Clustered Booster	Single Tank Booster
Engine	H-1	H-1
Propellant	LOX/RP-1	LOX/RP-1
Thrust, lb	8 x 188K	8 x 188K
I_{sp}, sec	257 s.l.	257 s.l.
Stage Diameter, in.	260	260
W_2, Guid. & Control, lb	2,500	1,000
W_3, Fuselage, lb	52,000	24,300
W_4, Propulsion, lb	22,000	18,700
W_5, Recovery Equip., lb	12,000	5,000
W_6, Trapped Prop., lb	15,000	11,000
W_7, Unusable Residuals, lb*	6,500	3,250
W_8, Prop. Consumption, lb*	650,000	650,000
W_s, Structure Wt., lb	88,500	49,000
W_n, Structure Net Wt., lb	110,000	63,250
W_a, Stage Wt., lb	760,000	713,250

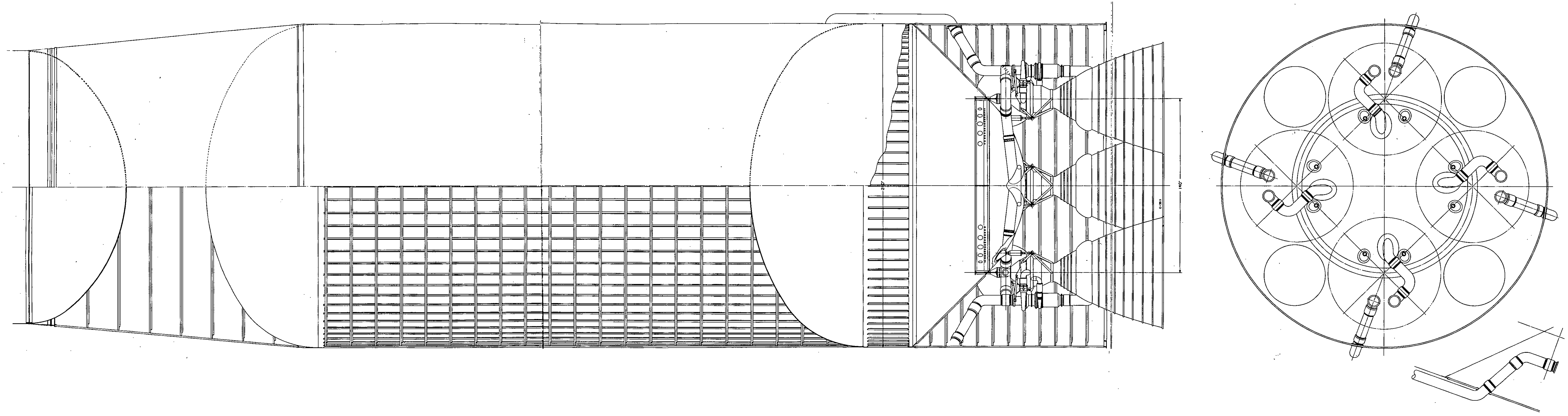
* These values are for maximum capacity. For most missions the propellant consumption will be 600,000 lbs.



SATURN C-2
SINGLE TANK BOOSTER
FIG. 15



SATURN C-2
FOUR STAGE - VEHICLE
ESCAPE PAYLOAD
FIG. 16



SATURN C-2
SECOND STAGE
INBOARD PROFILE
FIG. 17

TABLE VII

SATURN C-2 SECOND STAGE WEIGHT BREAKDOWN

W_2, Guidance and Control			500
W_3, Fuselage and Equipment			14,770
W_{31}, Propellant Container		9,465	
Upper hydrogen bulkhead	600		
Adapter ring	750		
Hydrogen cylinder skin	2,640		
Internal stiffners	545		
Center bulkhead 2 @ 300	600		
Adapter ring	100		
Oxygen cylinder skin	375		
Internal stiffners	45		
Ring frames in hydrogen cylinder	525		
Lower bulkhead	1,155		
Adapter ring	1,100		
Miscellaneous	1,030		
W_{32}, Structural Frame		4,405	
Thrust structure	650		
Lower bulkhead stiffners	695		
Forward transition corrugation	1,090		
Ring frames	275		
End ring	160		
Aft transition corrugation	650		
Ring frames	225		
End ring	160		
Miscellaneous	500		
W_{37}, Measuring Equipment		500	
W_{39}, Miscellaneous		400	
W_4, Propulsion System and Accessories			9,630
W_{41}, Engines 4 @ 1787.5		7,150	
$W_{42,43}$, Propellant Container Equipment		2,070	
Hydrogen suction lines and pre-valves	680		
Oxygen suction lines and pre-valves	200		

TABLE VII (CONTD)

Hydrogen vent lines and valves	110		
Oxygen vent lines and valves	110		
Hydrogen fill lines and valves	60		
Oxygen fill lines and valves	60		
Pneumatic control system	110		
Helium bottles and attachments	180		
Helium system plumbing	120		
Oxygen heat exchangers	300		
Helium controls	30		
Propellant utilization system	110		
W_{44}, Thrust Vector Control Equipment		410	
W_6, Unusable Propellants and Gas Residuals			3,290
W_{61}, Trapped Oxidizer		880	
Tanks	100		
Engines	300		
Lines	480		
W_{62}, Trapped Fuel		230	
Tanks	150		
Engines	40		
Lines	40		
W_{63}, Gas Residuals in Oxidizer Tank		1,780	
W_{64}, Gas Residuals in Fuel Tank		360	
W_{68}, Other Residuals		40	
W_7, Usable Propellant Residuals			1,650
W_8, Propellant Consumption			330,000
W_I, Insulation (jettisonable)			1,620
W_s, Dry Structure Weight			24,900
W_n, Effective Net Structure Weight			29,840
W_a, Stage Weight (without insulation)			359,840
W_a, Stage Weight (with insulation)			361,460

a. Structural Design - The aft transition piece which remains with the stage is a cylinder made of corrugated 7075 T-6 aluminum. The corrugation is optimized to achieve maximum allowable compression strength in all structural components of the cylinder. Ring frames are used internally to control the elastic buckling of the corrugated elements. An aluminum end ring on the rear of the transition piece provides the mating surface for separation and the connection devices necessary to effect separation. The upper end of the transition piece is riveted to the lower adapter ring which is made from 301 stainless steel. This ring provides sufficient area and stiffness to overcome stress concentrations which occur at the junction of the aft bulkhead, the lox tank cylindrical section, and aft transition piece.

The lower lox bulkhead is made up of three major parts:

1. A frustum of a cone
2. A spherical cap
3. A thrust beam structure

The spherical cap experiences only the internal pressure loads. The thrust beam structure is made up of an external I-beam and an internal cylinder. The depth and section modulus of the I-beam and internal cylinder is sufficient to distribute the four point loads of the engines into loads which are nearly uniformly distributed loads at the bulkhead. The extended flange of the I-beam acts as a doubler for the butt weld joint of the spherical cap and conical section of the bulkhead. The web of the internal cylinder is stiffened locally by shear ties at the engine mount points. The conical part of the bulkhead is stiffened externally by hat sections to prevent local buckling. Spot welding is used to attach the hat sections to the bulkhead. The complete bulkhead is fabricated from 301 stainless steel.

The cylindrical section of the lox tank is fabricated from 301 stainless steel and uses a butt weld joint to provide a pressure seal and spot welded doublers to provide strength across the weld. Hat sections are spot welded to the skin in the longitudinal direction to provide sufficient rigidity for ground handling and pad loading without internal pressure. An adapter ring is used to make the junction of the lox cylinder, the hydrogen cylinder, and the intermediate bulkhead and to provide the stiffness required to overcome stress concentrations.

The intermediate bulkhead is elliptical in shape and has double walls with fiberglas matting between the walls to provide the necessary insulation between the lox and hydrogen propellants. One of the walls of the bulkhead carries the pressure load while the other is used for sealing of the insulation.

Construction of the cylindrical section of the hydrogen tank is the same as described for the lox cylinder. Internal ring frames prevent elastic instability. An adapter ring is used at the junction of the hydrogen cylinder, the upper hydrogen bulkhead, and the forward transition piece to take care of stress concentrations and welding efficiencies. The shape of the upper bulkhead is the same as the intermediate bulkhead. The forward transition section is a frustum of a cone constructed from corrugations. Diameter at the lower end is 260 in. and at the upper end 220 in. Internal ring frames control elastic instability of the corrugated elements. Material for the transition section is 7075 T-6 aluminum. An end ring provides the mating surface for connection to upper stages or payload. Slosh and vortex problems were not investigated in detail, but an estimated weight for anti-vortex devices and slosh baffles was added to the stage structural weight.

As proposed, the stage structure is optimized for handling and reliability, and it is about 1250 lb heavier than the lightest structure found, which was a pure pressure stabilized shell with supercooling of the propellants while in the standby and hold condition on the pad. The proposed stage design is only 500 lb heavier than the pressure stabilized shell when supercooling of the propellants is not used.

b. Propulsion System - The second stage of the C-2 vehicle will use four 200K lox-hydrogen engines. At the initiation of this study, the engine specifications were in the hands of the potential engine developers. Since the contractor for the engine development, as well as a number of design details of the engine, were unresolved, it was concluded that, for the purposes of preliminary design, a hypothetical engine layout would be used until the specific details and manufacturer of the engine were known.

Studies were made from which it was determined that an expansion ratio of 35 to 1 would be used for this stage. It was this expansion ratio, when used with a chamber pressure of about 600 psia, that determined the original ABMA specification of a maximum exit diameter of 90 in. for the proposed engine. The influence of engine packaging within the limits of a 260-in. interstage adapter section also dictated that the engine exit diameter be kept within reasonable limits. The length of the bell nozzle was kept as short as consistency with the current state of the art would allow in order to minimize the weight penalty of the relatively heavy adapter section.

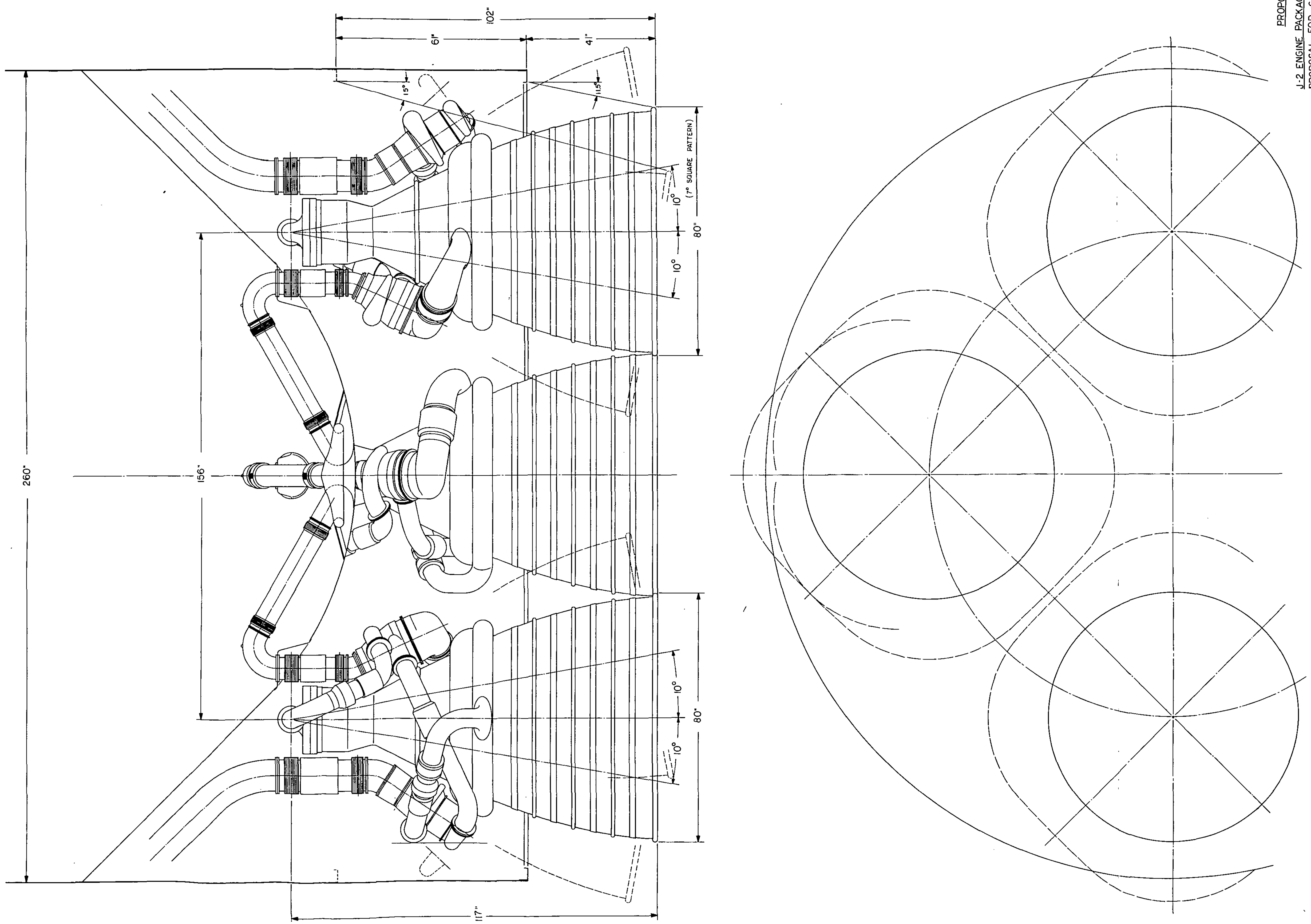
The engine gimbal blocks are attached near the lower bulkhead on a circular I-beam and internal thrust structure, resulting in a compact arrangement with a minimum interstage adapter length. This was considered an important design criterion for this vehicle since the high bending moments imposed by the flight loads result in a relatively heavy interstage structure.

The engine mounting diameter selected for this stage is 140 in., with about 9 in. between the adjacent 90-in. nozzle exits. With this relatively small clearance, it must be assumed that all engines will gimbal in unison during pitch, yaw, and roll control commands in order to avoid interference between the adjacent nozzles. A failure in the gimbal actuating system of any engine could result in interference between the adjacent engines. An interbleed between the hydraulic systems of the four engines might be considered here in order to insure that gimballing of all engines in unison can still occur even in the event that one engine fails and is shut down. The individual engine actuators are attached to locally reinforced positions on the spherical portion of the lower bulkhead. The upper pivot point of each actuator is located in the gimbal plane of the engine and can allow full angular movement of the actuator geometry during engine gimballing.

The turbopump arrangement shown on the stage layout (Fig.17) is a preliminary design speculation for the reasons stated earlier. Figure 18 shows a layout based on discussions with Rocketdyne, the selected engine contractor; our suggested repackaging of the turbopumps is shown mounted in a piggy-back fashion in a low position relative to the thrust chamber. This was done to keep the engine installation compact and to reduce interstage length.

The suction line arrangement for this stage is unique in certain respects owing to the compact placement of the engine gimbal block near the lower bulkhead surface. This placement results in a very low lox level relative to the engine and turbopump at the end of stage burning. In order to accomplish a nearly complete depletion of the lox within the spherical bottom of the tank, an internal plumbing arrangement is used in this area consisting of an inverted cone to which are attached the four suction lines. The function of the cone is to provide a common inverted sump for the four suction lines and to insure that the level of lox within the tank can be very low at time of depletion, thus minimizing residuals. The lox level within the tank may be dropped to the lower lip of the cone which is only about 2 in. from the lower bulkhead in this design. The propellant enters the inverted sump through the cylindrical annular area between the cone and the bulkhead surface. This area is slightly greater than the total cross sectional area of the four suction lines, thus permitting an unrestricted flow into the conical sump. In addition, the edge of the cone has a 3-in. rounded lip to lessen entry restrictions. The cone is supported by a center column which may be geometrically shaped to further reduce flow restrictions into the suction lines. To reduce the effects of vortexing at the sump entrance, a screen extending from the conical surface to the bulkhead is used. This conceptual design of internal suction line and sump arrangement is proposed in view of the weight advantages over an externally mounted bulkhead sump and plumbing arrangement for this particular case. These expected weight savings would be manifested by the shorter pipe lengths involved and by the lighter design of the internal cone as compared to an external sump. The lox lines of the internal system shown here pass through the bulkhead at a locally reinforced area giving a rigid reference at this point, with respect to the missile, for attaching the flexible lox suction lines associated with the engine and turbopump package. The flexible bellows in these suction lines are displaced by 90 degrees and are located in the gimbal plane to permit complete gimbal capability of the engine. The hydrogen suction lines have one gimbal bellows in the gimbal plane along a radial axis between the

PROPOSAL A
J-2 ENGINE PACKAGING & INSTALLATION
PROPOSAL FOR SATURN S-II STAGE



SUGGESTED J-2 ENGINE
INSTALLATION
FIG 18

bellows and the engine gimbal point. Perpendicular to this axis, gimballing of the suction line is permitted by the articulated motion afforded by the lower bellows and two additional bellows in the upper portion of the hydrogen suction lines.

Boost pumps were not considered for this stage since it was anticipated that the engine developers could meet the low NPSH values specified by NASA and determined by stage requirements.

A preliminary pressurization study was made for this stage to determine those pressurization values affecting structural design and the overall pressurization system design (Appendix A). It was assumed for this stage that pressurization of the hydrogen tank would be provided by a gaseous hydrogen bleed from the engine system during stage operation. For the requirements of lox tank pressurization, a heated helium system is proposed which, in view of specific studies on other cases (Appendix B), is relatively light.

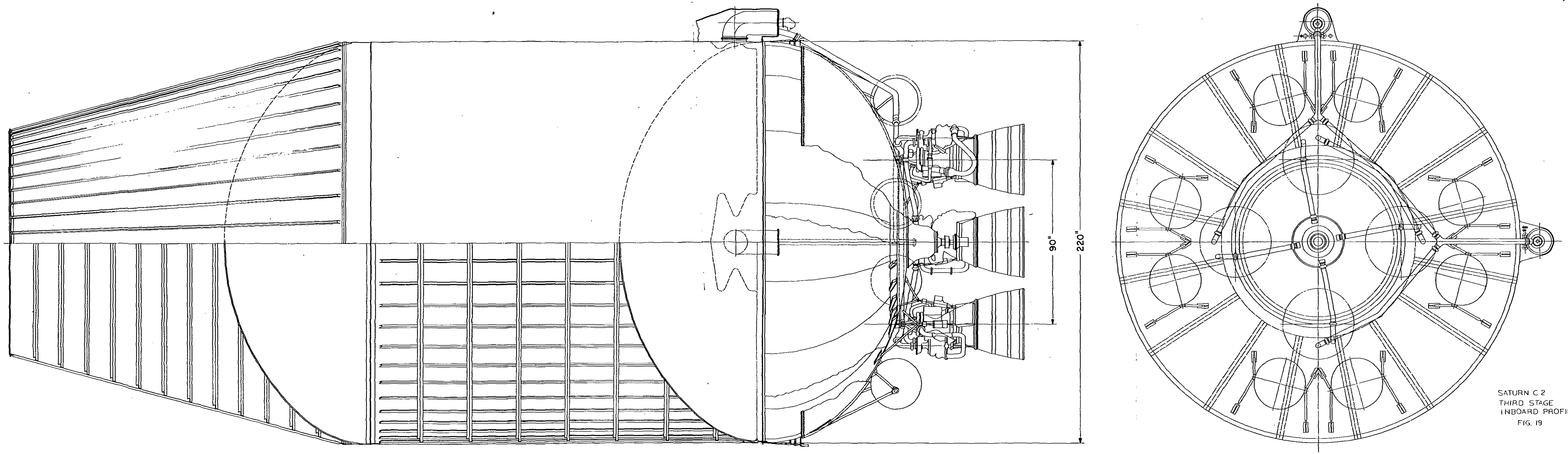
The pressurization sphere location and sizes shown on the layout for this stage are based on an early study which would have used unheated helium as the lox tank pressurant. However, in view of the weight advantages to be gained by using a heated helium system for lox tank pressurization, the unheated helium system is not considered. The number and location of the pressurant spheres to be used in the heated helium system will be determined in a later design and layout study for this stage.

3. Third Stage

The proposed third stage used for the preliminary studies of the SATURN C-2 vehicle is shown in Fig. 19 and a detailed weight breakdown in Table VIII. Propulsion is furnished by four LR-115 engines (20,000 lb vácuum thrust each). Consumable propellant capacity is 100,000 lb. The propellant tanks are designed for maximum pressures of 23 psia in the hydrogen tank and 30 psia in the oxygen tank. During the course of the SATURN C-2 studies a contract was awarded to Douglas Aircraft Co. for the design and fabrication of this stage. For completeness, the Douglas proposed stage is covered in item 5 of this section. This will be the second stage of the C-1 vehicle and the third stage of the C-2 vehicle.

a. Structural Design - The lox tank is made from two elliptical bulkheads joined by an adapter ring. The adapter ring serves also as the connector of the hydrogen cylinder and the mating surface for the aft separation plane. The lower lox bulkhead contains the thrust structure which is made up of an external I-beam with internal stiffeners and doublers to distribute the thrust loads into the bulkhead. The intermediate bulkhead is similar to the bulkhead described for the second stage.

The cylindrical portion of the hydrogen tank uses the same type of construction as the second stage with internal hat sections



SATURN C 2
THIRD STAGE
INBOARD PROFILE
FIG. 19

SATURN B-1 THIRD STAGE ENGINE LAYOUT PROPOSAL
USING UPRATED P&W RL-10 ENGINES

TABLE VIII

SATURN C-2 THIRD STAGE WEIGHT BREAKDOWN

W_2, Guidance and Control				500
W_3, Fuselage and Equipment				4,990
W_{31}, Propellant Container			3,090	
Upper hydrogen bulkhead		250		
Insulation		70		
Hydrogen cylinder skin		770		
Hydrogen cylinder corrugation		590		
Ring frames		140		
Center bulkhead 2 @ 180		360		
Insulation		180		
Lower bulkhead		530		
Forward transition adapter		200		
W_{32}, Structural Frame			1,210	
Thrust structure		300		
Doublers	100			
Channel stiffeners	40			
Thrust beam	160			
Forward transition piece		910		
Skin	300			
Hat sections	250			
Ring frames	180			
End rings	180			
W_{37}, Measuring Equipment			500	
W_{39}, Miscellaneous			190	
W_4, Propulsion System and Accessories				3,120
W_{41}, Engines 4 @ 320			1,280	
$W_{42,43}$, Propellant Container Equipment			1,300	
Boost pumps		340		
Suction lines, valves, etc.		160		
Helium bottles and attachments		550		
Helium system plumbing		100		
Helium controls		30		

TABLE VIII (CONTD)

Fill and drain system	20		
Propellant utilization system	60		
Start system	40		
W_{44}, Thrust Vector Control Equipment		160	
Hydraulic system and actuators	120		
Lube system	40		
W_{47}, Stage Attitude Control System Equipment		380	
Hydrogen peroxide bottles	100		
Thrust chambers	180		
Plumbing	100		
W_6, Unusable Propellants and Gas Residuals			1,040
W_{61}, Trapped Oxidizer		90	
Tanks	60		
Lines	30		
W_{62}, Trapped Fuel		110	
Tanks	100		
Lines	10		
W_{63}, Gas Residuals in Oxidizer Tank		530	
W_{64}, Gas Residuals in Fuel Tank		170	
W_{66}, Monopropellant Residuals		140	
W_7, Usable Propellant Residuals			500
W_8, Propellant Consumption			100,340
$W_{81,82}$, Oxygen-Hydrogen		100,000	
W_{85}, Hydrogen Peroxide		340	
W_I, Insulation (jettisonable)			600
W_{sep}, Separation Propellant (hydrogen peroxide)			130
W_s, Dry Structure Weight			8,610
W_n, Effective Net Structure Weight			10,150
W_a, Stage Weight at Ignition			110,490
W_a, Stage Weight with Insulation and Separation Propellant			111,220

and ring frames. An adapter ring joins the hydrogen cylinder, the upper hydrogen bulkhead, and the forward transition section. The forward transition section is a frustum of a cone with external hat sections and internal ring frames to prevent local buckling and elastic instability. An end ring provides the necessary mating surface for separation devices and attach points for upper stages or payload.

Material for the entire stage, including the transition section, is 301 stainless steel. Rather high aerodynamic heating is expected in the transition due to the steep angle; however, the predicted temperature allows the use of the stainless steel at a reasonable strength level.

b. Propulsion System - The third, or S-IV stage, will use four Pratt and Whitney engines which will be identical to the engines used in the S-V or CENTAUR stage. The target thrust level for this standard engine will be 20,000 lb*with a minimum guaranteed specific impulse of 420 sec and a nominal mixture ratio of 5 to 1. This engine, designated as RL 10B-3, will use a standard expansion ratio of 40 to 1 as in the S-V or CENTAUR, and also will use a regenerative cycle and cool-down sequence which will be discussed later.

The tank diameter for the third stage will be 220 in. with the gimbal blocks of the four engines bolted directly to a circular I-beam and internally stiffened hat section thrust structure. The thrust structure is integrally designed into the lower bulkhead which is a 45 degree semi-ellipsoied. The mounting of the engine gimbal blocks close to the lower bulkhead by this arrangement was chosen to minimize the interstage structural length in order to avoid a high weight penalty attributable to this relatively heavy structure. An engine mounting diameter of 90 in. was selected. An increase in this mounting diameter would shorten the adapter section, however, the actual shortening of the structure would be relatively small with respect to any moderate increase in the mounting diameter since the elliptical surface is relatively flat in this area. With the selected mounting diameter, the angular relationship between bulkhead surface and engine axis is approximately the same as for the CENTAUR or S-V stage, while still permitting a greater distance (64 in.) between any two adjacent engines than the 50 in. permitted in the CENTAUR configuration. In addition, the 90-in. mounting diameter results in a fairly close concentration of engines which is favorable from the control standpoint, and also facilitates separation by providing a generous clearance between engines and interstage structure at time of separation.

A pressurization study (Appendix B) was made for this stage to determine the most feasible system from the standpoint of overall stage weight, multiple re-start capability, and propellant utilization. In summary, the study revealed that, considering all of those factors which affect the weight of the overall pressurization system, the system using no boost pumps will yield the lightest basic pressurization system

* Thrust level has since been changed to 17.5 K.

for the simplified case wherein no engine re-start is required. However, when consideration is given to the additional weight in the propellant tanks due to the higher tank pressures required in the no boost pump system, the advantages of a boost pump system are then somewhat apparent from the standpoint of overall stage weight. Although these advantages are not significant enough to decisively justify the use of the boost pump system for the case where no re-start capability is required, the boost pumps were considered in this proposal based on the fact that, for certain missions of the C-2 vehicle, re-start capability will be necessary. When re-start capability is required, the necessity for a re-pressurization of the partially filled propellant tanks following a long coasting period makes that system attractive which requires less total tank pressure. The additional pressurant and pressurant container requirements for this re-pressurization phase will influence the total system weight and thus justify the use of a boost pump system which can permit a lower overall tank pressure. In view of these studies, and in consideration of cost and the reliability of using proven components, it is proposed that two of the CENTAUR type hydrogen boost pumps be used. Each pump will then supply the hydrogen requirements of two engines, as in the CENTAUR case. The two hydrogen boost pumps are located at a 90-degree displacement on the missile periphery in a radial alignment with two of the four engines. This permits an arrangement wherein one boost pump feeds two diametrically opposed engines through geometrically symmetrical suction lines. Any fluctuations in engine thrust level attributable to output differences between the two boost pumps will not then adversely affect missile control since diametrically opposed engines will feel the same fluctuations. Suction line displacement during gimballing is achieved by using three flexible joints in each line for both lox and fuel plumbing arrangements.

The pressurization gas requirements of the third stage were based on the assumption that hydrogen gas bleed from the engine system would be available for hydrogen tank pressurization. For oxidizer tank pressurization a comparison study was made between a heated helium system and a gox system (Appendix B). Based on the results of this study, the heated helium system is proposed for lox tank pressurization in conjunction with an unheated helium system for pressurization of the peroxide bottles, if used, and for the pneumatic requirements of the engine and vent valving systems. Ground pressurization of the ullage volumes in both the hydrogen and the lox tanks will be provided by refrigerated helium from a ground source prior to lift-off.

The pressurization sphere location and sizes shown on the layout for this stage are based on an early study which would have used unheated helium as the lox tank pressurant. However, in view of the weight advantages to be gained by using a heated helium system for lox tank pressurization, the unheated helium system is not considered. The number and location of the pressurization spheres to be used in the heated helium system would be subject to a further design and layout analysis for this stage.

Since these studies were made, Pratt and Whitney has proposed to develop a geared inducer for the hydrogen pump which will reduce the engine NPSH requirement from 8 psia to 0.6 psia. This geared inducer, which is essentially a boost pump, would be preferred for this third stage proposal, if now available, since it would then be possible to eliminate the separate boost pump system as proposed in this report and still have the low NPSH values that would permit the low hydrogen tank pressures desirable for those instances requiring multiple re-start capability.

Various methods of separation are currently being studied for this stage to determine the most feasible approach. The results of these studies will also determine the maximum gimbal angle requirements for the engines.

The exact requirements for attitude control are unknown at the present time. It was assumed, for the purposes of the pressurization study (Appendix B), that a hydrogen-peroxide system would be used for this purpose. If this is the case, the experience and some of the hardware used for the CENTAUR attitude control system might be utilized for this stage.

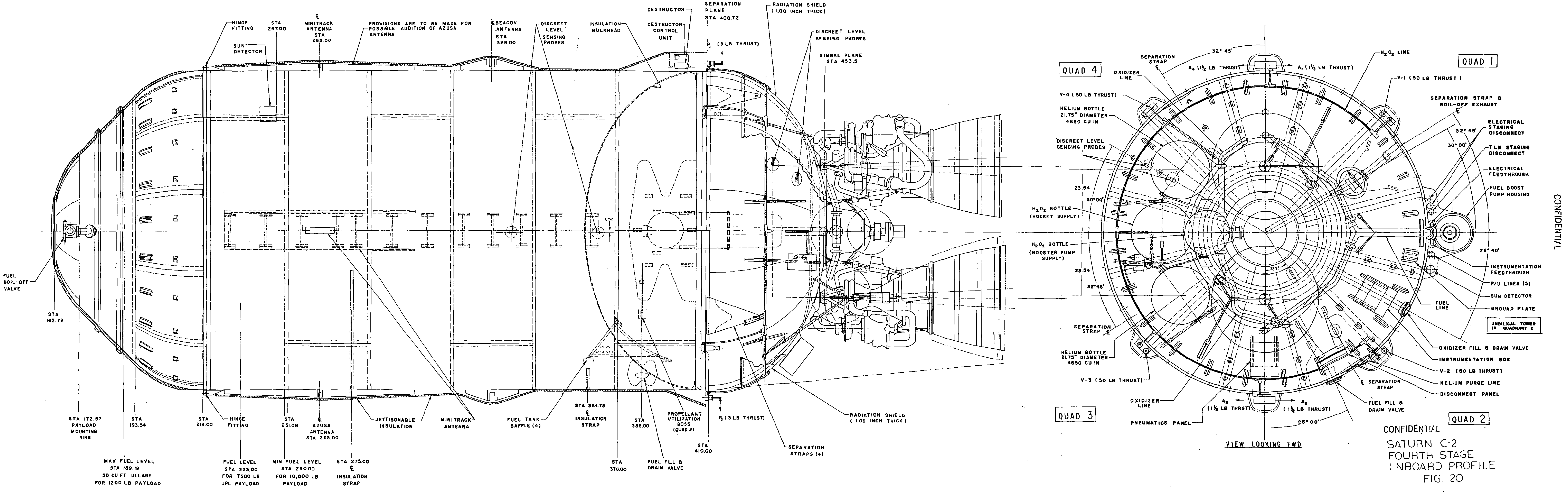
4. Fourth Stage

The fourth stage of the SATURN C-2 vehicle is quite similar to the CENTAUR stage being built by Convair for the ATLAS-CENAAUR program except for skin gages, etc. which are strengthened to provide the necessary stiffness and load-carrying ability required for use on the C-2 vehicle. Figure 20 shows the CENTAUR stage and Table IX presents a detailed weight breakdown. The consumable propellant capacity is 29,000 lb. Propellant tank structure is of the pure pressure stabilized shell type and is designed for maximum tank pressures of 59 psia in the hydrogen tank and 66 psia in the oxygen tank.

Propulsion System - From the propulsion standpoint this stage will consist of the same standard Pratt and Whitney RL 10B-3 engine used in the S-IV stage. The thrust level is 20,000 lb, with a minimum guaranteed specific impulse of 420 sec and a nominal mixture ratio of 5 to 1.

The tank diameter for this stage is 120 in. with the engines mounted diametrically opposed on a mounting diameter of 50 in. The gimbal blocks of the engines are attached to a circular I-beam and internal can thrust structure on a 45-degree semi-ellipsoidal bulkhead.

The engine expansion ratio is fixed at 40 to 1 for all applications, and both chambers are gimballed for a $\pm$ 2-degree travel in a square vectoring pattern. The turbopump is composed of a direct driven two stage, back-to-back centrifugal fuel pump with axial inducer. The fuel pump NPSH requirement is 8 psia or about 265 ft of head. The



CONFIDENTIAL

SATURN C-2
FOURTH STAGE
INBOARD PROFILE
FIG. 20

[redacted]

TABLE IX

SATURN C-2 FOURTH STAGE WEIGHT BREAKDOWN

W_3, Fuselage and Equipment			1,500
W_{31}, Propellant Container		1,141	
Upper hydrogen bulkhead	90		
Hydrogen cylinder skin	534		
Center bulkhead	67		
Lower bulkhead	185		
Frames	105		
Insulation (fixed)	160		
W_{32}, Structural Frame		123	
W_{35}, Cable Duct		13	
W_{36}, Measuring Equipment		130	
Gyro	20		
Programmer	50		
Servo amplifier	20		
Tracking unit	20		
Harness	20		
W_{37}, General Network		70	
Electrical harness	10		
Go-no-go- checkout	25		
Guidance harness	30		
Electrical system mounts	5		
W_{38}, Connection Elements		11	
Separation equipment	4		
Tank-mounted equipment mounts	7		
W_{39}, Miscellaneous		12	
W_4, Propulsion System and Accessories			1,200
W_{41}, Engines		540	
$W_{42,43}$, Propellant Container Equipment		420	
Boost pump system	140		
Boost pump system mounts	10		
Fill and drain	10		
Propellant stabilization	15		

TABLE IX (CONTD)

	Helium storage bottle	110		
	Propellant utilization system	50		
	Propellant loading system	20		
	Hydrogen tank system	10		
	Hydrogen system mounts	30		
	Oxygen tank system	20		
	Oxygen system mounts	5		
W_{41},	Thrust Vector Control Equipment		60	
	Hydraulic system	50		
	Hydraulic system mounts	10		
W_{45},	Control Equipment		30	
	Pneumatics system mounts	5		
	Pressure system controls	25		
W_{47},	Stage Attitude Control System		110	
	Attitude control system	75		
	Attitude control system mounts	35		
W_{49},	Miscellaneous		40	
W_6,	Unusable Propellants and Gas Residuals			440
W_{61},	Trapped Oxidizer		29	
	Tanks	15		
	Lines	14		
W_{62},	Trapped Fuel		63	
	Tanks	60		
	Lines	3		
$W_{63,64}$,	Gas Residuals		310	
	Oxygen	183		
	Hydrogen	121		
	Helium	6		
W_{66},	Monopropellant Residual		22	
W_{68},	Helium System Residual		16	
W_7,	Usable Propellant Residuals			150
W_8,	Propellant Consumption			29,000
W_I,	Insulation (jettisonable)			300

TABLE IX (CONTD)

W_s, Dry Structure Weight	2,700
W_n, Effective Net Structure Weight	3,290
W_a, Stage Weight (without insulation)	32,290
W_a, Stage Weight (with insulation)	32,590

The oxidizer pump NPSH requirement is 15 psia or about 30 feet of head. The pumps are driven by a two stage impulse turbine.

The RL 10B-3 engine uses the regenerative or "boot strap" cycle wherein the pumped fuel, after cooling the thrust chamber, is expanded through the turbine which drives the propellant pumps. The fuel is then injected into the combustion chamber. The pumped oxidizer is supplied directly to the propellant injector.

The S-V propulsion stage utilizes boost pumps for both fuel and oxidizer. The basic reason for use of boost pumps by Convair was to achieve a low overall stage weight by virtue of the low tank weight and pressurization requirements made possible by the low NPSH values of the boost pump system. These reasons were especially justified, as in the S-IV stage, by the requirement for multiple re-start capability in certain applications of the C-2 vehicle. This stage will be modified to use the geared inducer.

The RL 10B-3 engine used in this stage requires a 20 sec cooldown period prior to start. This is accomplished by allowing the propellants to flow through the propellant supply system, thus lowering the temperature of these parts to operating conditions. During this pre-start phase, the liquid oxygen flows through the oxidizer pump, the mixture ratio adjustment valve, all oxidizer plumbing, the propellant injector, and overboard through the thrust chamber. The fuel flows through the fuel pump and is directed overboard by the pump cooldown valve.

The CENTAUR features an attitude control system for orientation of the stage during coasting periods. This attitude control is achieved with six laterally-directed, fixed thrust, hydrogen-peroxide engines located 180 degrees apart on the tank periphery. These engines produce thrust in the order of 1.5 to 3 lb, with a specific impulse of about 150 sec at altitude.

Four rearward-facing hydrogen-peroxide engines, with a fixed thrust of 50 lb each, are used for separation from the previous stage, final velocity adjustment, and for attitude control and propellant bottoming during re-start after a coasting period.

5. Douglas Third Stage Design (S-IV)

Figure 21 presents the second stage for the SATURN C-1 as proposed by Douglas Aircraft Co. This stage will be used as the third stage of SATURN C-2. As mentioned earlier several decisions were made in a preliminary design meeting with Douglas. These decisions were:

1. To incorporate a geared inducer on the hydrogen side of the turbopump which reduces the NPSH requirement to about 0.4 psi.

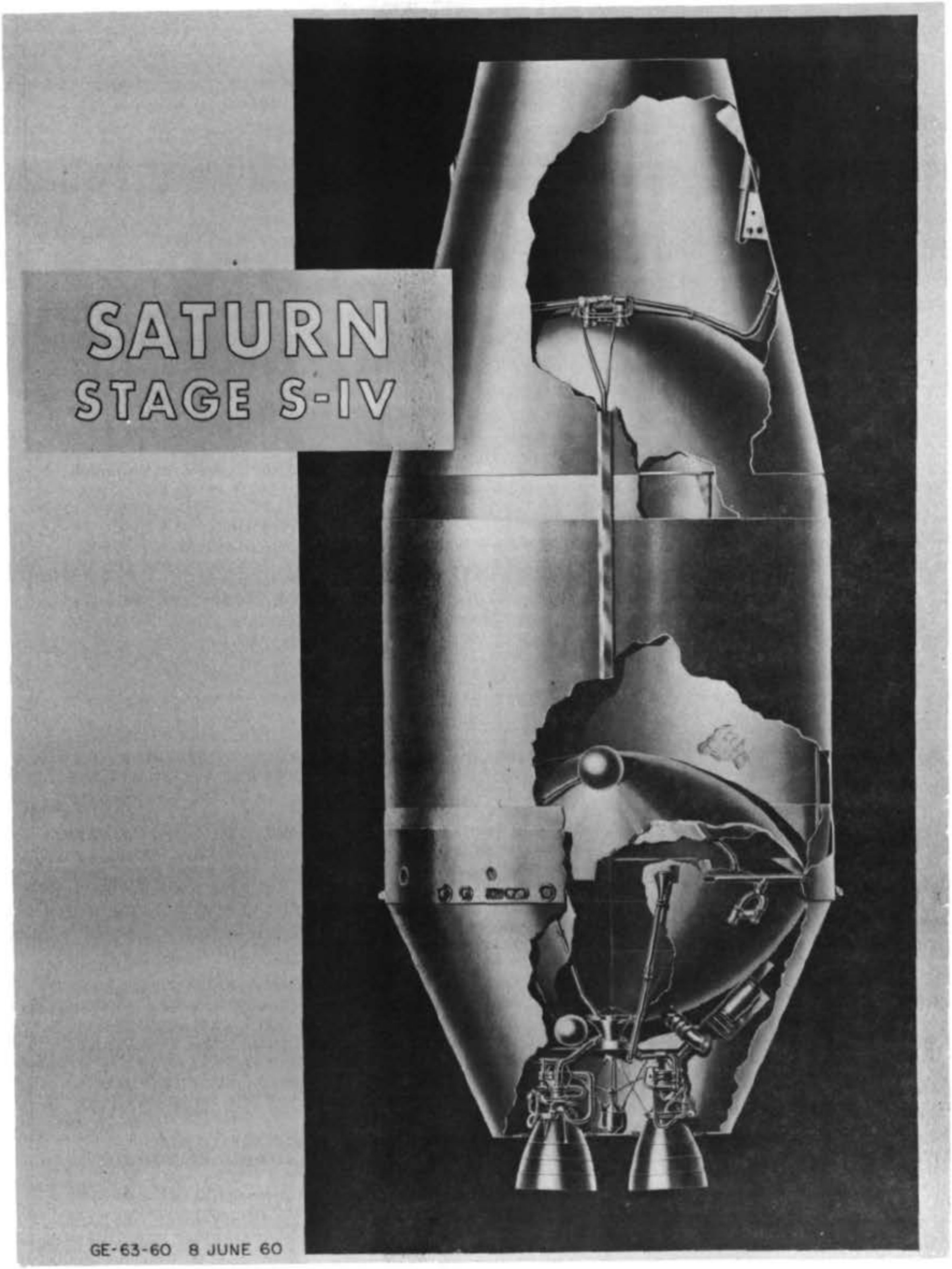


Figure 21

2. To design the intermediate bulkhead so that it will withstand the weight and pressure in the loaded hydrogen tank with no pressure in the lox tank.

In a joint NASA Headquarters - MSFC meeting. is was decided to lower the engine thrust to 17,500 lb primarily in order to reduce program costs.

Structural Design - The propellant tanks are made up of a 220-in. diameter cylinder constructed of 2014 T-6 aluminum with hemispherical ends having a 110-in. radius. Waffling of the cylindrical skin is accomplished by machine milling and the two hemispherical bulkheads are chemically etched. An intermediate bulkhead is used to separate the hydrogen from the oxygen and provide insulation between them. The intermediate bulkhead is attached to the lower bulkhead by a compression ring. Tentative construction of the intermediate bulkhead calls for a honeycomb design with a perforated phenolic resin core purged with helium and evacuated to provide the necessary insulation properties between the two propellants. This design allows either face of the bulkhead to leak without mixing the two propellants.

The engine mounting structure is a conical skin and stringer frustum tangentially attached to the aft bulkhead. The forward transition is a truncated cone of aluminum honeycomb attached tangentially to the forward bulkhead and adapter ring.

The engines are mounted at a 4-degree cant angle to minimize the effects of starting and shutdown transients. The engines are gimballed in a 3-degree square pattern. Gimbal actuation is provided by four independent hydraulic systems. In the absence of hydraulic power the engine is retained in a centered position.

Table X presents a detailed weight breakdown of the initially proposed Douglas stage modified to reflect the decisions stated previously.

In Appendix E there are six of the more likely C-2 configurations. The Douglas stage has been incorporated into these configurations. Summary weights, center of gravity, and moment of inertia during booster flight are also included.

TABLE X

DOUGLAS STAGE (S-IV) WEIGHT BREAKDOWN

W_2, Guidance and Control			500
W_3, Fuselage and Equipment			6,080
W_{31}, Propellant Container		3,190	
Forward hydrogen bulkhead	530		
Common bulkhead	600		
Aft bulkhead	780		
Container wall	860		
Insulation	250		
Sumps	20		
Antislosh	120		
Antivortex	30		
W_{32}, Structural Frame		1,980	
Basic thrust frame	350		
Forward skirt	260		
Aft skirt	520		
Interstage structure S-IV to S-V	850		
W_{33}, Tail Section		170	
Flame shield	140		
Flame shield supports	10		
Local insulation	20		
W_{34}, Structural Attachments		70	
Tunnels	35		
Conduits	35		
W_{35}, Control Elements		50	
Environmental control	15		
Malfunction computer	20		
Miscellaneous	15		
W_{36}, Stage network		180	
W_{37}, Measuring equipment		330	
W_{38}, Connection elements		30	
W_{39}, Miscellaneous		80	

TABLE X (CONTD)

W_4, Propulsion System and Accessories			2,500
W_{41}, Engines and Accessories		1,290	
Engines	1,200		
Pneumatic supply bottle	30		
Pneumatic supply lines and valves	30		
P. U. valves and controls	20		
Pneumatic system mounting	10		
W_{42}, Fuel Container Equipment		200	
Suction lines and valves	80		
Fill lines and valves	10		
Vent lines and valves	60		
Pressurization lines and valves	30		
Miscellaneous	20		
W_{43}, Oxidizer Container Equipment		300	
Suction lines and valves	60		
Fill lines and valves	30		
Vent lines and valves	60		
Gas generator	20		
Heat exchanger	10		
Helium bottle	50		
Miscellaneous	70		
W_{44}, Thrust Vector Control Equipment		260	
W_{47}, Stage Attitude Control System		100	
Attitude control rockets	30		
Propellant bottles	40		
Plumbing	30		
W_{48}, Separation Rockets		350	
Ullage rockets	120		
Propellant bottles	70		
Plumbing	50		
Brake rockets	110		

TABLE X (CONTD)

W_6, Unusable Propellants and Gas Residuals			260
W_{61}, Oxidizer trapped in engines		100	
W_{62}, Fuel trapped in engines		10	
W_{63}, Gas residual in oxidizer tank		90	
W_{64}, Gas residual in fuel tank		35	
W_{66}, H_2O_2 residual		20	
W_{68}, Helium residual		5	
W_7, Usable Propellant Residuals			1,350
W_{71-74}, Mixture ratio shift		500	
W_{79}, Attitude control and restart propellant		850	
Fine attitude control	350		
Coarse attitude	300		
Restart	200		
W_8, Propellant Consumption			100,070
W_{81}, Oxidizer		83,333	
W_{82}, Fuel		16,667	
W_{85}, Gas Generator Propellants		70	
W_9, Other Items of Intrest			410
W_{91-92}, Propellant Consumed at Thrust Buildup		190	
W_{93}, Chill-down Propellant		120	
W_{97}, H_2O_2 Consumed During Stage Separation		100	
W_s, Dry Structure Weight			9,080
W_n, Effective Net Structure Weight			10,690
W_a, Stage Weight at Ignition			110,950
W_a, Stage Weight With Chill-down and Separation Propellants			111,170

VI. A PARAMETRIC STUDY OF EARTH ORBIT-LAUNCHED VEHICLES BASED UPON A SATURN CLASS VEHICLE

The SATURN vehicle is quite capable for the manned lunar circumnavigation missions. The next order of missions is manned lunar landings, manned planetary circumnavigation, and manned planetary landings. The manned lunar soft landing and return to earth requires a booster of 9 to 12 million lb thrust, depending on whether liquid hydrogen or kerosene is used for second stage fuel. This booster vehicle could perform Mars or Venus circumnavigation. Manned planetary landing will require connection of multiple stage units in an earth orbit, or refueling of large vehicles in orbit, if accomplished with chemical vehicles. Direct flights for Mars manned landings could be accomplished only with boosters of about 12 million lb thrust with nuclear rockets in all upper stages. The nuclear second stage would require a thrust of about 4 million lb.

Orbital connection of stages and refueling of vehicles in orbit are extremely attractive since the mission capability would no longer be limited by the size of the vehicle making the ascent into orbit. Essentially, a SATURN size vehicle could accumulate the required power units to perform just about any mission in our solar system. The development of all the required techniques will not be an easy problem and the system may be expensive relative to larger vehicles, but the capability is essentially limited only by production rates and available launch and transport facilities. This parametric study indicates the concepts of orbit-launched vehicles and compares the orbit-launched vehicle to an all chemical NOVA class vehicle of 12 million lb thrust. Figure 22 shows the basic concepts. The loaded and unloaded volumes at different phases of the mission are indicated by the legend. Essentially, there are two methods for obtaining a orbital-launch vehicle in orbit. In Fig. 22 (A) shows a scheme using the full payload of the ascent vehicle to supply fully loaded rocket units that are connected to form the vehicle structure; (B) shows a system where the full payload consists of the dry structure of the departing vehicle which is loaded from tanks brought up by successive flights. C_1 and C_2 show schemes where assembly and refueling are combined.

Specific vehicle proposals based on SATURN C-2 and applying orbital techniques are discussed in the following paragraphs.

In Fig. 23 vehicle I and II are proposals for a 10,000-lb re-entry payload, which is considered suffieient for the initial lunar exploration operations. With Vehicle I the orbit is reached in three stages. The third stage of the orbit vehicle is used again to act as the departing stage of the space vehicle. Approximately five refill missions are necessary in order to fill the space vehicle. Departing and breaking stages are two individual stages. For the return vehicle storable propellants have been chosen which can be fueled on the ground in the cases, where just 10,000-lb of re-entry payloads are requested.

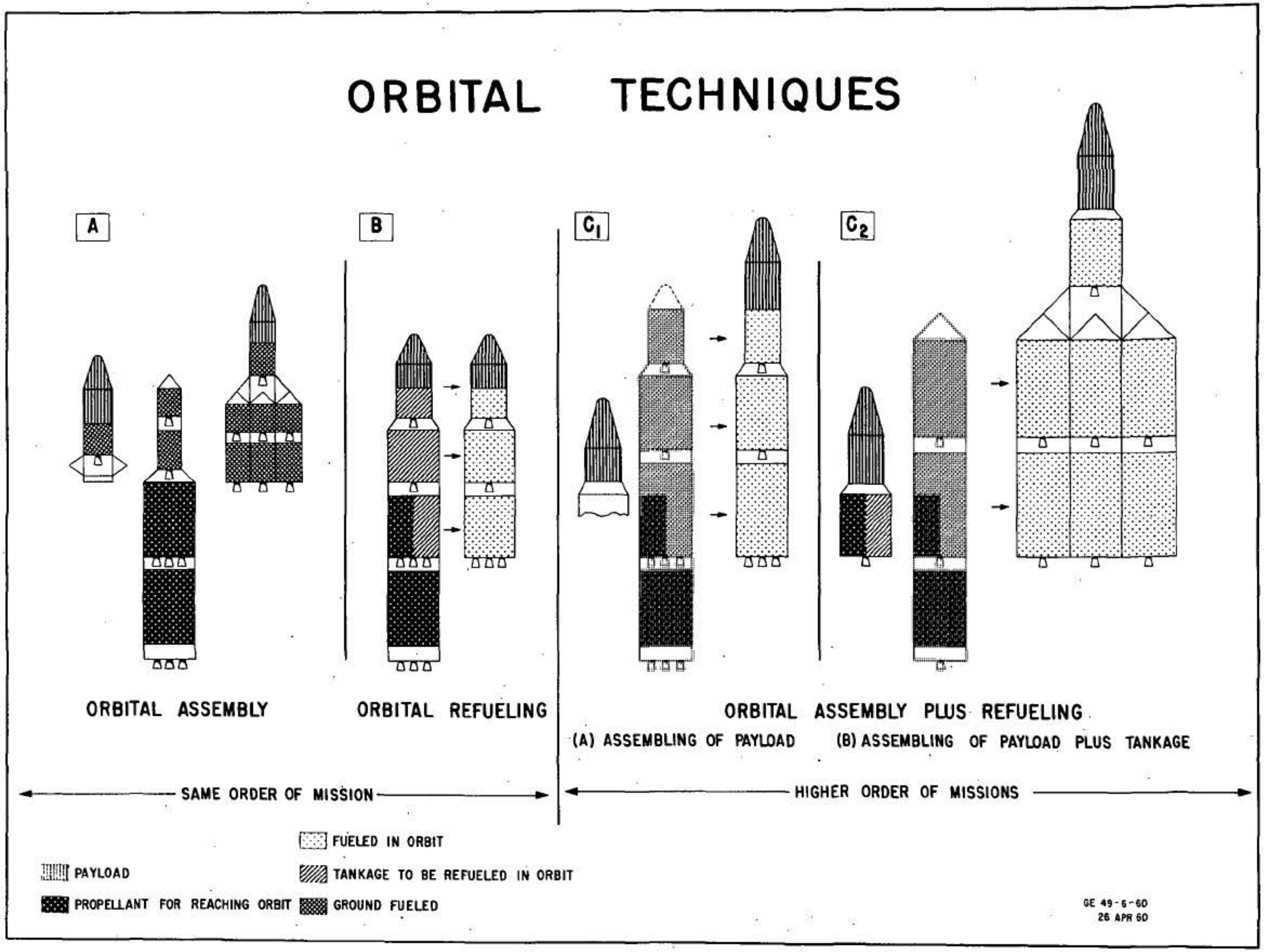

ORBITAL TECHNIQUES
A
B
C1
C2
ORBITAL ASSEMBLY
ORBITAL REFUELING
ORBITAL ASSEMBLY PLUS REFUELING
(A) ASSEMBLING OF PAYLOAD
(B) ASSEMBLING OF PAYLOAD PLUS TANKAGE
SAME ORDER OF MISSION
HIGHER ORDER OF MISSIONS
PAYLOAD
PROPELLANT FOR REACHING ORBIT
FUELED IN ORBIT
TANKAGE TO BE REFUELED IN ORBIT
GROUND FUELED
GE 49-6-60
26 APR 60

Figure 22

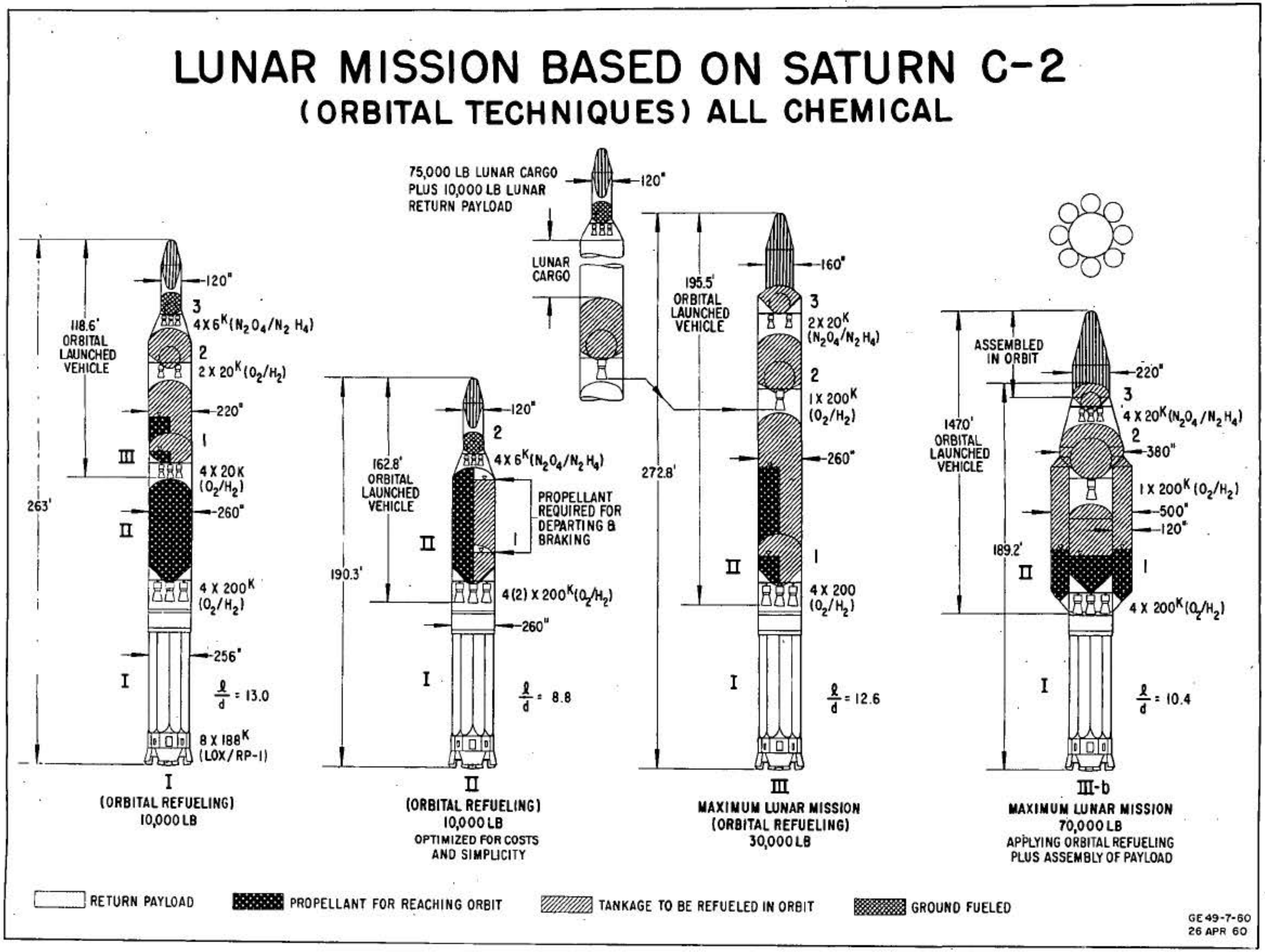

LUNAR MISSION BASED ON SATURN C-2
(ORBITAL TECHNIQUES) ALL CHEMICAL
75,000 LB LUNAR CARGO
PLUS 10,000 LB LUNAR
RETURN PAYLOAD
120"
LUNAR
CARGO
118.6'
ORBITAL
LAUNCHED
VEHICLE
263'
120"
3
4 X 6K (N2O4/N2H4)
2
2 X 20K (O2/H2)
220"
1
III
4 X 20K
(O2/H2)
260"
II
4 X 200K
(O2/H2)
256"
I
l/d = 13.0
8 X 188K
(LOX/RP-1)
I
(ORBITAL REFUELING)
10,000 LB
162.8'
ORBITAL
LAUNCHED
VEHICLE
190.3'
120"
2
4 X 6K (N2O4/N2H4)
PROPELLANT
REQUIRED FOR
DEPARTING & BRAKING
1
II
4(2) X 200K (O2/H2)
260"
I
l/d = 8.8
II
(ORBITAL REFUELING)
10,000 LB
OPTIMIZED FOR COSTS
AND SIMPLICITY
195.5'
ORBITAL
LAUNCHED
VEHICLE
272.8'
160"
3
2 X 20K
(N2O4/N2H4)
2
1 X 200K
(O2/H2)
260"
1
II
4 X 200
(O2/H2)
I
l/d = 12.6
III
MAXIMUM LUNAR MISSION
(ORBITAL REFUELING)
30,000 LB
ASSEMBLED
IN ORBIT
147.0'
ORBITAL
LAUNCHED
VEHICLE
189.2'
220"
3
4 X 20K (N2O4/N2H4)
2
380"
1 X 200K (O2/H2)
500"
120"
1
II
4 X 200K (O2/H2)
I
l/d = 10.4
III-b
MAXIMUM LUNAR MISSION
70,000 LB
APPLYING ORBITAL REFUELING
PLUS ASSEMBLY OF PAYLOAD
RETURN PAYLOAD
PROPELLANT FOR REACHING ORBIT
TANKAGE TO BE REFUELED IN ORBIT
GROUND FUELED
GE 49-7-60
26 APR 60

Figure 23

Vehicle II is an alternate proposal for the same mission, where simplicity and overall vehicle configuration is the main concern. Here the orbit is reached in two stages in order to utilize the second stage of the orbit vehicle for the space vehicle. Because of the larger volume requirements for the second stage of the orbit vehicle, this stage can be used for the departing and breaking maneuvers of the space vehicle. The return vehicle is the same as before. Approximately 6.5 refill missions will be required in this case to perform the 10,000-lb lunar mission; however, the overall vehicle configuration is simpler. Optimization studies, previously made, actually showed that for a payload of this magnitude the orbit should be reached in three stages if an efficient vehicle is the goal. The acceleration characteristic for the departing and breaking period is also considerably higher compared to Vehicle I, since the propulsion system of the second stage of the orbit vehicle is used for the same mission. Disconnection of two engines in orbit could be accomplished resulting in some increase in payload capability.

Vehicle III is the resulting vehicle for the maximum lunar return mission, based on SATURN C-2 applying the pure orbital refueling technique. In this case, the orbit should be reached in two stages. The fuel required for the space vehicle will have to be supplied in orbit. A return payload in the order of 30,000 lb can be achieved for the lunar landing and return mission. The slenderness of the vehicle reaches 12.6 based on 260-in. diameter. Fifteen tanker flights are required for this mission. In case a 30,000-lb return payload would not be required, a 75,000-lb cargo could be sent to the moon in addition to a 10,000-lb return payload. In this case, the propellant for the return vehicle again could be supplied on the earth.

Vehicle III b shows schematically the maximum lunar mission vehicle assuming that orbital assembly and refueling is allowed. A return paylaod of 70,000 lb resulted from this investigation; this also could be converted into 220,000 lb lunar cargo plus 10,000-lb return payload. A clustered configuration has been proposed for the second or departing stage, which lends itself to parallel tank staging within the departing period and results in performance increase. In order to perform this mission 33 additional flights are necessary if based on chemical propulsion systems and C-2 as the vehicle from earth to orbit.

The vehicles discussed above are represented by the circular areas in the payload capability chart Fig. 24 . In Table XI summary system characteristics are compiled for these vehicles together with comparison information on the nuclear vehicles discussed below. In order to get some feeling of the efficiency of the vehicles, the ratio of payload per number of booster flights required for the particular mission is plotted in the chart.

V c will move up increasing the area of payload capabilities for nuclear engines applied to SATURN C-2 with orbital techniques. In the

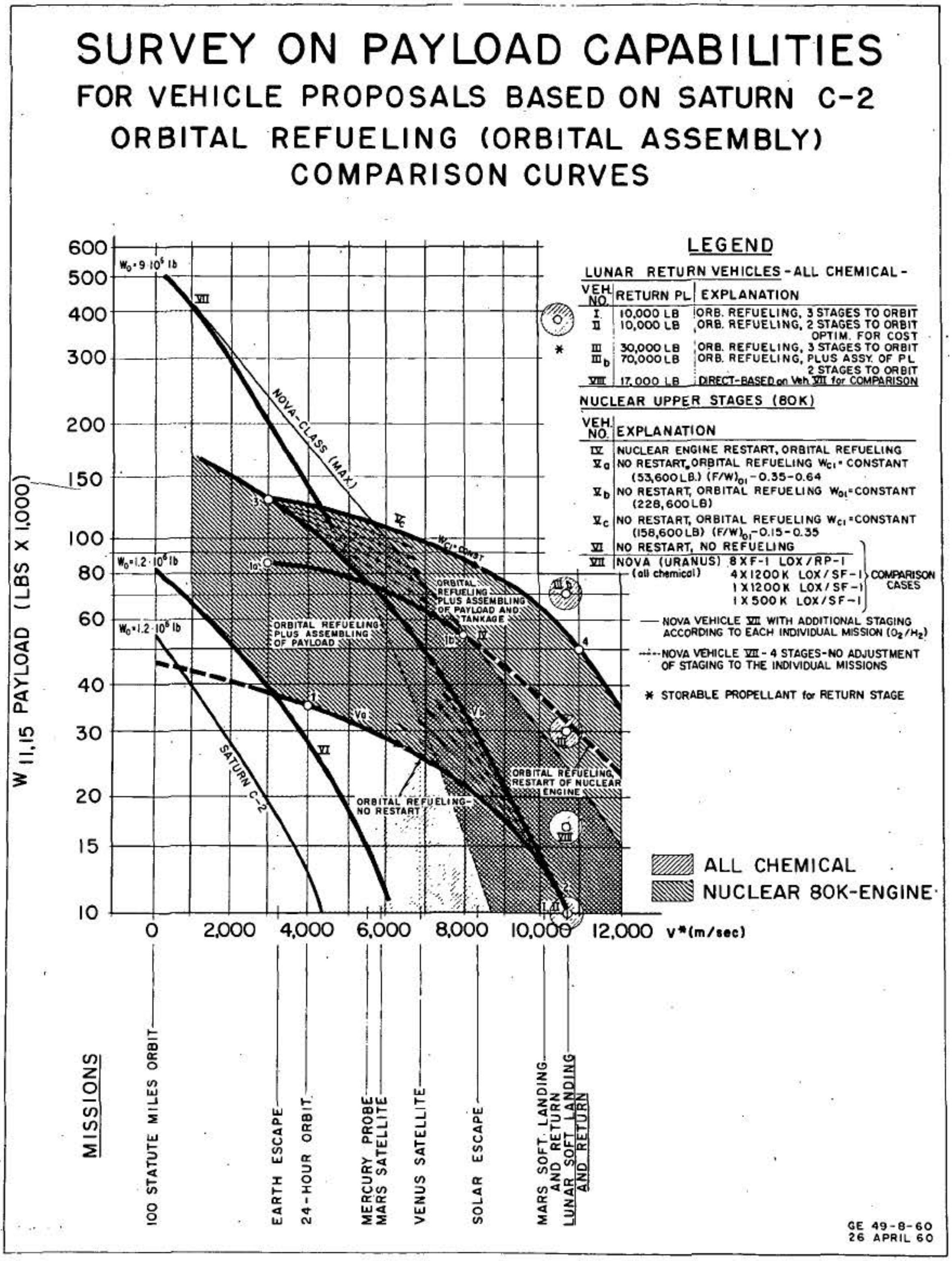

SURVEY ON PAYLOAD CAPABILITIES
FOR VEHICLE PROPOSALS BASED ON SATURN C-2
ORBITAL REFUELING (ORBITAL ASSEMBLY)
COMPARISON CURVES
LEGEND
LUNAR RETURN VEHICLES - ALL CHEMICAL -
VEH. NO. | RETURN PL | EXPLANATION
I | 10,000 LB | ORB. REFUELING, 3 STAGES TO ORBIT
II | 10,000 LB | ORB. REFUELING, 2 STAGES TO ORBIT OPTIM. FOR COST
III | 30,000 LB | ORB. REFUELING, 3 STAGES TO ORBIT
IIIb | 70,000 LB | ORB. REFUELING, PLUS ASSY. OF PL 2 STAGES TO ORBIT
VIII | 17,000 LB | DIRECT-BASED on Veh. VII for COMPARISON
NUCLEAR UPPER STAGES (80K)
VEH. NO. | EXPLANATION
IV | NUCLEAR ENGINE RESTART, ORBITAL REFUELING
Va | NO RESTART, ORBITAL REFUELING Wc1 = CONSTANT (53,600 LB.) (F/W)01 - 0.35-0.64
Vb | NO RESTART, ORBITAL REFUELING W01 = CONSTANT (228,600 LB)
Vc | NO RESTART, ORBITAL REFUELING Wc1 = CONSTANT (158,600 LB) (F/W)01 - 0.15-0.35
VI | NO RESTART, NO REFUELING
VII | NOVA (URANUS) 8XF-1 LOX/RP-1 (all chemical) 4X1200K LOX/SF-1 1X1200K LOX/SF-1 1X500K LOX/SF-1
COMPARISON CASES
NOVA VEHICLE VII WITH ADDITIONAL STAGING ACCORDING TO EACH INDIVIDUAL MISSION (O2/H2)
NOVA VEHICLE VII - 4 STAGES - NO ADJUSTMENT OF STAGING TO THE INDIVIDUAL MISSIONS
* STORABLE PROPELLANT for RETURN STAGE
W 11,15 PAYLOAD (LBS X 1,000)
600
500
400
300
200
150
100
80
60
40
30
20
15
10
W0 = 9·10^6 lb
W0 = 1.2·10^6 lb
W0 = 1.2·10^6 lb
NOVA-CLASS (MAX)
Wc1 = CONST
ORBITAL REFUELING PLUS ASSEMBLING OF PAYLOAD AND TANKAGE
ORBITAL REFUELING PLUS ASSEMBLING OF PAYLOAD
ORBITAL REFUELING NO RESTART
ORBITAL REFUELING, RESTART OF NUCLEAR ENGINE
SATURN C-2
ALL CHEMICAL
NUCLEAR 80K-ENGINE
0
2,000
4,000
6,000
8,000
10,000
12,000
v*(m/sec)
MISSIONS
100 STATUTE MILES ORBIT
EARTH ESCAPE
24-HOUR ORBIT
MERCURY PROBE
MARS SATELLITE
VENUS SATELLITE
SOLAR ESCAPE
MARS SOFT LANDING AND RETURN
LUNAR SOFT LANDING AND RETURN
GE 49-8-60
26 APRIL 60

Figure 24

COMPARISON DATA FOR VEHICLES BASED ON SATURN C-2
(ORBITAL TECHNIQUES)

A. LUNAR RETURN MISSION (all chemical)
Payload for 100 st. mi. orbit = 54,800 lb

VEHICLE	PAYLOAD (lb)	No. OF REFILL MISSIONS	PL/No. OF BOOSTER FLIGHTS (lb)	LIFT-OFF WEIGHT (lb) ORB. LAUNCHED Veh./a_0	SLENDERNESS	REMARKS:
I	10,000	5	1,660	358,140 / 0.22	12	
II	10,000	6.4	1,360	400,000 / 1.0	8.8	OPTIMIZED FOR SIMPLICITY
III	30,000 or 75,000 lun. cargo plus 10,000 return pl	15	1,880	880,000/0.9	12.6	
III b	70,000 or 220,000 lun. cargo plus 10,000 return pl	31.5 1.5 33.0 assembly	2,060	1.790,000/0.45	10.4	ASSEMBLY of PAYLOAD

4,500 - Comparison Figure for Lunar Mission by Nuclear Vehicle (IV)

B. Various Missions for Vehicles applying Nuclear 80k-engine and ORBITAL TECHNIQUES
Payload for 100 st. mi. orbit = 80,000 lb

VEHICLE	MISSIONS	No. OF REFILL MISSIONS	LIFT-OFF WEIGHT (lb) ORB. LAUNCHED Veh./a_0	SLENDERNESS OF VEHICLE CONFIG. EXCLUDING ASSEMBLED PARTS BASED ON 260-in.diam	REMARKS:	
IV	VARIOUS Ia - Ib	1.3-3.4 (chem) (5.8) [1.0-2.5 (nucl.)] (4.4)	172,900-285,000 0.46 - 0.28	11.4 - 14.5 (>14.5)	W_{c_e} = 110,000 - const.	RESTART
V a.	VARIOUS 1 - 2	1.3 - 3.4	126,200 - 228,600 0.64 - 0.35	11.4 - 14	W_{c_e} = 53,600 - const.	NO RESTART
V b.	VARIOUS 2 - 3	1.3 - 3.4	228,600 0.35	12. - 14	W_{c_e} = 158,600-53,600	NO RESTART ASSEMBLY of PL 0-1.5 missions
V c.	VARIOUS 3 - 4	3.4 - 8.5	228,600-523,600 0.35 - 0.15		W_{c_e} = 158,600 1.5 MISSIONS	NO RESTART {ASSEMBLY of PL plus " of tankage
VI	VARIOUS 1 - 2	0	—	12 - 12.2	COMPARISON CASE, NO ORBITAL TECHNIQUES	

GE 49-13-60 26 APR 60

Table XI

case of a non-restartable nuclear engine it seems that a slenderness of 14 will not likely be exceeded for the orbit vehicle since, for higher mission requirements, orbital assembly operations (payload and tankage) are necessary. However, there will still be problems with respect to controllability of such a configuration if the present SATURN booster should be used.

In Fig. 25 vehicle VI represents the comparison case to the orbital techniques showing a nuclear 80K engine third stage on the SATURN C-2. Payloads and tank volumes are shown for the respective missions.

In the direct approach, payload capabilities of SATURN C-2 can be extended by applying nuclear upper stages to the booster, and by a next generation type vehicle in the NOVA class; also nuclear upper stages can be applied here. Lift-off weights in the 10 million lb class can be foreseen for such vehicles and still the mission capabilities would be limited. Orbital techniques, however, permit performance of nearly unlimited missions based on any reliable booster vehicle in a reasonable thrust class. This thrust class should be investigated when more is known about the future missions.

In order to compare the two specific investigations - all chemical approach and vehicles applying nuclear engines including orbital refueling technique - the lunar landing and return mission is considered even though the all chemical proposals use storable propellants for the return stage, whereas the nuclear curves are derived for various missions where the nuclear stage is used for the total velocity requirement (no staging, but considerably higher specific impulse for the return stage).

The 30,000-lb lunar return mission can be performed with 15 orbital refill operations in the all chemical approach to be compared with 6 refill missions in the nuclear case, according to Vehicle IV. The tanker flights are based on an optimized SATURN C-2 vehicle. In the nuclear case, the tanker missions could also be performed by a SATURN vehicle with a nuclear last stage. This would cut down the number of refill missions to 4.

For the 10,000-lb lunar landing and return mission, 5 to 6.5 refill missions would be required, according to Vehicles I and II in the all chemical approach, compared with 3.4 refill missions in the nuclear case, which could be reduced to 2.5 if based on a nuclear tanker vehicle.

Approximately a 17,000-lb return payload could be achieved for the lunar landing and return mission with a maximum NOVA class vehicle in the direct approach and storable propellants for the return stage.

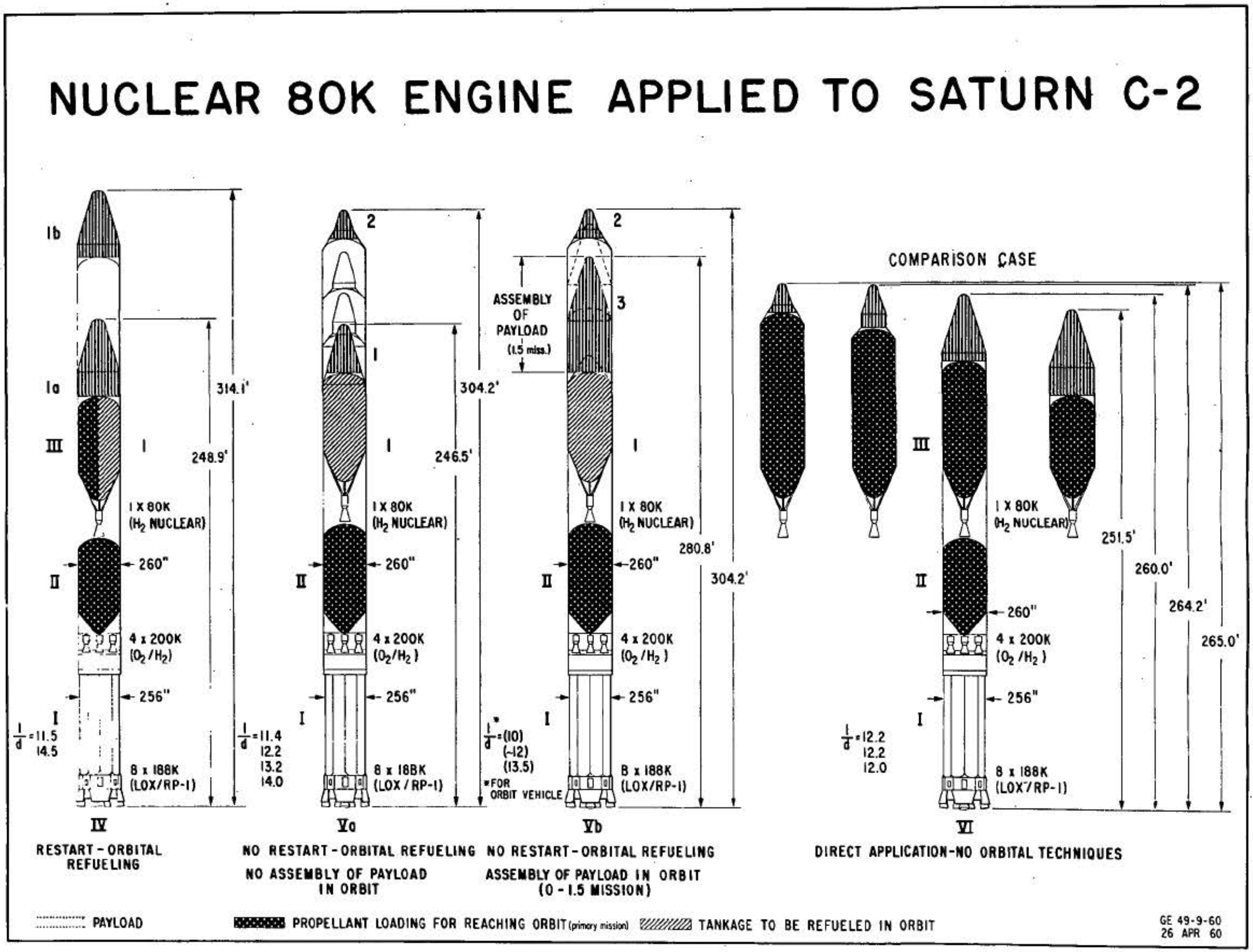

NUCLEAR 80K ENGINE APPLIED TO SATURN C-2
Ib
Ia
III
II
I
1 X 80K (H2 NUCLEAR)
260"
4 x 200K (O2/H2)
256"
8 x 188K (LOX/RP-1)
l/d = 11.5 14.5
314.1'
248.9'
IV
RESTART - ORBITAL REFUELING
2
304.2'
246.5'
l/d = 11.4 12.2 13.2 14.0
Va
NO RESTART - ORBITAL REFUELING
NO ASSEMBLY OF PAYLOAD IN ORBIT
ASSEMBLY OF PAYLOAD (1.5 miss.)
3
l*/d = (10) (~12) (13.5)
*FOR ORBIT VEHICLE
280.8'
304.2'
Vb
NO RESTART - ORBITAL REFUELING
ASSEMBLY OF PAYLOAD IN ORBIT (0 - 1.5 MISSION)
COMPARISON CASE
l/d = 12.2 12.2 12.0
251.5'
260.0'
264.2'
265.0'
VI
DIRECT APPLICATION-NO ORBITAL TECHNIQUES
PAYLOAD
PROPELLANT LOADING FOR REACHING ORBIT (primary mission)
TANKAGE TO BE REFUELED IN ORBIT
GE 49-9-60
26 APR 60

Figure 25

From this parametric study it can be concluded that:

1. Almost unlimited extension of mission capabilities of a SATURN class vehicle by orbital techniques and nuclear propulsion systems is possible.
2. The present SATURN booster would have to be redesigned to control most of the configurations indicated.
3. For the orbital refueled vehicles,ascent should be in two stages, with the second stage refueled as the initial orbit departing stage to insure configurations controllable during ascent.
4. The engines will require re-start capability and longer burning times (2,000 secs for the chemical propulsion systems and approximately 4,000 secs for the nuclear systems).
5. Nuclear propulsion systems are very attractive for space vehicle applications. Compared to the chemical systems a relatively low number of earth-to-orbit tanker flights per mission is required.
6. Further investigations on orbital refueling versus orbital assembly, concerning the techniques as such, should be undertaken; furthermore, cost investigations should be made for orbital refueling versus direct approach (NOVA class vehicle).
7. Parallel tank staging should be applied in nuclear vehicles for missions with high velocity requirements since significant performance increases can be achieved.

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APPENDIX A

SATURN C-2

PRELIMINARY DESIGN

Second Stage Pressurization Study

A. Propellant Tank Pressures

1. General Assumptions

P_v = Vapor pressure
= 15.5 psia (Lox)
= 17.0 psia (Hydrogen)

P_n = Net positive suction head requirement
= 12.5 psia (Lox)
= 4.0 psia (Hydrogen)

P_f = Friction losses and valve tolerance allowances
= 2.0 psia (Vent Valves)
= 2.0 psia (Vortex Screen)
= 3.0 psia (Line Losses)
7.0 psia (Total for Lox System)
= 2.0 psia (Vent Valves)
= 2.0 psia (Vortex Screen)
= 2.0 psia (Line Losses)
6.0 psia (Total for Hydrogen System)

a_{1c} = First stage cutoff acceleration = 2.83 g
a_{21} = Second stage lift-off acceleration = 1.634 g
a_{2c} = Second stage cutoff acceleration = 5.021 g
h_1 = Height of liquid column above turbopump inlet at lift-off
= 15.8 ft (Lox)
= 51.7 ft (Hydrogen)

h_c = Height of liquid column above turbopump inlet at cutoff
= -1.0 ft (Lox)
= 11.6 ft (Hydrogen)

ρ_o = Lox density = 71.2 lb/ft^3
ρ_h = Hydrogen density = 4.37 lb/ft^3

2. Hydrogen Tank

P_t = Minimum gas pressure in tank

At Stage 1 Cutoff:

P_t = 17 psia (For vapor suppression for 1°F temp. rise)

At Stage 2 Lift-off:

$$P_t = P_v + P_n + P_f - (a_{21})(h_1)(\rho_h)/144$$
$$= 17 + 4 + 6 - 2.6$$
$$= 24.4 \text{ psia}$$

At Stage 2 Cutoff:

$$P_t = 17 + 4 + 6 - 1.8$$
$$= 25.2 \text{ psia}$$

P_b = Total pressure on LH_2 side of common bulkhead

At Stage 1 Cutoff:

$$P_b = P_t + (a_{1c})\ (h_1 - h_c)\ (\rho h)\ /144$$
$$= 17 + 3.4$$
$$= 20.4 \text{ psia}$$

At Stage 2 Lift-off:

$$P_b = P_t + (a_{21})\ (h_1 - h_c)\ (\ h)\ /144$$
$$= 24.4 + 2$$
$$= 26.4 \text{ psia}$$

At Stage 2 Cutoff:

$$P_b = P_t$$
$$= 25.2 \text{ psia}$$

3. Oxygen Tank

P_t = Minimum gas pressure in lox tank. This pressure must at all times be at least 5 psi greater than the total pressure on the LH_2 side of the common bulkhead to maintain structural integrity of the bulkhead.

At Stage 1 Cutoff:

$$P_t = P_b(LH_2) + 5$$
$$= 20.4 + 5$$
$$= 25.4 \text{ psia}$$

Note: This value exceeds the minimum vapor pressure requirement of 15.5 psia also.

At Stage 2 Lift-off:

$$P_t = P_v + P_n + P_f - (a_{21})\ (h_1)\ (\rho_o)\ /144$$
$$= 15.5 + 12.5 + 7 - 12.8$$
$$= 22.2 \text{ psia}$$

Note: This pressure only satisfies the engine requirements. To satisfy the minimum structural requirement of a 5 psi bulkhead differential pressure, the pressure in the lox tank must be:

$$P_t = P_b\ (LH_2) + 5 = 26.4 + 5$$
$$= 31.4 \text{ psia}$$

At Stage 2 Cutoff:

$$P_t = P_v + P_n + P_f - (a_{2c})\ (h_c)\ (\rho_o)\ /144$$
$$= 15.5 + 12.5 + 7 + 2.5$$
$$= 37.5 \text{ psia}$$

Note: This pressure also exceeds the minimum bulkhead differential pressure

P_b = Total pressure on lower bulkhead

At Stage 1 Cutoff:

$$P_b = P_t + (a_{1c})\ (h_1 - h_c)\ (\rho_o)\ /144$$
$$= 25.4 + 23.5$$
$$= 48.9 \text{ psia}$$

At Stage 2 Lift-off:

$$P_b = P_t + (a_{21})\ (h_1 - h_c)\ (\rho_o)\ /144$$
$$= 31.4 + 13.6$$
$$= 45.0 \text{ psia}$$

At Stage 2 Cutoff:

$$P_b = P_t$$
$$= 37.5 \text{ psia}$$

4. Tank Pressure Summary

(a) Absolute Gas Pressure Within Tank, psia

	Lox	Hydrogen
At Stage 1 Cutoff	25.4	17.0
At Stage 2 Lift-off	31.4	24.4
At Stage 2 Cutoff	37.5	25.2

(b) Total Pressure At Bottom of Tank, psia

	Lox	Hydrogen
At Stage 1 Cutoff	48.9	20.4
At Stage 2 Lift-off	45.0	26.4
At Stage 2 Cutoff	37.5	25.2

(c) Pressure Differential Across Common Bulkhead, psi

At Stage 1 Cutoff	5.0
At Stage 2 Lift-off	5.0
At Stage 2 Cutoff	12.3

5. Tank Pressure Conclusions

a. Oxygen Tank

During stage operation, pressurization of the lox tank will be supplied by the pressurant in response to a pressurization sensing and programming system which will require a gas pressure above the lox of about 31.4 psia at stage lift-off and which will increase to about 37.5 psia at stage cutoff. Considering a tolerance on the order of $\pm$ 0.5 psia, a nominal value for these pressures, for the purpose of pressurant requirement studies, is set at 32 psia at lift-off and 38 psia at cutoff of the stage.

Since the lox tank pressure at cutoff of the first stage must be about 25.4 psia in order to provide a minimum value of 5 psi across the common bulkhead for structural reasons, it is assumed that the lox ullage volume will be ground-pressurized with sufficient refrigerated helium to provide a nominal lox tank pressure of 25 psia at first stage cutoff.

b. Hydrogen Tank

During stage operation, pressurization of the hydrogen tank will be supplied by gaseous hydrogen from the engines in response to a pressurization sensing and programming system which will require a gas pressure above the hydrogen of about 24.4 psia at stage lift-off and about 25.2 psia at stage cutoff. Considering a tolerance for the overall pressurization system, nominal value for these pressures, for the purposes of pressurant requirement studies, will be set at 26 psia.

The hydrogen tank ullage pressure, supplied by refrigerated helium during the filling operation, is assumed to be 1 psi above the atmospheric pressure during the filling operation. An expected liquid hydrogen temperature increase of about 1° F, during first stage flight, results in a vapor pressure of about 17 psia at the end of first stage burning. The gas pressure in the hydrogen tank is increased to the value required by the engine system (24.2 psia) by gaseous hydrogen bled from the engine system following stage ignition.

APPENDIX B

SATURN C-2

PRELIMINARY DESIGN

Third Stage Pressurization Study

A. Propellant Tank Pressures

1. Hydrogen Tank

During Stages I and II operation the minimum tank pressure necessary to meet structural requirements = 20 psia

During Stage III operation it is assumed that:
P_v = Vapor pressure = 17.0 psia (1.0°F temperature rise)
P_n = Net positive suction head requirement
= 1.0 psia (With boost pumps)
= 8.0 psia (Without boost pumps)
P_f = Friction losses
= 2.0 psia (Vent valves)
= 1.0 psia (Vortex screen)
= 0.0 psia (Lines)
3.0 psia (Total for boost pump system)
= 2.0 psia (Vent valves)
= 1.0 psia (Vortex screen)
= 2.0 psia (Lines)
5.0 psia (Total for system without boost pumps)
a_l = Lift-off acceleration = 0.610 g
a_c = Cutoff acceleration = 2.523 g
h_l = Height of liquid column at lift-off
= 18.33 ft (With boost pumps)
= 25.00 ft (Without boost pumps)
h_c = Height of liquid column at cutoff
= 0.0 ft (With boost pumps)
= 8.33 ft (Without boost pumps)
ρ_h = Hydrogen density = 4.18 lb/ft^3

With Boost Pumps:
P_t = Total minimum gas pressure in tank necessary to meet boost pump requirements

At Lift-off:
$$P_t = P_v + P_n + P_f - (a)(h_l)(\rho_h)/144$$
$$= 17 + 1 + 3 - 0.3$$
$$= 20.7 \text{ psia}$$

At Cutoff:
$$P_t = P_v + P_n + P_f - (a)(h_c)(\rho_h)/144$$
$$= 17 + 1 + 3 - 0$$
$$= 21.0 \text{ psia}$$

Without Boost Pumps:

P_t = Total minimum gas pressure in tank necessary to meet turbopump requirements

At Lift-off:

$$P_t = P_v + P_n + P_f - (a_l)\,(h_l)\,(\rho_h)\,/144$$
$$= 17 + 8 + 5 - 0.4$$
$$= 29.6 \text{ psia}$$

At Cutoff:

$$P_t = P_v + P_n + P_f - (a_c)\,(h_c)\,(\rho_h)\,/144$$
$$= 17 + 8 + 5 - 0.6$$
$$= 29.4 \text{ psia}$$

2. Oxygen Tank

During Stages I and II operation the minimum tank pressure necessary to meet structural requirements = 30 psia. This includes a surge margin of at least 10 psia across the integrated bulkhead for bulkhead support.

During Stage III operation it is assumed that:

P_v = Vapor pressure = 15.5 psia (1.0°F temperature rise)

P_n = Net positive suction head requirement
= 2.5 psia (With boost pumps)
= 15.0 psia (Without boost pumps)

P_f = Friction losses
= 2.0 psia (Vent valves)
= 1.0 psia (Vortex screen)
= 0.0 psia (Lines)
3.0 psia (Total for boost pump system)
= 2.0 psia (Vent valves)
= 1.0 psia (Vortex screen)
= 2.0 psia (Lines)
5.0 psia (Total for system without boost pumps)

a_l = Lift-off acceleration = 0.610 g

a_c = Cutoff acceleration = 2.523 g

h_l = Height of liquid column at lift-off
= 10 ft (With and without boost pumps)

h_c = Height of liquid column at cutoff
= 0 ft (With and without boost pumps)

ρ_o = Oxidizer density = 71.2 lb/ft^3

With Boost Pumps:

P_t = Total minimum gas pressure in tank necessary to meet boost pump requirements

At Lift-off:

$$P_t = P_v + P_n + P_f - (a_c)\,(h_l)\,(\rho_o)\,/144$$
$$= 15.5 + 2.5 + 3.0 - 3.0$$
$$= 18.0 \text{ psia}$$

At Cutoff:

$$P_t = P_v + P_n + P_f - (a_c)\,(h_c)\,(\rho_o)\,/144$$
$$= 15.5 + 2.5 + 3.0 - 0$$
$$= 21.0 \text{ psia}$$

Without Boost Pumps:

P_t = Total minimum gas pressure in tank necessary to meet turbopump requirements

At Lift-off:

$$P_t = P_v + P_n + P_f - (a_1)(h_1)(\rho_o)/144$$
$$= 15.5 + 15.0 + 5.0 - 3.0$$
$$= 32.5 \text{ psia}$$

At Cutoff:

$$P_t = P_v + P_n + P_f - (a_c)(h_c)(\rho_o)/144$$
$$= 15.5 + 15.0 + 5.0 - 0$$
$$= 35.5 \text{ psia}$$

3. Summary

a. Hydrogen Tank With Boost Pump

Gas pressure in the hydrogen tank will be maintained at 21.0 psia. This value will satisfy the structural (20.0 psia) as well as the lift-off (20.7 psia) and cutoff (21.0 psia) pressurization requirements of the the engine boost pump system. During stage operation approximately 0.233 lb/sec total hydrogen bleed flow will be required for fuel tank pressurization. Sonic venturis will be built into each engine bleed system to prevent excessive bleed from any one engine. Prior to launch the pressurant will be ground-supplied helium to obtain the tank ullage pressure of 21.0 psia required for structural regidity during launch, and stage I and Stage II operation. Vent valves operated from a helium requlated supply will be used to maintain the tank pressure within the 21.0 $\pm$ 1.0 psia limits throughout vehicle flight.

b. Hydrogen Tank Without Boost Pump

Gas pressure will be maintained at 30.0 psia. This will satisfy the structural miminum requirement of 20.0 psia. During stage operation approximately 0.321 lb/sec total H_2 bleed flow will be required for fuel tank pressurization. Sonic venturis, vent valves, and pre-start helium pressurization will be used, as with the boost pump system described previously, but for a tank pressure of 30.0 psia in the case of the "no boost pump" system.

c. Lox Tank With Boost Pump

Gas pressure in the lox tank will be maintained at 31.0 psia if both lox and hydrogen boost pumps are used. This value, which more than meets the minumum pressure necessary for supplying the boost pump (21.0 psia), is necessary to maintain the structural rigidity of the center bulkhead

(10 psia differential). The use of a lox boost pump, in case a hydrogen boost pump is not used, does not make sense since the lox tank pressure would have to be about 40.0 psia for a pressure differential in the tanks and the lox boost pumps would only need about one-half this amount of tank pressure for operation. If the center bulkhead were reversed, in this case, then the pressurization values would be 30.0 psia for the hydrogen tank and 21.0 psia for the lox tank which would be the more reasonable configuration.

d. Lox Tank Without Boost Pump

Gas pressure in the lox tank will be maintained at 36.0 psia if a lox boost pump is not used. This will meet lox pump requirements as well as give an acceptable pressure differential across the center bulkhead. Gas pressure in the lox tank will be maintained at 40.0 psia in the event that no boost pumps are used in either the lox or the hydrogen tanks. This gives a pressure differential of 10 psia across the center bulkhead since the hydrogen tank pressure in this case is 30.0 psia.

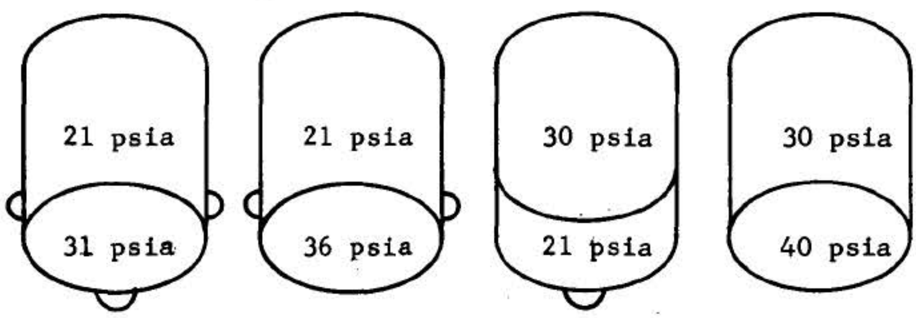
In view of the foregoing analysis, four different propellant tank and boost pump arrangements seem feasible and will be studied further from the overall pressurization standpoint to determine the relative feasibility or merits of each system. These four cases are as follows:

Case A: Both lox and hydrogen boost pumps are used.

Case B: Only a hydrogen boost pump is used.

Case C: Only a lox boost pump is used.

Case D: No boost pumps are used.



Case A   Case B   Case C   Case D

B. Pressurization System Comparison

In order to have a relative comparison of the four pressurization cases previously discussed, the following evaluation criteria will be considered:

1. Weight influence on specific configuration due to:
   (a) Gaseous Hydrogen System Requirements
   (b) Gaseous Oxygen System Requirements
   (c) Hydrogen Peroxide System Requirements
   (d) Helium System Requirements
   (e) Boost Pump System Requirements

Propellant Tank Volume Assumptions

W_8 = 99,397 lb (Ref. 3)

Oxidizer Tank:
   Total tank volume displaced by lox at lift-off = 1,170 ft^3
   Ullage volume at lift-off = 443 ft^3 (Case A, B, & D)*
   = 30 ft^3 (Case C)

*This ullage volume is large due to excess volume in the 45 degree ellipsoidal container for the given W_8.

Hydrogen Tank:
   Total tank volume displaced by LH_2 at lift-off = 4.072 ft^3
   Ullage volume at lift-off = 130 ft^3

Evaluation Criteria:
   (a) Gaseous Hydrogen System Requirements
      (1) Amount of LH_2 converted to GH_2

      Total tank volume pressurized by GH_2 = 4,072 ft^3
      Assuming that gas temperature in tank at burnout has an average value of 200°R,

      w = weight of LH_2 converted to GH_2
      = 96 lb for 21 psia tanks with a 20% allowance for boil-off venting
      = 137 lb for 30 psia tanks with a 20% allowance for boil-off venting

      t_b = stage burning time = 525 sec

      w = average pressurant flow rate
      = 0.183 lb/sec for 21 psia tanks
      = 0.261 lb/sec for 30 psia tanks

      (2) Plumbing and valve weight allowance

      Tubes and fittings = 12 lb
      Valves, regulators, etc., = 42 lb

(b) Gaseous Oxygen System Requirements

(1) Amount of lox converted to gox

Total tank volume pressurized by gox = 1,170 ft^3
Assuming that gas temperature in tank at burnout has an average value of 300°R when lox is converted to gox by heat exchanger and used as tank pressurant, weight of lox used for this purpose will then be:

w = 432 lb (Case A)
= 503 lb (Case B)
= 293 lb (Case C)
= 558 lb (Case D)

A 20% margin for boil-off venting is included in these values.

(2) Heat Exchanger

Using the lox boost pump turbine exhaust as a heat source for the gox heat exchanger:

Heat Exchanger Weight	16 lb
Tubing and Fittings	6 lb
Valves, Regulators, etc.	43 lb

If no boost pumps are used then a separate hydrogen peroxide gas generator is used for heat exchanger. Weight assumptions for this approach are:

Heat Exchanger Weight	50 lb
Tubing and Fittings	8 lb
Valves, Regulators, etc.	50 lb

(c) Hydrogen Peroxide System Requirements

(1) Hydrogen-peroxide used for boost pump drive

Boost pumps, lox and LH_2 = 140 lb
Boost pumps, lox only = 66 lb
Boost pumps, LH_2 only = 74 lb

(2) Attitude Control System

Hydrogen peroxide weight estimate for attitude control reaction jets = 50 lb

(3) Lox Heat Exchanger

Hydrogen peroxide weight estimate for lox heat exchanger = 50 lb

(4) Helium Heat Exchanger

Hydrogen peroxide weight estimate for helium heat exchanger = 50 lb

Summary of Usable Hydrogen-Peroxide Requirements

Case A:	Boost pump drive	140 lb
	Attitude Control	50 lb
	Total	190 lb

Note: In this case heat source from boost pump exhaust is available for heating either the lox or helium for tank pressurization so no hydrogen peroxide requirement allowance is made for heat exchanger.

Case B:	Hydrogen boost pump drive	74 lb
	Attitude control	50 lb
	Total	124 lb

Note: Same as above

Case C:	Lox boost pump drive	66 lb
	Attitude control	50 lb
	Total	116 lb

Note: Same as above

Case D:	Attitude control	50 lb
	Lox or helium heat exchanger	50 lb
	Total	100 lb

(5) Residuals in Hydrogen Peroxide System

Assuming 10% residual in the hydrogen peroxide system, the total hydrogen peroxide to be contained is:

Case A: 190 + 19 = 209 lb
Case B: 124 + 12 = 136 lb
Case C: 116 + 12 = 128 lb
Case D: 100 + 10 = 110 lb

(6) Bottle Weight

V_b = Hydrogen peroxide container volume based on 5% ullage allowance and hydrogen peroxide density of 0.0498 lb/in.^{3}
= 4,407 in.^{3} (Case A)
= 2,867 in.^{3} (Case B)
= 2,699 in.^{3} (Case C)
= 2,319 in.^{3} (Case D)

d_i = Inside diameter of hydrogen peroxide sphere
= 20.3 in. (Case A)
= 17.6 in. (Case B)
= 17.3 in. (Case C)
= 16.4 in. (Case D)

Hydrogen peroxide sphere conditions and assumptions:

Pressurant: Helium from engine supply
at 450 ± 50 psia
Material: Aluminum Alloy 6061
Yield Strength: 35,000 psi
Factor of Safety: 2
Material Density: 0.1 lb/in.^{3}

t_s = Wall thickness of sphere
= 0.145 in. (Case A)
= 0.126 in. (Case B)
= 0.124 in. (Case C)
= 0.117 in. (Case D)

W_s = Weight of spherical bottle
= 18.8 lb (Case A)
= 12.3 lb (Case B)
= 11.7 lb (Case C)
= 9.9 lb (Case D)

(7) Hydrogen peroxide plumbing, valves, fittings, etc.

Allowance = 50% of bottle weight
= 9.4 lb (Case A)
= 6.2 lb (Case B)
= 5.6 lb (Case C)
= 5.0 lb (Case D)

(d) Helium System Requirements

(1) Lox Tank Pre-start Pressurization

Assuming that the mean temperature of the ground-supplied, refrigerated helium in the lox tank ullage volume is 173°R,

w = Weight of helium required for lox tank pre-start pressurization
= 29.6 lb (Case A)
= 34.4 lb (Case B)
= 1.4 lb (Case C)
= 38.2 lb (Case D)

(2) Hydrogen Tank Pre-start Pressurization

Assuming that the mean temperature of the ground-supplied, refrigerated helium in the LH_2 tank ullage volume is 60°R,

w = Weight of helium required for LH_2 tank pre-start pressurization
= 22.2 lb (Case A, Case B)
= 31.7 lb (Case C, Case D)

(3) Lox Tank Pressurization During Stage Operation

Total volume to be pressurized during flight = 1,170 ft^3. Assuming that the mean temperature of the heated helium pressurant in the tank at burnout is 300°R,

w = Weight of helium in tank at burnout
= 45.1 lb (Case A)
= 52.3 lb (Case B)
= 30.5 lb (Case C)
= 58.2 lb (Case D)

(4) Pressurization of Hydrogen Peroxide Tanks

Pressure in hydrogen peroxide tanks = 500 psia. Assuming final helium temperature in hydrogen peroxide sphere = 500°R

w = Helium weight in peroxide tank at burnout
= 1.0 lb (Case A)
= 0.6 lb (Case B)
= 0.6 lb (Case C)
= 0.5 lb (Case D)

(5) Engine and Vent Valve Operation

Engine valve operation helium requirement	2.0 lb
Vent valve operation helium requirement	1.0 lb

Summary of unsubmerged helium system requirements for engine and hydrogen peroxide pressure circuit:

Initial bottle conditions, T_i = 575°R, P_i = 3000 psia
Final bottle conditions, T_f = 500°R, P_f = 1900 psia

Based on these conditions, the total volumes of the helium bottles used in this application are:

7.6 ft^3 (Case A)
6.8 ft^3 (Case B)
6.8 ft^3 (Case C)
6.6 ft^3 (Case D)

From this it is determined that the total weight of the helium initially charged into the bottles is:

14.8 lb (Case A)
13.2 lb (Case B)
13.2 lb (Case C)
12.8 lb (Case D)

The weight of the residual helium in the bottles, at the end of stage operation, is the weight of the initially charged helium minus the weight of helium used for engine and vent valve operation and the weight used for hydrogen peroxide tank pressurization. This residual helium weight is then:

10.8 lb (Case A)
9.6 lb (Case B)
9.6 lb (Case C)
9.3 lb (Case D)

Summary of submerged helium system requirements for lox tank pressurization circut:

Initial bottle conditions, T_i = 163°R, P_i = 3000 psia
Final bottle conditions, T_f = 110°R, P_f = 500 psia

Based on these conditions, the total volumes of the helium bottles used in this application are:

8.7 ft^3 (Case A)
10.1 ft^3 (Case B)
5.9 ft^3 (Case C)
11.3 ft^3 (Case D)

From this it is determined that the total weight of the helium initially charged into the bottles is:

59.7 lb (Case A)
69.3 lb (Case B)
40.5 lb (Case C)
77.5 lb (Case D)

The weight of the residual helium in the submerged bottles, at end of stage operation, is the weight of the initially charged helium minus the weight of helium used for pressurization of the lox tank during stage operation. This residual helium weight is:

14.6 lb (Case A)
17.0 lb (Case B)
10.0 lb (Case C)
19.3 lb (Case D)

(6) Bottle Weight of Un submerged Bottles Used for Engine and Hydrogen Peroxide Pressurization Circuit

d = Diameter of 1 bottle when 4 bottles are used
= 18.4 in. (Case A)
= 17.8 in. (Case B)
= 17.8 in. (Case C)
= 17.6 in. (Case D)

Bottle material: Titanium 6 AL-4V
F_{tv} = 100,000 psi
P_{max} = 3000 psi
Factor of safety = 2
Density = 0.16 lb/in.^{3}
t = Wall thickness of sphere
= 0.276 in. (Case A)
= 0.267 in. (Case B)
= 0.267 in. (Case C)
= 0.264 in. (Case D)

W_s = Weight of 1 spherical bottle
W_t = Total weight of 4 spherical bottles

W_s = 47.0 lb	W_t = 188 lb	(Case A)
= 42.5 lb	= 170 lb	(Case B)
= 42.5 lb	= 170 lb	(Case C)
= 41.1 lb	= 164 lb	(Case D)

(7) Un submerged Sphere Support Bracketry Weight

= 24 lb (Case A)
= 21 lb (Case B, C, D)

(8) Bottle Weight of Submerged Bottles Used for Lox Tank Pressurization Circuit

d = Diameter of 1 bottle when 4 bottles are used
= 19.3 in. (Case A)
= 20.3 in. (Case B)
= 16.9 in. (Case C)
= 21.0 in. (Case D)

Bottle material: Titanium 6 AL-4V
F_{ty} = 175,000 psi
P_{max} = 3000 psi
Factor of safety = 2
Density = 0.16 lb/in.^{3}
t = Wall thickness of sphere
= 0.165 in. (Case A)
= 0.174 in. (Case B)
= 0.145 in. (Case C)
= 0.180 in. (Case D)

W_s = Weight of 1 spherical bottle
W_t = Total weight of 4 spherical bottles

W_s = 30.9 lb	W_t = 123.6 lb	(Case A)
= 36.0 lb	= 144.0 lb	(Case B)
= 20.8 lb	= 83.2 lb	(Case C)
= 39.9 lb	= 159.6 lb	(Case D)

(9) Submerged Sphere Support Bracketry Weight

= 15 lb (Case A)
= 18 lb (Case B)
= 10 lb (Case C)
= 20 lb (Case D)

(e) Boost Pump System Requirements

(1) Lox Boost Pump System

Boost Pump and Drive		155 lb
Pump (including turbine and gearbox)	120 lb	
Pump Mounts	35 lb	

(2) Fuel Boost Pump System

Boost Pump and Drive		135 lb
Pump (including turbine and gearbox)	120 lb	
Pump Mounts	15 lb	

(3) Plumbing and Valving

For Lox or Fuel Boost Pumps = 20 lb

TABLE I-B

SUBSYSTEM WEIGHT SUMMARY FOR THIRD STAGE PRESSURIZATION SYSTEMS

WEIGHT INFLUENCE DUE TO SPECIFIC PRESSURIZATION SYSTEM CHOICE	CASE A	CASE B	CASE C	CASE D
(a) Gaseous Hydrogen System				
1. Amount of LH_2 converted to GH_2	96.0	96.0	137.0	137.0
2. Plumbing and valve weight	54.0	54.0	54.0	54.0
(b) Gaseous Oxygen System				
1. Amount of lox converted to gox	432.0	503.0	293.0	558.0*
2. Heat exchanger	65.0	108.0	65.0	108.0*
(c) Hydrogen Peroxide System				
1. H_2O_2 for boost pump drive	140.0	74.0	66.0	----
2. Attitude control system H_2O_2	50.0	50.0	50.0	50.0
3. Lox heat exchanger	---	---	---	50.0*
4. Helium heat exchanger	---	---	---	50.0**
5. H_2O_2 residuals	19.0	12.0	12.0	10.0
6. H_2O_2 bottle weight	18.8	12.3	11.7	9.9
7. H_2O_2 plumbing, valves, etc.	9.4	6.2	5.6	5.0
(d) Helium system requirements				
1. Lox tank pre-start pressurization	29.6	34.4	1.4	38.2
2. LH_2 tank pre-start pressurization	22.2	22.2	31.7	31.7
3. Lox tank stage pressurization	45.1	52.3	30.5	58.2**
4. H_2O_2 bottle pressurization	1.0	0.6	0.6	0.5
5. Engine and vent valve helium req.	3.0	3.0	3.0	3.0
Residuals in un submerged circuit	10.8	9.6	9.6	9.3
Residuals in submerged circuit	14.6	17.0	10.0	19.3**
6. Bottle weight of un submerged sys.	188.0	170.0	170.0	164.0
7. Un submerged bottle support struc.	24.0	21.0	21.0	21.0
8. Bottle weight of submerged system	123.6	144.0	83.2	159.6**
9. Submerged bottle support structure	15.0	18.0	10.0	20.0**
(e) Boost Pump System Requirements				
1. Lox boost pump system	155.0	---	155.0	---
2. Fuel boost pump system	135.0	135.0	---	---
3. Boost pump plumbing and valving	40.0	20.0	20.0	---

* These weights in all four cases are attributable only to that lox tank pressurization system which uses gox for pressurization.

** These weights in all four cases are attributable only to that lox tank pressurization system which uses heated helium pressurization.

Total Weight Influence of Specific Pressurization System Choice on Overall Stage Weight Excluding the Influence Due to Propellant Tank Weight Differences for the Various Systems:

Lox Tank Pressurization by Gox

	CASE A	CASE B	CASE C	CASE D
(a) Gaseous Hydrogen System	150.0	150.0	191.0	191.0
(b) Gaseous Oxygen System	497.0	611.0	358.0	666.0
(c) H_2O_2 System	237.2	154.5	145.3	124.9
(d) Helium System	278.6	260.8	237.3	267.7
(e) Boost Pump System	330.0	155.0	175.0	-----
Total, lb	1492.8	1331.3	1106.6	1249.6

Lox Tank Pressurization by Heated Helium

	CASE A	CASE B	CASE C	CASE D
(a) Gaseous Hydrogen System	150.0	150.0	191.0	191.0
(b) Gaseous Oxygen System	-----	-----	-----	-----
(c) H_2O_2 System	237.2	154.5	145.3	124.9
(d) Helium System	476.9	492.1	371.0	524.8
(e) Boost Pump System	330.0	155.0	175.0	-----
Total, lb	1194.1	951.6	882.3	840.7

2. Conclusions

Although this study is based on a number of orbitrary assumptions and simplifications which influence the absolute weight values obtained, it is, nevertheless, felt that the results represent valid parametric data by which the different pressurization systems may be compared on a relative basis.

From the results obtained it is concluded that, of the two lox tank pressurization systems studied, the heated helium system will yield the lowest overall stage weight in all of the four cases. It is also concluded that, from the pressurization system standpoint only, the lightest system will be case D, the system which uses no boost pumps. It must be stressed, however, that from the overall stage weight standpoint, each of the four cases studied will be strongly influenced by the propellant tank weights involved. The selection of a particular system would depend on the overall stage weight based on the combined results of this study and the results of a tank weight study for each of the four cases.

In addition to the overall weight considerations, the final selection will be influenced by other factors such as adaptability to multiple restart requirements, attitude control requirements, propellant utilization requirements, cost, and reliability.

APPENDIX C

MASS CHARACTERISTICS FOR SELECTED SATURN C-2 VEHICLES

The center of gravity and moment of inertia for both the single tank and clustered tank four stage SATURN C-2 configurations are given in this appendix in Figs. 1-C through 4-C. Basic data used to obtain these curves were given in Tables VI through VIII. Summary weight and propulsion data for the two configurations are contained in Tables I-C and II-C. Propellant loading and flight performance reserves were obtained from these tables.

FOUR STAGE SATURN C-2 (CLUSTERED BOOSTER)
ESCAPE PAYLOAD

CENTER OF GRAVITY VS TIME DURING FIRST STAGE BURNING

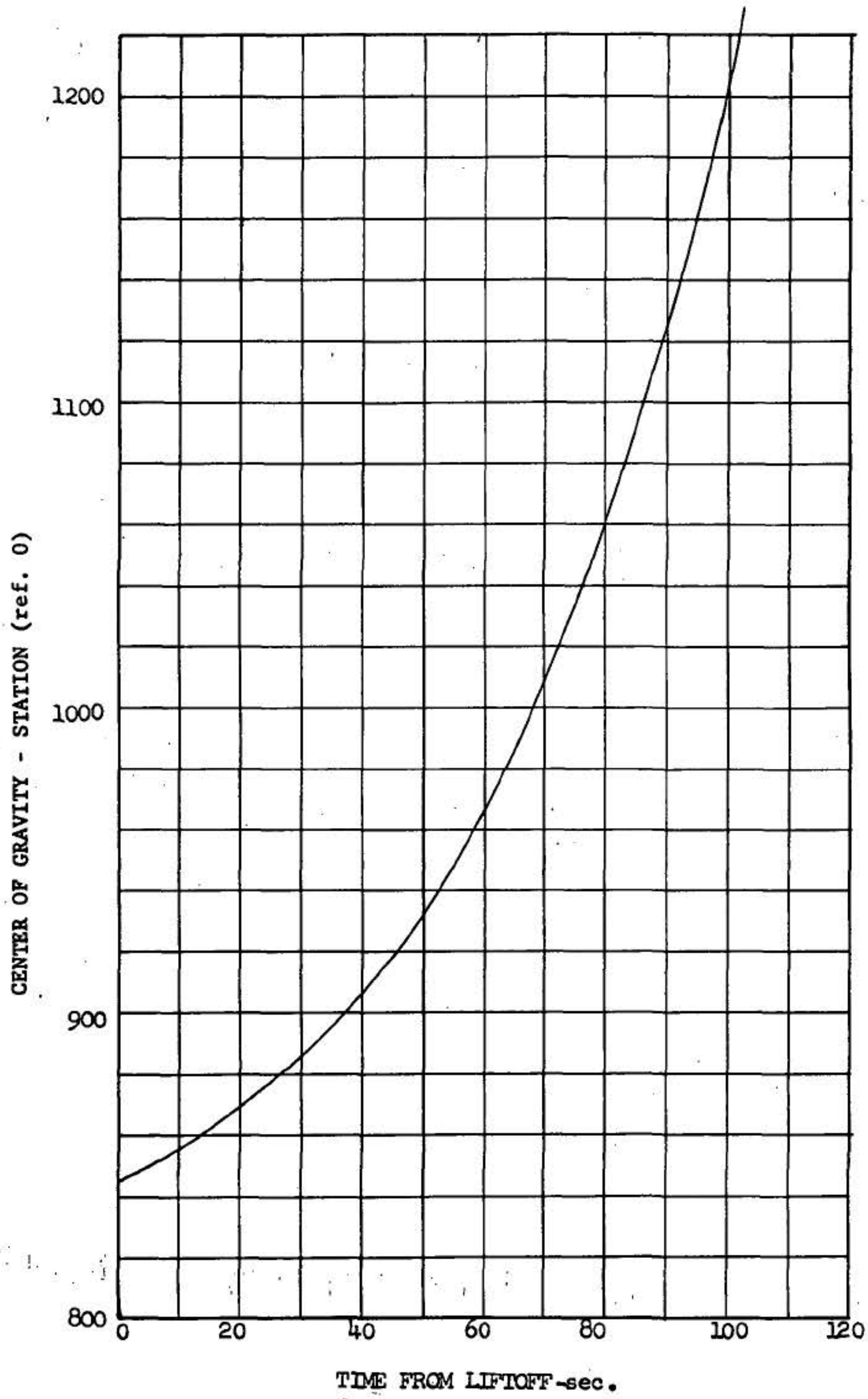
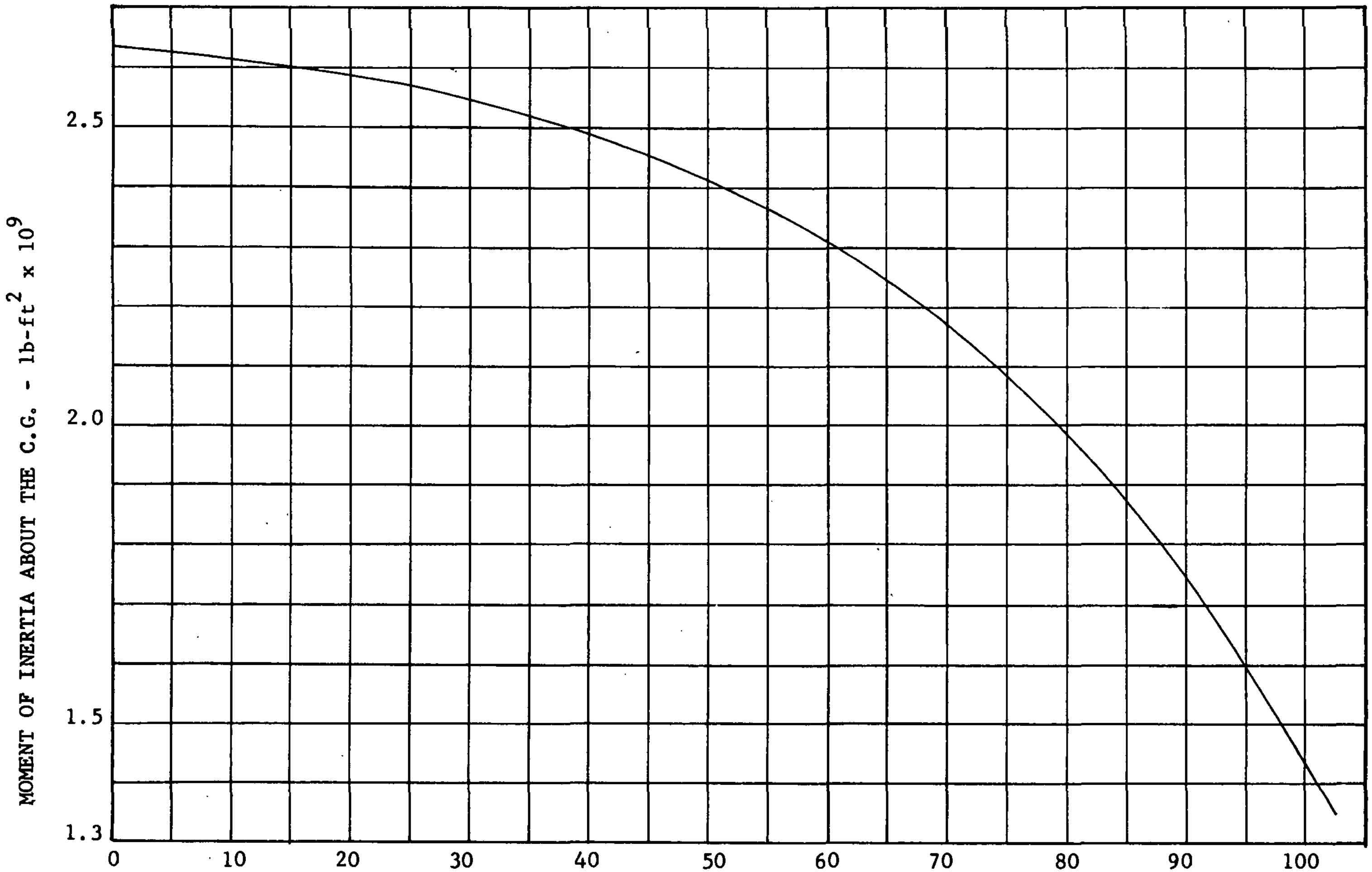


Figure 1-C

FOUR STAGE SATURN C-2 (CULSTERED BOOSTER)
ESCAPE PAYLOAD

PITCH MOMENT OF INERTIA VS TIME DURING FIRST STAGE BURNING



TIME FROM LIFTOFF - sec.
Figure 2-C

FOUR STAGE SATURN C-2 (SINGLE TANK BOOSTER)
ESCAPE PAYLOAD

CENTER OF GRAVITY VS TIME DURING FIRST STAGE BURNING

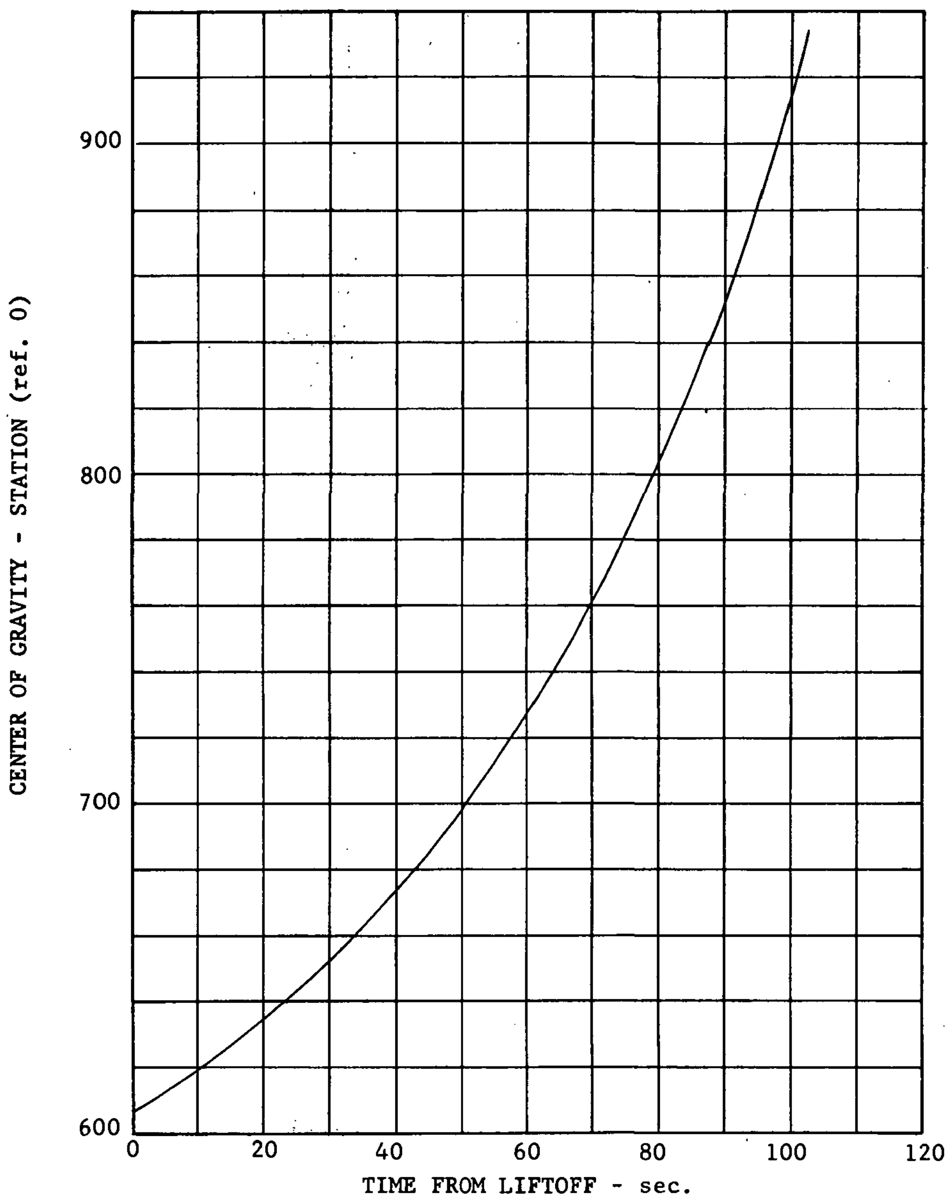


Figure 3-C

FOUR STAGE SATURN C-2 (SINGLE TANK BOOSTER)
ESCAPE PAYLOAD

PITCH MOMENT OF INERTIA VS TIME DURING FIRST STAGE BURNING

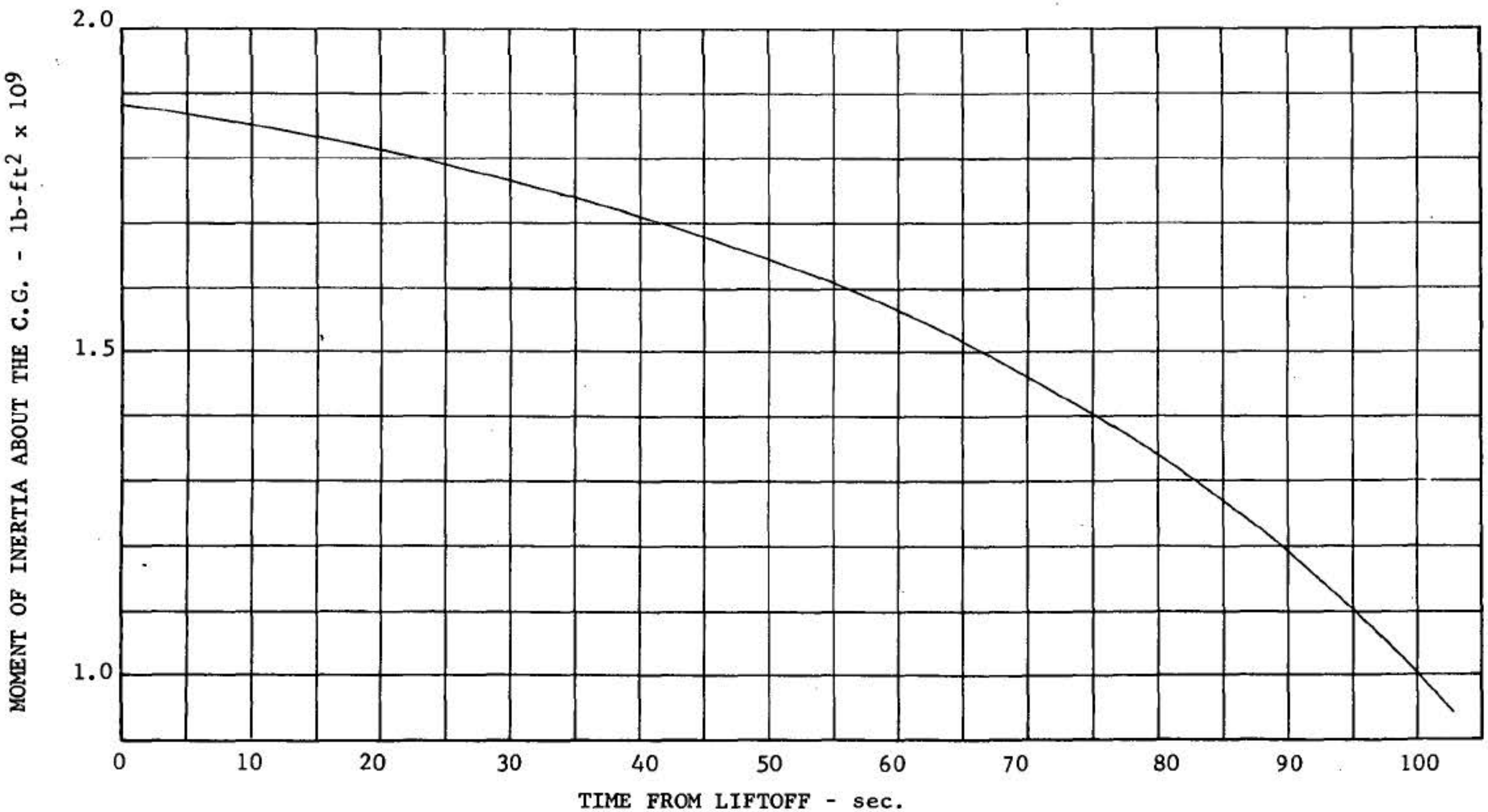


Figure 4-C

TABLE I-C

FOUR STAGE SATURN C-2 (Clustered Booster)

Escape Payload

SUMMARY WEIGHT AND PROPULSION DATA

Stage	I	II	III	IV
Engine	H-1	J-2	LR-115	LR-115
Propellant	LOX/RP-1	LOX/LH	LOX/LH	LOX/LH
Thrust, lb	8 x 188K	4 x 200K	4 x 17.5K	2 x 17.5K
I_{sp}, sec	257 s.l.	420 vac.	420 vac.	420 vac.
Burning Time, sec	102.53	168.00	425.82	341.16
Missile Diameter, in.	260	260	220	120
$W_{11,15}$, Payload, lb	--	--	--	19,620
W_{16}, Guid. Compartment, lb	--	--	--	500
W_2, Guid. & Control, lb	2,500	500	500	1,500
W_3, Fuselage, lb	52,000	14,770	4,990	1,500
W_I, Insulation,* lb	--	2,520	--	--
W_4, Propulsion, lb	22,000	9,630	3,120	1,200
W_5, Recovery Equip., lb	12,000	--	--	--
W_6, Trapped Prop., lb	15,000	3,290	1,040	440
W_7, Usable Residuals, lb	6,000	1,600	MRS 380 FPR 4,560	MRS 150 FPR 570
W_8, Prop. Consumption, lb	600,000	320,000	70,900	28,430
$W_{s,16}$, Structure Wt., lb	88,500	24,900	8,610	4,700
$W_{n,16}$, Structure Net Wt., lb	109,500	29,790	14,590	5,860
$W_{a,16}$, Stage Wt., lb	709,500	352,310*	85,490	34,290
W_o, Liftoff Wt., lb	1,201,340	491,840*	139,530**	53,910
W_c, Cutoff Wt., lb	601,340	169,320	68,500	25,480
r, Mass Ratio	1.998	2.887	2.035	2.116
Δu, Charac. Vel., m/sec	1,873	4,360	2,922	3,082
F_o/W_o	1.25	1.63	0.50	0.65
F_{vac}/W_c	2.81	4.72	1.02	1.37

* Insulation jettisoned at 150 sec $W_{I_{II}}$ = 1620 lb., $W_{I_{III}}$ = 600 lb., and $W_{I_{IV}}$ = 300 lb.

** Includes 130 lbs of H_2O_2 expended at separation.

TABLE II-C

FOUR STAGE SATURN C-2 (Single Tank Booster)

Escape Payload

SUMMARY WEIGHT AND PROPULSION DATA

Stage	I	II	III	IV
Engine	H-1	J-2	LR-115	LR-115
Propellant	LOX/RP-1	LOX/LH	LOX/LH	LOX/LH
Thrust, lb	8 x 188K	4 x 200K	4 x 17.5K	2 x 17.5K
I_{sp}, sec.	257 s.l.	420 vac	420 vac	420 vac
Burning Time, sec.	102.53	173.25	570.96	340.80
Missile Diameter, in.	260	260	220	120
$W_{11,15}$, Payload, lb	---	---	---	21,500
W_{16}, Guid. Compartment, lb	---	---	---	500
W_2, Guid. & Control, lb	1,000	500	500	1,500
W_3, Fuselage, lb	24,300	14,770	4,990	1,500
W_I, Insulation, lb*	---	2,520	---	---
W_4, Propulsion, lb	18,700	9,630	3,120	1,200
W_5, Recovery Eq., lb	5,000	---	---	---
W_6, Trapped Prop., lb	11,000	3,290	1,040	440
W_7, Usable Residuals, lb	3,000	1,650	MRS 500 FPR4,840	MRS 150 FPR 600
W_8, Prop. Consumption, lb	600,000	330,000	95,160	28,400
$W_{s,16}$, Structure Wt, lb	49,000	24,900	8,610	4,700
$W_{n,16}$, Struc. Net Wt, lb	63,000	29,840	14,990	5,890
$W_{a,16}$, Stage Wt, lb	663,000	362,360*	110,150	34,290
W_o, Liftoff Wt, lb	1,191,430	528,430*	166,070**	55,790
W_c, Cutoff Wt, lb	591,430	195,910	70,780	27,390
r, Mass Ratio	2.014	2.682	2.344	2.037
Δu, Charac. Vel. (m/sec)	1896	4057	3504	2926
F_o/W_o	1.26	1.51	0.42	0.63
F_{vac}/W_c	2.87	4.08	0.99	1.28

* Insulation jettisoned at 150 sec $W_{I_{II}}$ = 1620 lb, $W_{I_{III}}$ = 600 lb, and $W_{I_{IV}}$ = 300 lb.

** Includes 130 lb of H_2O_2 expended at separation.

APPENDIX D

SEPARATION STUDIES USING THE J-2 ENGINE THRUST BUILD-UP CURVE

Included in this appendix are the time history plots of angle of attack after separation of the clustered booster from the upper stages. The thrust build-up curve for the J-2 engine as shown in Fig. 1-D was used throughout and the constants used were given in Table V. Figures 2-D through 8-D present these time histories for several delay times of the engine start signal. The start signal here is assumed to have been followed immediately by first thrust from the engine. According to Rocketdyne there is approximately 0.5 seconds from start signal until first thrust. This will cause no problem since there will be an extensive timing system to control the various signals during separation. The procedure used to compute these data is discussed in Refs. 5 and 7.

THRUST BUILD-UP FOR J-2 ENGINE

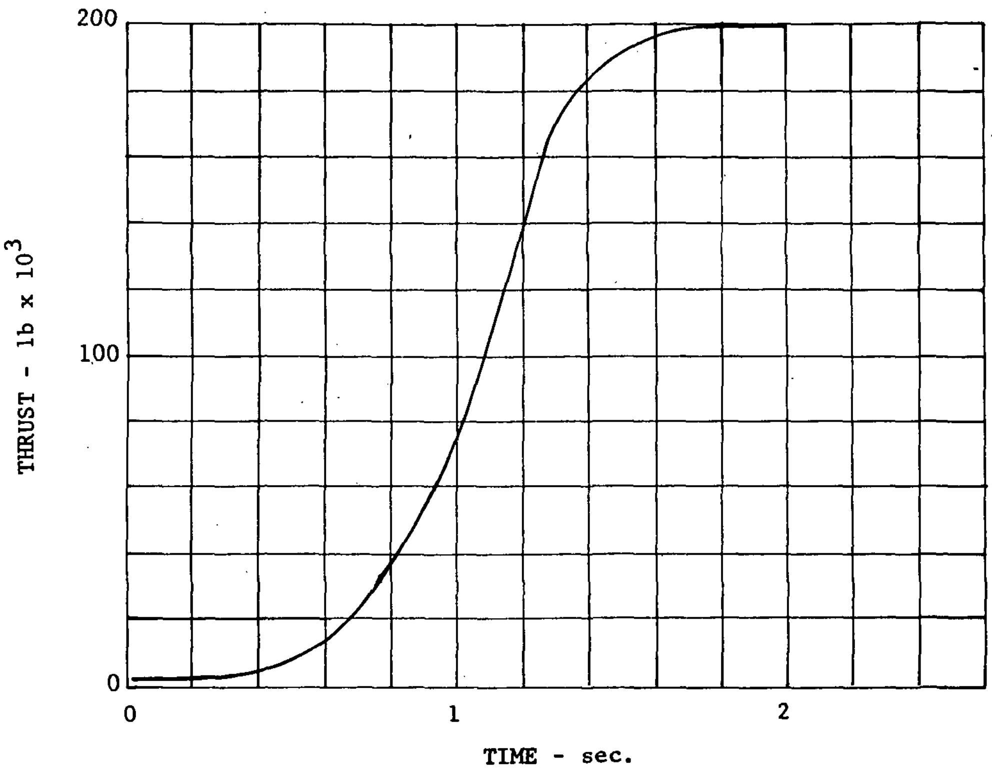


Figure 1-D.

SATURN C-2

ANGLE OF ATTACK TIME HISTORY DURING BOOSTER SEPARATION FOR 0 DELAY IN ENGINE START SIGNAL

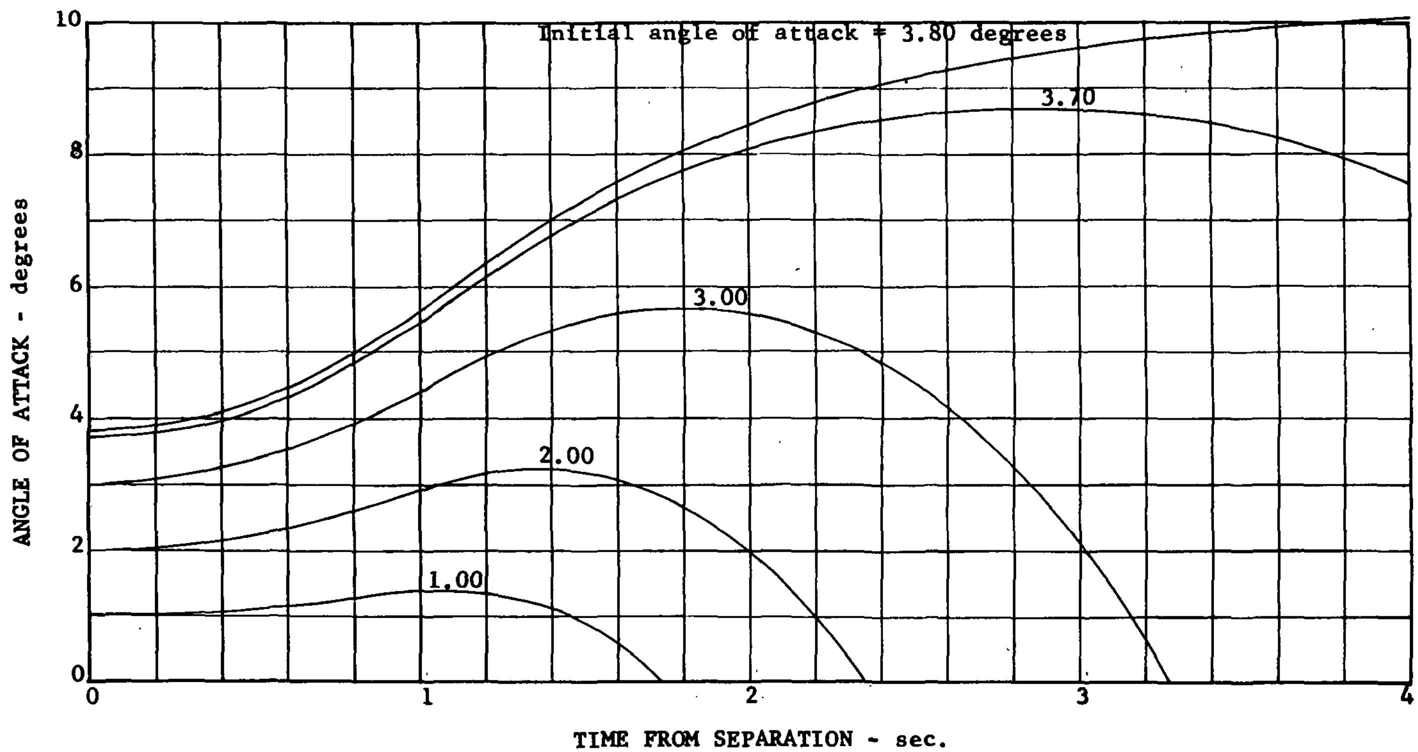


Figure 2-D

SATURN C-2

ANGLE OF ATTACK TIME HISTORY DURING BOOSTER SEPARATION FOR 0.2 DELAY IN ENGINE START SIGNAL

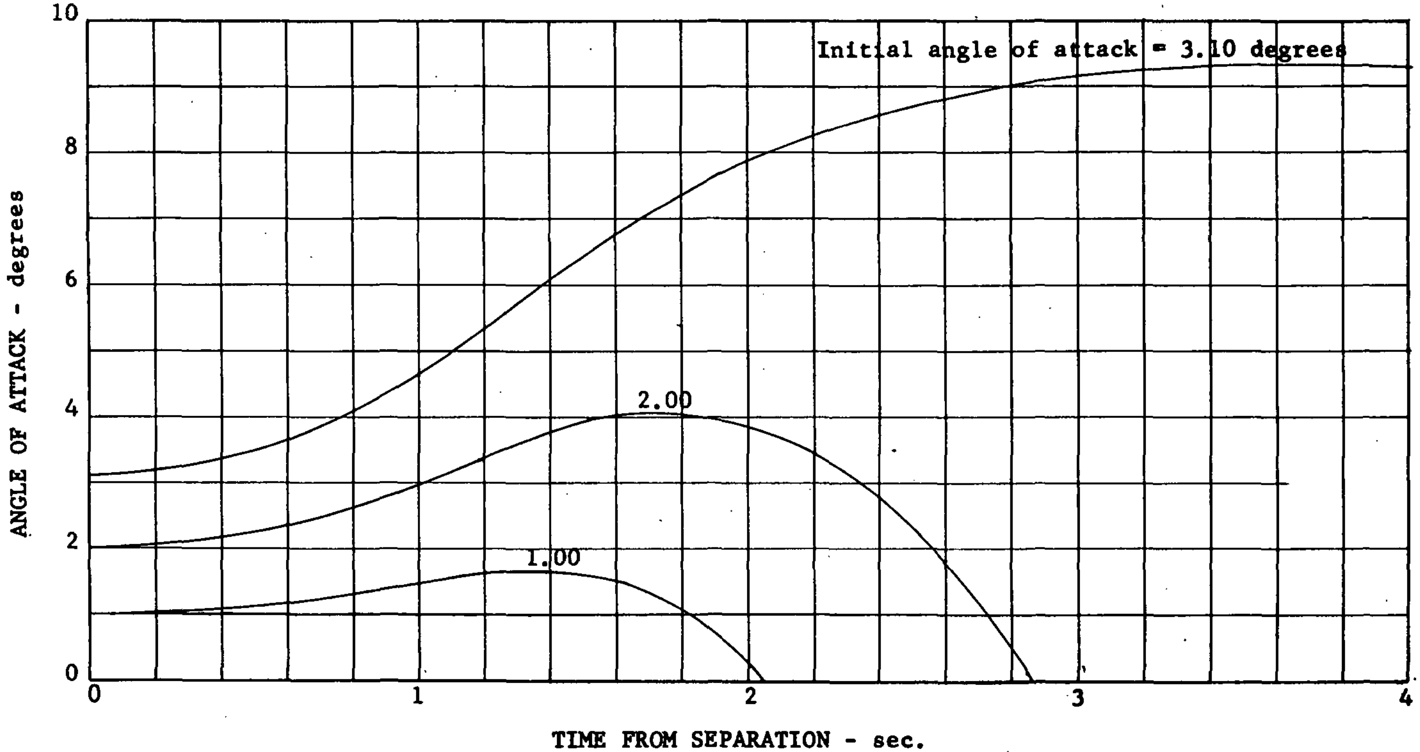


Figure 3-D

SATURN C-2

ANGLE OF ATTACK TIME HISTORY DURING BOOSTER SEPARATION FOR .4 DELAY IN ENGINE START SIGNAL

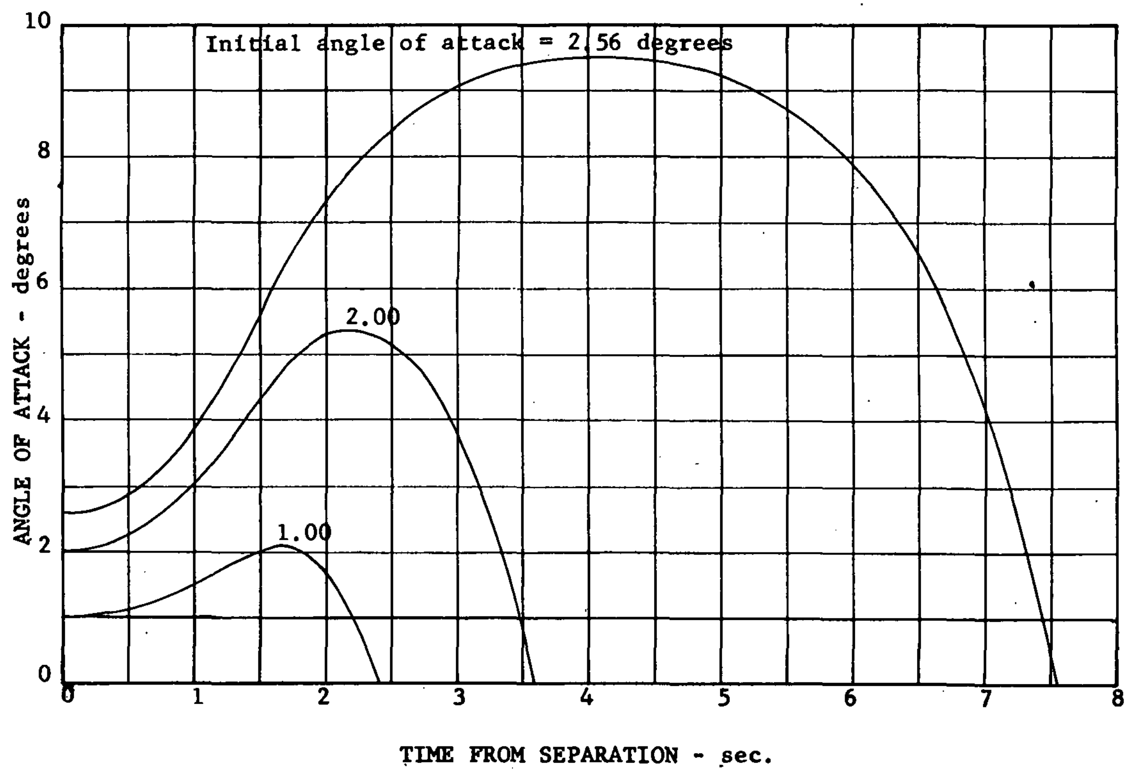


Figure 4-D

SATURN C-2

ANGLE OF ATTACK TIME HISTORY DURING BOOSTER SEPARATION FOR 0.6 DELAY IN ENGINE START SIGNAL

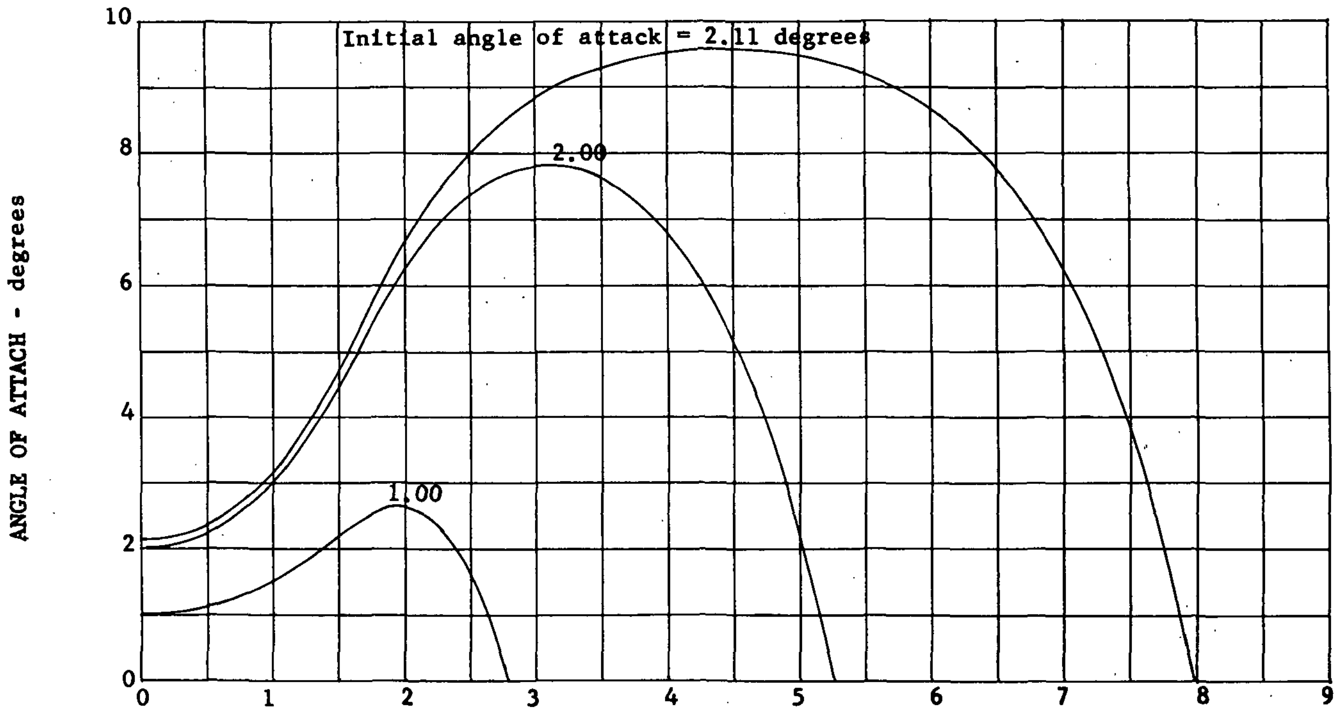


Figure 5-D

SATURN C-2

ANGLE OF ATTACK TIME HISTORY DURING BOOSTER SEPARATION FOR. 0.8 DELAY IN ENGINE START SIGNAL

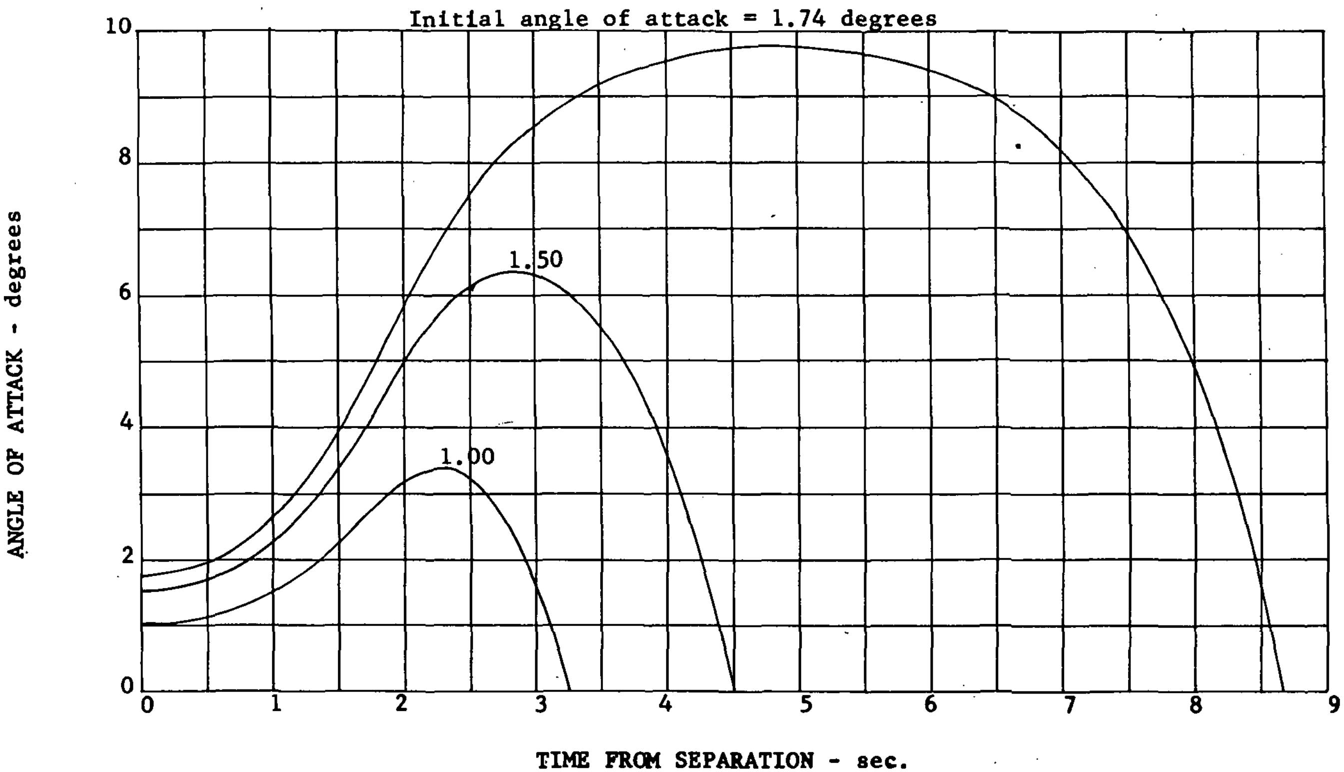


Figure 6-D

SATURN C-2

ANGLE OF ATTACK TIME HISTORY DURING BOOSTER SEPARATION FOR 1.0 DELAY IN ENGINE START SIGNAL

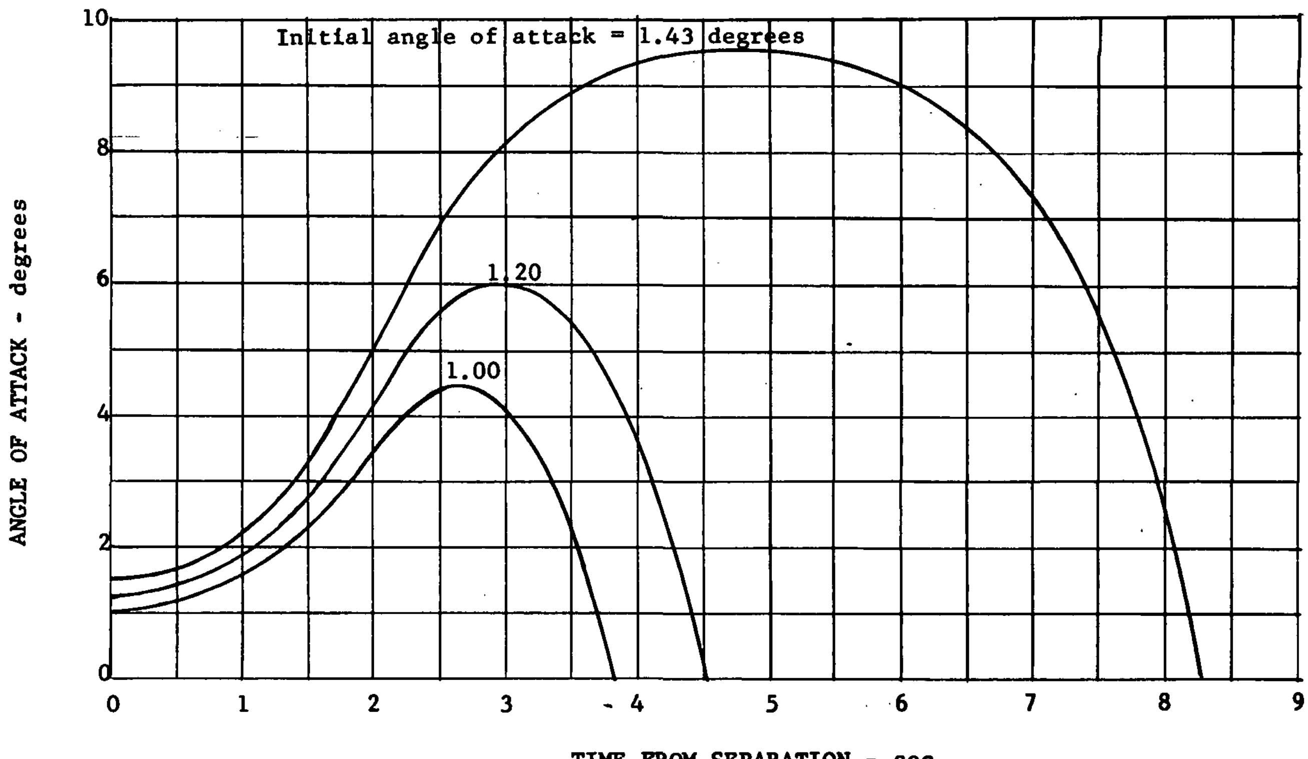


Figure 7-D

SATURN C-2

ANGLE OF ATTACK TIME HISTORY DURING BOOSTER SEPARATION FOR 1.2 DELAY IN ENGINE START SIGNAL

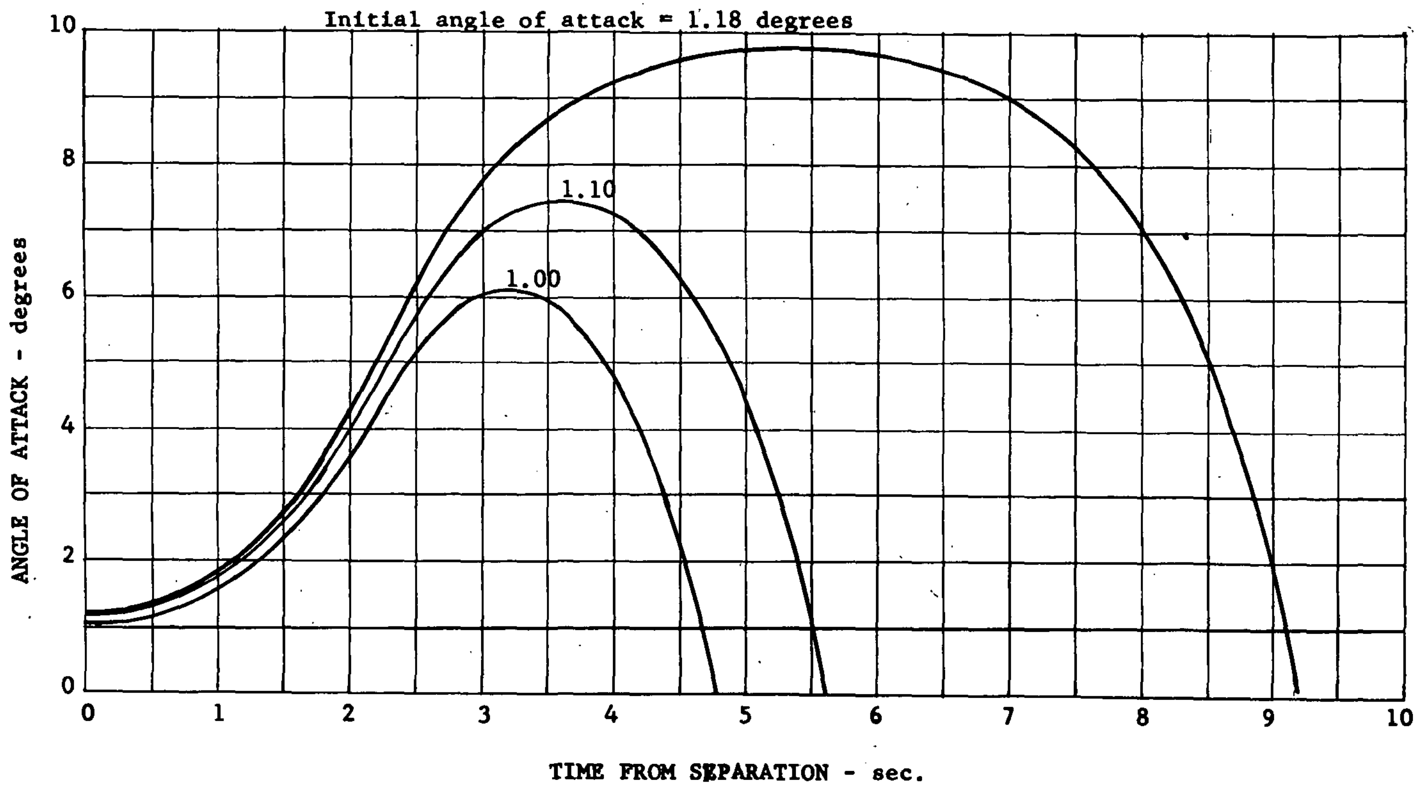


Figure 8-D

APPENDIX E

POSSIBLE SATURN C-2 CONFIGURATIONS

Presented in this appendix are six of the possible SATURN C-2 configurations. These six configurations represent a cross section of the most probable configurations conceivable. The primary purpose of these data is to provide the basic information necessary to establish maximum gimbal angle and load requirements. Studies to determine these items are now underway. Results from these studies will be published at a later date. The following paragraphs give brief discussions of the six configurations.

Configuration 1 - This vehicle consists of the S-I, and S-II stages. The payload is a nuclear stage which will be tested in orbit. Figure 1-E is a line drawing of this configuration and Table I-E presents the summary weight and propulsion data. Center of gravity and pitch moment of inertia versus time during first stage burning are given in Figs. 2-E and 3-E.

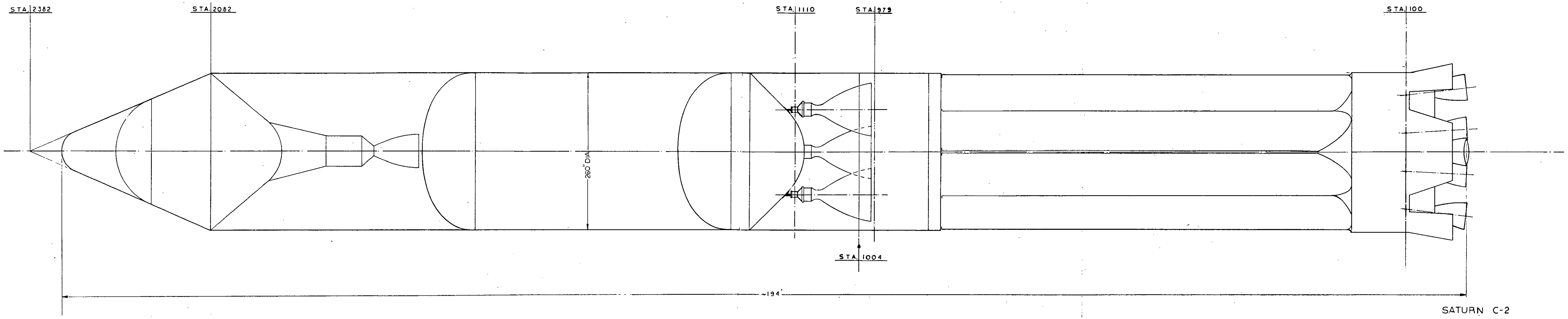
Configuration 2 - This vehicle consists of the S-I and S-II stages with a cargo-type payload. The line drawing for this configuration is shown in Fig. 4-E and the summary weight and propulsion data in Table II-E. Center of gravity and pitch moment of inertia versus time during first stage burning are given in Figs. 5-E and 6-E.

Configuration 3 - This vehicle consists of the S-I, S-II and S-IV stage with a cargo-type low orbit payload. Figure 7-E presents the line drawing and Table III-E the summary weight and propulsion data. Center of gravity and pitch moment of inertia versus time during first stage burning are given in Figs. 8-E and 9-E.

Configuration 4 - This vehicle is identical to number 3 except that it has the escape payload. Figure 10-E presents the line drawing and Table IV-E the summary weight and propulsion data. Center of gravity and pitch moment of inertia versus time during first stage burning are given in Figs. 11-E and 12-E.

Configuration 5 - This vehicle consists of the S-I, S-II, S-IV and S-V stage with a cargo-type escape payload. Figure 13-E presents the line drawing and Table V-E the summary weight and propulsion data. Center of gravity and pitch and roll moment of inertia versus time during first stage burning are shown in Figs. 14-E, 15-E and 16-E.

Configuration 6 - This vehicle consists of the S-I, S-II and nuclear stages with a cargo-type escape payload. The line drawing is shown in Fig. 17-E and the summary weight and propulsion data in Table VI-E. Center of gravity and pitch moment of inertia versus time during first stage burning are shown in Figs. 18-E and 19-E.



SATURN C-2
TWO STAGE VEHICLE
NUCLEAR TEST STAGE PAYLOAD
FIG-1E

TABLE I-E

TWO STAGE SATURN C-2 (Clustered Booster)

Nuclear Test Stage Payload

SUMMARY WEIGHT AND PROPULSION DATA

Stage	I	II
Engine	H-1	J-2
Propellant	LOX/RP-1	LOX/LH
Thrust, lb	8 x 188K	4 x 200K
I_{sp}, sec.	257 sl	420 vac
Burning Time	116.99	170.58
Missile Diameter, in.	260	260
$W_{11,15}$, Payload, lb	---	40,000
W_{16}, Guid. Compartment, lb	---	---*
W_2, Guid. & Control, lb	2,500	500
W_3, Fuselage, lb	55,000	16,390
W_4, Propulsion, lb	22,000	9,630
W_5, Recovery Eq., lb	12,000	---
W_6, Trapped Prop., lb	15,000	3,290
W_7, Usable Residuals, lb	6,500	MRS 1,650 FPR 5,080
W_8, Propellant Consumption, lb	684,620	324,920
W_s, Structure Wt, lb	91,500	26,520
W_n, Structure Net Wt, lb	113,000	36,540
W_a, Stage Wt, lb	797,620	361,460
W_o, Liftoff Wt, lb	1,199,080	401,460
W_c, Cutoff Wt, lb	514,460	76,540
r, Mass Ratio	2.331	5.245
Δu, Charac. Vel. (m/sec)	2291	6816
F_o/W_o	1.25	1.99
F_{vac}/W_c	3.29	10.45

* 2000 lb of W_{16} assumed to be in the payload.

TWO STAGE SATURN C-2 (CLUSTERED BOOSTER)

Nuclear Test Payload

CENTER OF GRAVITY VS TIME DURING FIRST STAGE BURNING

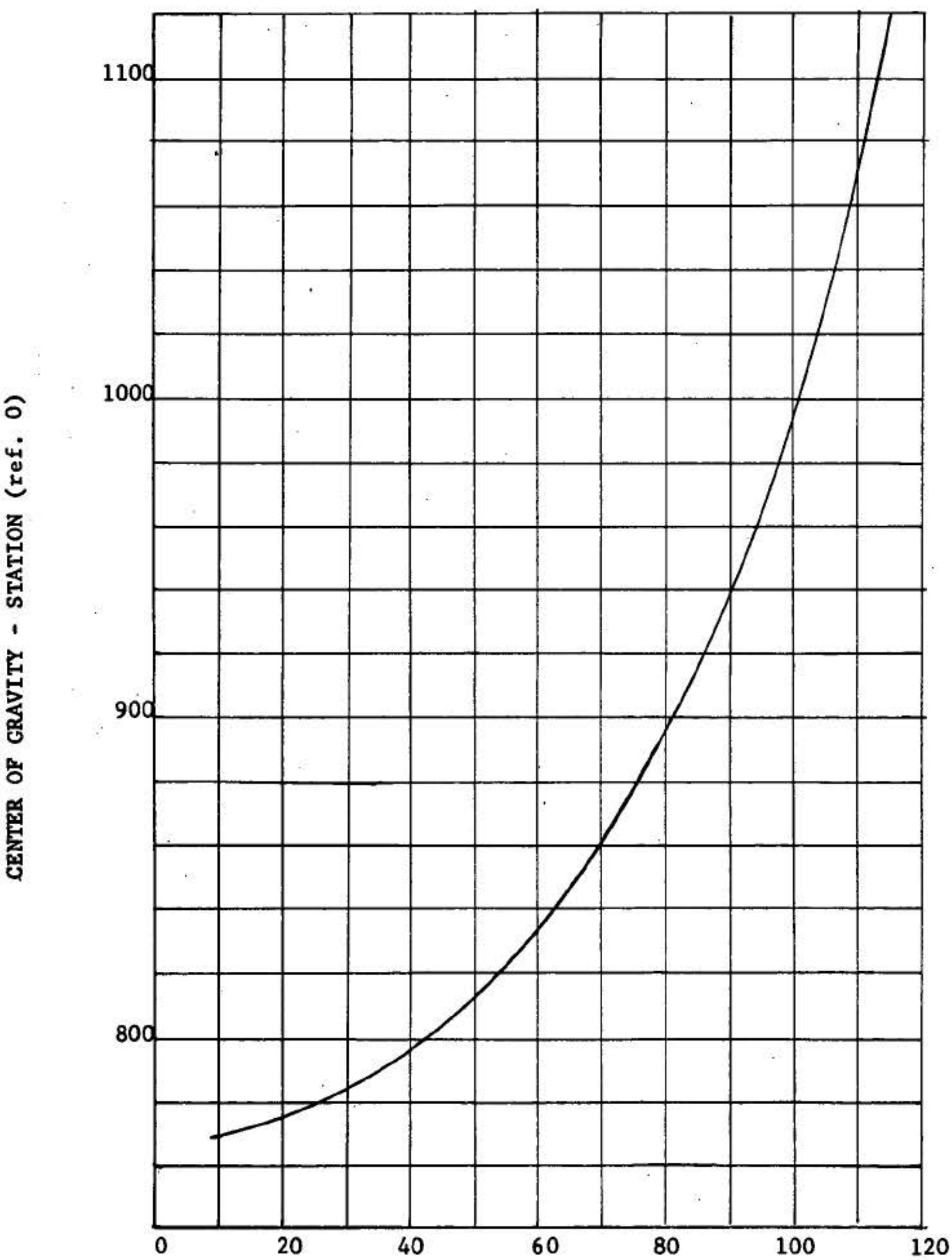


Figure 2-E

TWO STAGE SATURN C-2 (CLUSTERED BOOSTER)

Nuclear Test Payload

PITCH MOMENT OF INERTIA VS TIME DURING FIRST STAGE BURNING

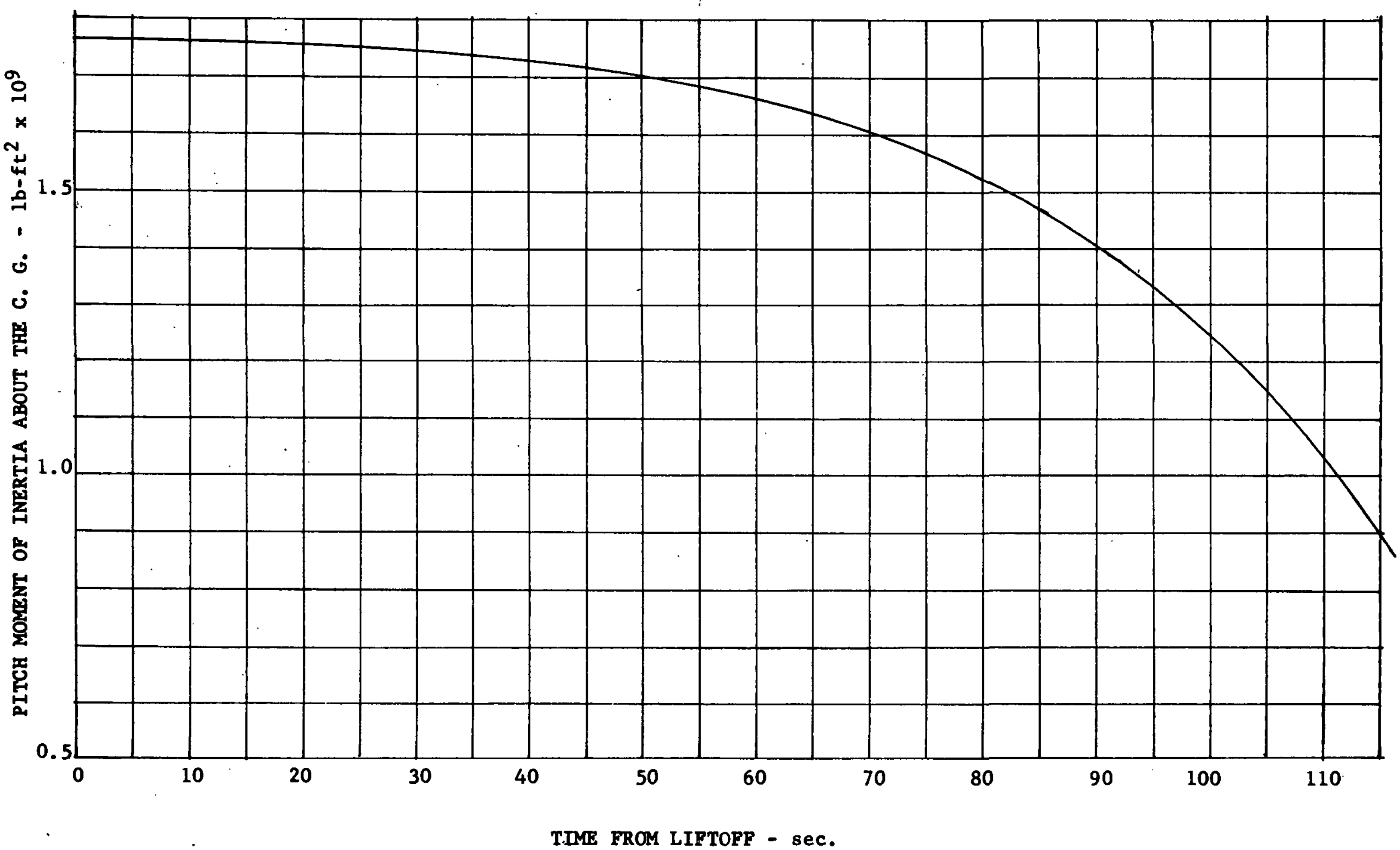
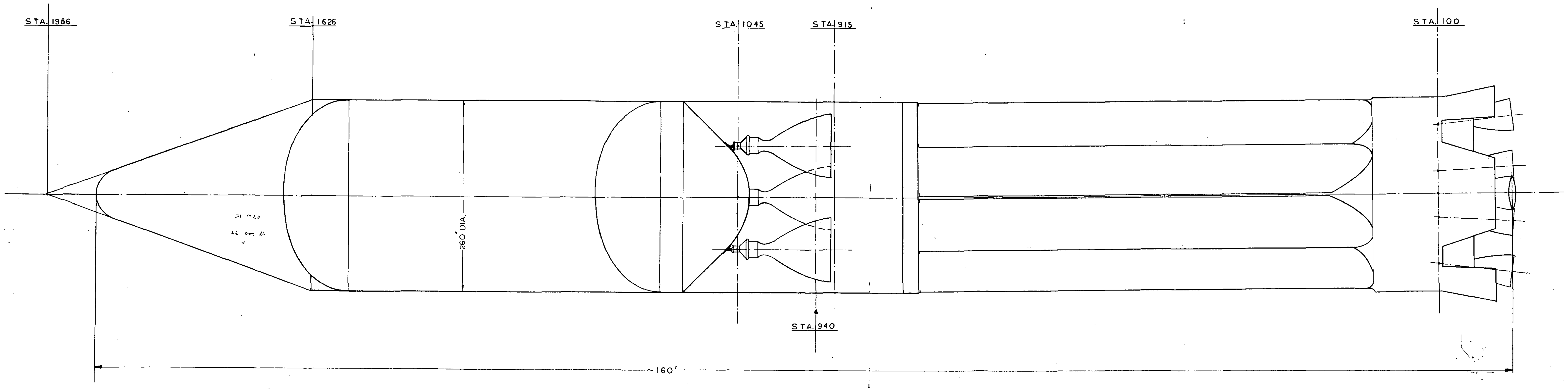


Figure 3-E



SATURN C-2
TWO STAGE-VEHICLE
LOW-ORBIT PAYLOAD
FIG 4E

TABLE II-E

TWO STAGE SATURN C-2 (Clustered Booster)

Low Orbit Payload

SUMMARY WEIGHT AND PROPULSION DATA

Stage	I	II
Engine	H-1	J-2
Propellant	LOX/RP-1	LOX/LH
Thrust, lb	8 x 188K	4 x 200K
I_{sp}, sec.	257 sl	420 vac
Burning Time, sec.	111.07	171.04
Missile Diameter, in.	260	260
$W_{11,15}$, Payload, lb	---	20,000
W_{16}, Guid. Compartment, lb	---	500
W_2, Guid. and Control, lb	2,500	2,000
W_3, Fuselage, lb	52,000	16,390
W_4, Propulsion, lb	22,000	9,630
W_5, Recovery Eq., lb	12,000	---
W_6, Trapped Prop., lb	15,000	3,290
W_7, Usable Residuals, lb	6,500	MRS 1,650 FPR 4,200
W_8, Prop. Consumption, lb	650,000	325,800
$W_{s,16}$, Structure Wt, lb	88,500	28,520
$W_{n,16}$, Structure Net Wt, lb	110,000	37,660
$W_{a,16}$, Stage Wt, lb	760,000	363,460
W_o, Liftoff Wt, lb	1,143,460	383,460
W_c, Cutoff Wt, lb	493,460	57,660
r, Mass Ratio	2.317	6.650
Δu, Charac. Vel. (m/sec)	2274	7792
F_o/W_o	1.31	2.09
F_{vac}/W_c	3.43	13.87

TWO STAGE SATURN C-2 (CLUSTERED BOOSTER)
LOW ORBIT PAYLOAD

CENTER OF GRAVITY VS TIME DURING FIRST STAGE BURNING

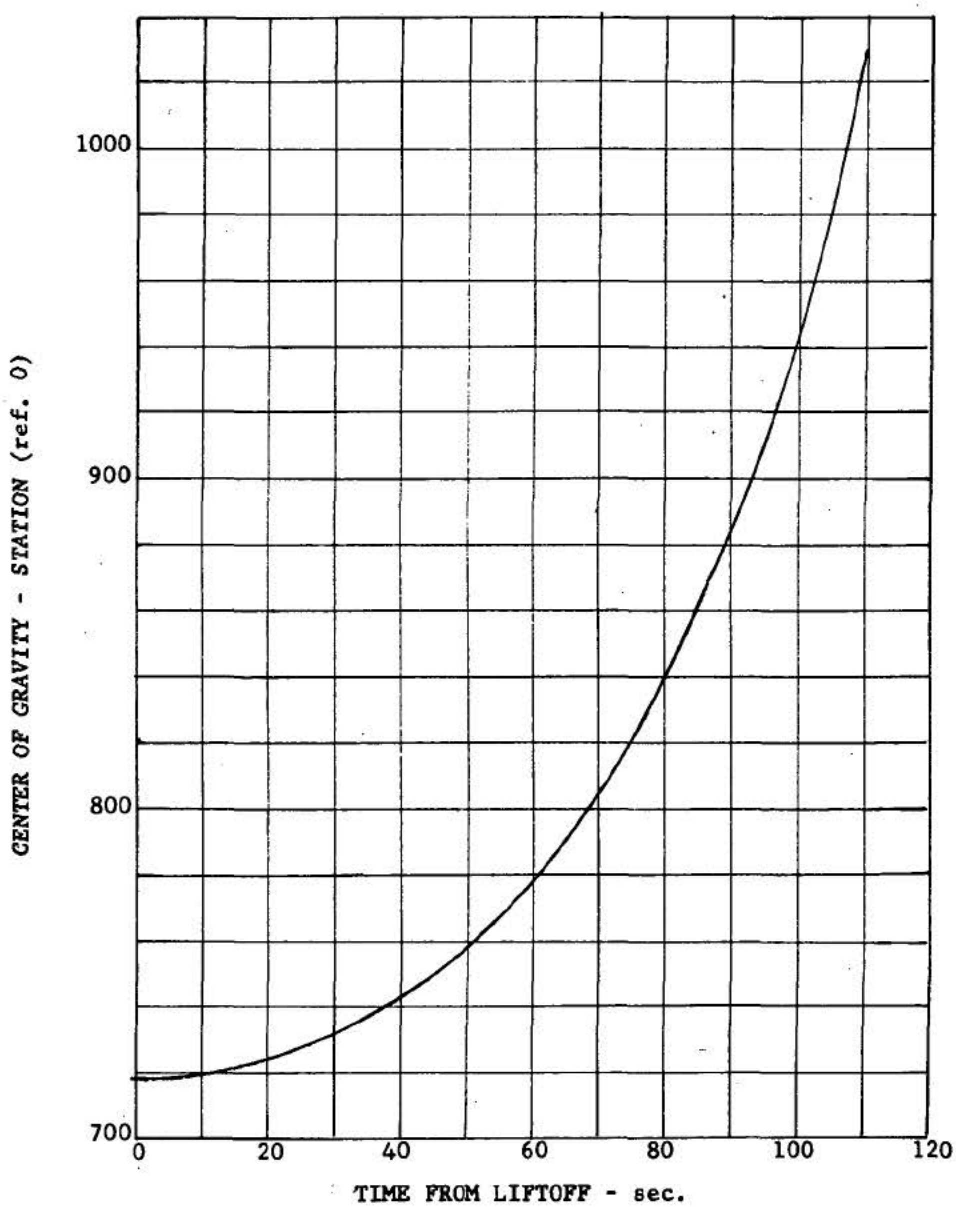


Figure 5-E

TWO STAGE SATURN C-2 (CLUSTERED BOOSTER)

Low Orbit Payload

PITCH MOMENT OF INERTIA VS TIME DURING FIRST STAGE BURNING

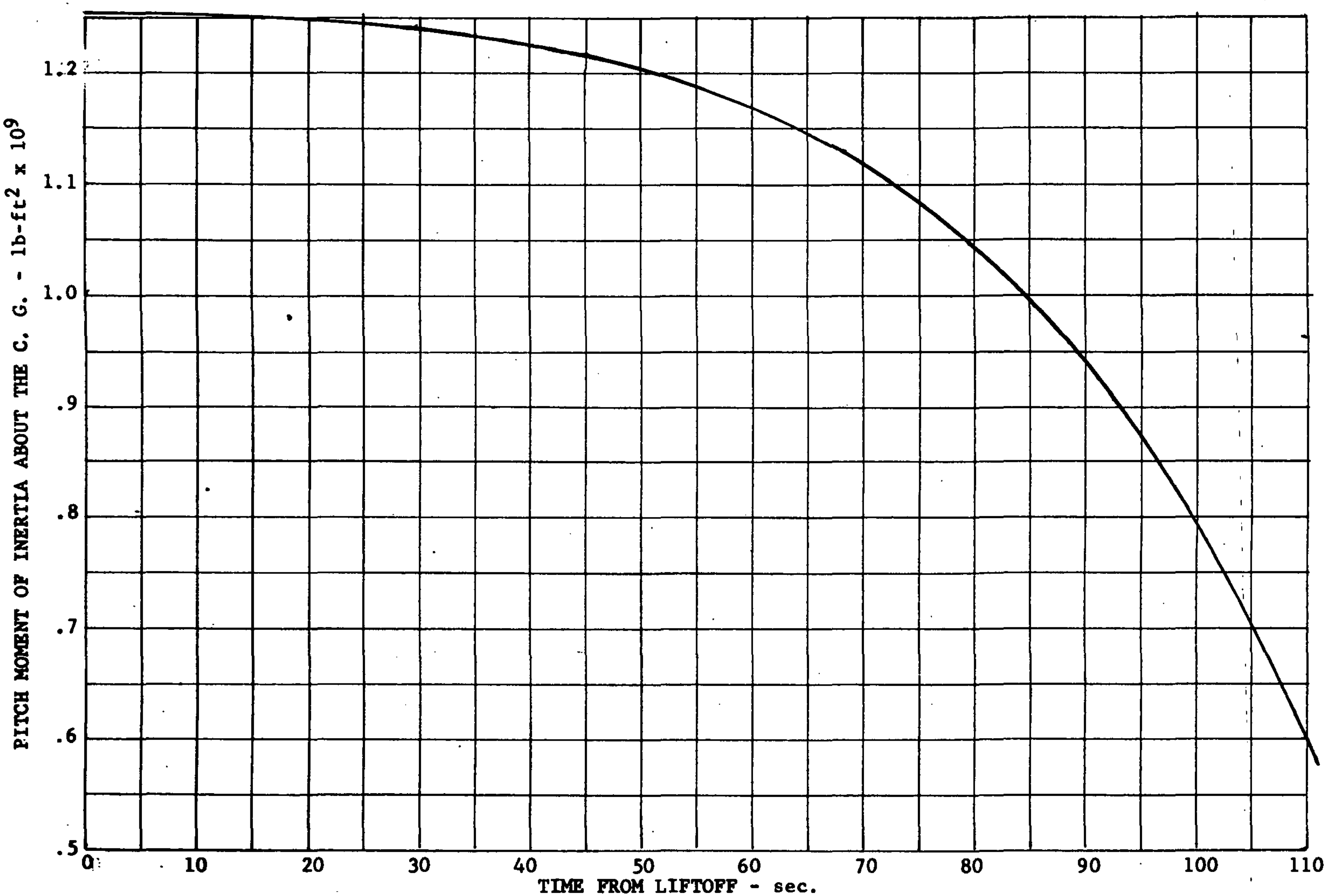
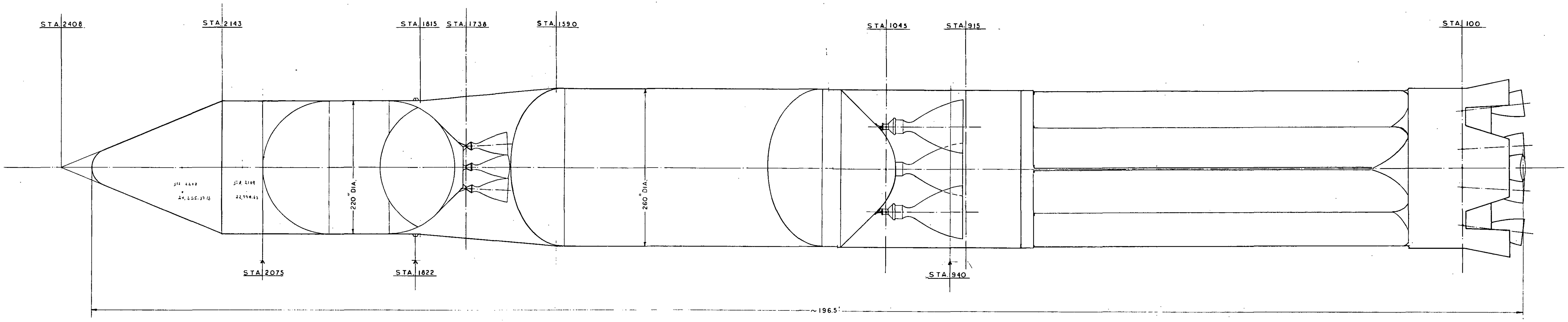


Figure 6-E



SATURN C-2
THREE STAGE-VEHICLE
LOW-ORBIT PAYLOAD
FIG. 7E

TABLE III-E

THREE STAGE SATURN C-2 (Clustered Booster)

Low Orbit Payload

SUMMARY WEIGHT AND PROPULSION DATA

Stage	I	II	III
Engine	H-1	J-2	LR-115
Propellant	LOX/RP-1	LOX/LH	LOX/LH
Thrust, lb	8 x 188K	4 x 200K	4 x 17.5K
I_{sp}, sec	257 sl	420 vac	419.7*vac
Burning Time, sec	102.53	170.62	424.26
Missile Diameter, in.	260	260	220
$W_{11,15}$, Payload, lb	---	---	45,000
W_{16}, Guid. Compartment, lb	---	---	500
W_2, Guid. and Control, lb	2,500	500	2,000
W_3, Fuselage, lb	52,000	16,390**	6,080
W_4, Propulsion, lb	22,000	9,630	2,500
W_5, Recovery Eq., lb	12,000	---	---
W_6, Trapped Prop., lb	15,000	3,290	260
W_7, Usable Residuals, lb	6,000	1,620	MRS 1,350 FPR 4,270
W_8, Prop. Consumption, lb	600,000	325,000	70,780
W_9, Chill-down and Separation, lb	---	---	410***
$W_{s,16}$, Structure Wt, lb	88,500	26,520**	11,080
$W_{n,16}$, Stage Net Wt, lb	109,500	31,430**	16,960
$W_{a,16}$, Stage Wt, lb	709,500	356,430**	87,740
W_o, Liftoff Wt, lb	1,199,080	489,580**	132,740
W_c, Cutoff Wt, lb	599,080	162,960	61,960
r, Mass Ratio	2.002	2.993	2.142
Δu, Charac. Vel. (m/sec)	1878	4508	3131
F_o/W_o	1.25	1.63	0.53
F_{vac}/W_c	2.83	4.91	1.13

* Reduced to account for gas generator propellant flow.

** Includes 1620 lb of second stage insulation jettisoned at 150 sec.

*** Expended during separation and thrust build-up.

THREE STAGE SATURN C-2 (CLUSTERED BOOSTER)
LOW ORBIT PAYLOAD

CENTER OF GRAVITY VS TIME DURING FIRST STAGE BURNING

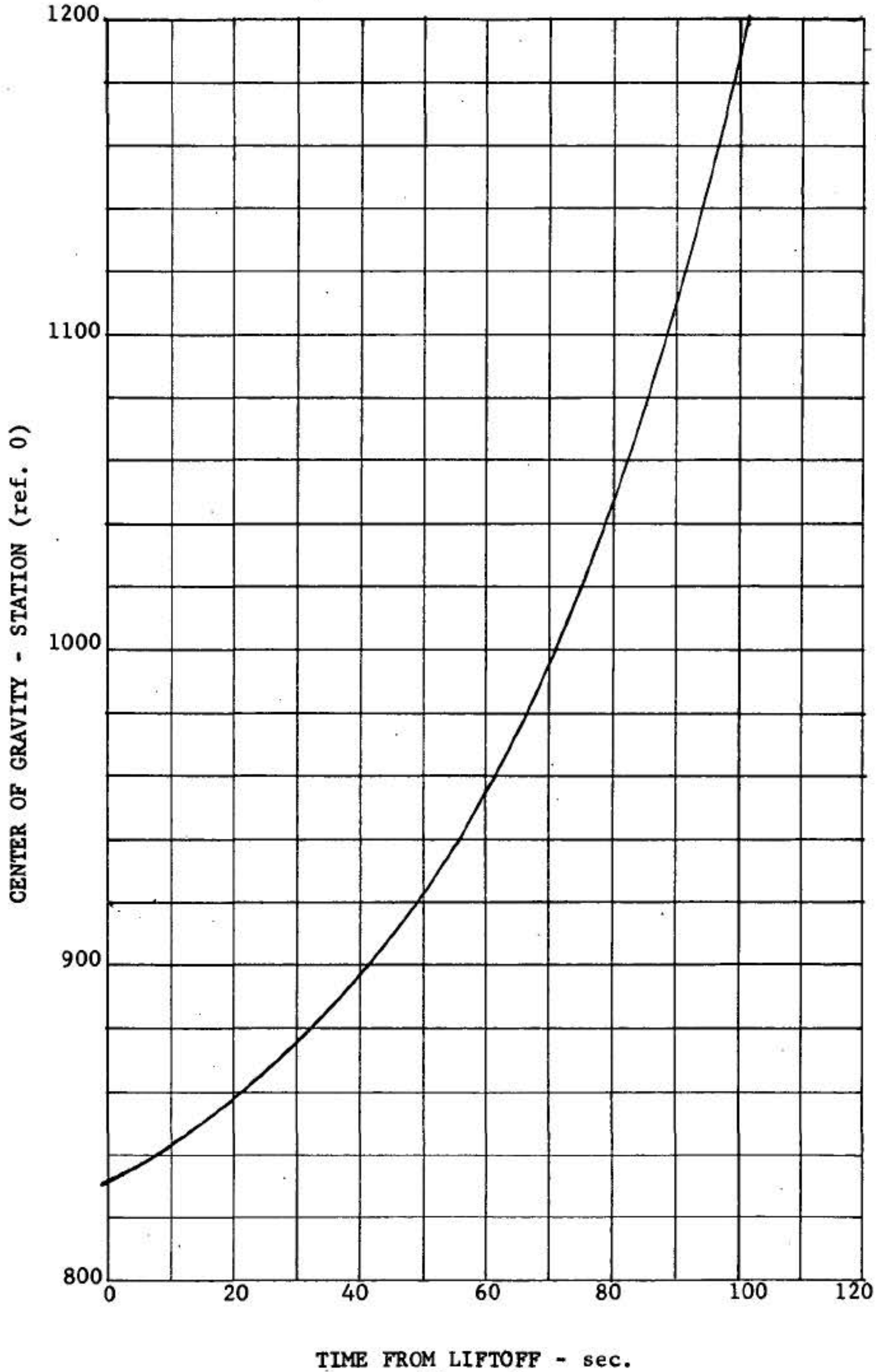


Figure 8-E

THREE STAGE SATURN C-2 (CLUSTERED BOOSTER)
LOW ORBIT PAYLOAD

PITCH MOMENT OF INERTIA VS TIME DURING FIRST STAGE BURNING

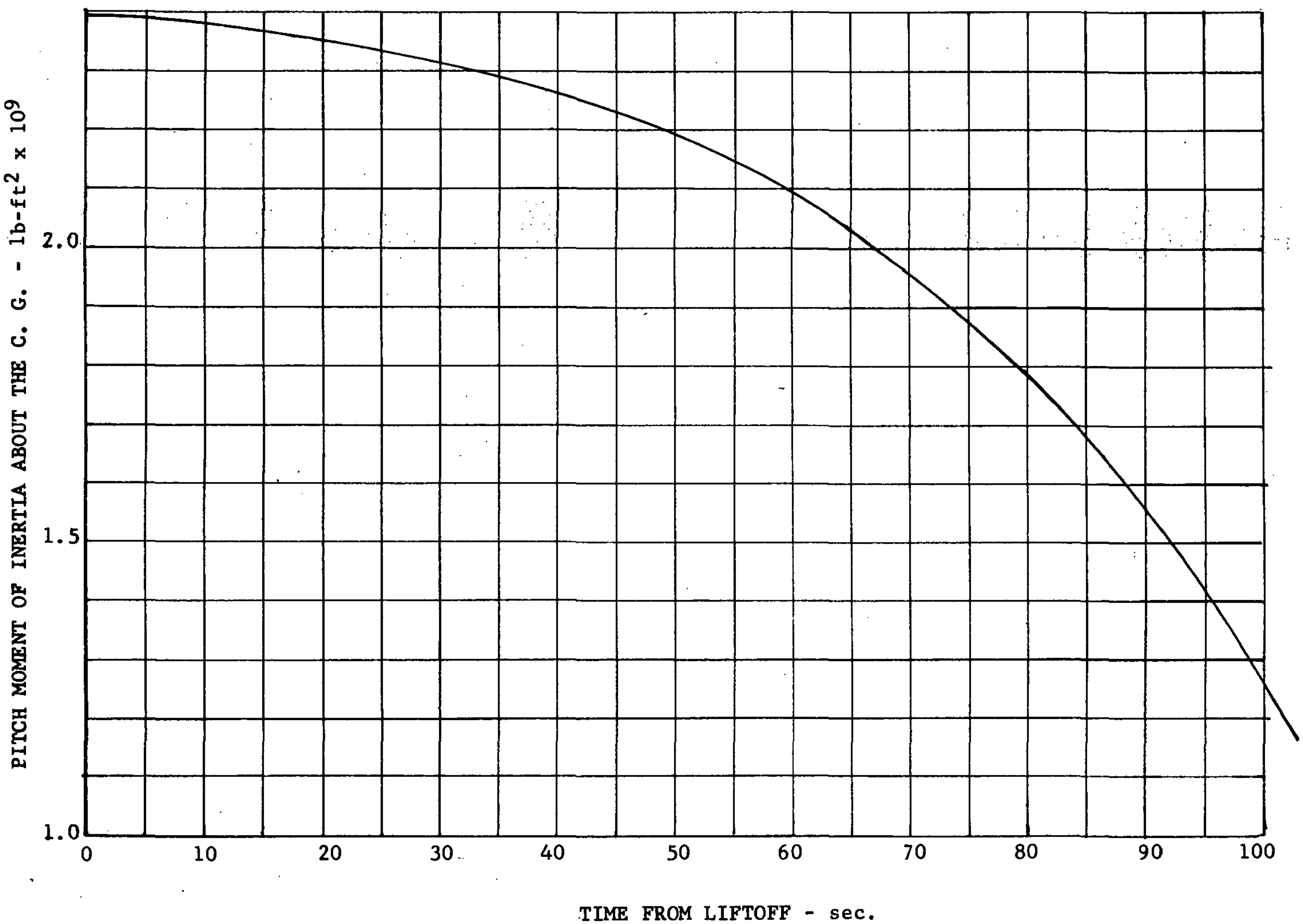
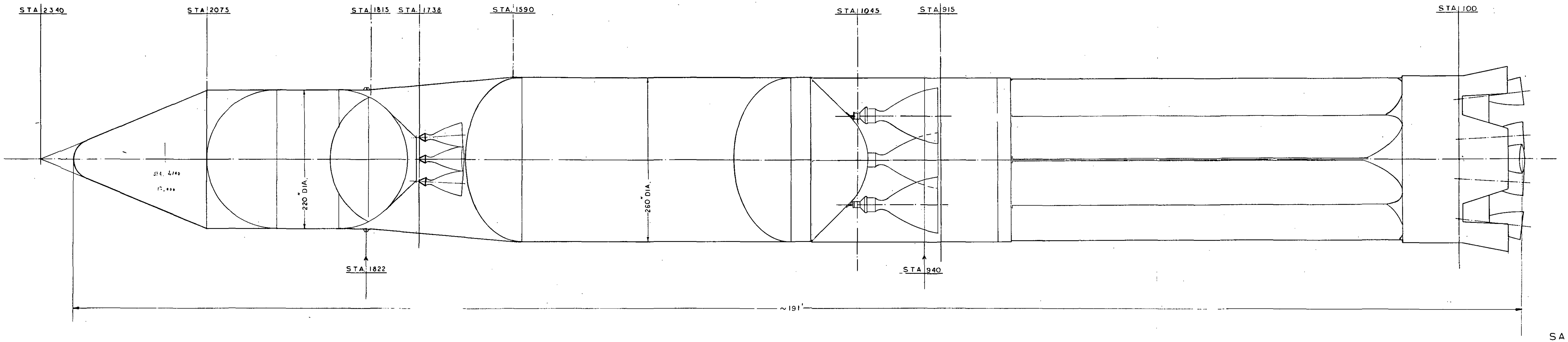


Figure 9-E



SATURN C-2
THREE STAGE-VEHICLE
ESCAPE PAYLOAD
FIG. 10E

TABLE IV-E

THREE STAGE SATURN C-2 (Clustered Booster)

Escape Payload

SUMMARY WEIGHT AND PROPULSION DATA

Stage	I	II	III
Engine	H-1	J-2	LR-115
Propellant	LOX/RP-1	LOX/LH	LOX/LH
Thrust, lb	8 x 188K	4 x 200K	4 x 17.5K
I_{sp}, sec	257 sl	420 vac	419.7 vac
Burning Time, sec	102.53	173.16	583.56
Missile Diameter, in.	260	260	220
$W_{11,15}$, Payload, lb	---	---	15,000
W_{16}, Guid. Compartment, lb	---	---	500
W_2, Guid. and Control, lb	2,500	500	2,000
W_3, Fuselage, lb	52,000	16,390*	6,080
W_4, Propulsion, lb	22,000	9,630	2,500
W_5, Recovery Equipment	12,000	---	---
W_6, Trapped Propellant, lb	15,000	3,290	260
W_7, Usable Residuals, lb	6,000	1,650	MRS 1,350 FPR 2,740
W_8, Prop. Consumption, lb	600,000	329,950	97,330
W_9, Chill-down and Separation, lb	---	---	410**
$W_{s,16}$, Structure Wt, lb	88,500	26,520*	11,080
$W_{n,16}$, Structure Net Wt, lb	109,500	31,460*	15,430
$W_{a,16}$, Stage Wt, lb	709,500	361,410*	113,170
W_o, Liftoff Wt, lb	1,199,080	489,580*	127,760
W_c, Cutoff Wt, lb	599,080	158,010	30,430
r, Mass Ratio	2.002	3.083	4.198
Δu, Charac. Vel. (m/sec)	1878	4631	5896
F_o/W_o	1.25	1.63	0.55
F_{vac}/W_c	2.83	5.06	2.30

* Includes 1620 lb of second stage insulation jettisoned at 150 sec.

** Expended during separation and thrust build-up.

*** Reduced to account for gas generator propellant flow.

THREE STAGE SATURN C-2 (CLUSTERED BOOSTER)
ESCAPE PAYLOAD

CENTER OF GRAVITY VS TIME DURING FIRST STAGE BURNING

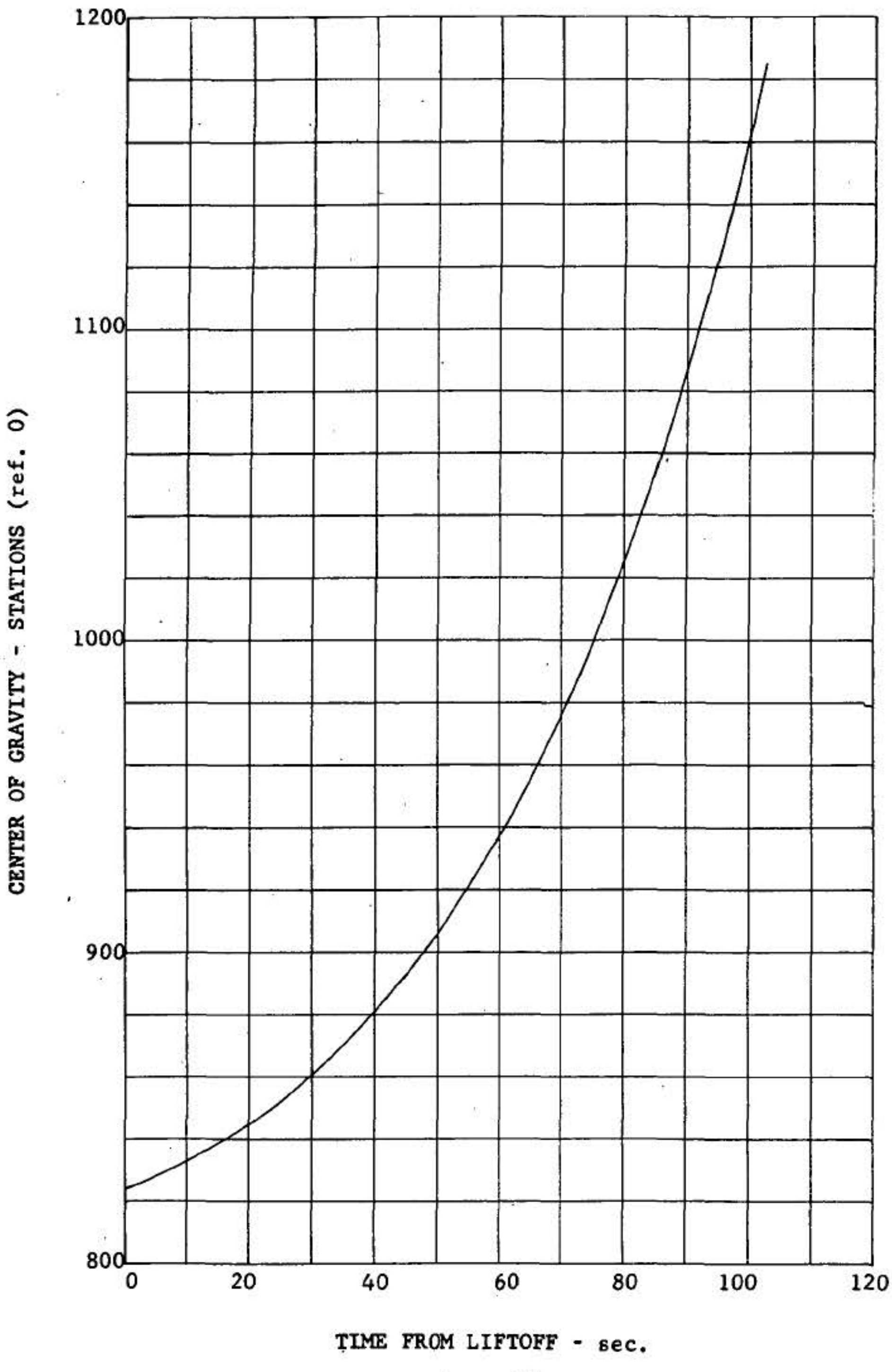


Figure 11-E

THREE STAGE SATURN C-2 (CLUSTERED BOOSTER)
ESCAPE PAYLOAD

PITCH MOMENT OF INERTIA VS TIME DURING FIRST STAGE BURNING

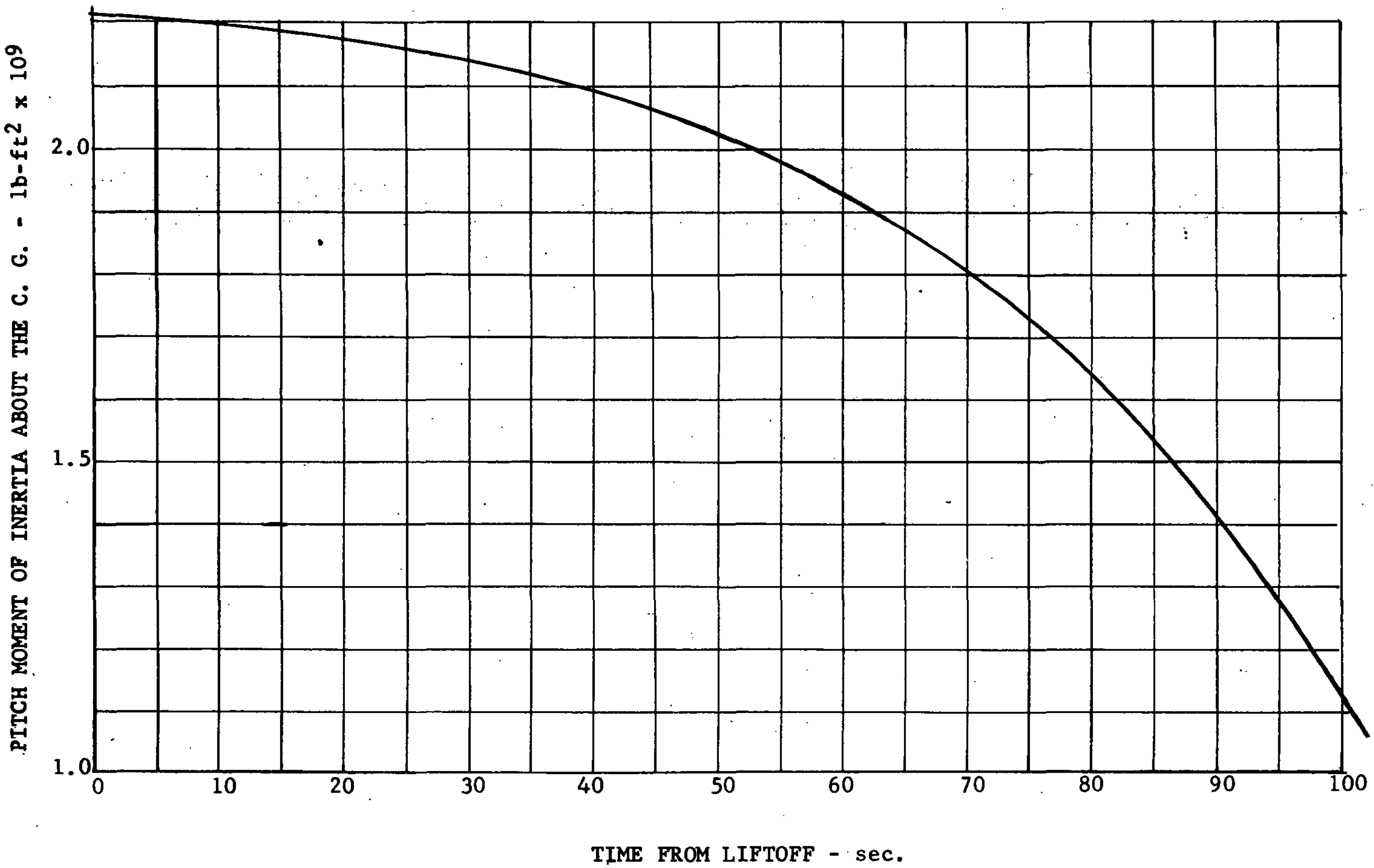
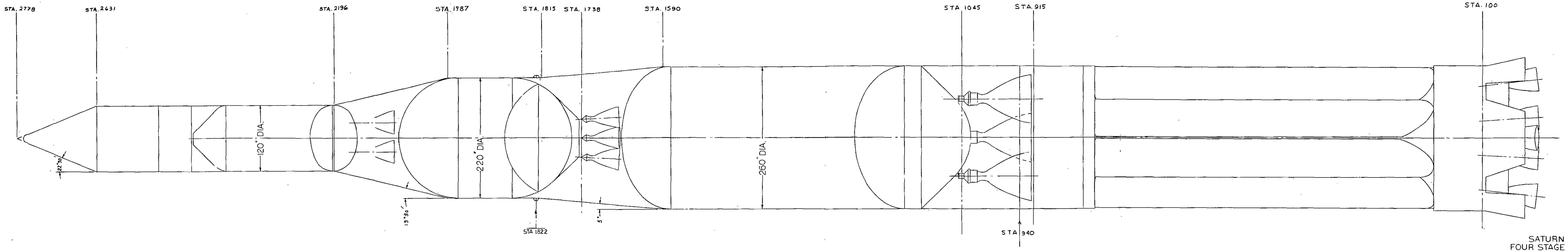


Figure 12-E



SATURN C-2
FOUR STAGE VEHICLE
ESCAPE PAYLOAD
FIG. 13E

TABLE V-E

FOUR STAGE SATURN C-2 (Clustered Booster)

Escape Payload

SUMMARY WEIGHT AND PROPULSION DATA

Stage	I	II	III	IV
Engine	H-1	J-2	LR-115	LR-115
Propellant	LOX/RP-1	LOX/LH	LOX/LH	LOX/LH
Thrust, lb	8 x 188K	4 x 200K	4 x 17.5K	2 x 17.5K
I_{sp}, sec	257 sl	420 vac	420 vac	420 vac
Burning Time, sec	102.53	168.00	425.70	341.04
Missile Diameter, in.	260	260	220	120
$W_{11,15}$, Payload, lb	---	---	---	18,000
W_{16}, Guid. Compartment, lb	---	---	---	500
W_2, Guid. & Control, lb	2,500	500	500	1,500
W_3, Fuselage, lb	52,000	16,390*	6,080	1,500
W_4, Propulsion, lb	22,000	9,630	2,500	1,200
W_5, Recovery Eq., lb	12,000	---	---	---
W_6, Trapped Prop., lb	15,000	3,290	260	440
W_7, Usable Residuals, lb	6,000	1,600	MRS1,030 FPR4,150	MRS 150 FPR 580
W_8, Prop. Consumption, lb	600,000	320,000	70,950	28,420
W_9, Chill-down & Separation	---	---	410**	---
$W_{s,16}$, Structure Wt, lb	88,500	26,520*	9,080	4,700
$W_{n,16}$, Structure Net Wt, lb	109,500	31,410*	14,520	5,870
$W_{a,16}$, Stage Wt, lb	709,500	351,410*	85,880	34,290
W_o, Liftoff Wt, lb	1,199,080	489,580*	137,760	52,290
W_c, Cutoff Wt, lb	599,080	167,960	66,810	23,870
r, Mass Ratio	2.002	2.902	2.062	2.191
Δu, Charac. Vel. (m/sec)	1878	4382	2974	3226
F_o/W_o	1.25	1.63	0.51	0.67
F_{vac}/W_c	2.83	4.76	1.05	1.47

* Includes 1620 lb of second stage insulation jettisoned at 150 sec.

** Expended during separation and thrust build-up.

FOUR STAGE SATURN C-2 (CLUSTERED BOOSTER)
ESCAPE PAYLOAD

CENTER OF GRAVITY VS TIME DURING FIRST STAGE BURNING

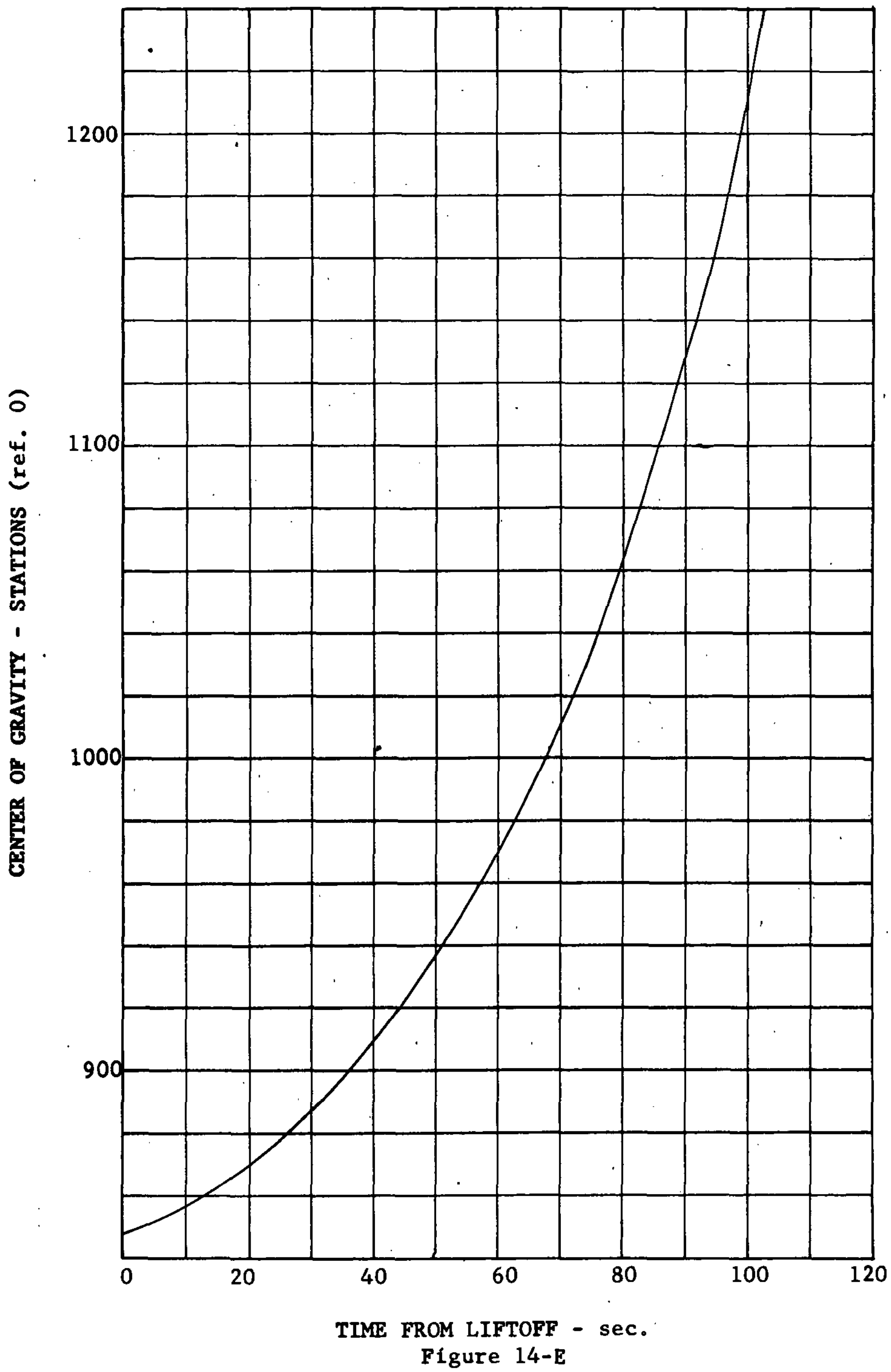


Figure 14-E

FOUR STAGE SATURN C-2 (CLUSTERED BOOSTER)
ESCAPE PAYLOAD

PITCH MOMENT OF INERTIA VS TIME DURING FIRST STAGE BURNING

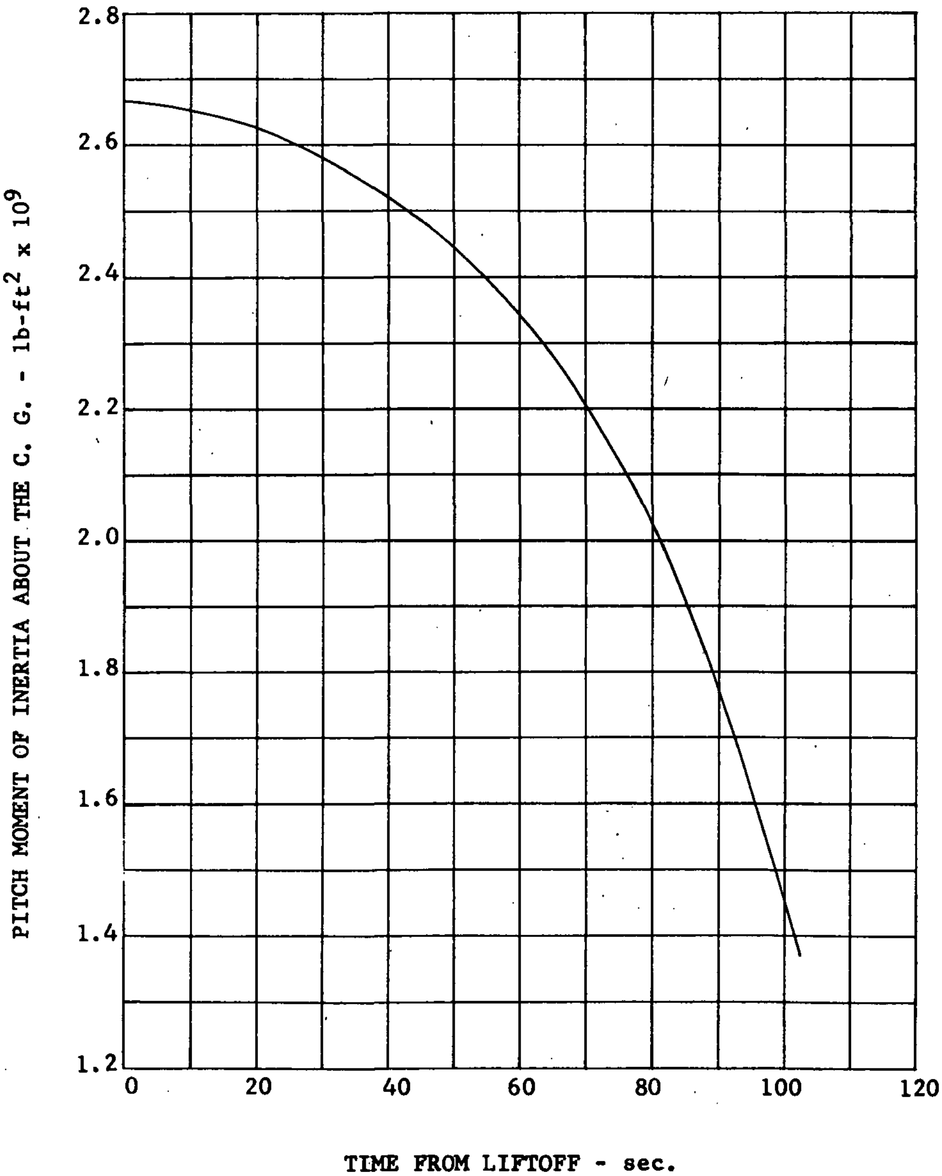


Figure 15-E

FOUR STAGE SATURN C-2 (CLUSTERED BOOSTER)
ESCAPE PAYLOAD

ROLL MOMENT OF INERTIA VS TIME DURING FIRST STAGE BURNING

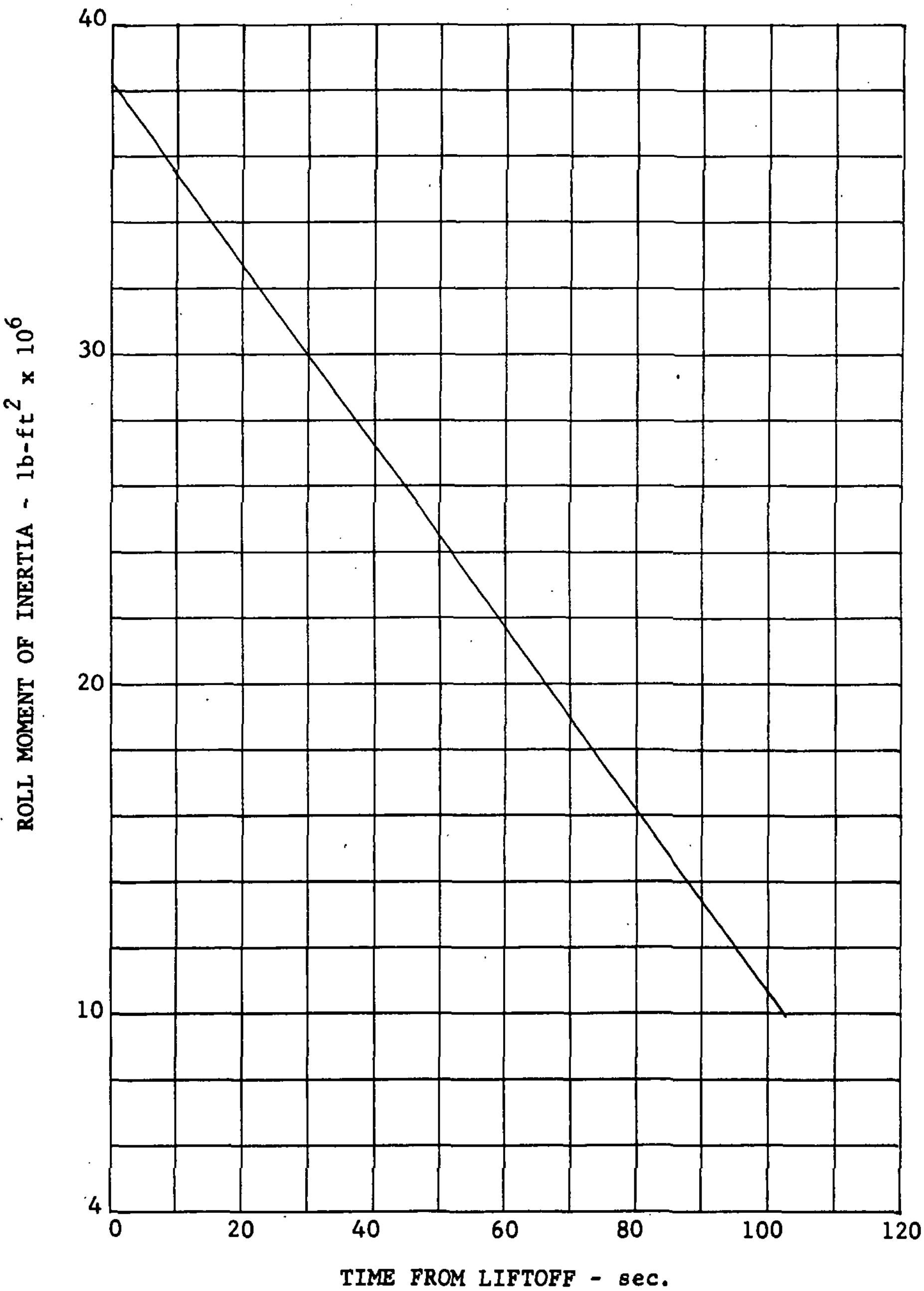
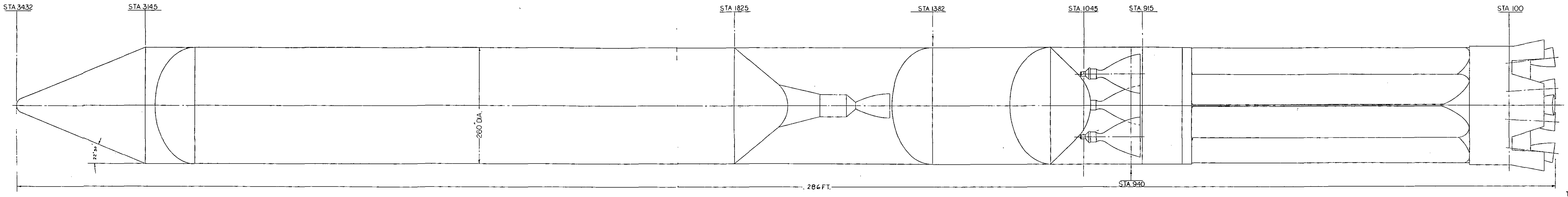


Figure 16-E



SATURN D-2
THREE STAGE VEHICLE
NUCLEAR THIRD STAGE
ESCAPE PAYLOAD
FIG. 17E

TABLE VI-E

THREE STAGE SATURN D-2 (Clustered Booster)

Nuclear Third Stage and Escape Payload

SUMMARY WEIGHT AND PROPULSION DATA

Stage	I	II	III
Engine	H-1	J-2	4000 MW
Propellant	LOX/RP-1	LOX/LH	LH
Thrust, lb	8 x 188k	4 x 200k	200k
I_{sp}, sec	257 (sl)	420 (vac)	800 (vac)
Burning Time, sec	102.53	107.73	674
Missile Diameter, in.	260	260	260
$W_{11,15}$, Payload, lb	---	---	45,400
W_{16}, Guid, Compartment, lb	---	---	500
W_2, Guid. and Control, lb	2,500	500	2,000
W_3, Fuselage, lb	52,000	16,920*	22,850
W_4, Propulsion, lb	22,000	9,630	10,000
W_5, Recovery Equipment, lb	12,000	---	---
W_6, Trapped Propellant, lb	15,000	2,320	840
W_7, Usable Residuals, lb	6,000	1,030	3,890
W_8, Propellant Consumption, lb	600,000	205,200	168,500
$W_{s,16}$, Structure Weight, lb	88,500	27,050*	35,350
$W_{n,16}$, Structure Net Weight, lb	109,500	30,400*	40,080
$W_{a,16}$, Stage Weight, lb	709,500	235,600*	208,580
W_o, Liftoll Weight, lb	1,199,080	489,580*	253,980
W_c, Cutoff Weight, lb	599,080	283,350	85,480
r, Mass Ratio	2.002	1.724	2.971
Δu, Characteristic Velocity (m/sec)	1878	2239	8530
F_o/W_o	1.25	1.63	0.79
F_{vac}/W_c	2.83	2.82	2.34

* Includes 1030 lb of second stage insulation jettisoned at 150 sec.

THREE STAGE SATURN D-2 (CLUSTERED BOOSTER)
NUCLEAR THIRD STAGE WITH ESCAPE PAYLOAD

CENTER OF GRAVITY VS TIME DURING FIRST STAGE BURNING

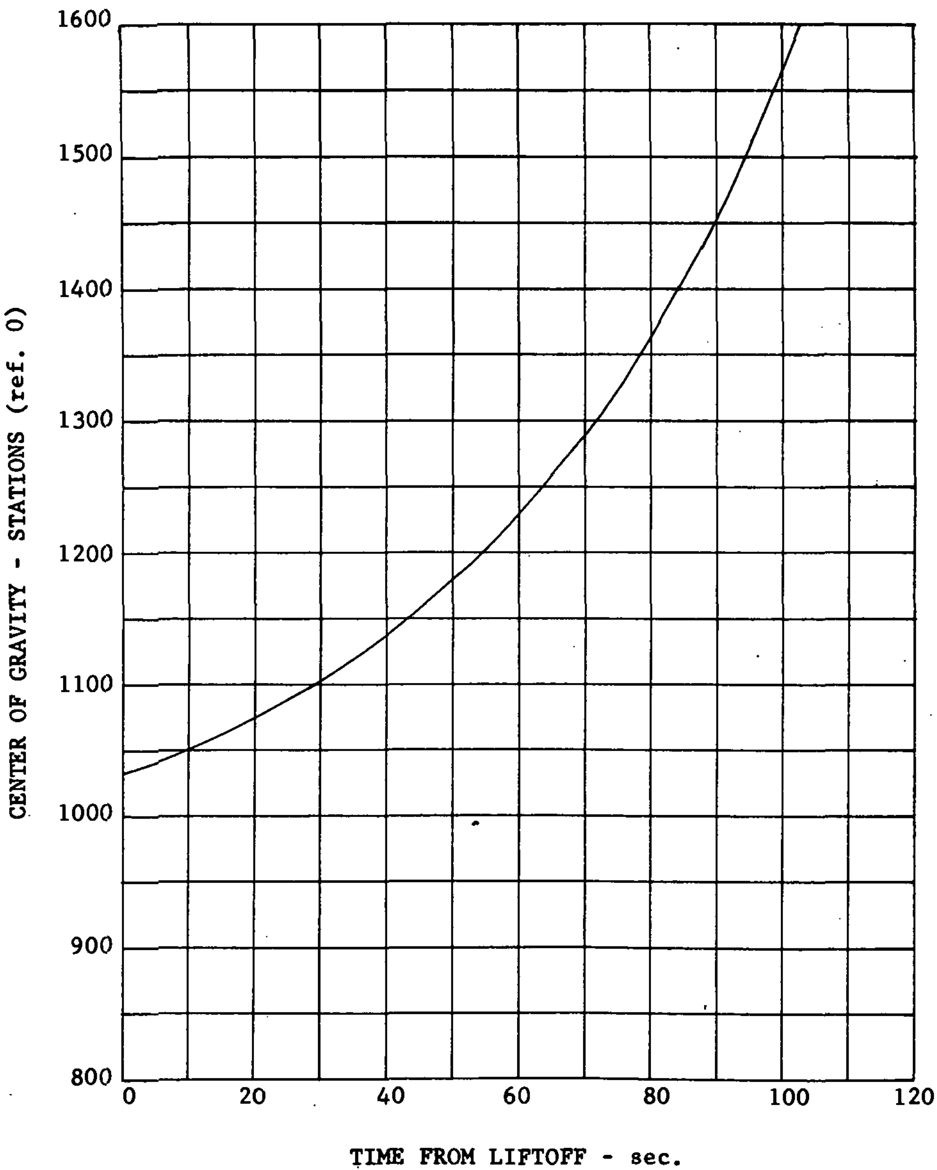


Figure 18-E

THREE STAGE SATURN D-2 (CLUSTERED BOOSTER)
NUCLEAR THIRD STAGE WITH ESCAPE PAYLOAD

PITCH MOMENT OF INERTIA VS TIME DURING FIRST STAGE BURNING

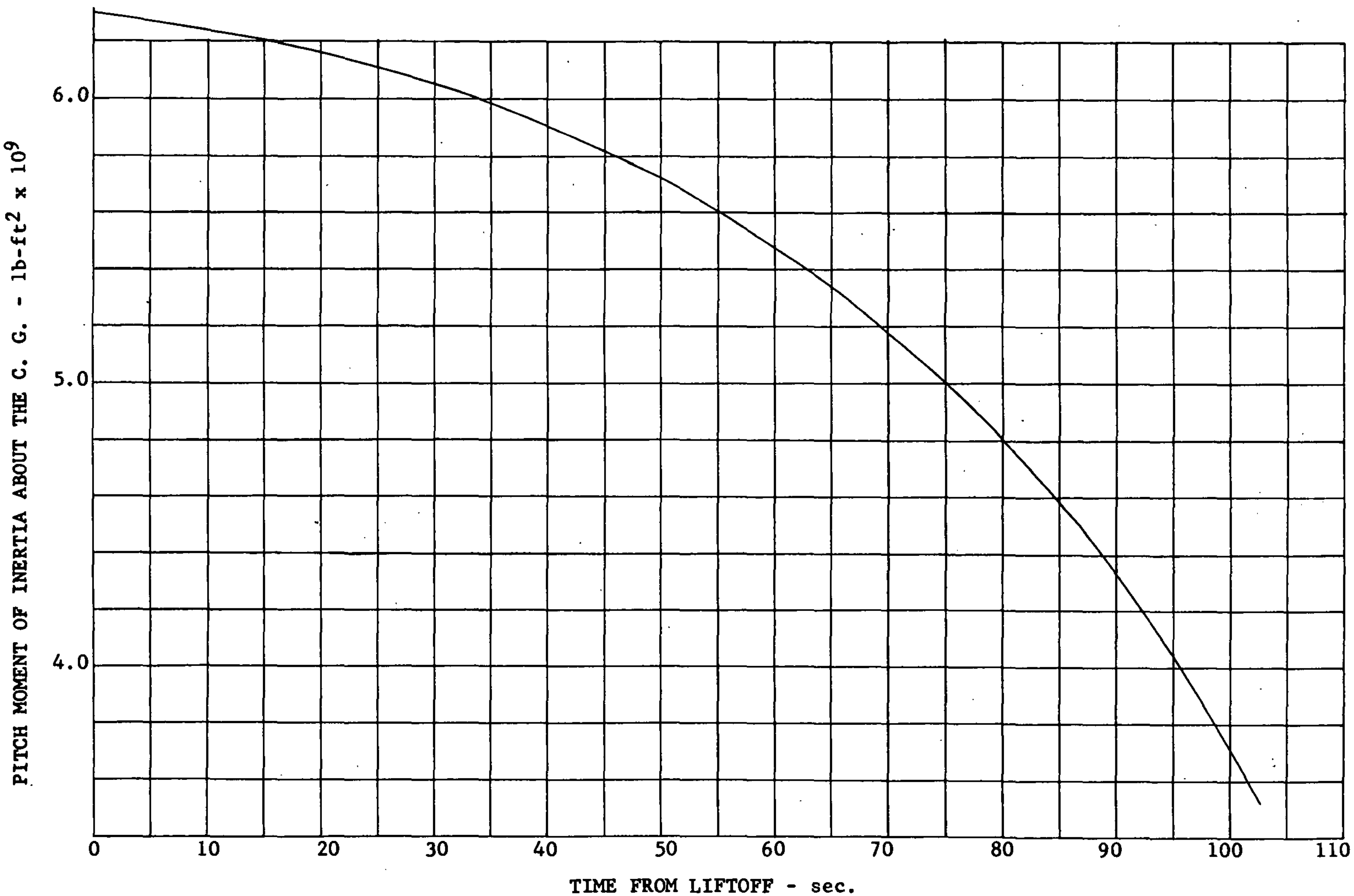


Figure 19-E

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