AD 328 952

Reproduced by the

ARMED SERVICES TECHNICAL INFORMATION AGENCY
ARLINGTON HALL STATION
ARLINGTON 12, VIRGINIA

CONFIDENTIAL

CONFIDENTIAL

Copy No.

Issue Date: 18 April 1962

(Unclassified Title)

DESIGN STUDY OF A LARGE UNCONVENTIONAL LIQUID PROPELLANT ROCKET ENGINE AND VEHICLE

Prepared by

AEROJET-GENERAL CORPORATION
Liquid Rocket Plant
Sacramento 9, California

Final Report
Report No. LRP 257

Volume 5: Advanced Engine-Vehicle Integration Study
(The Boeing Company)

Contract NAS 5-1025

Prepared for

OFFICE OF LIQUID ROCKETS
NASA HEADQUARTERS
Code MLPL (Mr. H. Burlage)
400 Maryland Avenue, S. W.
Washington 25, D. C.

EXCLUDED FROM AUTOMATIC REGRADING; DOD DIR 5200.10 DOES NOT APPLY

Aerojet-General CORPORATION
LIQUID ROCKET PLANT • SACRAMENTO, CALIFORNIA
THE GENERAL TIRE

CONFIDENTIAL

CONFIDENTIAL

GROUP 4
DOWNGRADED AT 3 YEAR INTER-
VALS; DECLASSIFIED AFTER
12 YEARS.
DOD DIR 5200.10

FINAL REPORT

ADVANCED ENGINE - VEHICLE

INTEGRATION STUDY

FOR REFERENCE ONLY AT EACH OF THE ASTIA OFFICES. THIS REPORT CANNOT BE SATISFACTORILY REPRODUCED; ASTIA DOES NOT FURNISH COPIES.

for

AEROJET GENERAL CORPORATION

Purchase Order A290298

August 25, 1961

THE BOEING COMPANY
AERO SPACE DIVISION
SEATTLE, WASHINGTON

DOCUMENT D2-12072

US 4070 7080 (WAS BAC 1946 P-R3)

CONFIDENTIAL

FOREWORD

This document contains the results of airframe studies conducted by the Boeing Company in fulfillment of Aerojet General Corporation Purchase Order A290298. The studies were conducted over a period ending Aug. 25, 1961, in support of Aerojet General Corporation work on Task I of the NASA GS-1541 study. The Aerojet General Corporation work was conducted under NASA Contract Number NAS 5-1025. As such, the contents of this document supplement that contained in Aerojet General Corporation Document No AGC LRP 234.

NOTICE

TABLE OF CONTENTS

1.0 SUMMARY

1.1 SCOPE

The end point objective of the study covered herein was directed toward determining the effects of advanced engine design concepts on the cost performance parameter (dollars per pound of payload) of a total airborne vehicle and ground support system. ~~Major~~ emphasis was placed on use of a 2 million [redacted] pounds sea level force deflection (F-D) engine, ~~chosen from several potential engine designs developed by the Aerojet General Corporation~~. This engine was used in two basic vehicle configurations:

~~Model 900-1~~: A two stage vehicle with a thrust to weight ratio (T/W) of 1.1 and using the F-D engine in both stages operating at Po of 1000 psi; and

~~Model 900-4~~: A single-stage-to-orbit vehicle with a T/W of 1.4 and using one F-D engine operating at Po of 3000 psi.

Emphasis was placed on the engine installation, the engine influence on connecting subsystems, and the engine mounting structure. ~~The performance of the Model 900-1 and Model 900-4 vehicles and the cost of the vehicles and support systems were developed. These data were compared with like design data for baseline vehicles using conventional bell type engines of the same thrust level and LO_2/LH_2~~ [redacted] ~~and LO_2/RP-1~~ [redacted] ~~propellants~~. This was accomplished on the basis of comparative cost performance including predicted reliability effects for launch rates of 25, 100 and 400 over a six year period.

[redacted]

1.1 Cont.

Conventional tankage arrangements using semi-monocoque structure were used for each of the four vehicle designs. The design in each case was based around "man rated" criteria with neutral stability required.

In addition, preliminary investigations were made to determine the potential performance of the Model 902-4 (single stage) when operating as a two-stage vehicle.

The payload and cost performance of the study vehicle systems were found to be as follows:

VEHICLE	ENGINE TYPE	PROPELLANT	PAYLOAD	COST PERFORMANCE - $/#* LAUNCH RATE 25/6 yrs	100/6 yrs	400/6 yrs
902-1 (2 stage Baseline)	Bell Pc=1000 psi	LO_2/LH_2	129,900	$661	$155	$ 67
902-2 (2 Stage Baseline)	Bell Pc=1000 psi	LO_2/RP-1	59,700	$1224	$287	$126
902-3 (2 Stage)	P-D Pc=1000 psi	LO_2/LH_2	134,400	$618	$145	$ 62
902-4 (1 Stage) T/W_o=1.4	P-D Pc=3000 psi	LO_2/LH_2	113,200	$498	$125	$ 54
902-4A (2 Stage) T/W_o=1.4	P-D Pc-3000 psi	LO_2/LH_2	143,600	**	**	**

* Includes estimated cumulative system reliability.

** 902-4A not costed due to time limitation.

1.1 Cont.

The above reflects a 6.5% to 24.5% cost performance gain for the vehicle using the advanced F-D engines and LO_2/LH_2 propellants. This is attributed primarily to the estimated higher performance and the compatible thrust structure installation features offered by the F-D engine. From the standpoint of the airframe and the supporting system, no major problem areas were determined that would influence decisions regarding future consideration of the F-D engine.

1.2 RECOMMENDATIONS

It is recommended that the potential of the Model 902-4 single-stage to orbit vehicle, or variations thereof, be evaluated more thoroughly. From the quantitative standpoint, this configuration offers good comparative cost performance. In addition, it offers very desirable "no-fallout during launch" characteristics. Further, the use of this basic vehicle with other programmed upper stages should provide an economical method of achieving versatility.

It is further recommended that the practice of considering potential vehicles in parallel with investigation of future engine designs, be continued. The more significant interface problems can be established and resolved early, thereby reducing potential redesign requirements to a minimum.

2.0 STUDY OBJECTIVES

The prime objective of this study was to determine the relative merits of advanced engine concepts over conventional engine design where the engines are considered as an element of the total vehicle and supporting system. The primary comparison was to be based on the net effect of dollars per pound of payload in a 300 n. mi. orbit as influenced by Research and Development and hardware costs and the reliability and performance of the resulting total vehicles. This objective was to be pursued considering both the conventional and advanced engines when used with nominally conventional airframe design.

A secondary objective was to provide a conceptual review of potential advanced engine concepts when used with conceptual nonconventional airframe designs.

3.0 INTRODUCTION

3.1 GENERAL

Two-fold benefits are derived by analyzing potential new engine concepts in parallel with applicable airframes, as was done on a preliminary basis in this program.

The true net effect of the engine on total system ($/#) cost parameter is more evident than when the engine only is considered. Important interfaces exist between the engine and the airframe, that can be studied to the mutual design benefit of both.

Many potential design penalties can, thereby, be circumvented by considering the design of both early, rather than waiting and making the airframe "line" with a frozen engine design.

3.2 STUDY APPROACH

To meet the major objective of the study, as noted in Section 2.0, the following preliminary analytical and design efforts were completed:

Two conventional two-stage vehicles were developed. These used 2.0×10^6 pound sea-level thrust bell type engines on the first stage and optimized upper staging. The first used liquid oxygen (LO_2) and liquid hydrogen (LH_2); the second LO_2 and RP-1 fuel in both stages. Costs of these vehicles for production rates of 25, 100, and 400 over a six year period, their supporting system and the required research and development were determined. This was accomplished on the basis of $/# using predicted vehicle payload performance, and was used as the baseline to which similar data for vehicles using advanced engines was compared.

3.2 Cont.

In cooperation with Aerojet General several advanced engine concepts developed by Aerojet were reviewed from the standpoint of predicted weight, performance, cost, reliability, and installation characteristics. The engines considered and applicable characteristics are shown by table 8.1. More detailed information is provided in Aerojet General Document reference 15.3.

The Aerojet General force deflection engine (F-D) was selected for preliminary design into a two-stage and a single-stage to orbit vehicle. Both vehicles used LO_2/LH_2 propellants. The F-D engine used on the two stage vehicle operated at a P_c = 1000 psi, while the single stage used a P_c = 3000 psi.

Several design approaches for installation of the advanced F-D engine were developed. These were analyzed and the best from the standpoint of the engine and vehicle was chosed for weight, connecting subsystem and performance analysis.

Cost data was developed for both advanced vehicles using the F-D engines. This provided a basis for comparison with the conventional baseline vehicles.

Potential advanced vehicle concepts were developed to a limited degree. Various non-conventional vehicle arrangements using non-conventional engines were reviewed primarily from a qualitative standpoint.

It was desirable to concentrate on the advanced engine-vehicle aspect

3.2 Cont.

of the study. To achieve this, data developed previously by Boeing under Air Force Contract AF 04(611)-5970 "Advanced Propulsion System (APS) Study were relied upon for much of the conventional baseline vehicle work. Results of that work are contained in reference 152. To achieve good comparative data, the advanced engine-vehicles portion was also analyzed to the same assumptions and ground rules as the APS and baseline vehicle studies. The performance and cost anal included herein should be considered as applicable to the vehicles also covered herein. Such data when used for comparison with other studies must be corrected where the effect of different ground rules would be significant.

4.0 STUDY VEHICLE CONFIGURATIONS

4.1 GENERAL

A comparison of the physical size of the four engine-vehicle configurations that were developed during this study is shown by figure 4.1. These are essentially conventional airframe arrangements, to which two types of engine (Bell and Force Deflection) were applied. Other, non-conventional airframe arrangements with various engine types were considered briefly and are discussed in Section 14.0, "Unconventional Arrangements".

Basic criteria that influenced development of the study configurations are as follows:

A. Mission - 300 N.M. orbit - Easterly launch at Cape Canaveral

B. First Stage Thrust - 2×10^6 pound (Sea Level)

C. Man Rated

D. Neutral Stability Required

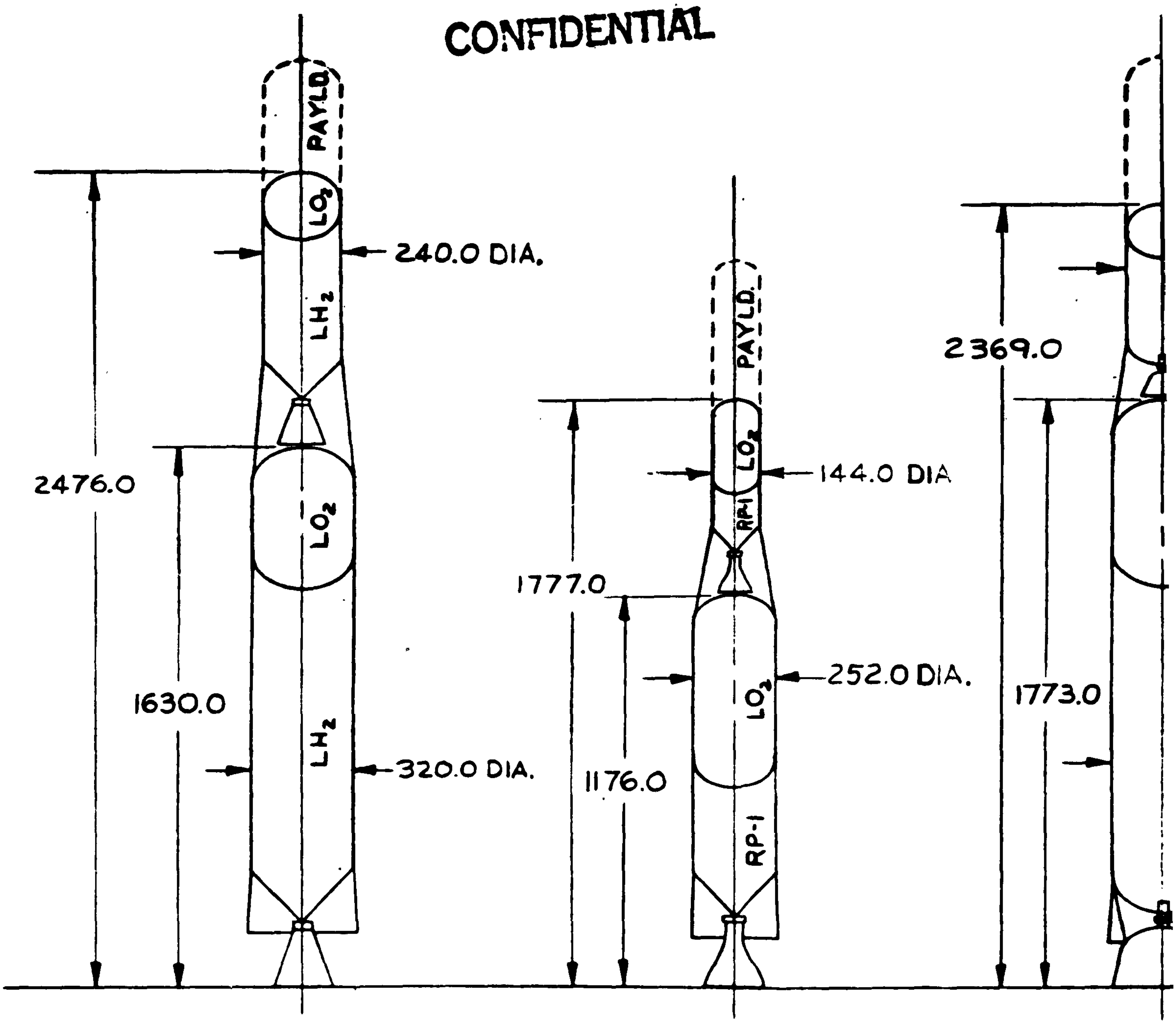
E. Self supporting on the launch pad, including condition with bottom tank empty and unpressurized with upper tanks full.

The performance, structural and subsystem criteria, weights and comparative economic analysis of the vehicles and support systems are covered in separate sections. General vehicle configuration descriptions are presented in the following paragraphs.

4.2 BASELINE LO_2/LH_2 VEHICLE (MODEL 902-1)

The general arrangement and principal design criteria for model 902-1 are shown by figure 4.2. Model 902-1 is conventional in concept. It was used directly to establish a factor for relating this report

CONFIDENTIAL



MODEL 902-1 MODEL 902-2

BASELINE VEHICLES
(CONVENTIONAL ENGINES)

MODEL

UNCC

1

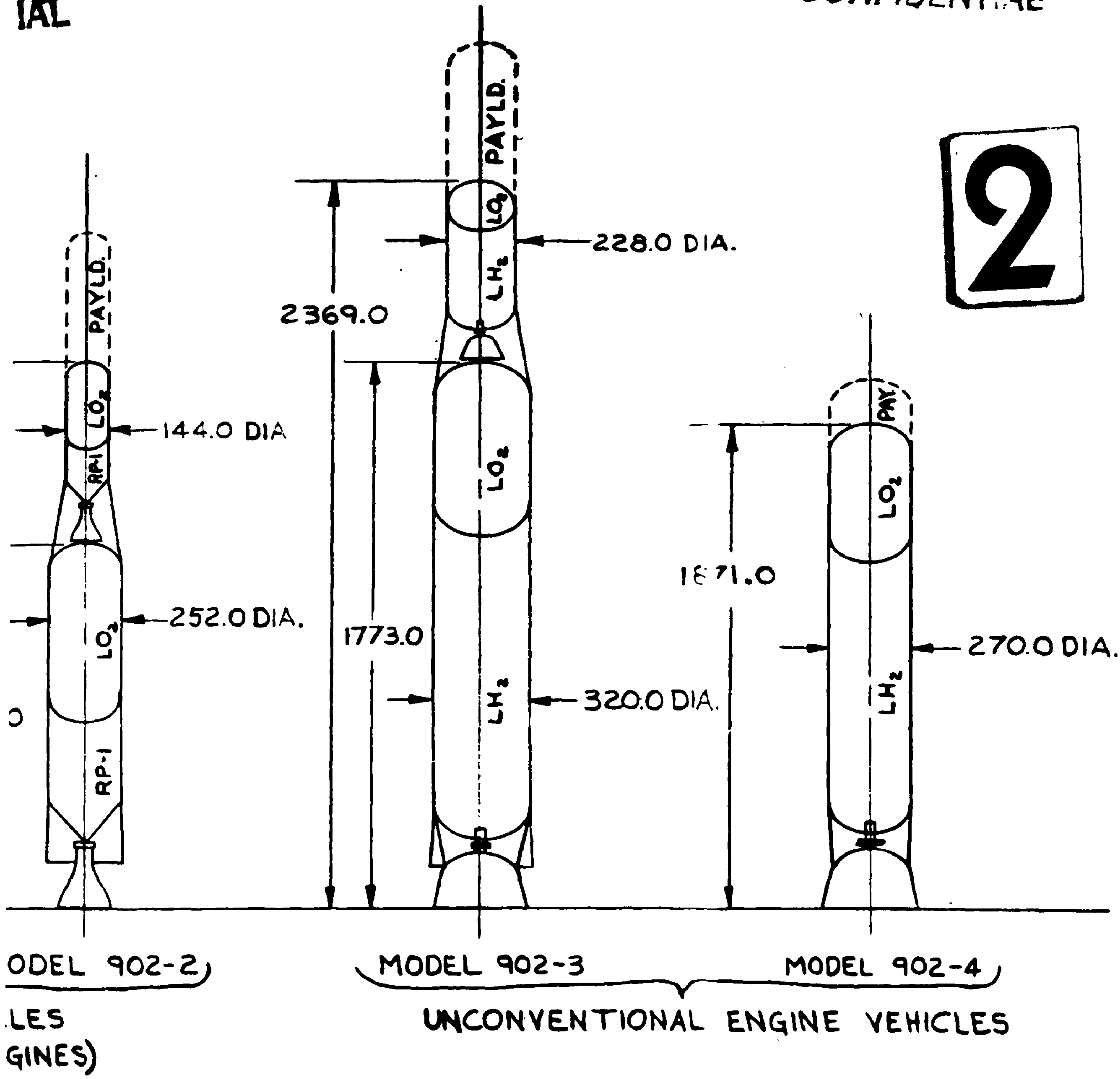
CALC	
CHECK	
APR.	
APR	
DRN	B.T. BROWN 8-14-61

BS 4078 7-60 (WAS BAC 1000-R2)

IAL

CONFIDENTIAL

2

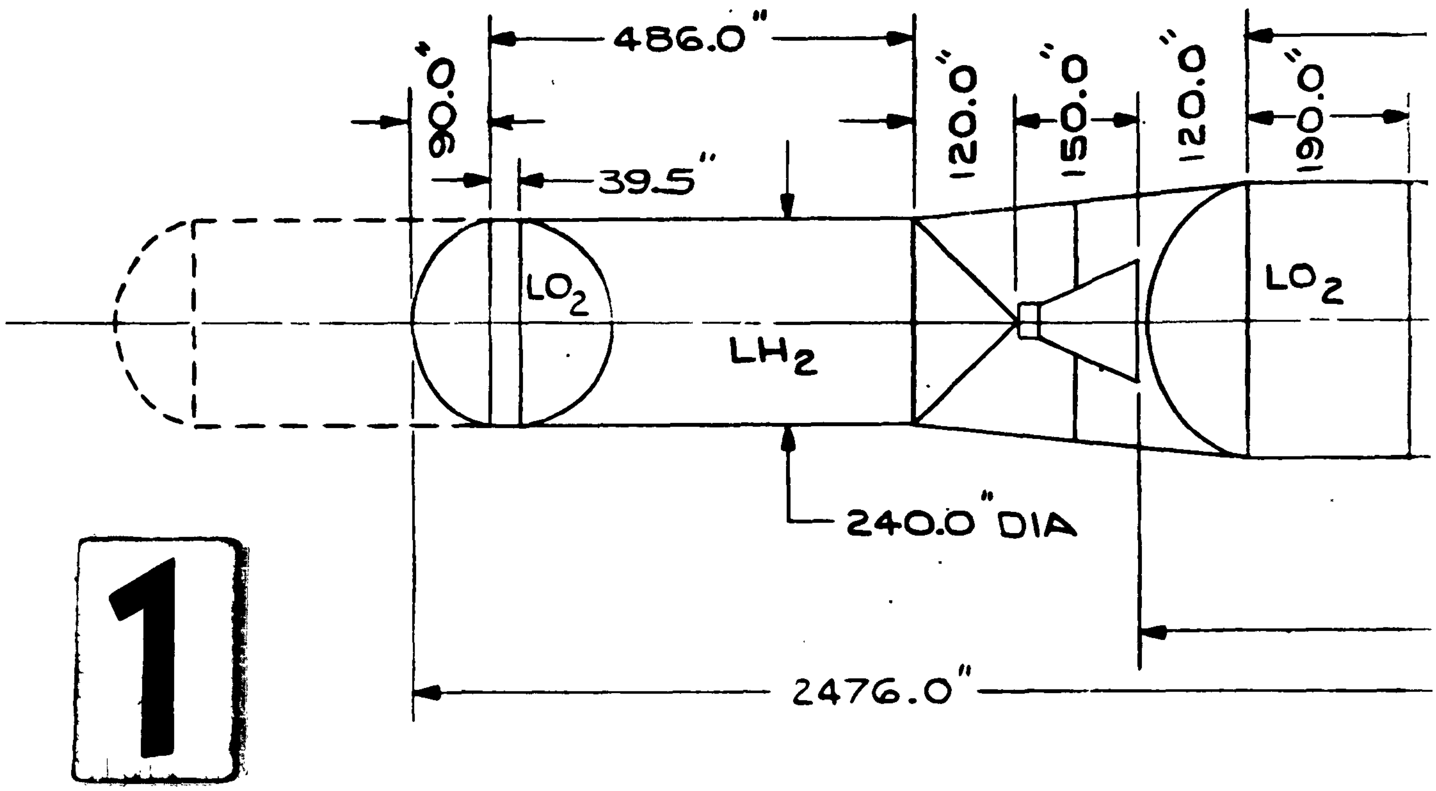


			REVISED	DATE		
CALC					VEHICLE COMPARISON	FIG. 4.1
CHECK						D2-12072
APR.					BOEING AIRPLANE COMPANY	PAGE
APR.					SEATTLE 24, WASHINGTON	9
DRN	B.T. BROWN 8-14-61					

U3 4070 7000 (WAS BAC 1600-R2)

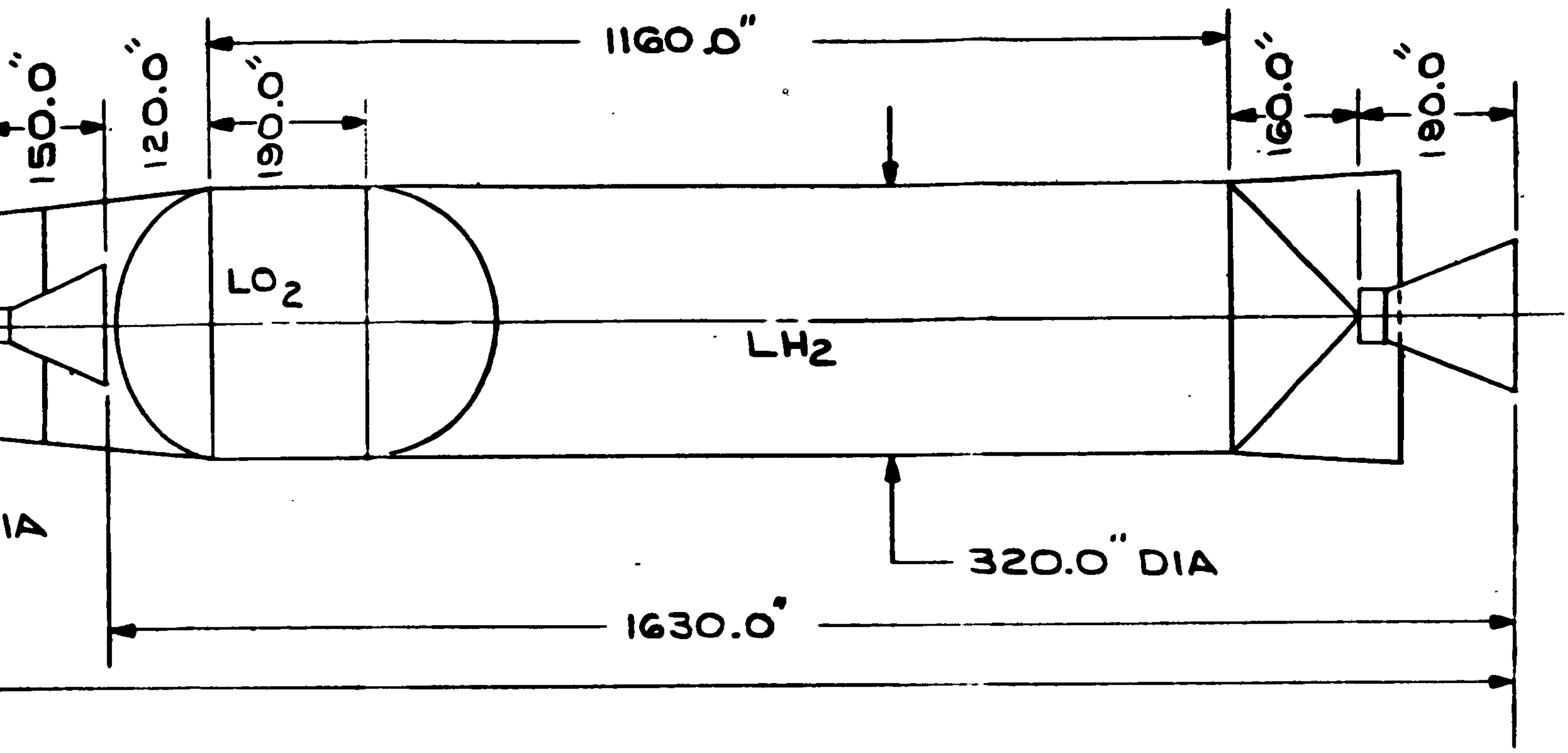
CONFIDENTIAL

CONFID[illegible]NTIAL



DESIGN CRITERIA

	1ST STAGE	2ND STAGE	PAYLOAD
V_{BO} =	10,000	25,260	W = 129,900
λ' =	.945	.940	P = 15#/FT^3
F/W =	1.1	1.1	
W_O =	1,847,600	477,000	
F =	2,032,400 (S.L.)	531,100 (VAC.)	
W_P =	1,295,200	326,300	
W_{LO_2} =	1,110,200	279,700	
W_{LH_2} =	185,000	46,600	
$M^R_{VEHICLE}$ =	6	6	
W_{BO} =	552,400	150,700	
W_{STEP} =	1,370,600	347,100	
ε =	.701	.684	



2

SCALE: 1/200TH SIZE

			REVISED	DATE		
CALC					BASE LINE VEHICLE MODEL 902-1	FIG. 4.2
CHECK						D2-12072
APR.						
APR.					BOEING AIRPLANE COMPANY SEATTLE 24, WASHINGTON	PAGE 10
DWN	K.OSBORNE	7-31-61				

4.2 Cont.

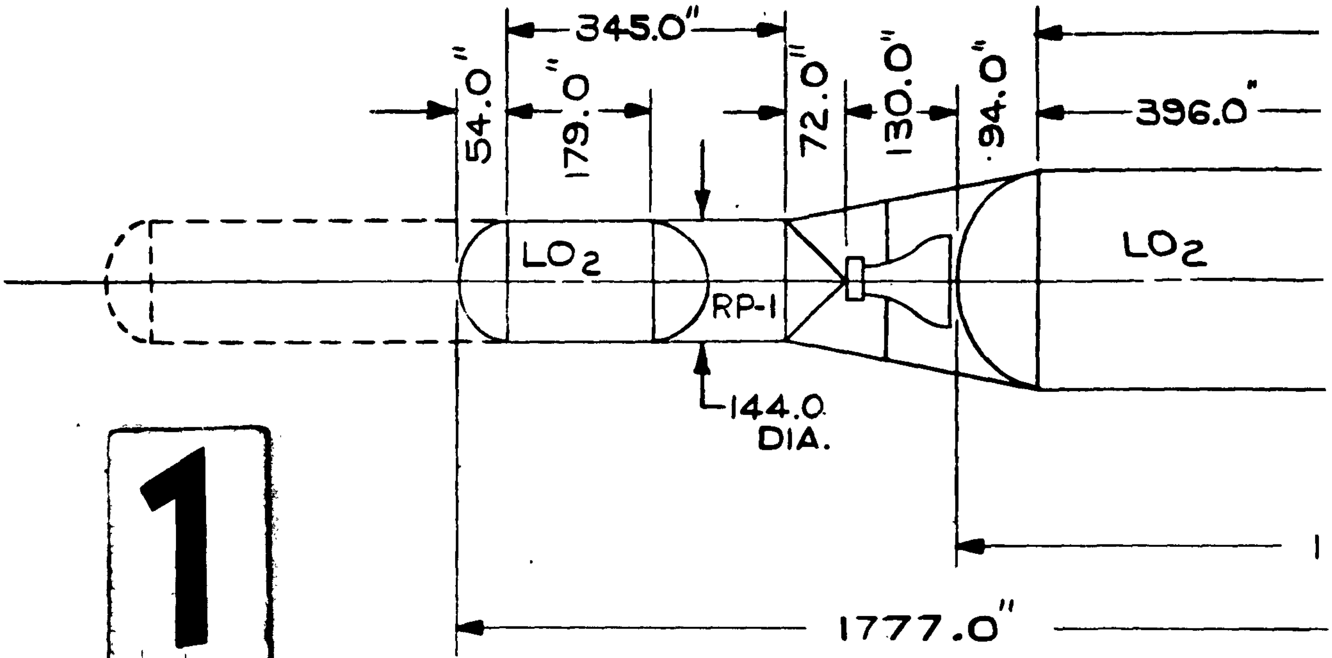
to work from previous Boeing studies (reference 15.2). In general, the vehicle is of aluminum semi-monocoque construction. Gimballed bell nozzle engines of 2,032,400 and 531,100 pounds thrust are used on the first and second stages respectively. The engines are supported by the conical tank ends. Interstage structure is conventional, separation being accomplished by a shaped explosive charge. Auxiliary power and guidance components are carried in the second stage or payload area depending on the mission. Gimball deflection can be accomplished by a hot gas servo control system. Electrical power is supplied by batteries. Location of the LO_2 tanks ahead of the LH_2 tanks aids control, and neutral stability is achieved during boost by a small degree of flare in the vehicle base skirt. This structure also serves to support the vehicle on the launching pad. Upper tank ends are .75 to 1 hemi-ellipsoids. Propellant tank septums are hemispherical.

4.3 BASELINE LO_2/RP-1 VEHICLE (MODEL 902-2)

In general, the description under 4.2 above applied to the Model 902-2 vehicle also. Exceptions are: the propellant, which is LO_2/RP-1, and the second stage thrust, which is 320,000 pounds. The general arrangement and principal design criteria are shown by figure 4.3.

4.4 UNCONVENTIONAL ENGINE LO_2/LH_2 VEHICLE (TWO STAGE) (MODEL 902-3)

For comparison of engine efficiencies, an Aerojet General engine of 2,000,000 pounds thrust utilizing the Force Deflection (F-D) concept was applied to a vehicle similar to Model 902-1, but with propellant



1

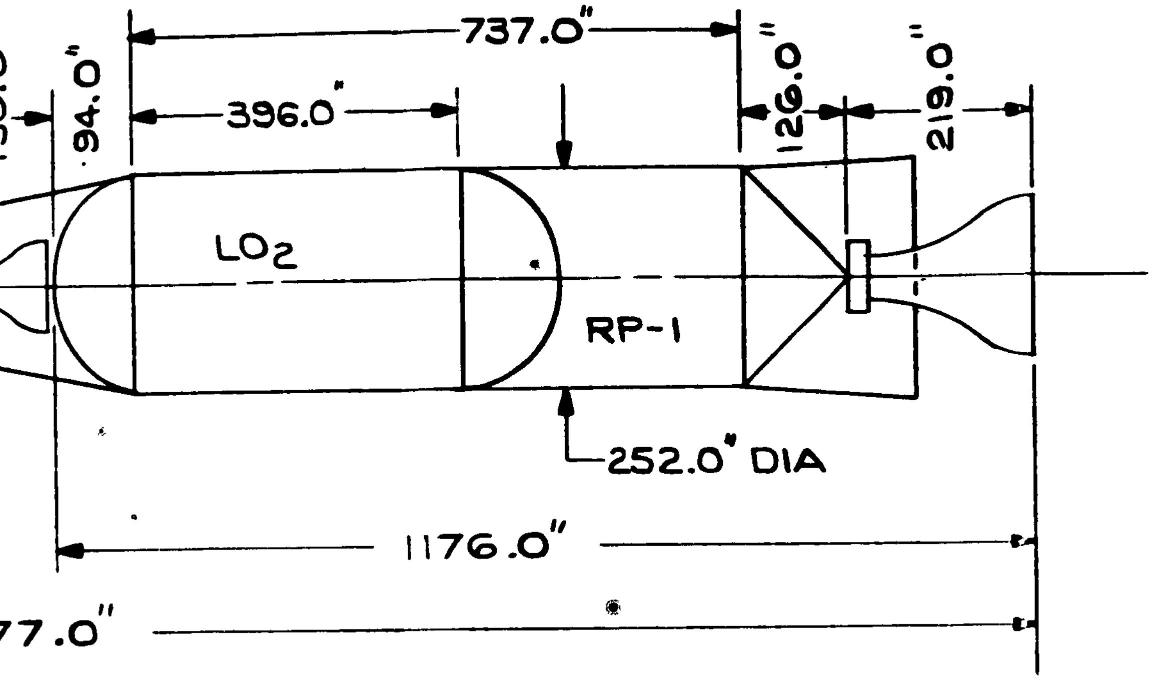
DESIGN CRITERIA

	1ST STAGE	2ND STAGE	PAYLOA
$V_{B.O}$	= 11,000	25,260	W = 59,8
λ'	= .956	.946	P = 15 #/
F/W	= 1.1	1.1	
W_o	= 1,818,200	300,500	
W_P	= 1,450,900	227,800	
W_{LOX}	= 1,024,200	160,800	
W_{RP-1}	= 426,700	67,000	
M.R.	= 2.4	2.4	
$W_{B.O.}$	= 367,300	72,700	
W_{STEP}	= 1,517,700	240,800	
F	= 2,000,000 (S.L.)	320,000 (VAC)	
&	= .798	.758	

CALC.
CHECK
APR.
APR.
DWN

CONFIDENTIAL

2



TERIA

GE

PAYLOAD
W = 59,800
P = 15 #/FT^3

O
OO
OO
OO

O
OO
OO(VAC)

SCALE: 1/200TH SIZE

			REVISED	DATE		
CALC					BASE LINE VEHICLE MODEL 902-2	FIG.4.3
CHECK						D2-12072
APR.					BOEING AIRPLANE COMPANY SEATTLE 24, WASHINGTON	PAGE 12
APR.						
DWN	K.OSBORNE	8-1-61				

U3 4078 7000 (WAS BAC 1650-R2)

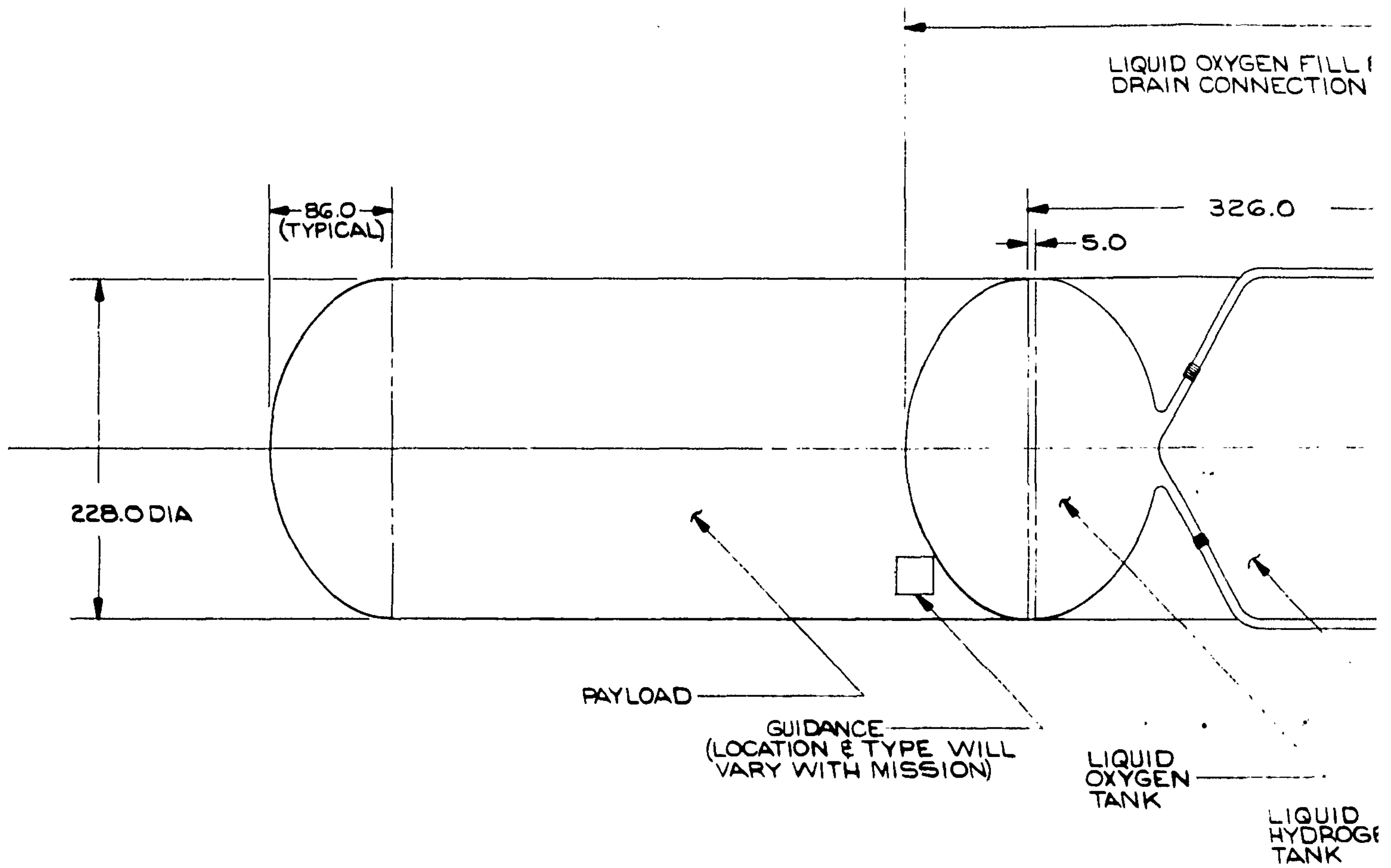
CONFIDENTIAL

4.4 Cont.

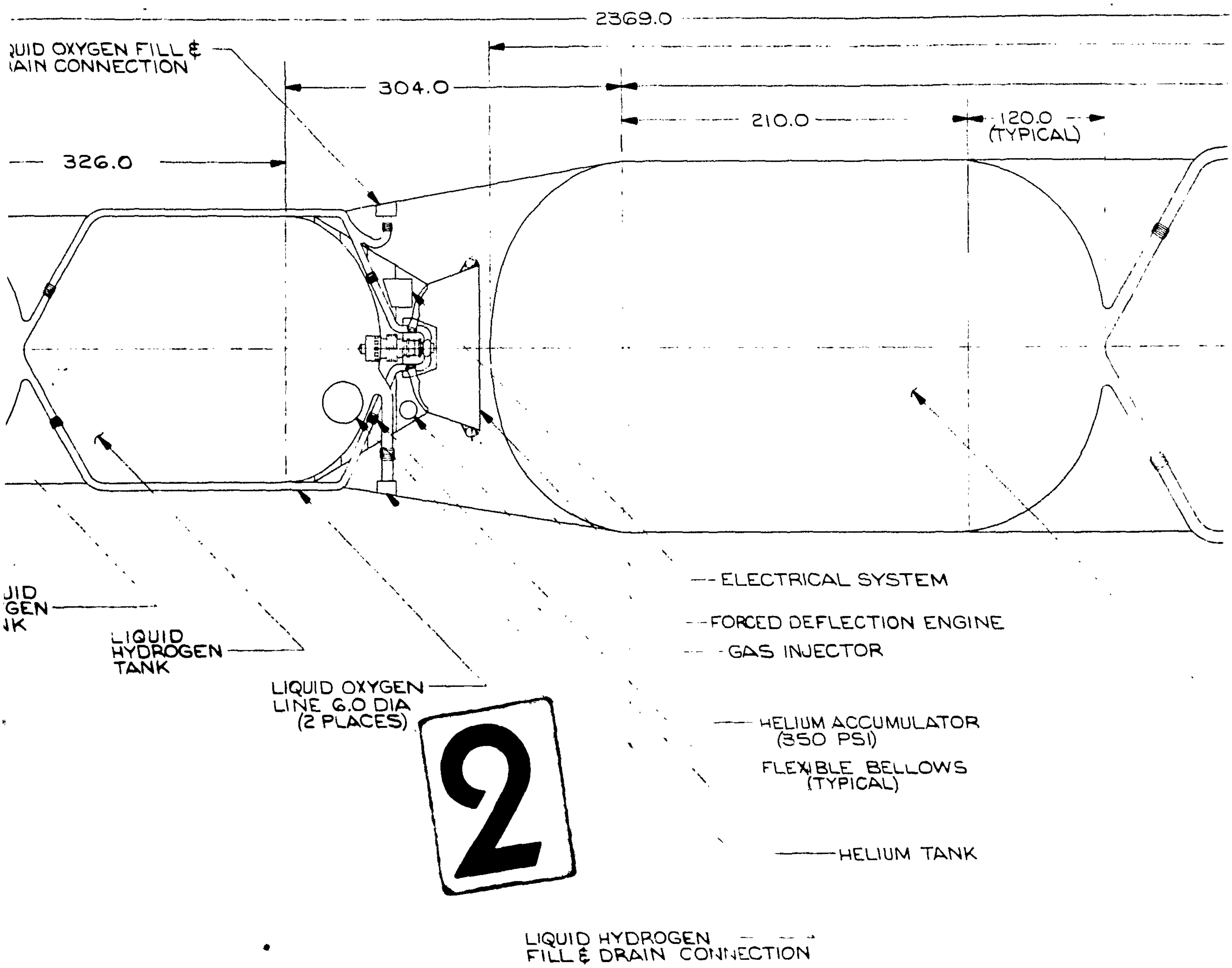
quantities optimized for the P-D engine. A general arrangement of the vehicle, Model 902-3, is shown by figure 4.4. Principal design criteria are also included.

4.5 SINGLE STAGE TO ORBIT VEHICLE (MODEL 902-4)

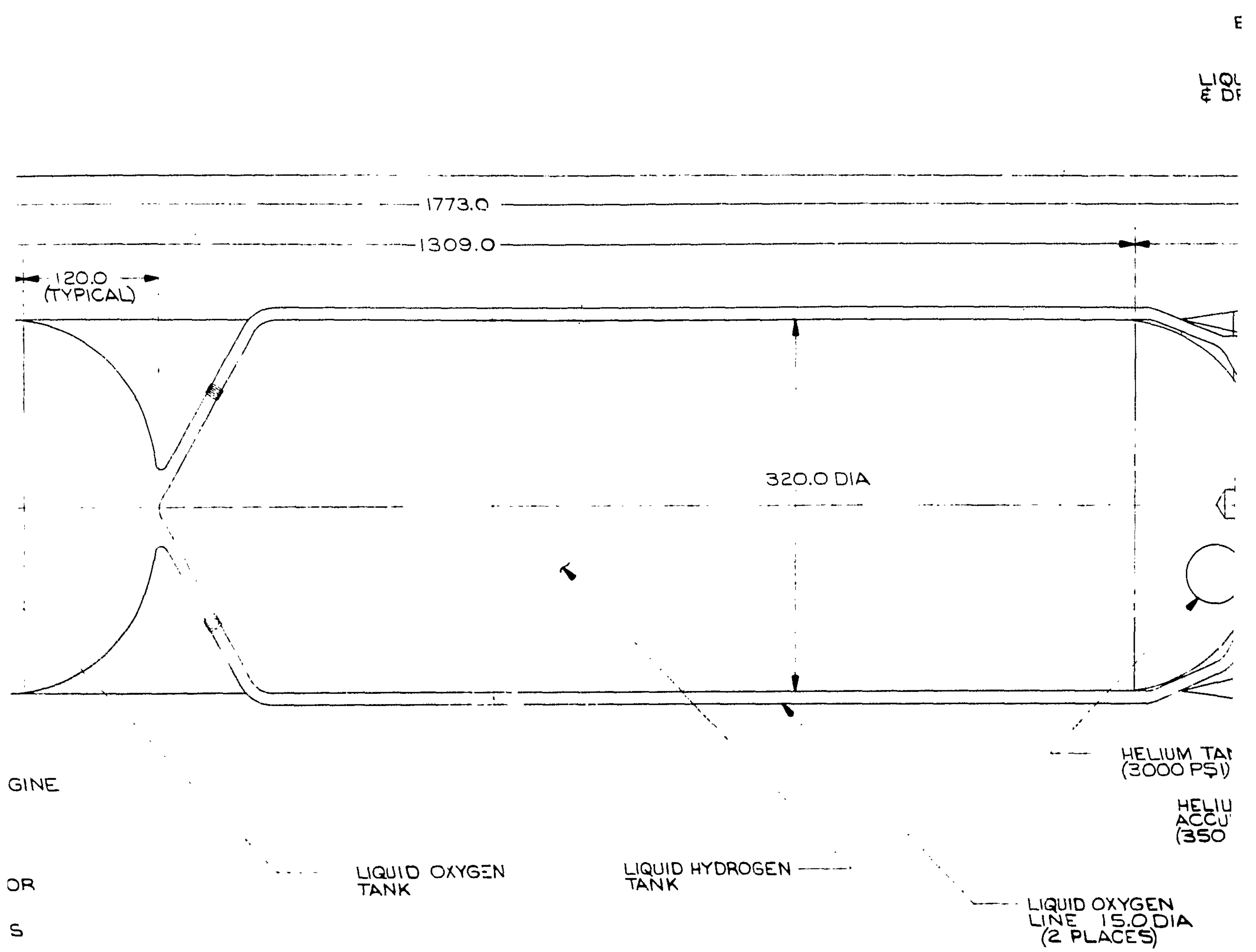
A promising application of the Force Deflection (P-D) engine is on a single stage vehicle capable of fulfilling the design mission. Model 902-4 is a conventionally arranged vehicle in this category and is shown in figure 4.5 together with principal design criteria. Construction is essentially similar to the first stage of the Model 902-3. The same 2,000,000 pound thrust P-D engine is used, except that chamber pressure is incre sed to 3000 psi. Propellant requirements for the Model 902-4 vehicle allows a tank diameter of 270 inches with a relatively short vehicle overall height. This permits the engine skirt to provide the base flare required for neutral stability during atmospheric flight. Support on the launch pad is achieved by ground pad structure extending upward inside the nozzle and through the air vents sufficiently to engage the vehicle engine support structure. Lateral stability on the launch pad is augmented by three retractable compression members engaging sockets near the vehicle center of pressure to form a tripod-like support.


LIQUID OXYGEN FILL
DRAIN CONNECTION
86.0
(TYPICAL)
326.0
5.0
228.0 DIA
PAYLOAD
GUIDANCE
(LOCATION & TYPE WILL
VARY WITH MISSION)
LIQUID
OXYGEN
TANK
LIQUID
TANK

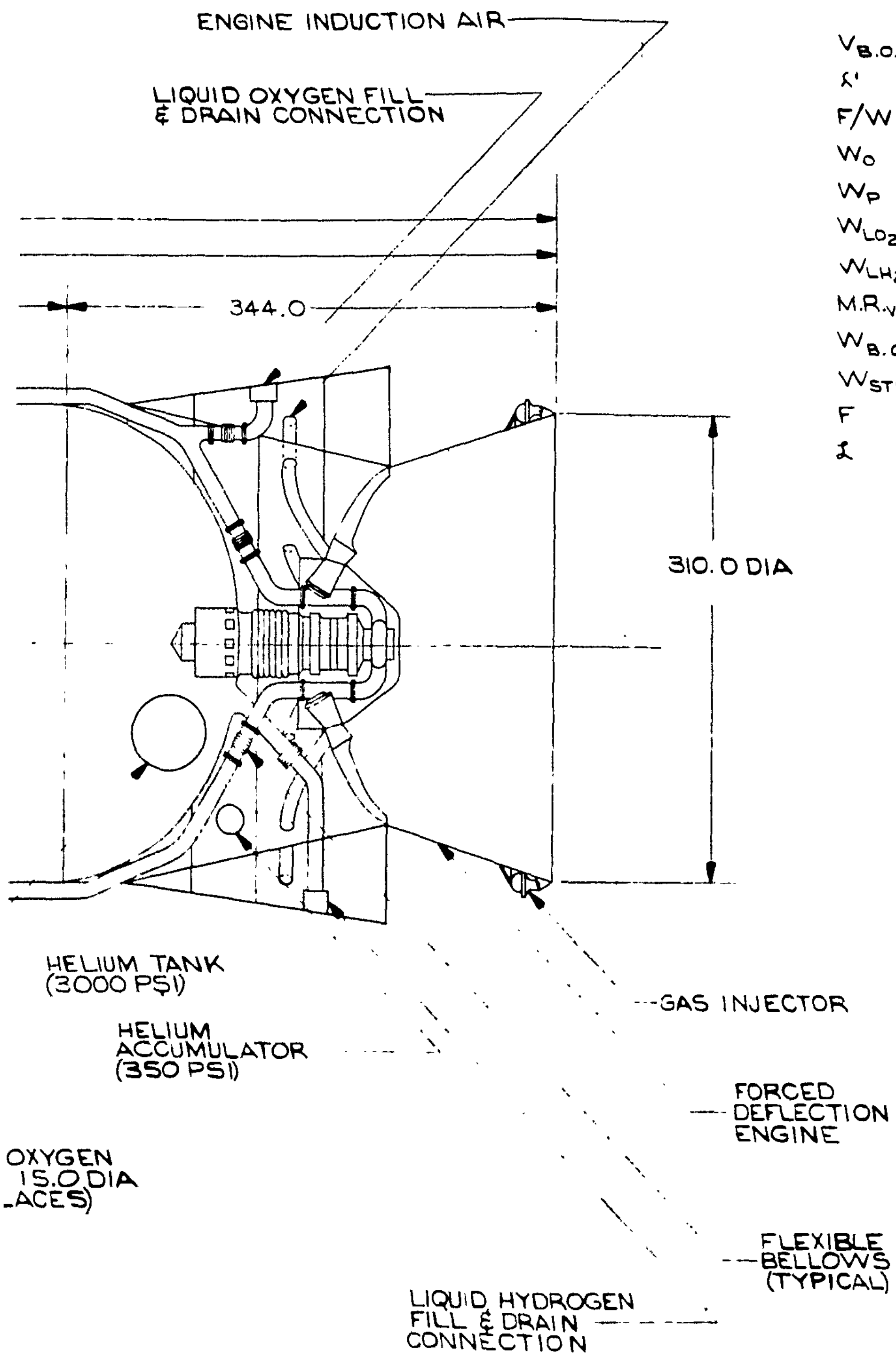
1


2369.0
UID OXYGEN FILL &
AIN CONNECTION
304.0
210.0
120.0
(TYPICAL)
326.0
UID
GEN
IK
LIQUID
HYDROGEN
TANK
LIQUID OXYGEN
LINE 6.0 DIA
(2 PLACES)
ELECTRICAL SYSTEM
FORCED DEFLECTION ENGINE
GAS INJECTOR
HELIUM ACCUMULATOR
(350 PSI)
FLEXIBLE BELLOWS
(TYPICAL)
HELIUM TANK
LIQUID HYDROGEN
FILL & DRAIN CONNECTION

2


E
LIQU
& DR
1773.0
1309.0
120.0
(TYPICAL)
320.0 DIA
HELIUM TAN
(3000 PSI)
HELIU
ACCU
(350
GINE
OR
S
NK
LIQUID OXYGEN
TANK
LIQUID HYDROGEN
TANK
LIQUID OXYGEN
LINE 15.0 DIA
(2 PLACES)

3



DESIGN CRITERIA

	1ST STAGE	2ND STAGE
$V_{B.O.}$ =	13,000	25,260
λ' =	.947	.937
F/W =	1.1	1.1
W_O =	1,818,200	372,500
W_P =	1,369,100	223,100
W_{LO_2} =	1,173,500	191,200
W_{LH_2} =	195,600	31,900
$M.R._{VEHICLE}$ =	6	6
$W_{B.O.}$ =	449,100	149,400
W_{STEP} =	1,445,700	238,100
F =	2,000,000(S.L.)	405,000(VAC)
λ =	.753	.599

PAYLOAD

W = 134,400

$P = 15 LBS/FT^3$

4

NOTE:
ALL DIMENSIONS IN INCHES

FIG. 4.4

D2-12072 PAGE 14

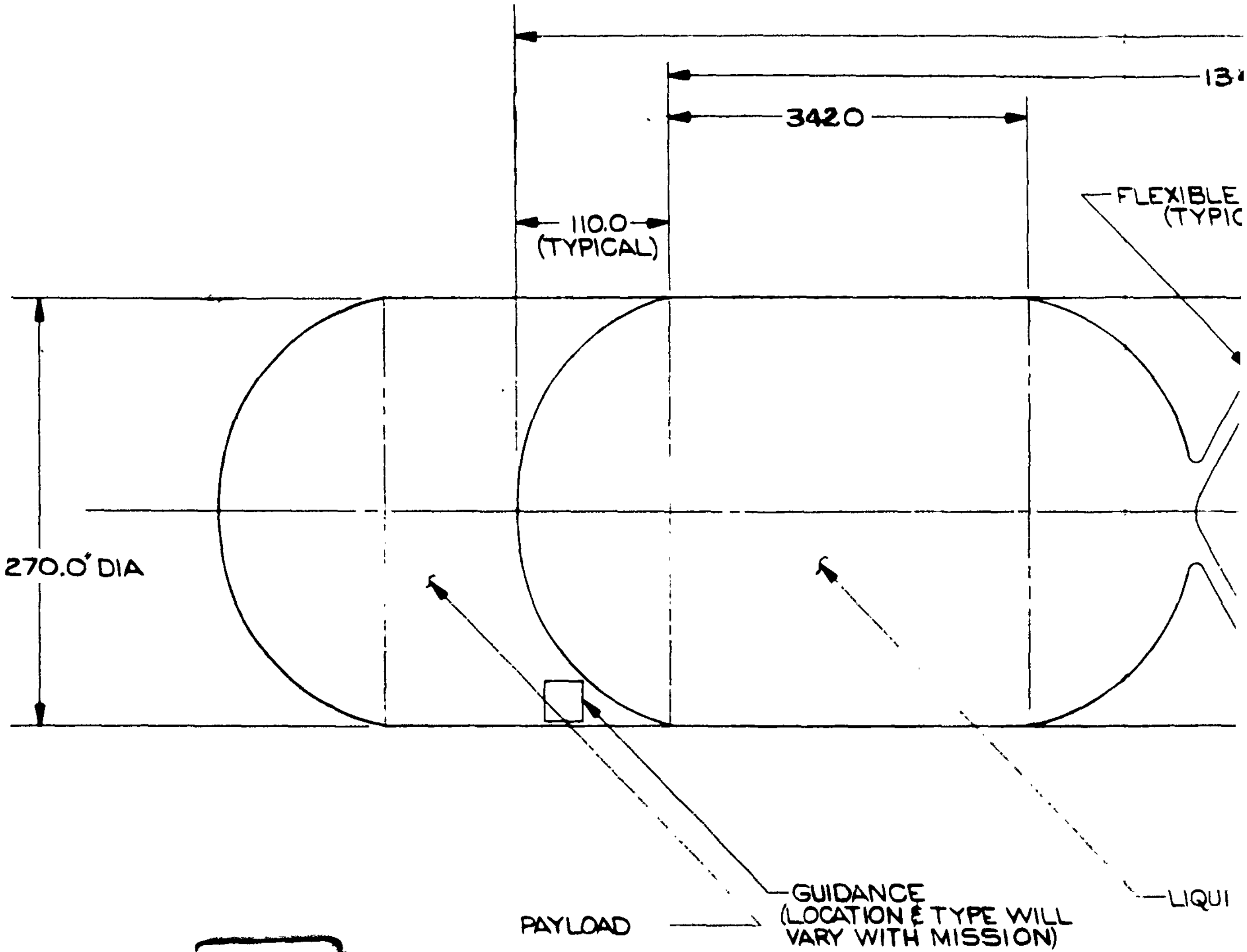
DRAWN K. OSBORNE 8-15-6[illegible]

UNCONVENTIONAL ENGINE VEHICLE (TWO STAGE)

MOD [illegible]

SCALE NONE

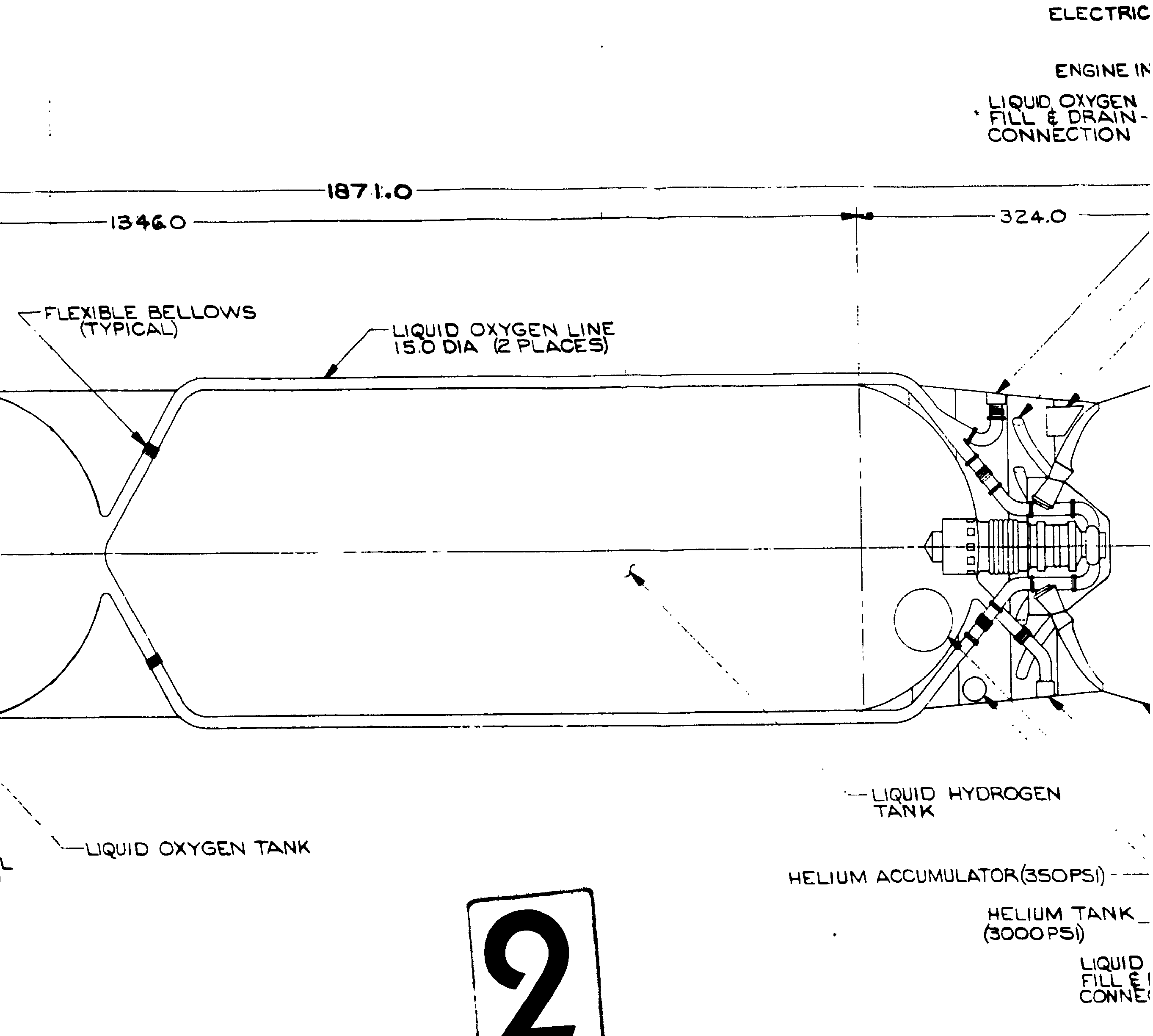
LO-105



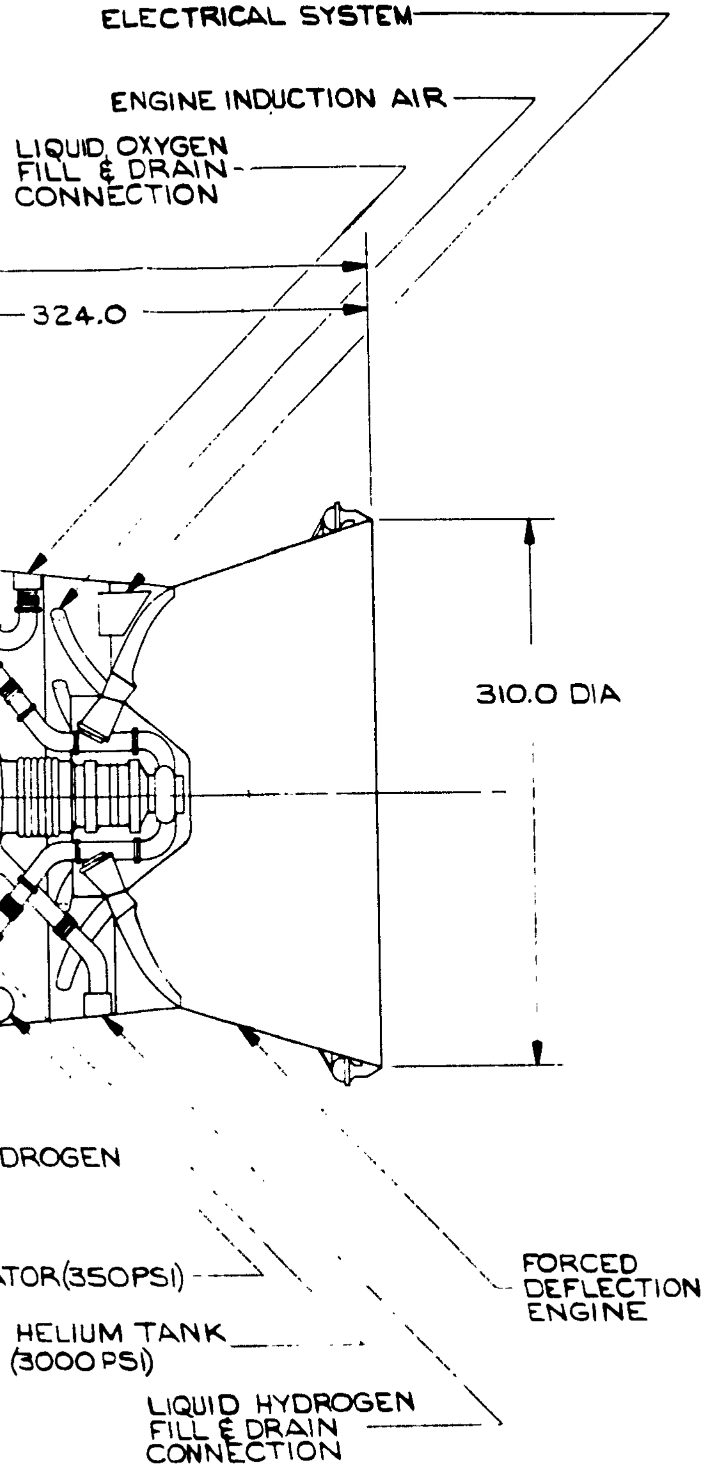
13
3420
FLEXIBLE
(TYPIC
110.0
(TYPICAL)
270.0' DIA
GUIDANCE
(LOCATION & TYPE WILL
VARY WITH MISSION)
PAYLOAD
LIQUI

1

CONFIDENTIAL



2



DESIGN CRITERIA

$V_{B.O.}$	=	25,260
λ'	=	.943
F/W	=	1.4
W_O	=	1,428,500
W_P	=	1,239,700
W_{LO_2}	=	1,084,700
W_{LH_2}	=	155,000
$M.R._{VEHICLE}$	=	7
$W_{B.O.}$	=	188,800
W_{STEP}	=	1,315,300
F	=	2,000,000 (S.L.)
ϵ	=	.8677

PAYLOAD

W = 113,200

P = 15 LBS/FT^3

3

NOTE:
ALL DIMENSIONS IN INCHES.

FIG 4.5 D2-12072

DRAWN K. OSBORNE 8-15-6	SINGLE STAGE TO ORBIT VEHICLE UNCONVENTIONAL ENGINE MODEL 902-4	BOEING AIRPLANE CO. AERO-SPACE DIV. SEATTLE WASH.
SCALE NONE		LO-106

5.0 PERFORMANCE

5.1 MISSION AND APPROACH

Performance analysis for all vehicles was based on a 300 n. mi. circular orbit with an easterly launch from Cape Canaveral. Performance calculations were conducted using IBM trajectory data with the following characteristics:

1. Vertical launch
2. Tilt at V = 400 fps
3. Gravity turn during the first stage
4. Thrust vectoring during the second stage to achieve constant angle of attack.

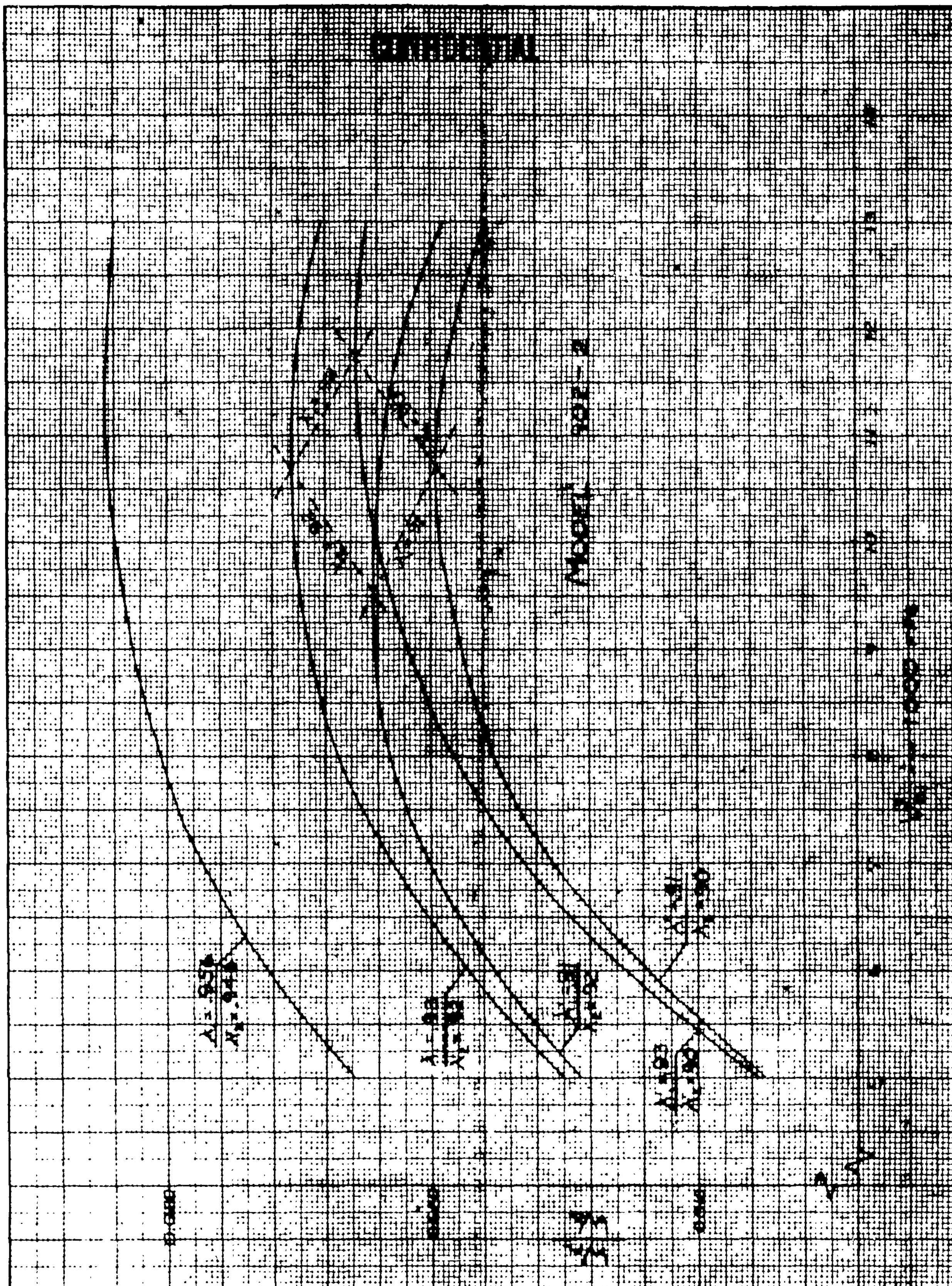
For all two-stage vehicles the first stage thrust to launch weight ratio, T/W_{o1}, was established at 1.1. Second stage thrust to weight ratio was also established at 1.1. Both are based on cost optimization trade studies conducted at Boeing as discussed in reference 15.2.

5.2 VEHICLE STAGING

For all two-stage vehicles, the staging velocity for a given combination of λ_1' and λ_2', was taken as that first stage burnout velocity which maximized the payload/launch weight ratio. Staging velocity was found relatively unaffected by the choice of λ_1'; and λ_2' within the range of 0.90 to 0.94. Fig. 5.1 shows curves giving payload/launch weight vs. burnout velocity for Model 902-2 (baseline LO_2/RP vehicle) using several combinations of λ_1': and λ_2'.

A staging velocity (V_{B1}) of 11,000 fps was established as valid for all λ' combinations for this vehicle. Maximum deviation of W_{pl}/W_o within the outlined area of Fig. 5.1 for V_{B1} = 11,000 ft/sec was only

CONFIDENTIAL



CALC			REVISED	DATE
CHECK				
APR				
APR				

Effect of First Stage Burnout Velocity & λ's on $\frac{\text{Payload Weight}}{\text{Initial Launch Weight}}$

Figure 5.1

D2-120[illegible]

BOEING AIRPLANE COMPANY

CONFIDENTIAL

5.2 Cont.

1.5%. Fig. 5.1 also shows curves for combinations of $(\lambda_1' - \lambda_2')$. This parameter remains unchanged for ($\lambda_1' = .91$ & $\lambda_2' = .92$, $\lambda_1' = .93$ & $\lambda_2' = .94$, $\lambda_1' = .956$ & $\lambda_2' = .946$) and the curves are displaced nearly vertically from each other. This leaves the staging velocity virtually unchanged. The final weight analysis of Model 902-2 established λ' values of .956 for λ_1' and .946 for λ_2'. A similar staging analysis was performed on Model 902-1 and Model 902-3. The results are summarized on Figure 5.2.

5.3 SINGLE STAGE VEHICLE

The problem of sizing the single stage to orbit vehicle differs from the two stage case. Here it is desireable to provide a proper balance between payload capability and the cost sensitive inert and propellant weights. This was found to be a function of the thrust/launch weight (T/W_o), with final determination made on the basis of the maximum payload for the least cost.

Curves showing the effect of thrust launch weight (T/W_o) on propellant weight and the weight at burnout/launch weight (W_{Bo}/W_o) for Model 902-4 (single stage) are given in figure 5.3. This data was generated from IBM single stage trajectories. It is seen for the fixed 2×10^6 sea level thrust, the propellant cost item decreases rapidly while the W_{Bo}/W_{o1}, which gives a measure of the inert weight cost factor, levels off at the higher T/W_o values. This would infer that lower costs would be involved at higher T/W_o than for a two stage case.

CONFIDENTIAL

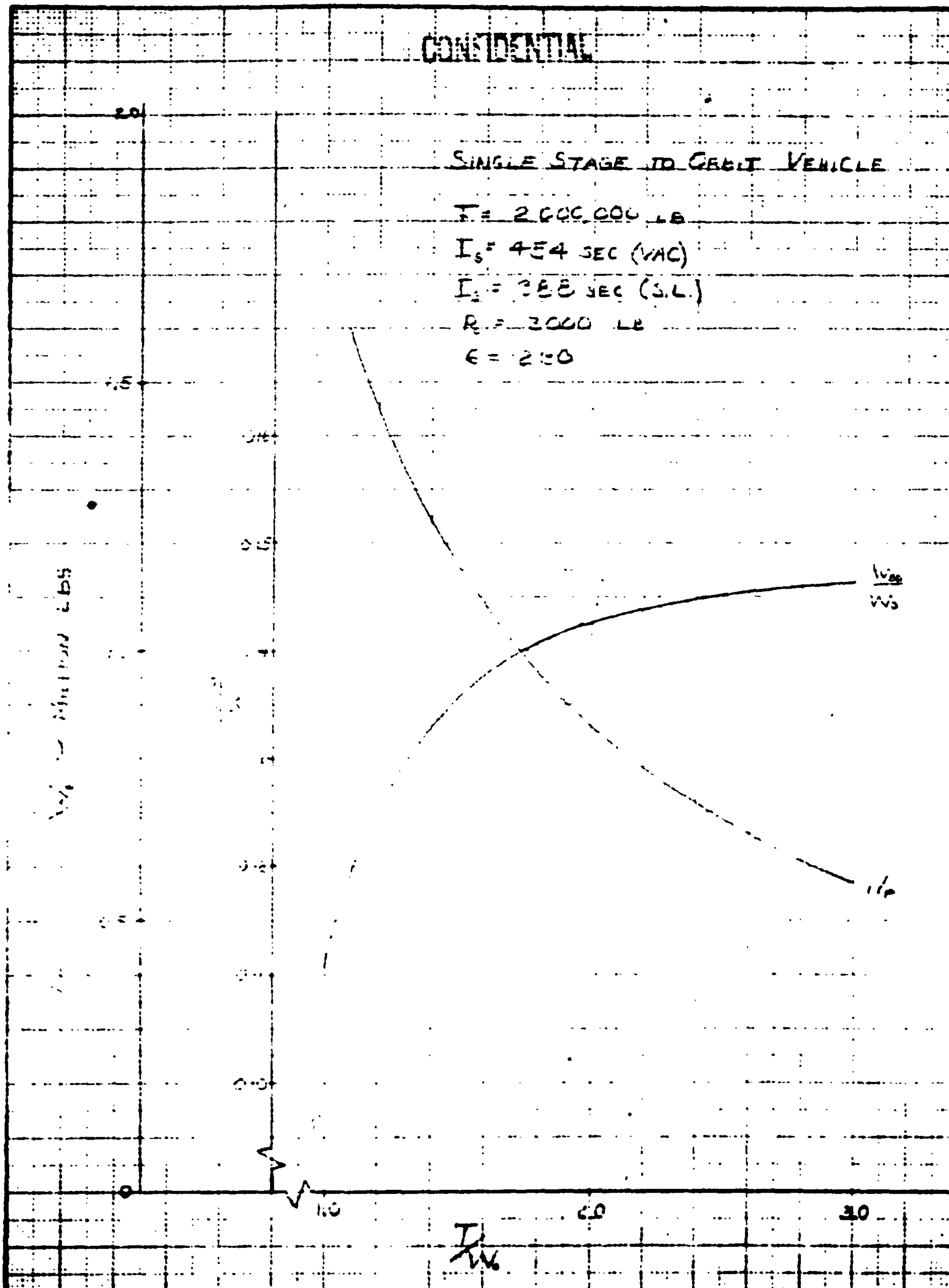
STAGING ANALYSIS RESULTS

	Model 902-1 LO_2/LH_2 Baseline		Model 902-2 LO_2/RP Baseline		Model 902-3 LO_2/LH_2 F.D. Nozzle Advanced Engine	
	Stage 1	Stage 2	Stage 1	Stage 2	Stage 1	Stage 2
Thrust (lb)	2,032-400(SL)	524,700(VAC)	2,000,000(SL)	330,500(VAC)	2,000,000(SL)	409,700(VAC)
Prop. Wt. (lb)	1,295,000	326,300	1,450,900	227,800	1,369,100	223,100
T/W_o	1.1	1.1	1.1	1.1	1.1	1.1
λ'	.945	.940	.956	.946	.947	.937
W_p/W_o	.701	.684	.798	.758	.753	.599
V_{Bo} (fps)	10,000	25,260	11,000	25,260	13,000	25,[illegible]60
I_s (sec)	345 (SL)	426 (VAC)	268 (SL)	324 (VAC)	361 (SL)	426 (VAC)
ϵ	20	40	16	40	40	40
P_c (psi)	1000	1000	1000	1000	1000	1000
MR	6	6	2.4	2.4	5.66	5.66
PL (lb)	129,900		59,700		134,400	

FIGURE 5.2

US 4070 7000 (WAS BAC 1546 F-R3)

CONFIDENTIAL



EFFECT OF $\frac{T}{W_0}$ ON $\frac{W_{BO}}{W_0}$ AND W_0

SINGLE STAGE TO ORBIT VEHICLE

BOEING AIRPLANE COMPANY

CALC		Figure 5.3
CHECK		D2-12072
APD		PAGE 30
APD		
DWN F W S	1-26-61	

5.3 Cont.

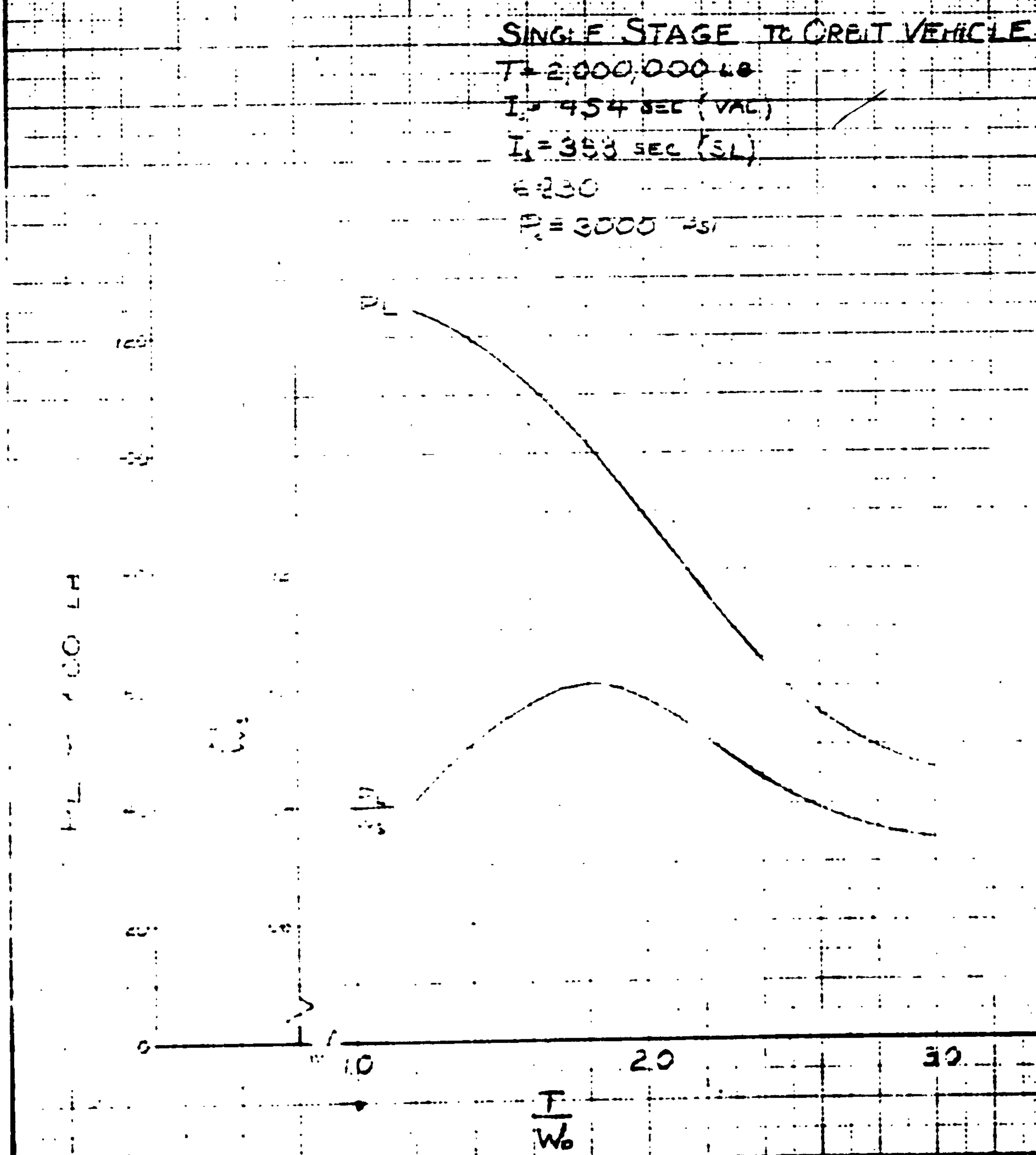
The above is borne out by Figure 5.4 which exhibits the effect of thrust/launch weight (T/W_o) on payload and payload/stage weight (PL/W_s). Payload was found to be the greatest for small values of T/W_o, and decreases rapidly with increasing T/W_o. From a cost performance standpoint a low T/W_o is desirable. Figure 5.4 also shows, however, that the payload/weight of stage (PL/W_s) is a maximum at $T/W_o = 1.8$.

5.4 VEHICLE COMPARISON

Table 5.2 compares models 902-1, 902-2, and 902-3. Two-stage vehicle weights, engine data, and staging data are presented. A comparison of the single stage vehicle, model 902-4 and the Model 902-1 LO_2/LH_2 baseline vehicle is given in Figure 5.5.

From Figure 5.2 it can be seen that a 3.5% payload advantage is indicated for the Model 902-3 two stage vehicle using the advanced F-D engine over the Model 902-1 conventional baseline design. This is attributed to both higher specific impulse of the first stage F-D engine and the better installation features as affecting structural weight.

Comparison of the single stage Model 902-4 vehicle to the Model 902-1 design by reference to Figure 5.5 shows a net reduction of from 3.8% to 29% from the standpoint of performance alone. As noted previously, however, evaluation of costs as covered in the Economic Analysis section of this report must be considered before conclusions are drawn.



CALC			REVISED	DATE	EFFECT OF $\frac{T}{W_0}$ ON PAYLOAD & $\frac{PL}{W_0}$ SINGLE STAGE VEHICLE	Figure 5.4
CHECK						D2-12072
APR					BOEING AIRPLANE COMPANY	PAGE 22
APR						
LAW	EWS	8-1-61				

FIGURE 5.5

MODELS 902-1 and 902-4 COMPARISON

	Model 902-1 LO_2/LH_2 Baseline		Model 902-4 Single Stage LO_2/LH_2 Advanced Engine	
	Stage 1	Stage 2	T/W_o = 1.1	T/W_o = 1.8
Thrust (lb)	2,032,400	524,700	2,000,000	2,000,000
Prop. Wt. (lb)	1,295,200	326,300	1,585,000	955,000
T/W_o	1.1	1.1	-	-
λ'	.945	.940	.9474	.937
W_p/W_o	.701	.684	.8816	.8594
V_{Bo} (fps)	10,000	25,260	25,260	25,260
I_s (sec)	345 (S.L.)	426 (Vac)	388 (S.L.)	383 (S.L.)
ϵ	20	40	230	230
P_c (psi)	1,000	1,000	3,000	3,000
PL (lb)	129,900		125,000	192,000

5.5 MODEL 902-4 - ALTERNATE USE

To determine the performance potential of the single stage Model 902-4 for use as the booster of a two stage vehicle, a limited study considering application of possible upper stages was conducted. In this study 1.2 to 2.0 T/W_0 versions of the Model 902-4 vehicle were modified by addition of estimated upper stage plus payloads weights to yield a T/W_0 of 1.1 for the resulting two stage vehicles. The resulting payloads are shown by Figure 5.6. A significant increase can be noted. After iterating with costing inputs, a T/W_0 = 1.4 was selected, providing a 27% increase in payload over the single stage 902-4. The two stage version is designated Model 902-4A.

It is interesting to note that the addition of an 816,000 pound thrust upper stage with [illegible] pounds of propellants for the T/W_0 = 1.6 version permits a payload of [illegible] lbs. This upper stage would be similar to the currently programmed Saturn S-II stage.

It is not possible in the time [illegible] to determine the effects on the $/# cost parameter, the prime intent being to establish whether the single stage vehicle offered growth and/or versatility characteristics.

FIGURE 5.6

ALTERNATE USE OF MODEL 902-4

First Stage	Second Stage
Basic Model 902-4	
LO_2/LH_4	LO_2/LH_2
T = 2,000,000 lb. (S.L.)	I_{s2} = 426 sec. (vac)
I_s = 388 sec. (S.L.)	T/W_{o2} = 1.2
ϵ = 230	ϵ = 40
P_c = 3000 psi	λ'_L = .92
T/W_{o1} = 1.1	P_c = 1000 psi

Single Stage Vehicle (Reference)

T/W	1.2	1.4	1.6	1.8	2.0
PL (lB)	124,700	113,200	101,800	92,000	82,000

Two-stage vehicle T/W_{o1}	1.1	1.1	1.1	1.1	1.1
WP_1 (lb)	1,458,400	1,247,000	1,083,000	954,800	863,000
WP_2 (lb)	128,300	328,200	492,500	625,200	723,000
second stage thrust	331,000	611,000	816,000	971,000	1,089,000
Two-stage vehicle payload (lb) (300 n. mi. orbit)	136,300	143,600	138,000	129,200	121,100

6.0 **WEIGHTS**

6.1 WEIGHT ANALYSIS MODEL 902-1 THROUGH 902-4

Two primary objectives of the weight study portion of the program were:

(a) To develop sufficient weight data for evaluation of vehicle performance and vehicle cost;

(b) To describe system weight differences between the use of "conventional" engines and the use of "advanced" engines.

To satisfy the first objective of this study, the four configurations as described in Section 4.0 were analyzed. Weight data generated for similar configurations (Reference 15.2), where applicable, was used for study of these configurations. There were, however, several differences between the criteria used for the Reference study and that criteria used for this study. The effect of these criteria differences on weight been incorporated.

The most significant criteria change affecting weight was that associated with the manned payload ground rule. Manned criteria requires a factor of safety of 1.4, increased from 1.25, and also requires vehicle neutral stability throughout the flight trajectory. To accomplish neutral stability the more dense oxidizer has been placed above the fuel, and a flared first step skirt has been added. The tank design assumes that it is self-supporting, unpressurized on the launch pad, with no restriction on the propellant loading sequence.

Another design difference which affects tank weight is the tank pressures which are specified at a slightly higher value for this study than for the previous Boeing studies.

Figure 6.1 provides weight statements of the four basic vehicle models as described in Section 4.0. Models 902-1, -2 and -3 were designed at a thrust-to-weight ratio of 1.1. Model 902-4 is shown prior to cost inputs at a thrust-to-weight ratio of 1.8 in this figure.

The method of determining engine weight was provided by the Aerojet General Corporation. Engine weights for 2 x 10^6 lb thrust were specified at 15,000 lb. for conventional engines and 14,000 lb for the forced-deflection engine. These engines have a chamber pressure of 1000 psi. The conventional engines had an expansion ratio of 20 and 15 for LO_2LH_2 and LO_2/RP-1 respectively. The forced-deflection engine for the two stage vehicle had an expansion ratio of 40. The single-stage-to-orbit engine had a chamber pressure of 3000 psi, an expansion ratio of 230, and was specified to weigh 20,000 lb. To estimate the effects of size and expansion ratio on engine weight, data from reference 15.4 was utilized.

It may be noted that the weight statements of Figure 6.1 may not adequately reflect discrete weight differences between systems using the "forced-deflection" and those using conventional "bell" nozzles. These discrete weight differences (as described below) were recognized to have a small effect on the mass efficiency. The step mass ratio (λ') values are

	MODEL 902-1 BASE LINE LO_2/LH_2		MODEL 902-2 BASE LINE LO_2/RP-1	
	STEP I	STEP II	STEP I	STEP
PROPELLANT TANKS	22,500	4,900	19,200	
THRUST STRUCTURE	3,800	600	3,000	
SKIRT	3,000	-	2,600	
INTERSTAGE STRUCTURE	2,500	2,200	2,200	
SEPARATION PROVISIONS	100	100	100	
SLOSH AND ANTI-VORTEX PROVISIONS	1,100	400	600	
EXTERNAL INSULATION	2,900	800	1,000	
MISCELLANEOUS STRUCTURE	1,200	400	1,300	
TOTAL STRUCTURE	(37,100)	(9,400)	(30,000)	(
EQUIPMENT	3,900	1,400	5,500	
ENGINE (WET)	15,000	3,900	15,000	
PROPELLANT SYSTEM	3,400	1,900	1,500	
PRESSURIZATION SYSTEM	6,200	1,900	4,100	
RESIDUALS	9,800	2,300	10,700	
TOTAL INERT WEIGHT	(75,400)	(20,800)	(66,800)	(1
PROPELLANT - FUEL	185,000	46,600	426,700	6
- OXIDIZER	1,110,200	279,700	1,024,200	16
TOTAL STEP WEIGHT	1,370,600	347,100	1,517,700	2
STEP MASS RATIO (λ')	.945	.940	.956	.
LAUNCH WEIGHT	1,847,600		1,818,200	
BURNOUT WEIGHT - STEP I	552,400		367,300	
STARTBURN WEIGHT - STEP II	477,000		300,500	
BURNOUT WEIGHT - STEP II	150,700		72,700	
PAYLOAD WEIGHT	129,900		59,700	

1

Calc			REVISED	DATE
Trac				
Chk				
Appr.				
Appr.				

BOOSTER SYSTEMS

BOEING AIRPLANE COMPANY
SEATTLE 24, WASHINGTON

DATA SHE

MODEL 902-2 BASE LINE LO_2/RP-1 STEP I	STEP II	MODEL 902-3 ADVANCED LO_2/LH_2 STEP I	STEP II	MODEL 902-4 ADVANCED LO_2/LH_2 SINGLE STEP		
19,200	3,200	23,700	3,500	16,300		
3,000	450	3,500	500	3,300		
2,600	-	3,000	-	2,000		
2,200	2,000	2,700	1,860	500		
100	100	100	100	100		
600	300	1,100	400	800		
1,000	150	3,000	600	2,200		
1,300	300	1,300	300	1,000		
(30,000)	(6,500)	(38,400)	(7,200)	(26,200)		
5,500	900	5,900	1,000	3,000		
15,000	2,600	14,000	2,750	20,000		
1,500	600	3,400	1,500	2,800		
4,100	700	6,600	1,050	5,200		
10,700	1,700	10,300	1,500	7,000		
(66,800)	(13,000)	(76,600)	(15,000)	(64,200)		
426,700	67,000	195,600	31,900	119,300		
1,024,200	160,800	1,173,500	191,200	835,500		
1,517,700	240,800	1,445,700	238,100	1,019,000		
.956	.946	.947	.937	.937		
1,818,200		1,818,200		1,111,000		
367,300		449,100		196,200		
300,500		372,500		-		
72,700		149,400		-		
59,700		134,400		92,800		

2

DATA SHEET

MODEL 902-4 ADVANCED LO_2/LH_2 SINGLE STEP		
16,300		
3,300		
2,000		
500		
100		
800		
2,200		
1,000		
(26,200)		
3,000		
20,000		
2,800		
5,200		
7,000		
(64,200)		
119,300		
835,500		
1,019,000		
.937		
1,111,000		
196,200		
-		
-		
92,000		

therefore typical for the configurations and are an adequate basis for performance and cost evaluation.

To satisfy the second objective of this study, weight evaluations were made of several arrangements of integrating conventional "bell" and advanced "forced-deflection" engines into the vehicle configuration. The primary components of significant weight differences are:

(1) Aft tank bulkhead

(2) Thrust structure

(3) Skirt or interstage

(4) Base heating provisions

(5) Engine

Other weight differences will be relatively minor and should not affect the trend of weight differences or significantly affect vehicle cost or performance.

Figure 6.2 compares these significant weight items for several arrangements of integrating conventional and advanced engines. These discrete weight differences reflect the design differences as shown by the drawings in Section 7.0. For either engine type, the various concepts of mounting the engines to carry the thrust loads is seen to have only a small effect on weight. The accuracy of weight estimates is not sufficient to indicate a definite conclusion from these small weight differences.

When comparing engine types however, a "bell" nozzle design

FIGURE 6.2 - WEIGHT COMPARISON OF DESIGN CONCEPTS
FIRST STAGE ENGINE INTEGRATION

	DWG. AJG-100 Dry Bay Long Skirt F.D. Engine	DWG. AJG-101 Dry Bay Short Skirt F.D. Engine	DWG. AJG-102 Head Mounted Short Skirt F.D. Engine	DWG. AJG-103 Head Mounted Semi-Short Skirt Bell Engine	DWG. AJG-104 Dry Bay Semi-Short Skirt Bell Engine
Aft Bulkhead	900	900	---	---	900
Thrust Structure	2550	2550	3650	4400	4000
Skirt	6400	4800	4950	5750	6150
Heatshield	---	1450	1450	2250	2250
Engine	14,000	14,000	14,000	15,000	15,000
TOTAL	23,850	23,700	24,050	27,400	28,300

for the first stage is seen to be approximately 4000 lb heavier than a design for a forced-deflection nozzle. Approximately 1000 lb is due to engine weight differences and 3000 lb is attributable to thrust structure, skirt, and the base heating provisions.

The change in performance is not primarily due to weight reduction, but rather, is due to engine low altitude performance characteristics. However, use of the advanced engine concept for second stage application may significantly improve vehicle performance due to weight reduction. These weight reductions occur as described below:

(1) As compared for the first stage, thrust structure and engine attachment is lighter;

(2) The relation of nozzle maximum diameter to interstage diameter results in less weight of base heating provisions;

(3) The shorter forced-deflection nozzle results in a shorter and lighter interstage.

(4) The shorter interstage causes a reduction in first step bending loads which results in a first step tank weight reduction.

Figure 6.3 compares some of these weight differences between use of conventional and advanced engine designs for second stage application. This table is a comparison of significantly affected items from Models 902-1 and 902-3. These two vehicles were optimized at different staging ratios and hence, part of

FIGURE 6.3

WEIGHT COMPARISON OF ENGINE CONCEPTS
SECOND STAGE ENGINE INTEGRATION

	Bell Engine	F-D Engine
Thrust - Lb	382,000	262,000
Aft Bulkhead	450	400
Thrust Structure	950	650
Interstage II	1,850	1,650
Engine	3,900	2,750
Total Stage II	7,150	5,450
Interstage I	2,700	2,150

the thrust structure weight difference is due to thrust level differences. The heat shield weight difference is due to engine concept and the interstage weight difference is due to a "forced-deflection" nozzle being shorter than the "bell" nozzle. An additional weight increment which has not been evaluated for this configuration is possible due to the resulting reduction in bending loads on the first step tank. A further increment might accrue for some configurations due to stability relationships.

6.2 PARAMETRIC TRADE STUDIES

The following parametric weight studies have been performed in support of the configuration evaluation weight studies described previously and the thrust versus cost analysis described in the economic evaluation section.

6.2.1 Thrust/Weight Ratio Single-Stage-To-Orbit Vehicle (Model 902-4)

The single-stage-to-orbit vehicle was iterated and designed at a thrust/launch weight ratio of 1.4 instead of 1.8. To establish this value, single-stage vehicles were analyzed at various values of thrust-to-weight ratio as shown in Figure 6.4. The step mass ratio (λ'), payload, and payload/launch weight parameters are illustrated in Figure 6.5. A thrust-to-weight ratio of 1.8 is shown to provide a maximum payload/launch weight ratio. Figure 6.5 also shows payload/inert weight (W_{pl}/W_{ip}). This is maximum at a T/W_o of approximately 1.4, and was considered to be a closer indication of economic efficiency.

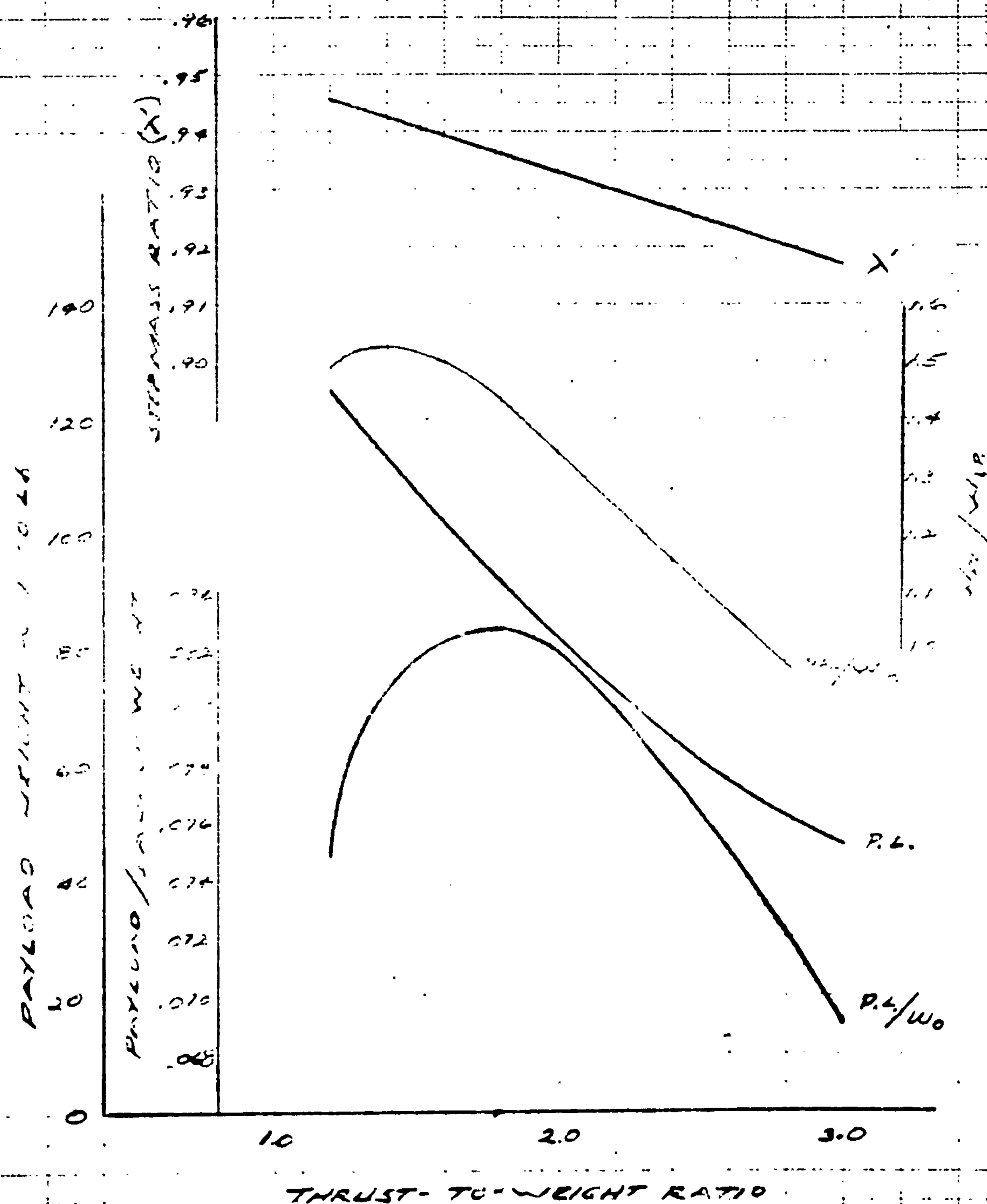
6.2.2 Vehicle Size Effects

Figure 6.6 and 6.7 provide a parametric evaluation of a LO_2/LH_2 vehicle at launch thrusts varying from 0.6×10^6 to 6.0×10^6 lb. These data are again based on interpolation of Reference 15.2 results with corrections for the design criteria differences as discussed in Section 6.1.

Figure 6.7 indicates that step mass ratio remains essentially

FIGURE 6.4
THRUST-TO-WEIGHT RATIO OPTIMIZATION
SINGLE-STAGE-TO-ORBIT VEHICLES
LO_2-LH_2
THRUST = 2×10^6 LB

T/W =	1.2	1.4	1.8	2.4	3.0
Propellant Tanks	25,500	21,500	16,300	13,000	11,000
Thrust Structure	4,000	3,750	3,300	3,100	2,900
Skirt	2,500	2,350	2,000	1,900	1,800
Interstage Structure	700	600	500	500	500
Separation Provisions	100	100	100	100	100
Slosh and Anti-Vortex Provisions	1,000	900	800	800	800
External Insulation	3,000	2,700	2,200	1,900	1,600
Miscellaneous Structure	1,400	1,300	1,000	700	600
Total Structure	(38,300)	(33,200)	(26,200)	(22,000)	(19,300)
Equipment	4,200	3,800	3,000	2,300	2,000
Engine (Wet)	20,000	20,000	20,000	20,000	20,000
Propellant System	3,400	3,200	2,800	2,500	2,300
Pressurization System	7,000	6,200	5,200	4,000	3,400
Residuals	10,700	9,200	7,000	5,300	4,500
Total Inert Weight	(83,600)	(75,600)	(64,200)	(56,100)	(51,500)
Propellant - LH_2	182,300	155,000	119,300	89,100	71,100
LO_2	1,276,100	1,084,700	835,500	623,500	498,000
Total Step Weight	1,542,000	1,315,300	1,019,000	768,700	620,600
Stage Mass Ratio	,875	.8276	.8594	.8552	.8536
Step Mass Ratio	.9458	.9425	.9367	.9270	.9170
Startburn	1,666,700	1,428,500	1,111,000	833,300	666,700
Burnout	208,300	188,800	156,200	120,700	97,600
Payload (Approx.)	124,700	113,200	92,000	64,600	46,100



SINGLE-STAGE-TO-ORBIT

$T = 2 \times 10^6$ LB

D2-12072

FIG. 6.5

THE BOEING COMPANY

	MODEL 902-1a $T=0.6\times10^6$ LO_2/LH_2 STEP I	STEP II	MODEL 902-1 (BASE LINE) $T=2\times10^6$ LO_2/LH_2 STEP I	STE[P II]
PROPELLANT TANKS	8,500	1,600	22,500	4,
THRUST STRUCTURE	800	200	3,800	
SKIRT	1,500	-	3,000	
INTERSTAGE STRUCTURE	600	450	2,500	2,
SEPARATION PROVISIONS	100	50	100	
SLOSH AND ANTI-VORTEX PROVISIONS	450	200	1,100	
EXTERNAL INSULATION	1,700	400	2,900	
MISCELLANEOUS STRUCTURE	450	100	1,200	
TOTAL STRUCTURE	(14,100)	(3,000)	(37,100)	(9
EQUIPMENT	1,500	400	3,900	1
ENGINE (WET)	4,800	1,300	15,000	3
PROPELLANT SYSTEM	1,900	1,000	3,400	1
PRESSURIZATION SYSTEM	2,000	400	6,200	1
RESIDUALS	3,000	700	9,800	2
TOTAL INERT WEIGHT	(27,300)	(6,800)	(75,400)	(20
PROPELLANT - FUEL				
PROPELLANT - FUEL	154,600	13,300	185,000	4
- OXIDIZER	327,800	79,600	1,110,200	27
TOTAL STEP WEIGHT	1,409,700	99,700	1,370,600	34
STEP MASS RATIO (λ′)	.933	.932	.945	
LAUNCH WEIGHT	545,500		1,847,600	
BURNOUT WEIGHT - STEP I	163,100		552,400	
STARTBURN WEIGHT - STEP II	135,800		477,000	
BURNOUT WEIGHT - STEP II	42,900		150,700	
PAYLOAD WEIGHT	36,100		129,900	

1

Calc			REVISED	DATE	EFFECT OF SIZE OR WEIGHT
Trac					
Chk					
Appr.					BOEING AIRPLANE COMPANY SEATTLE 24 WASHINGTON
Appr.					

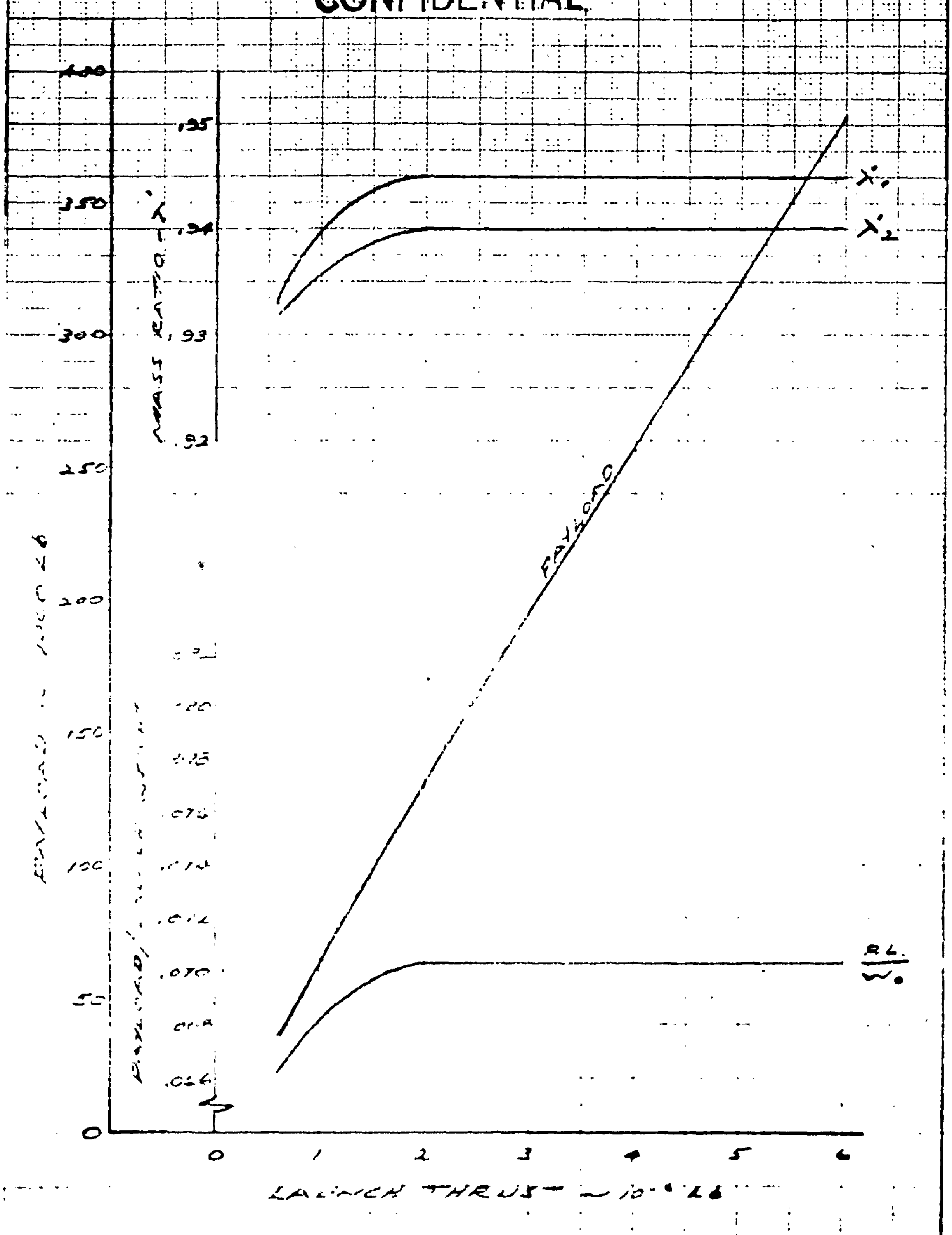
DATA SH[EET]

6 MODEL 902-1 (BASE LINE) LO_2/LH_2 P I	STEP II	MODEL 902-1b T=4x10⁶ STEP I	STEP II	MODEL 902-1e T=6.0x10⁶ LO_2/LH_2 STEP I	STEP II
2,500	4,900	47,500	11,400	75,000	20,500
3,800	600	9,000	1,400	15,000	1,500
5,000	-	6,500	-	10,000	-
2,500	2,200	6,000	5,900	10,000	9,000
100	100	200	100	200	200
1,100	400	1,900	750	2,800	1,100
2,900	800	4,200	1,500	5,500	2,200
1,200	400	2,500	450	4,000	1,000
7,100)	(9,400	(77,800)	(21,300)	(122,500)	(21,300)
3,900	1,400	7,300	2,300	11,000	3,800
5,000	3,900	26,800	7,300	38,800	7,300
3,400	1,900	4,700	2,300	5,900	3,200
5,200	1,900	12,000	3,000	18,000	4,500
9,800	2,300	19,000	4,600	28,000	7,000
5,400)	(20,800)	(147,600)	(40,800)	(224,200)	(61,200)
,000	46,600	364,100	91,800	546,300	137,500
,200	279,700	2,184,700	550,900	3,277,700	824,800
,600	347,100	2,696,400	683,500	4,048,200	1,023,500
45	.940	.945	.940	.945	.940
1,847,600		3,636,000		5,455,000	
552,400		1,087,200		1,631,000	
477,000		939,600		1,406,800	
150,700		296,900		444,500	
129,900		256,100		383,300	

2

DATA SHEET

MODEL 902-1e T=6.0x10[6] LO_2/LH_2 STEP I	STEP II
75,000	20,500
15,000	1,500
10,000	-
10,000	9,000
200	100
2,800	1,100
5,500	2,200
4,000	1,000
(122,500)	(21,300)
11,000	3,800
38,800	7,300
5,900	3,200
18,000	4,500
28,000	7,000
(224,200)	(61,200)
546,300	137,500
,277,700	824,800
,048,200	1,023,500
.945	.940
5,455,000	
1,631,000	
1,406,800	
444,500	
383,300	



EFFECT OF SIZE ON WEIGHT
LO2/LH2
CONVENTIONAL ENGINES

D2-12072

FIG. 6.7

THE BOEING COMPANY

constant when the vehicle size is greater than a launch thrust of approximately 2 x 10^6 lb. Therefore, the payload-to-launch weight ratio also remains essentially constant.

This trend of constant mass ratio for large vehicles is somewhat contradictory to weight data which may be observed in the Reference 15.5 study. That study indicates a reduction in step mass ratio as size is increased. This is due to the difference in engine concept. In the reference study the "plug" engine was an increasingly larger percent of propellant weight as thrust increased, causing the reduction in step mass ratio.

6.2.3 Tank Configuration

A ground rule established early in this study was that the LO_2 tank would be placed above the LH_2 tank to aid the stability problem. A study was subsequently performed to investigate the implications of reversing the location of the LO_2 and LH_2 tanks. Figure 6.8 shows that a tank weight saving of approximately 2400 lb may be realized with LO_2 below the LH_2. However, to maintain vehicle neutral stability approximately 7000 lb of fin weight must be added. Other weight differences such as propellant feed system are negligible.

Placing the lox tank above the hydrogen tank is therefore more optimum for this configuration to provide neutral stability.

FIGURE 6.8 - EFFECT OF TANK ARRANGEMENT ON COMPONENT WEIGHTS

	LO_2 FWD	LO_2 AFT
TANK CYLINDER	18,450	15,900
FWD BULKHEAD	800	800
INTERMEDIATE BULKHEAD	1,850	1,975
AFT BULKHEAD	900	925
STABILITY FINS	---	7,000
TOTAL	22,000	26,600

7.0 STRUCTURES

7.1 INTRODUCTION

This section presents the structural design studies conducted during this program. The various booster configurations are described and discussed. The results and conclusions of this study are based to a large extent on the results of the Boeing study covered by reference 15.2.

The major structural design effort during this program was concentrated on the comparison of bell and forced-deflection engine installations for a first stage booster using LH_2/LO_2 propellants. The design approach was to first establish a baseline vehicle and then study the various elements such as thrust structure, interstage structure, and ground support structure that are affected by the differences in the two engines.

The design study indicates that the installation weight for a forced deflection engine is significantly lighter than for a bell engine. This lighter weight results from the shorter length of the thrust structure and the elimination of engine gimbaling requirements with the forced deflection engine. However, since thrust structure is only a small fraction of total stage inert weight, the weight saving is not significant from an overall vehicle performance standpoint.

7.2 STRUCTURAL DESIGN CRITERIA

The criteria established for the study are outlined below:

7.2.1 Safety Factors

Ultimate factor of safety = 1.4

Yield factor of safety = 1.1

7.2.2 Ground Support

The vehicle shall be free standing on the launch pad without tank pressurization and with any combination of propellant tanks filled.

7.2.3 Ground Winds

The vehicle shall be capable of withstanding ground wind loads due to a 40 mph steady wind plus a 20 mph gust while free standing on the launch pad.

7.3 GENERAL DESCRIPTION

Baseline Configuration

Figure 4.1 presents a layout of the Model 902-1 LO_2/LH_2 baseline configuration. The fuel and oxidizer are contained in a single tank with the oxidizer located forward and separated from the fuel by a single bulkhead. The oxidizer is located forward to improve vehicle neutral stability and reduce the magnitude of the engine gimbal angles required for control. The tank length to diameter ratio is based on results of the reference 15.2 study.

The propellant tanks are of aluminum construction with an integrally stiffened, semi-monocoque cylindrical shell, a .75 to 1 elliptical upper bulkhead, and a hemispherical divider bulkhead. The lower bulkhead varies with the type of thrust structure and engine. The divider bulkhead design provides the required insulation between the hydrogen and oxygen portions of the tank and is capable of withstanding a collapse pressure. The LH_2 tank includes thermal protection to prevent excessive boiloff on the ground and during flight.

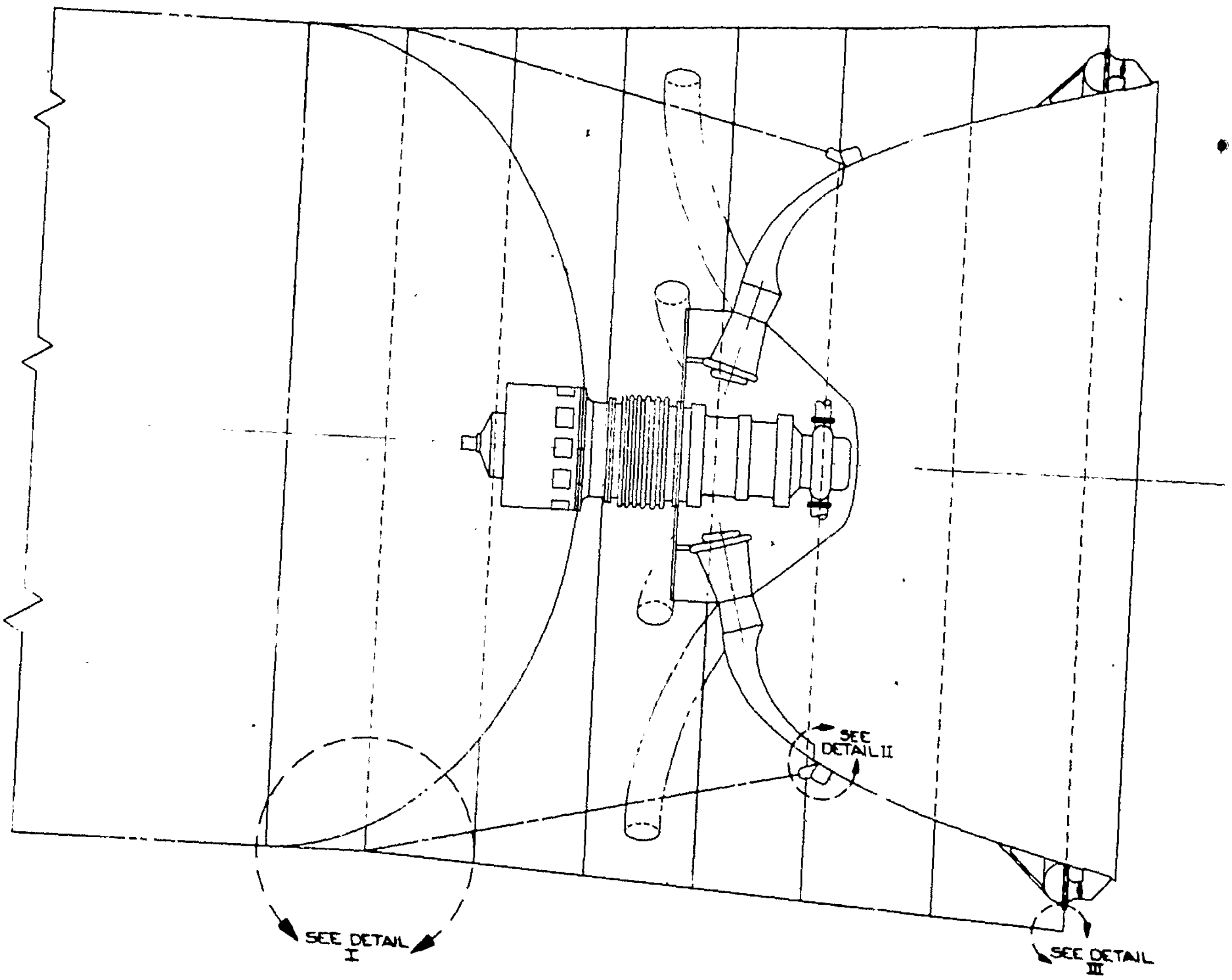
An aluminum semi-monocoque interstage design is used to join the first and second stages. The ground support skirt and thrust structure are of aluminum semi-monocoque type construction. The ground support skirt is skin-stringer design with an integral ground connecting ring. These are shown by Figures 7.4 and 7.5 and are applicable to Models 902-1 and 902-2. The bell nozzle engine skirt mounted thrust structure and the force-deflection engine, dry bay, skirt mounted thrust structure are skin-stringer construction. The head mounted thrust structure is a wet-bay, milled skin construction with either integral milled frame-stringer or waffle pattern design.

7.4 ENGINE MOUNT COMPARISONS

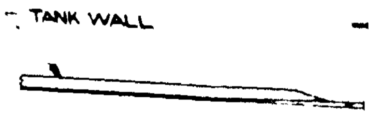
Five thrust structure designs were prepared for the bell nozzle and forced deflection engines. Figures 7.1 through 7.5 show proposed installations for both engines.

Three designs for installation of the forced deflection engine are shown by Figures 7.1 through 7.3 and would be applicable to both Model 902-3 and Model 902-4. Two additional designs for the bell nozzle engines were made for weight comparison with the forced deflection engine. These are shown by Figures 7.4 and 7.5 and are applicable to Models 902-1 and 902-2.

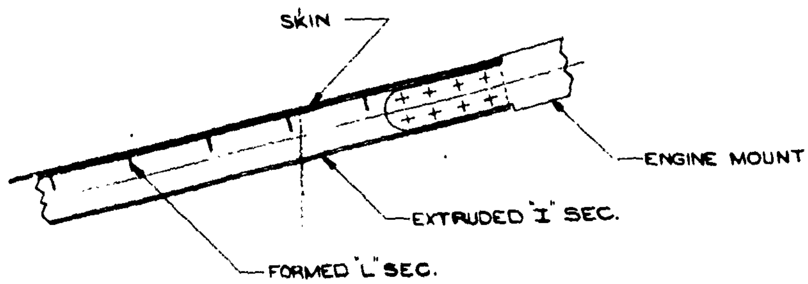
All configurations were designed with flared skirts in lieu of fins to attain neutral stability. The flared skirts also have structural capability for ground support, thus providing a dual function.



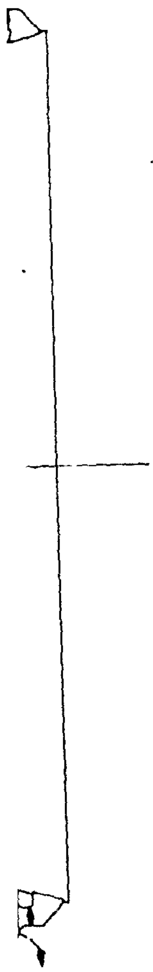
1



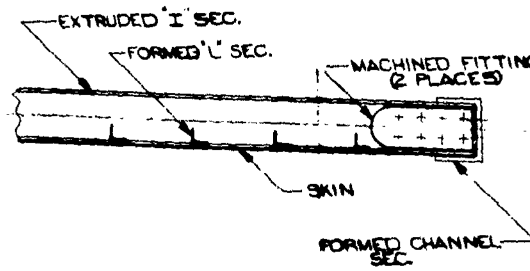
CONFIDENTIAL



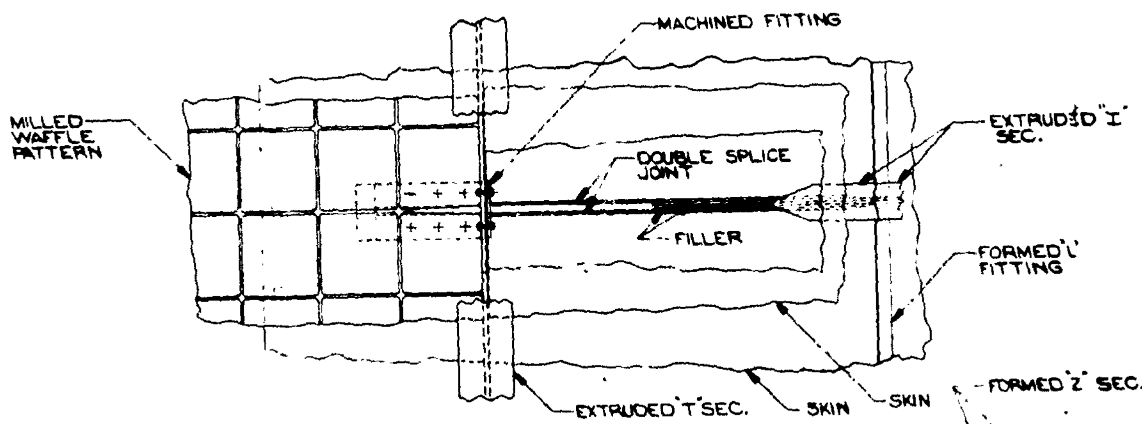
DETAIL II



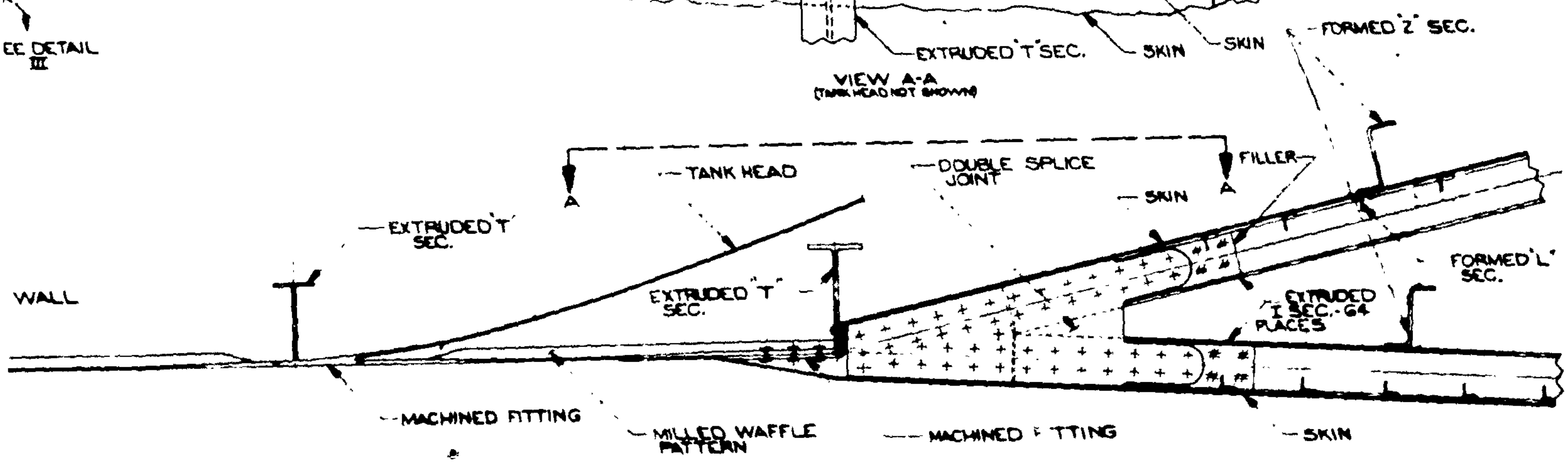
2



DETAIL III

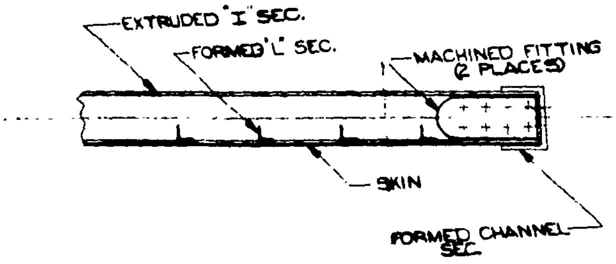


VIEW A-A
(TANK HEAD NOT SHOWN)

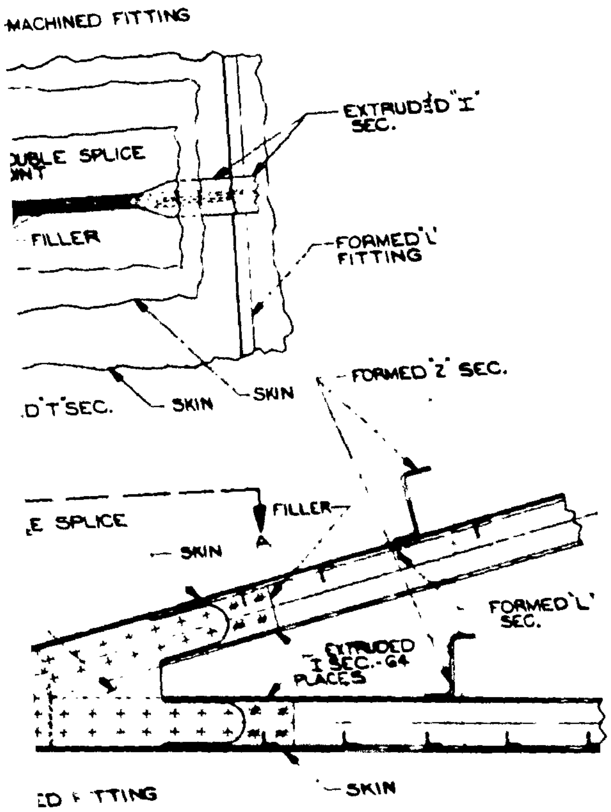


DETAIL I

NE MOUNT



DETAIL III



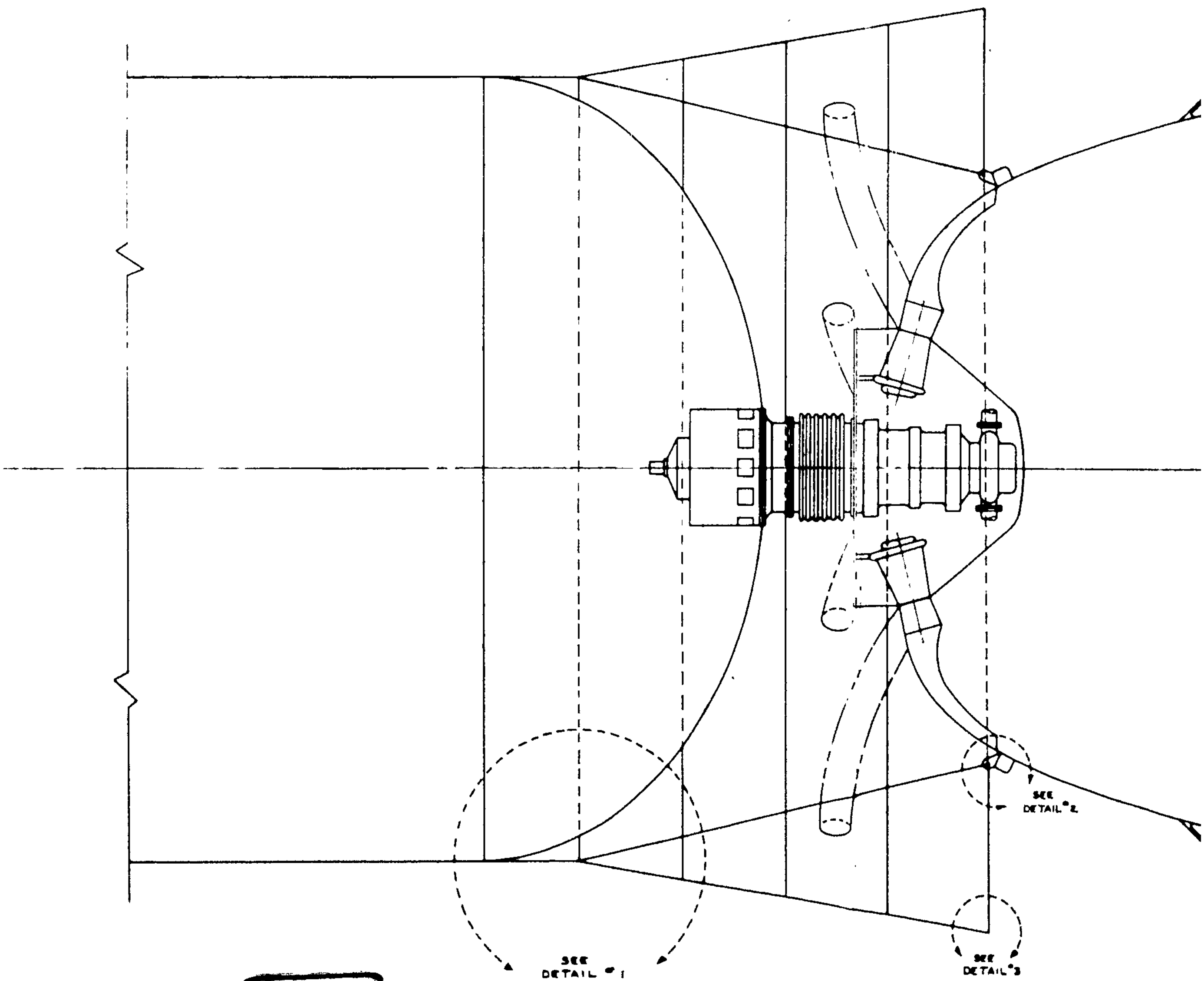
3

FIG. 7.1

D2-12072

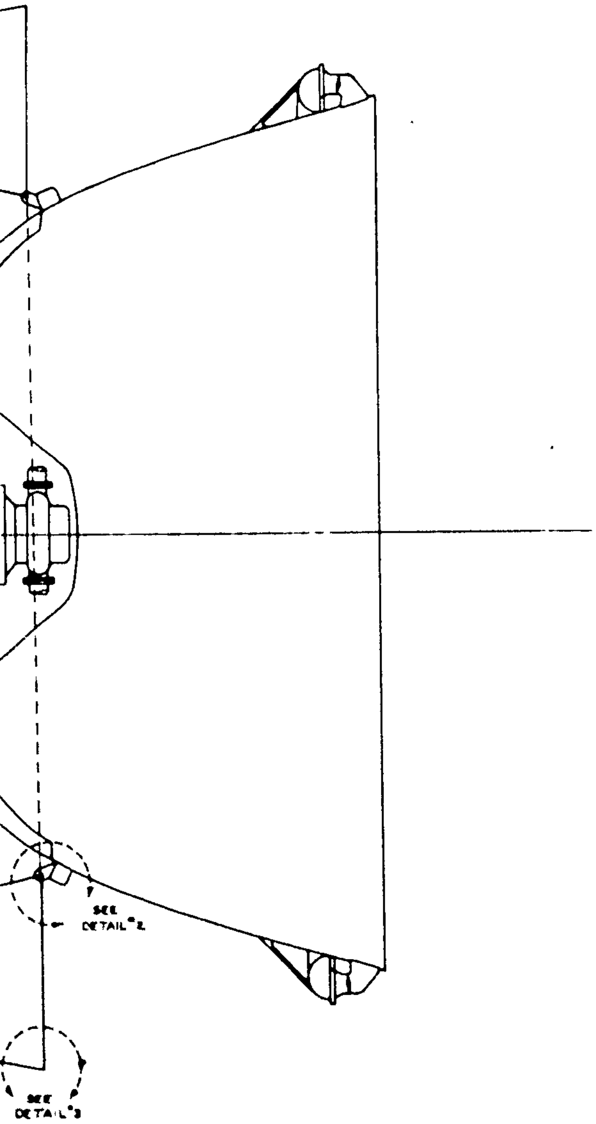
LONG SKIRT-DRY BAY, THRUST STRUCTURE FORCE DEFLECTION ENGINE

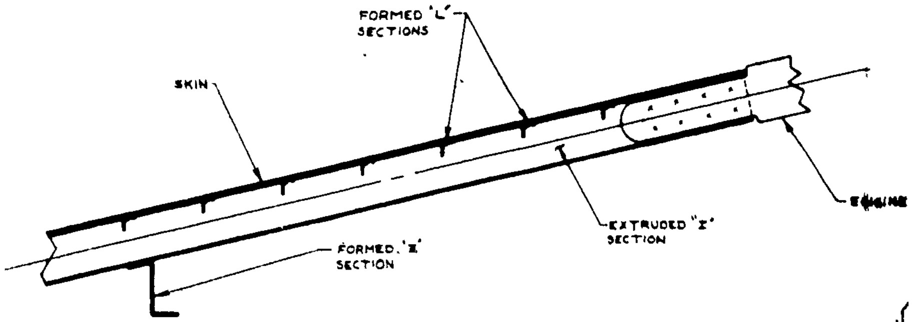
LO-100



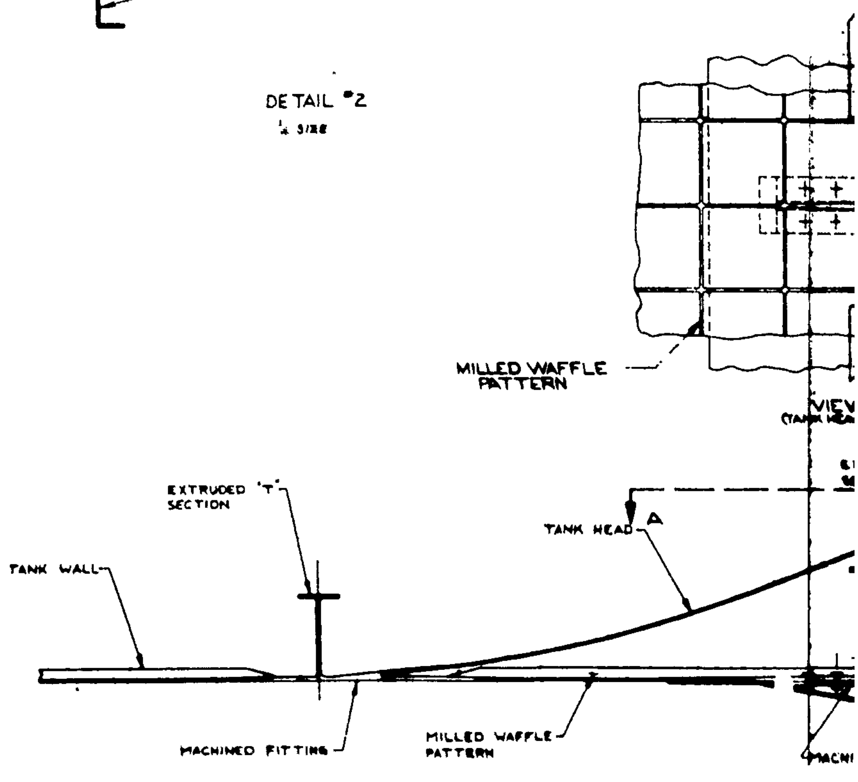
1

CONFIDENTIAL





DETAIL #2
¼ SIZE



DETAIL #1
¼ SIZE

2

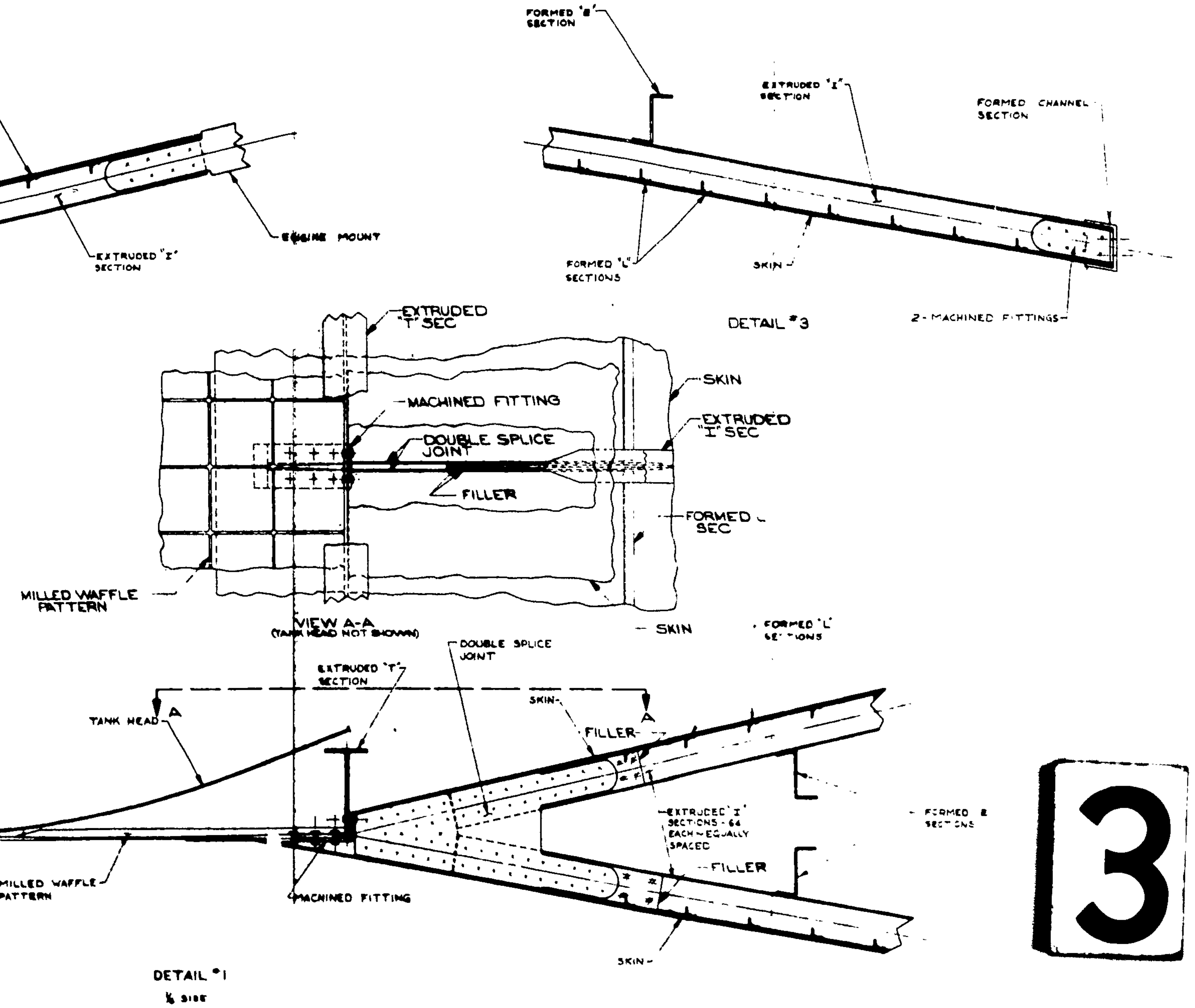
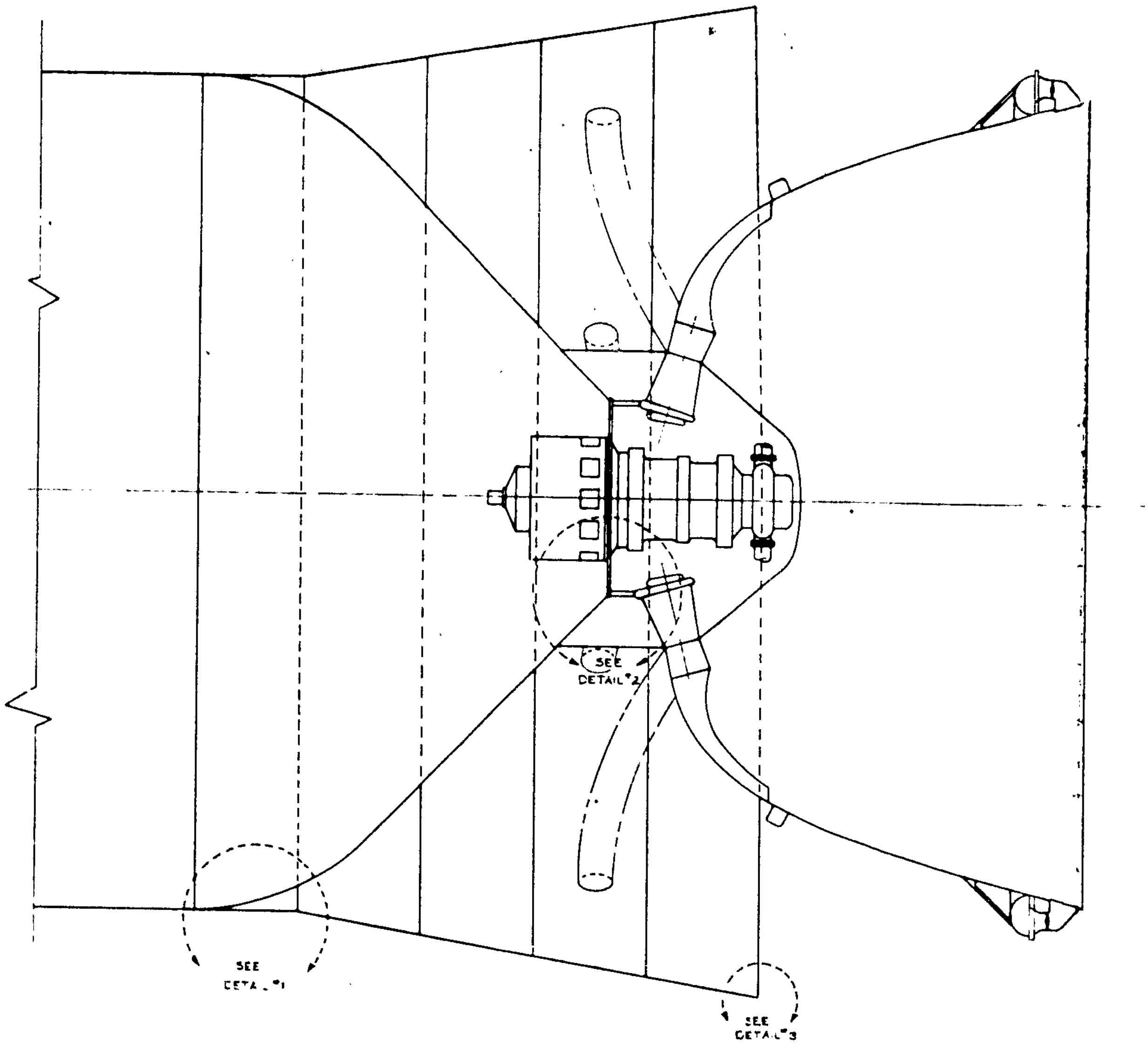


FIG. 7.2 D2-12072

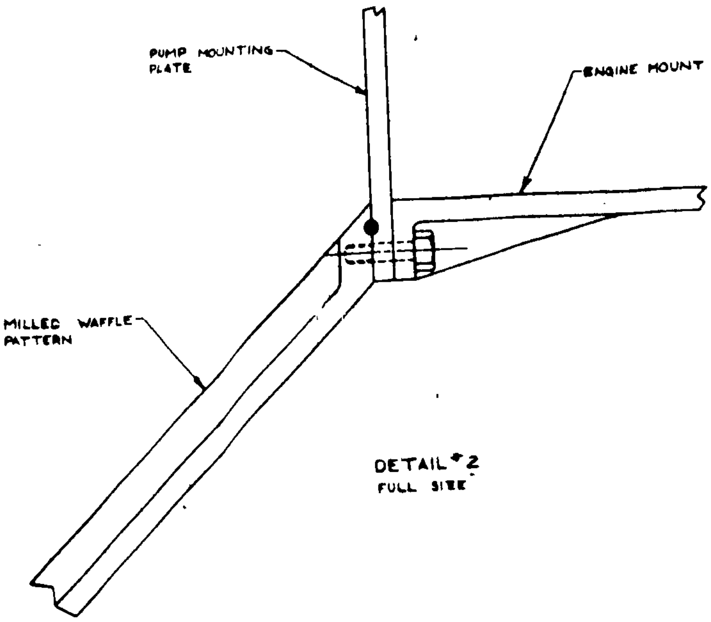
SHORT SKIRT-DRY BAY THRUST STRUCTURE FORCE DEFLECTION ENGINE	101



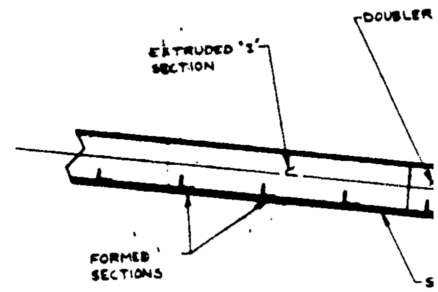
PLAN VIEW

1

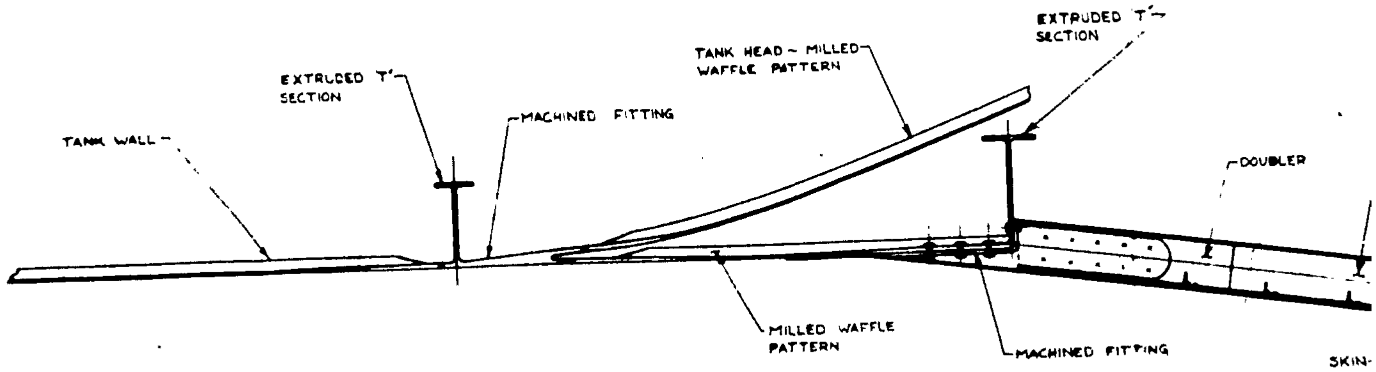
CONFIDENTIAL



DETAIL #2
FULL SIZE

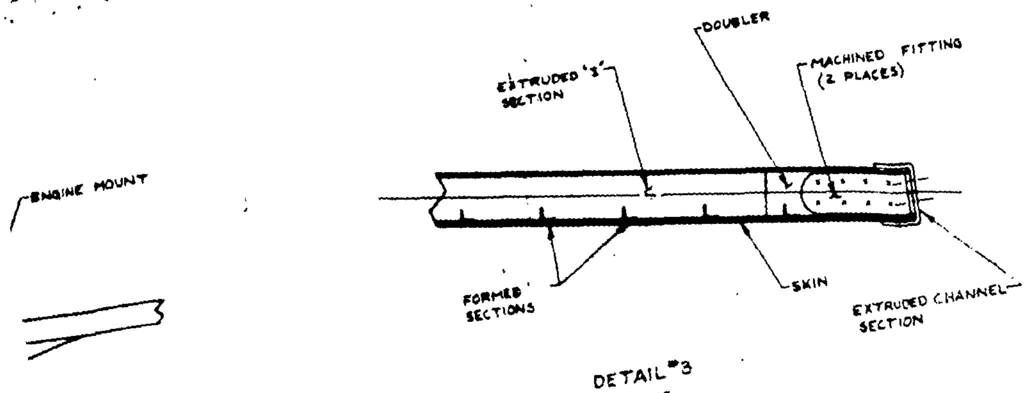


DETAIL #3
¼ SIZE



DETAIL #1
¼ SIZE

2



DETAIL #3
¼ SIZE

3

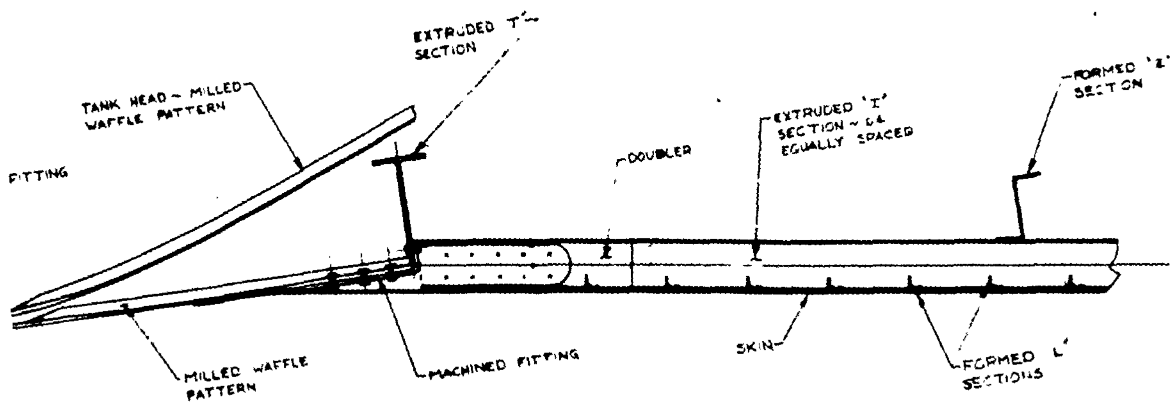
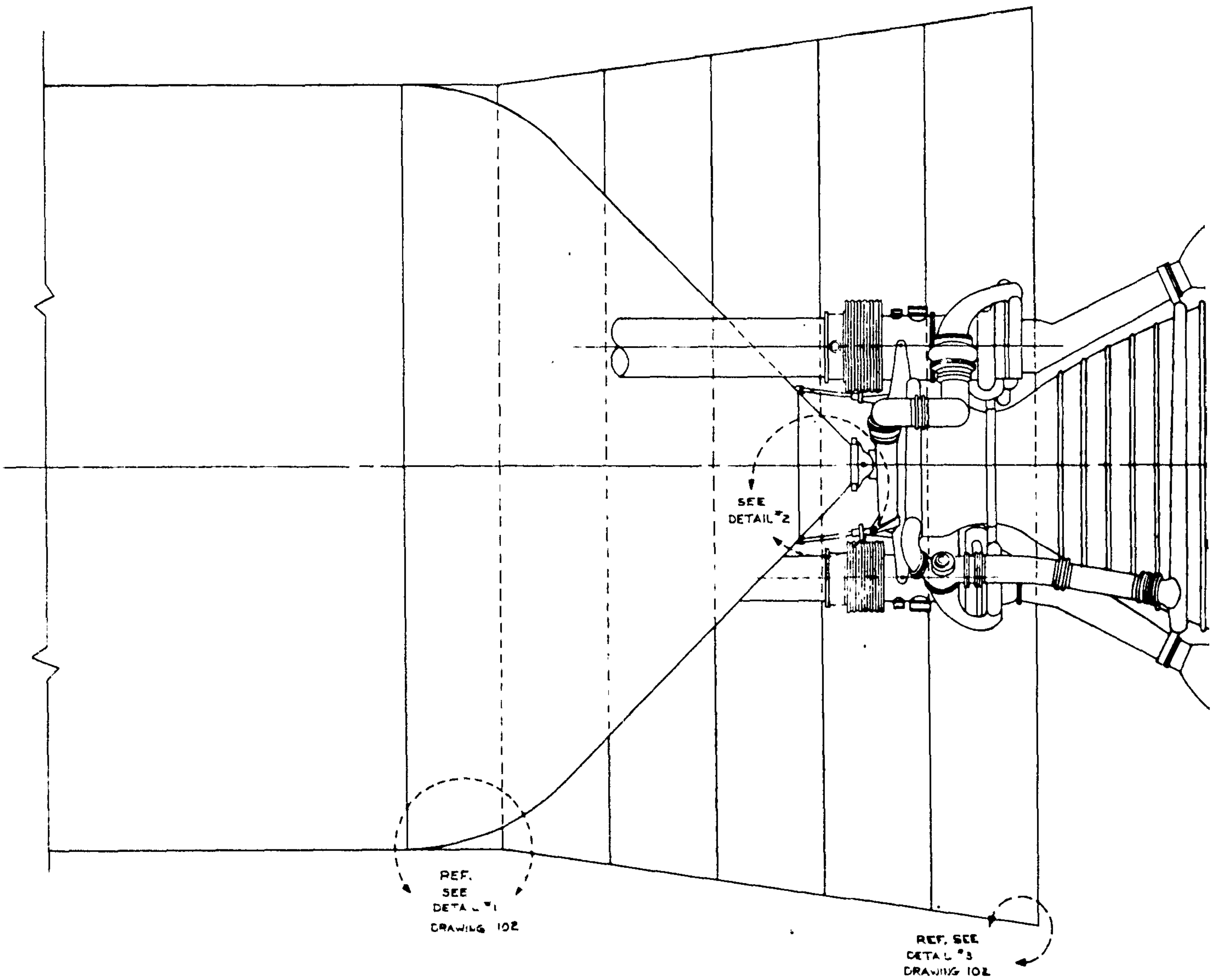


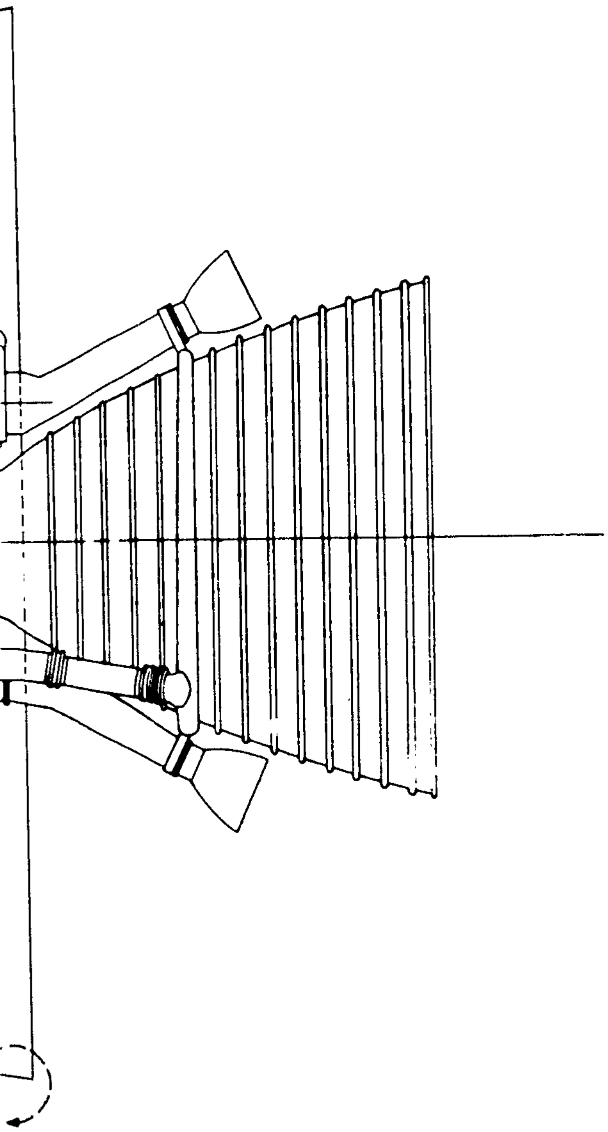
FIG. 7.3 D2-12072

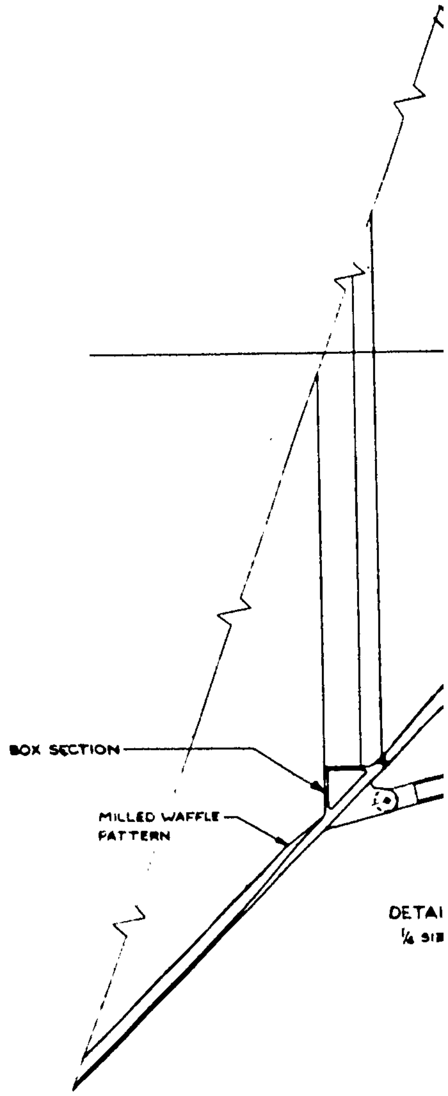
SHORT SKIRT ~ HEAD MOUNTED THRUST STRUCTURE FORCE DEFLECTION ENGINE

LO-102

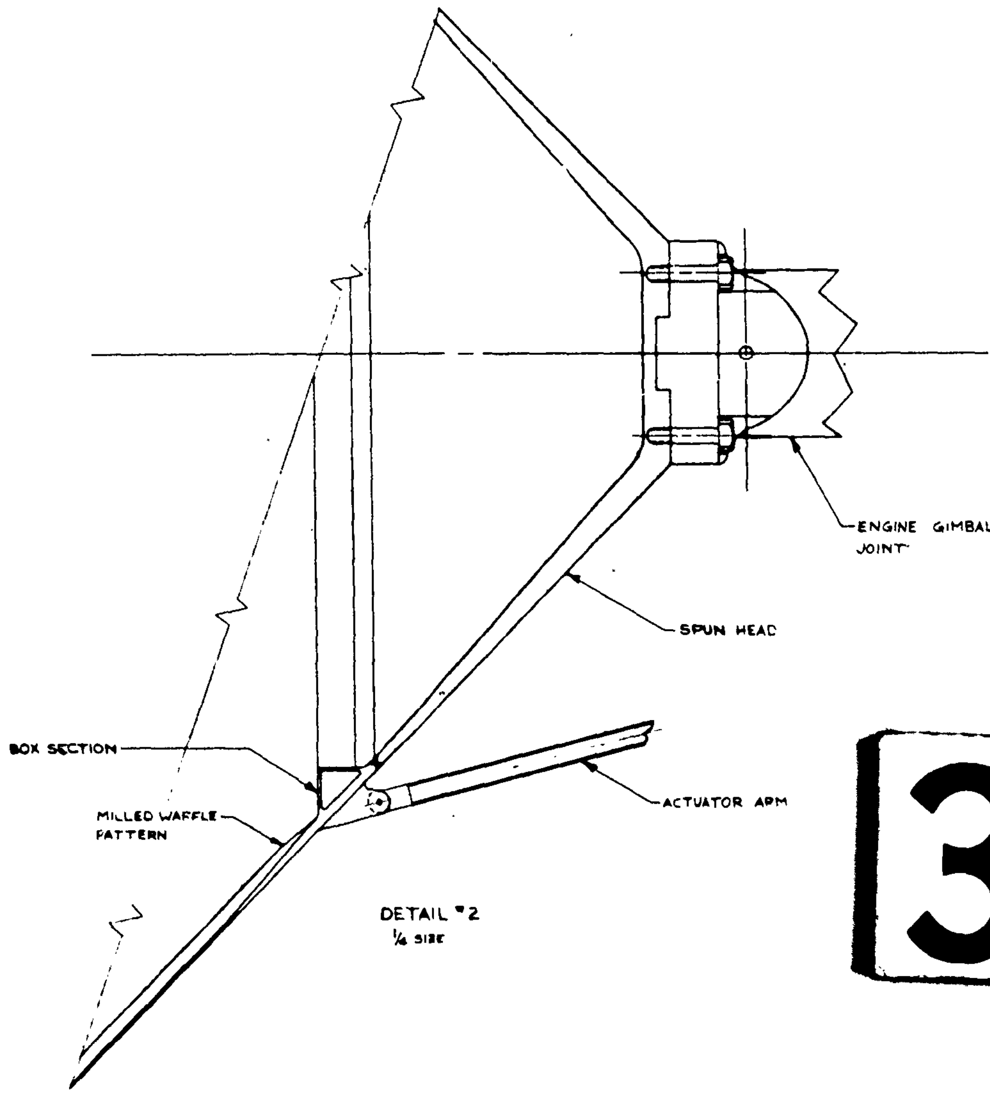

SEE
DETAIL #2
REF.
SEE
DETAIL #1
DRAWING 102
REF. SEE
DETAIL #3
DRAWING 102

1





2



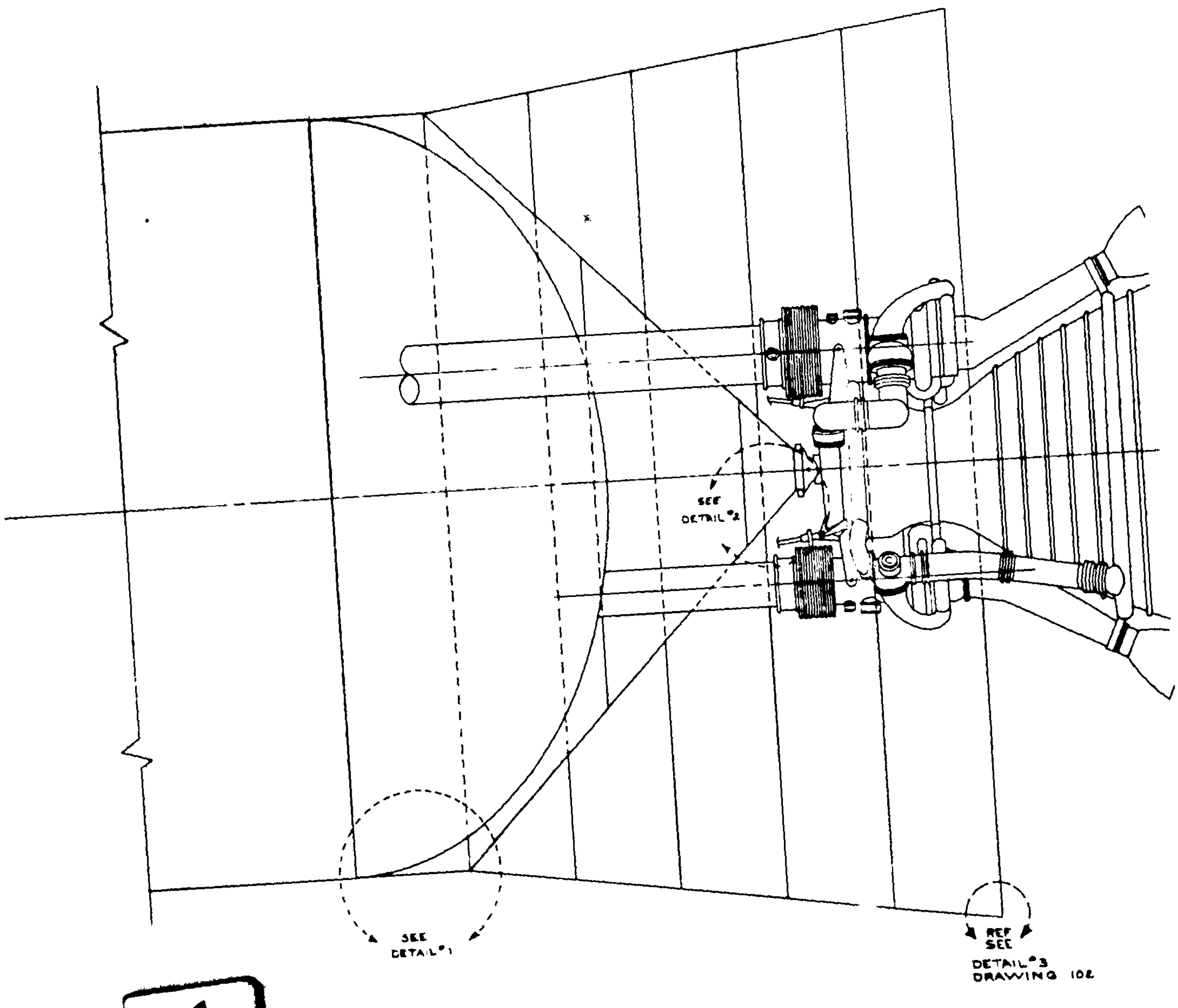
3

FIG. 7.4 D2-12072

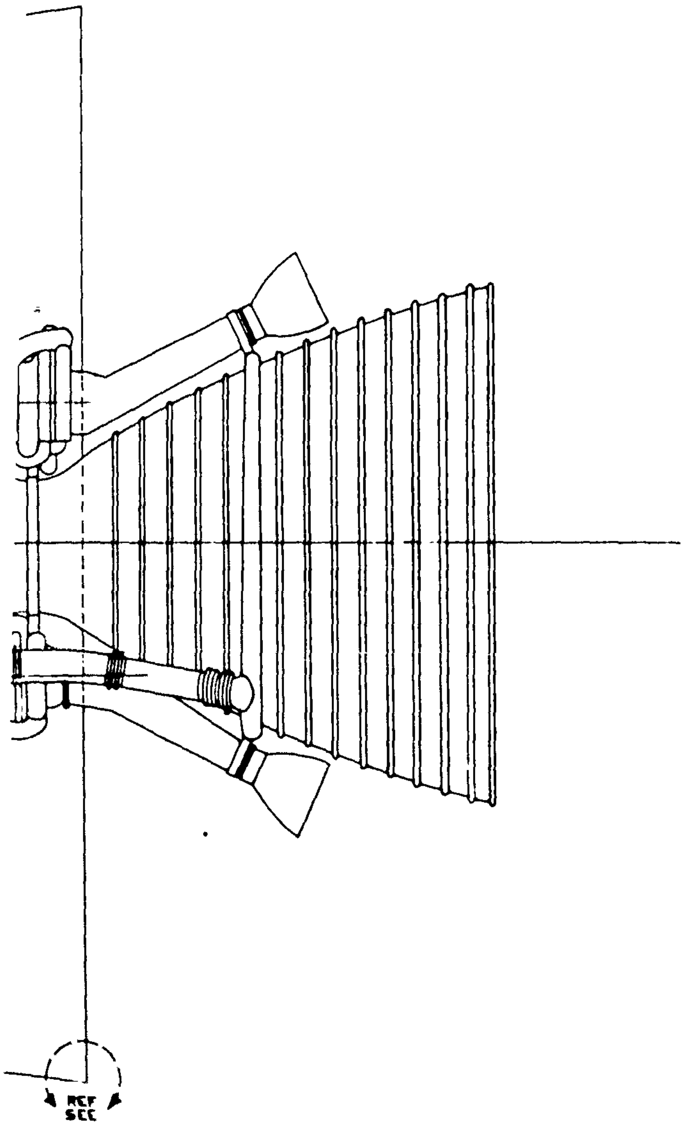
HEAD MOUNTED THRUST STRUCTURE BELL ENGINE

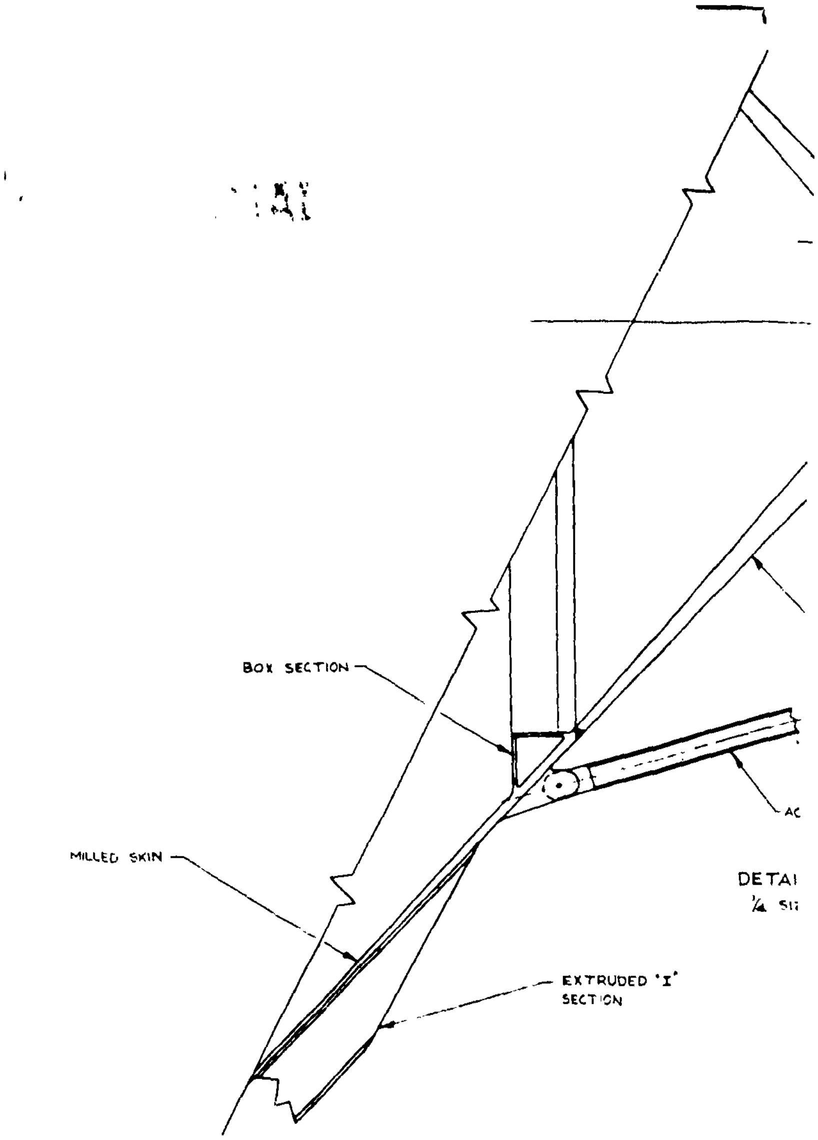
T. BROWN

LO-103

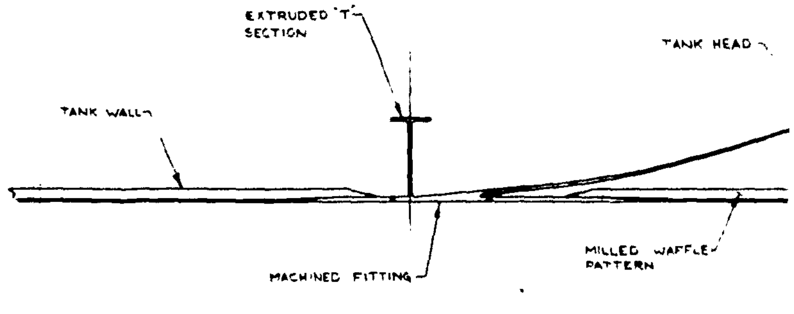


1





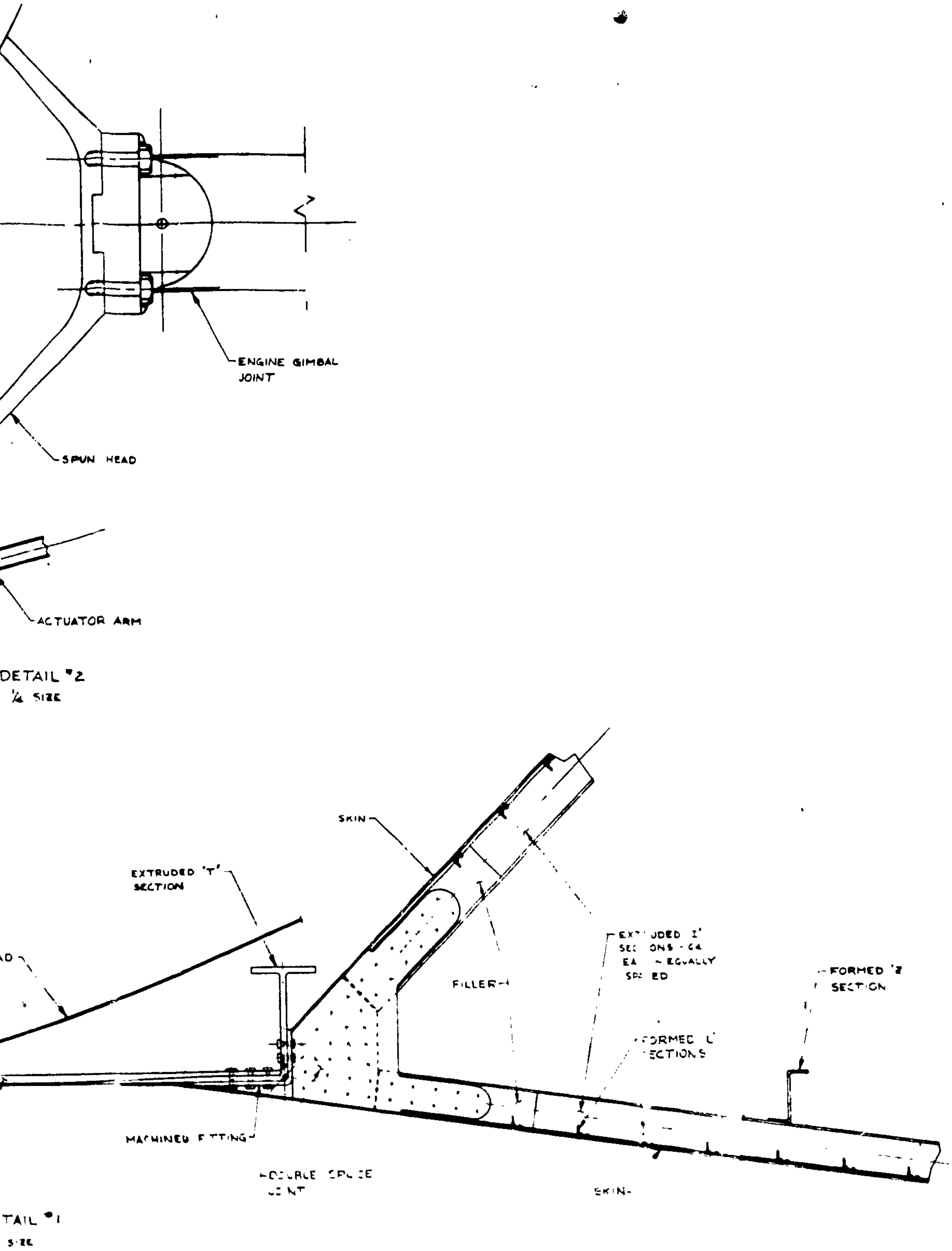
DETAI



DETAIL
1/4 SIZE

2

CONFIDENTIAL



3

CONFIDENTIAL

FIG. 75 D2-12072

DRY BAY
THRUST STRUCTURE
BELL ENGINE

Figure 7.1 shows the forced deflection engine mounted to the thrust structure at the engine C.P. This configuration has a full length flared skirt that serves the function of providing fin effect, heat shield and ground support. Figure 7.2 shows the forced deflection engine installed as above; this configuration has a short flared skirt that ends on a plane with the engine mounts. This skirt serves the same functions as the long skirt except a base heat shield is required.

Figure 7.3 shows the forced deflection engine installed to the head of the tank in a wet bay. The engine pick up is made on top of the engine instead of at the C.P. The flared skirt is identical with that of Figure 7.2 and also requires a heat shield. From an overall vehicle standpoint the long flared skirt design (Figure 7.1) appears most efficient. Ignoring weight effect on the engine, all thrust structure designs considered for the forced deflection engine appear nearly equal from a weight standpoint.

One bell nozzle design (Figure 7.4) installed the engine to the tank head also using the head for thrust structure. The second nozzle utilized a stiffened dry bay cone with a separate elliptical fuel tank head as shown by Figure 7.5. Bell nozzle engine thrust structure installation was found to be slightly heavier, reference sec 6 weight statement.

7.5 STRUCTURAL LOADS

Based on previous study programs, the critical loads for a vehicle of this type with a ballistic payload occur during ground wind, launch, or first stage burnout. These three loading conditions were investigated considering the effects of axial loads, bending moments, and internal pressure.

Tank pressurization was established by propellant utilization requirements and was not increased to help carry design loads.

7.6 EFFECT OF TANKAGE ARRANGEMENT

Neutral stability is enhanced by locating the center of gravity as far forward as possible. Locating the LO_2 forward tends to help this situation. A weight trade study was, therefore, conducted to determine the effect of propellant arrangement on stage inert weight. The tankage structure was sized for both the LO_2 forward and aft conditions. The LO_2 forward condition resulted in tankage 2400 pounds heavier than for the LO_2 aft condition. This weight increase was due to the higher axial loads in the LH_2 tank walls with the LO_2 forward. However, for neutral stability with the LO_2 aft, 1200 sq. ft. of fins are required at a weight of 7000 pounds. This fin requirement results in a net stage inert weight increase of 4600 pounds with the LO_2 aft.

8.0 PROPULSION

8.1 ENGINE SYSTEMS

Four advanced engine concepts were evaluated prior to choice of engine type to be used for the engine-vehicle integration study contained herein. These were the "Plug," the "Reverse Flow" (RF) and two versions of the "Force Deflection" (FD) engine. The two F-D engines differed only as influenced by the chamber pressure (P_c = 1000 psi and P_c = 3000 psi.) The more important characteristics of these engines as supplied by Aerojet General Corporation are shown by Figure 8.1 which also shows the characteristics of the bell engines used in this study on Model 902-1. More detailed descriptions of these engines are included in Aerojet General Corporation reports, Reference 1[illegible].

Reference to Figure 8.1 shows the predicted sea level and vacuum specific impulse to be approximately equal for the advanced engines when operating at P_c = 1000 psi. The main difference appears in the predicted weights, where the F-D engine shows the better characteristics. On this basis it was agreed with Aerojet General that Boeing would concentrate on integration of the F-D engine during this study; the P_c = 1000 psi mission to be used on a two stage vehicle (Model 902-3) with the P_c = 3000 psi version used on the single stage vehicle (Model 902-4.) Performance data used on all configurations as influenced by the engines are shown in Section 5.0.

Figure 8.1

ENGINE CHARACTERISTICS

Engine Type

	Bell P_c = 1000 psi	Plug P_c = 1000 psi	Reverse Flow P_c = 1000 psi	Forced Deflection P_c = 1000 psi	Forced Deflection P_c = 3000 psi
Propellents	LO_2/LH_2	LO_2/LH_2	LO_2/LH_2	LO_2/LH_2	LO_2/LH_2
Area Ratio	20			40	230
Oxidizer/Fuel Ratio	6.0	6.0	6.0	6.0	7
I_{sp} - Sea Level ($\frac{\#-Sec}{\#}$)	346	361	361	361	388
I_{sp} - Vacuum ($\frac{\#-Sec}{\#}$)	413	422	426	426	454
Weights - Lbs	15,000	19,000	15,000	14,000	20,000
Boeing Vehicle Model	902-1 (Also Model 902-2 using LO_2/RP-1 with applicable I_{sp} & mass ratio)	—	—	902-3	902-4 & 902-44

From a performance standpoint the F-D engine has an advantage over the bell engine with the same chamber pressure. That is, the bell engine is forced to use a low area - ratio nozzle because it is optimumly expanded at only one design altitude and the performance above the design altitude must be sacrificed to prevent separation at sea level. The F-D engine can use a higher area ratio nozzle at sea level because separation is prevented by the secondary air flow. Therefore, it has higher performance from sea level to altitude. The F-D engine appears to have a slight weight advantage, is shorter and offers the advantage of using a fixed structure installation since secondary gas injection rather than gimballing can be used for thrust vector control. This allows a lighter connecting structure between engine and airframe.

The high-pressure F-D engine has the advantages of better performance, smaller size, and less weight than the bell. Possible disadvantages include: higher temperatures, high pressure turbo pumps, and longer development times.

8.1.1 Development

Items to be developed on both the bell and F-D concepts include the turbo pumps, especially on the high chamber pressure versions, and the thrust vector control systems.

Peculiar to the bell are the injector design problems and flexible high pressure line connections. The F-D concept will require work in heat transfer, jet interaction, and secondary airflow design.

8.2 PROPELLANT SYSTEM

8.2.1 General Description

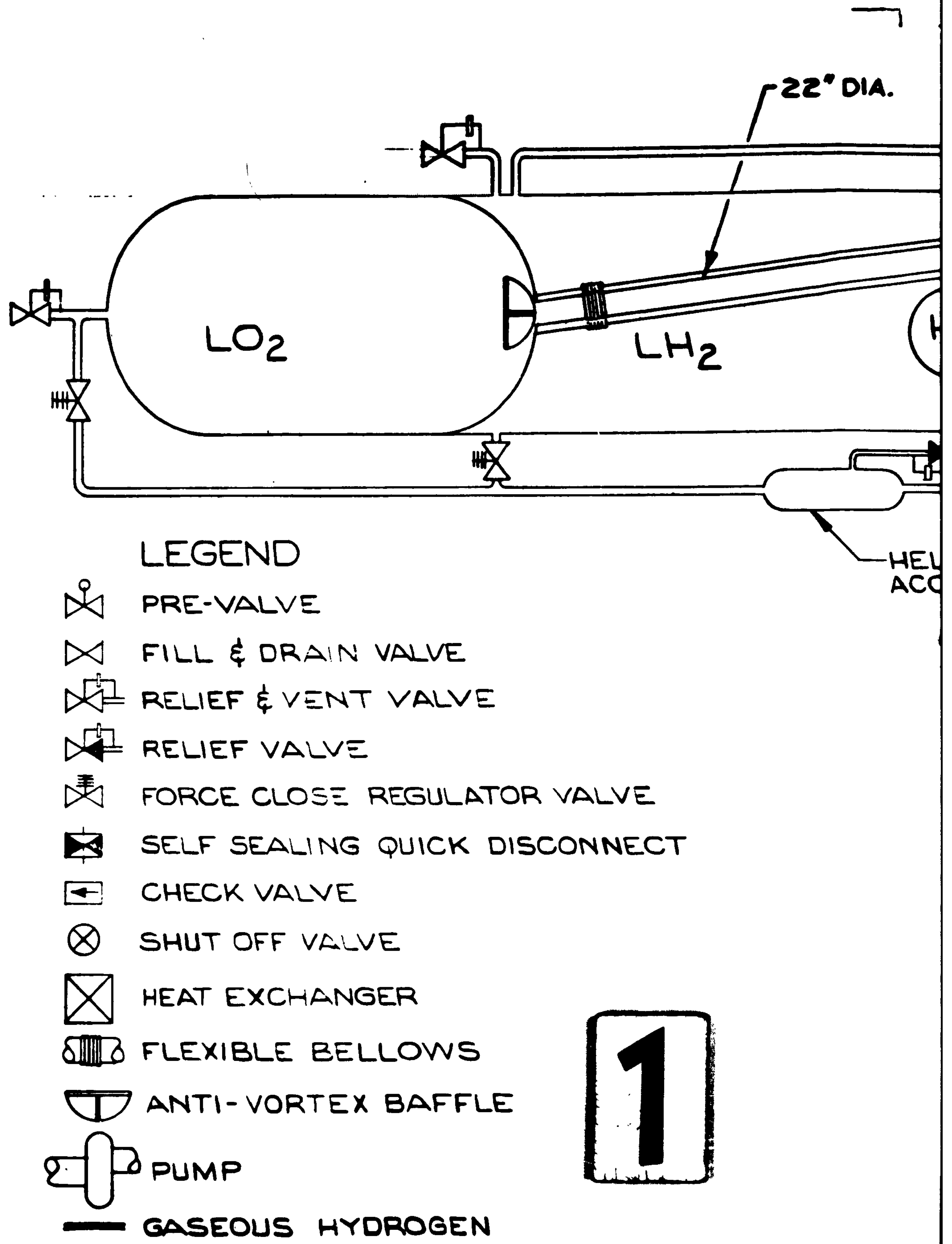
8.2.1.1 Model 902-1 Baseline (LO_2/LH_2 Bell)

The propellant subsystem diagram is shown in Figure 8.2. Both fuel and oxidizer are withdrawn from natural sumps in the bottom of the tanks and routed directly to the engine through pre-valves located immediately upstream of the engine gimbal bellows. The oxidizer line is routed through the hydrogen tank in a double-walled evacuated tube to provide the most direct route and to aid in sub-cooling the oxidizer. The hydrogen line is short and insulated to prevent air liquification. A stored gas helium system provides the expulsion media for both propellants during engine start. At engine start, liquid hydrogen is withdrawn from the high pressure side of the turbopump, vaporized, heated and injected into the hydrogen tank ullage space. Hydrogen gas pressure over-rides the helium flow to the oxidizer tank. Ullage pressure is maintained through standard primary and secondary regulators. A gas accumulator is installed between the two regulators to decouple the system and prevent hunting.

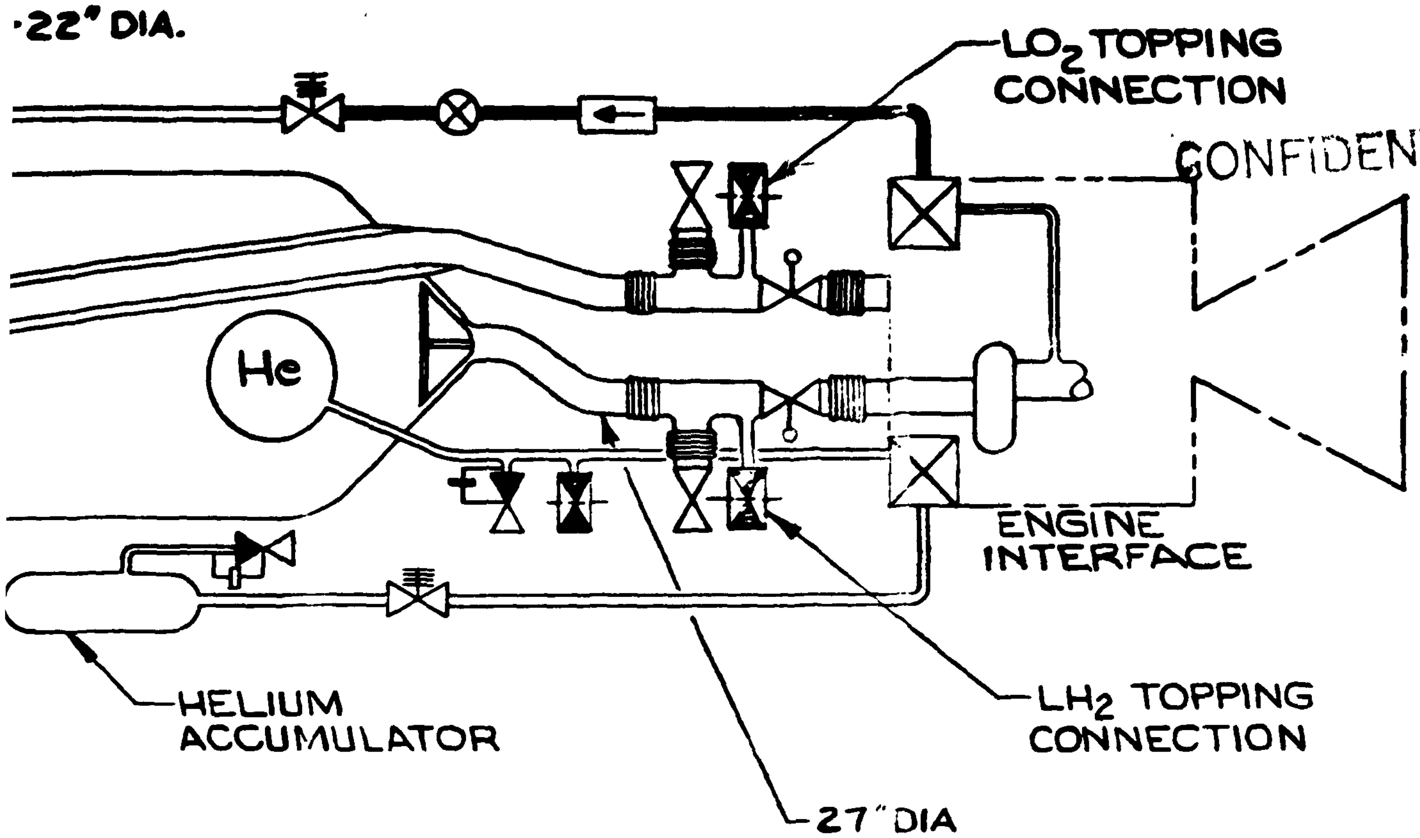
The helium bottle is stored in the hydrogen tank for minimum gas storage volume and bottle weight. Standard fill and topping connections, overpressure relief, check, and shut-off valving complete the system.

8.2.1.2 Model 902-2 Baseline (LO_2/RP-1-Bell)

The propellant subsystem diagram is shown in Figure 8.3. Fuel and oxidizer is withdrawn from the bottom of their respective tanks and routed directly to the engines. The oxidizer line is routed through


22" DIA.
LO2
LH2
LEGEND
PRE-VALVE
FILL & DRAIN VALVE
RELIEF & VENT VALVE
RELIEF VALVE
FORCE CLOSE REGULATOR VALVE
SELF SEALING QUICK DISCONNECT
CHECK VALVE
SHUT OFF VALVE
HEAT EXCHANGER
FLEXIBLE BELLOWS
ANTI-VORTEX BAFFLE
PUMP
GASEOUS HYDROGEN

1



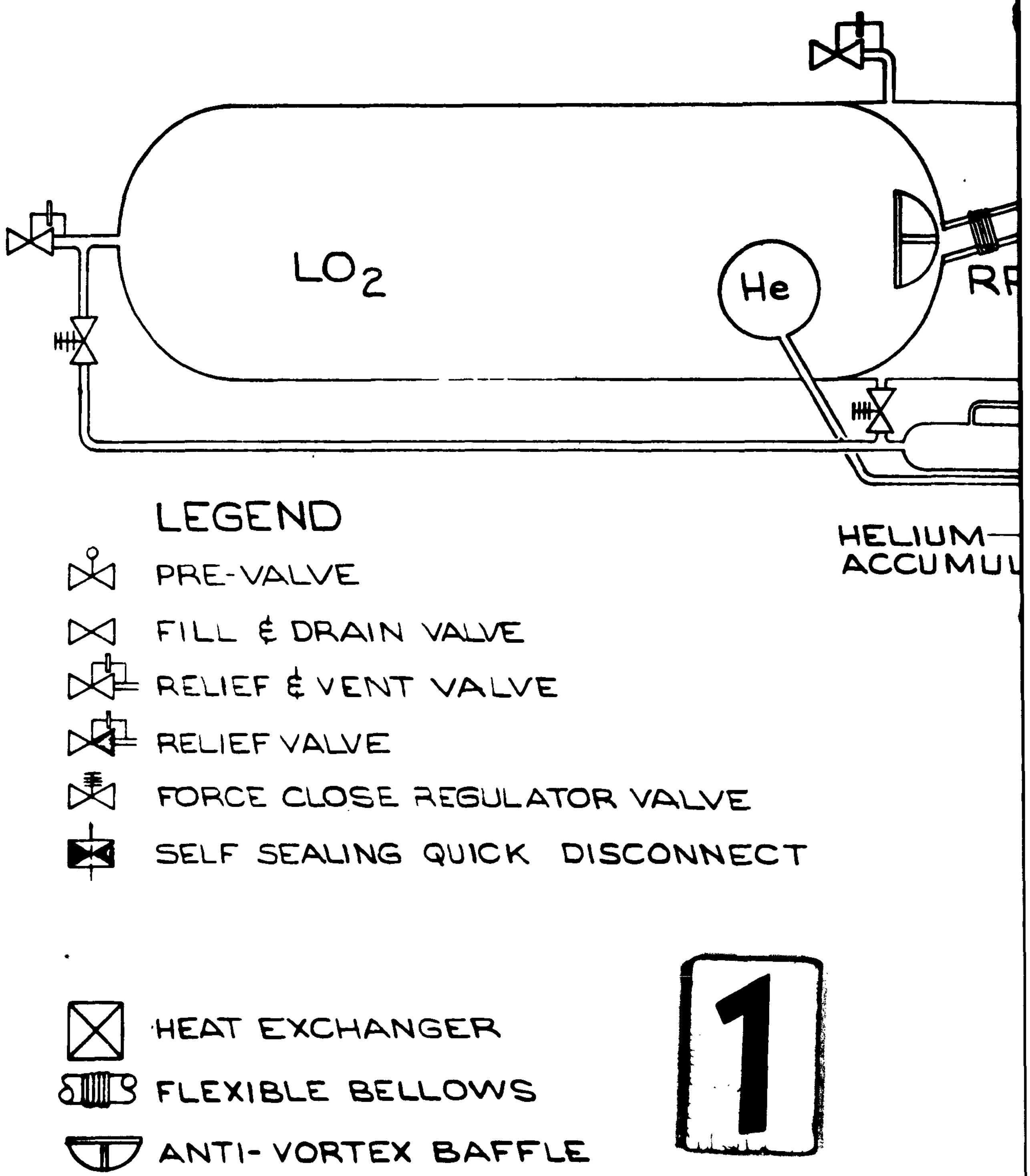
NOTE:
FIRST STAGE ONLY SHOWN-
SECOND STAGE SCHEMATICALLY
IDENTICAL.

2

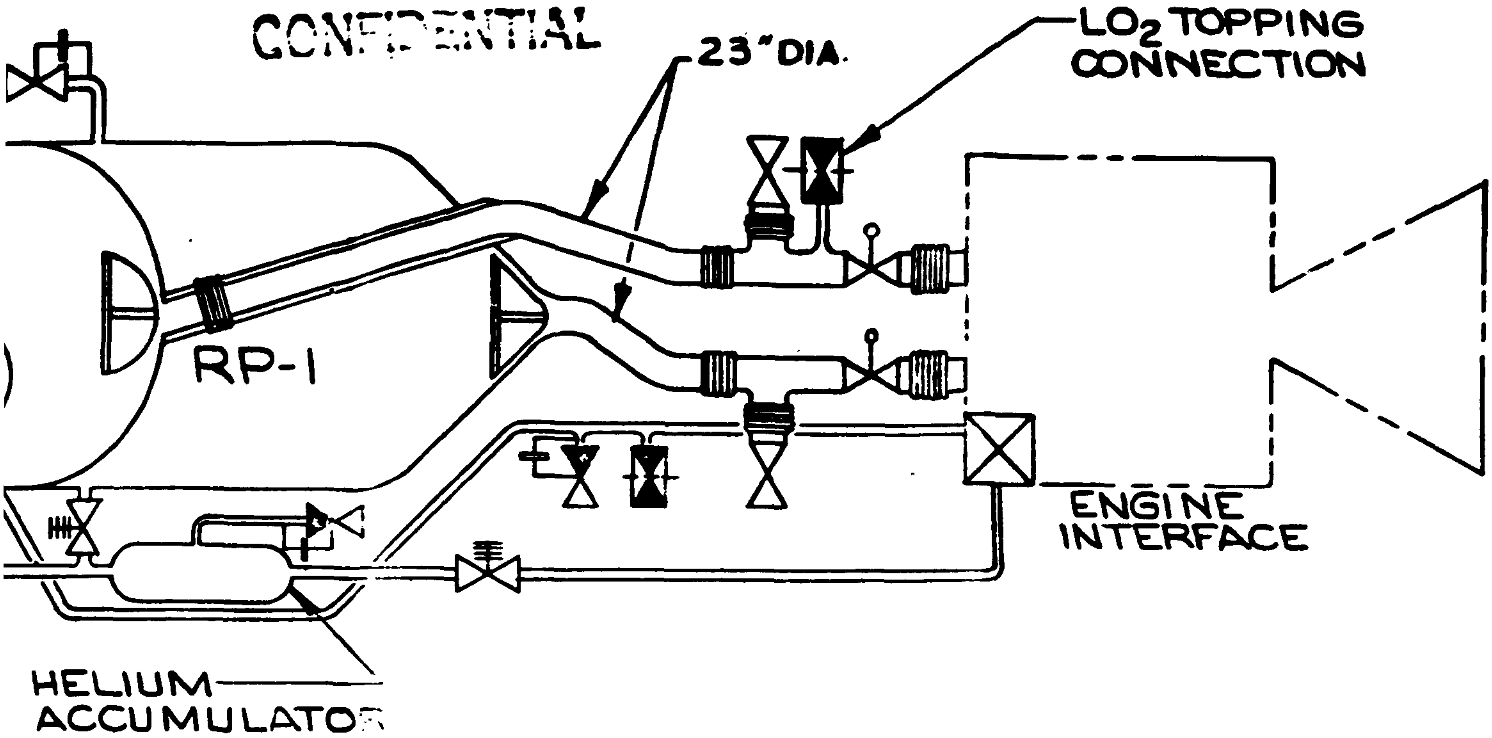
SCALE: NONE

		REVISED	DATE		
CALC				SCHEMATIC-PROPELLANT SYSTEM	FIG. 8.2
CHECK					
APR.				MODEL 902-1	D2-12072
APR				BOEING AIRPLANE COMPANY SEATTLE 24, WASHINGTON	PAGE 55
DWN. K. OSBORNE 8-7-61					

U3 4070 7000 (WAS BAC 1000-R2)


LO_2
He
RP
HELIUM
ACCUMUL
LEGEND
PRE-VALVE
FILL & DRAIN VALVE
RELIEF & VENT VALVE
RELIEF VALVE
FORCE CLOSE REGULATOR VALVE
SELF SEALING QUICK DISCONNECT
HEAT EXCHANGER
FLEXIBLE BELLOWS
ANTI-VORTEX BAFFLE

1



NOTE:
FIRST STAGE ONLY SHOWN
SECOND STAGE SCHEMATICALLY IDENTICAL.

2

SCALE: NONE

CALC			REVISED	DATE	SCHEMATIC-PROPELLANT SYSTEM	FIG. 8.3
CHECK						
APR.					MODEL 902-2	D2-12072
APR.						
DWN	K.OSBORNE 8-7-61				BOEING AIRPLANE COMPANY SEATTLE 24, WASHINGTON	PAGE 56

US 6876 7000 (WAS BAC 1000-R2)

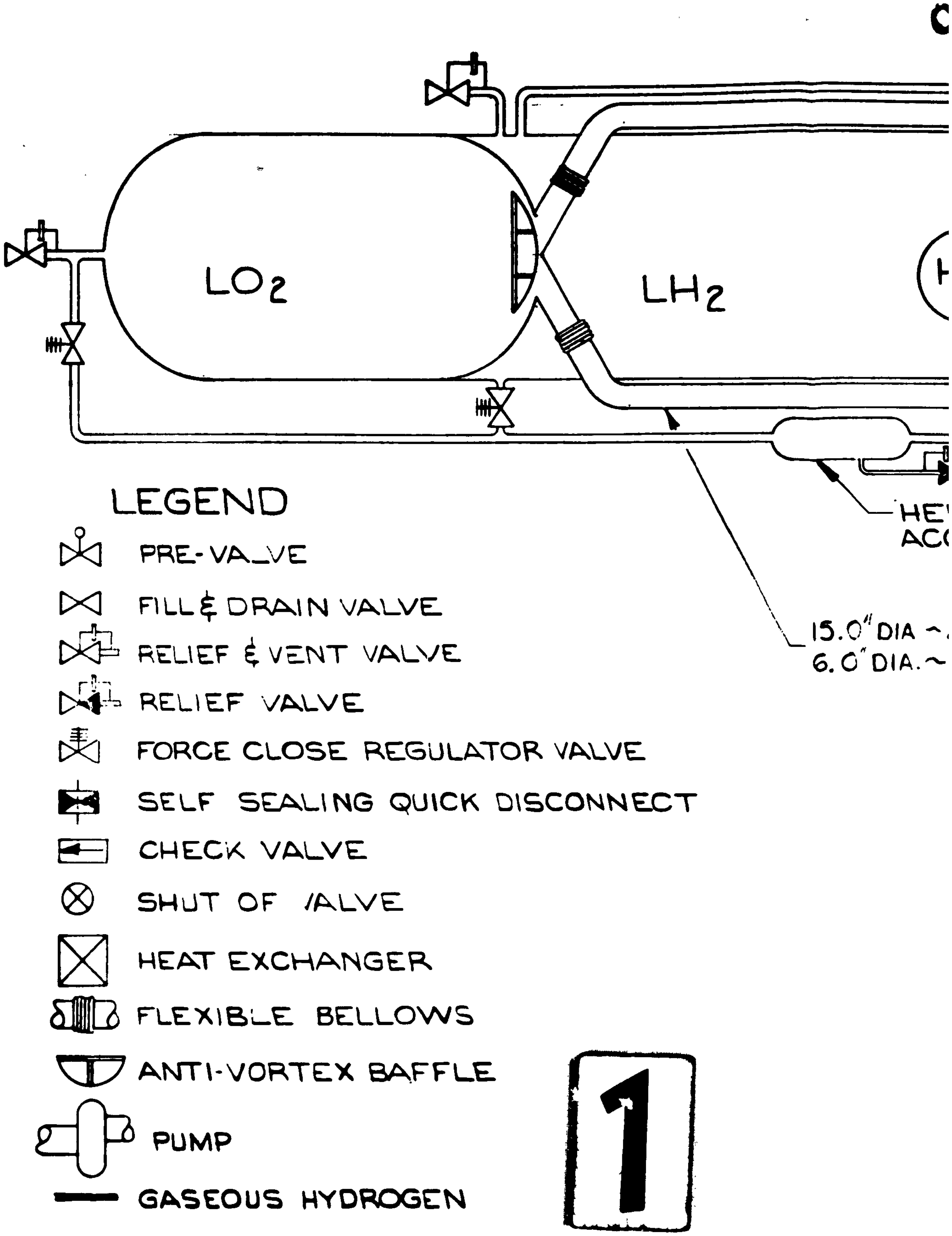
the fuel tank in a double wall evacuated tube to preclude fuel freezing and excessive heat leak to the oxidizer. A stored gas helium system provides the expulsion media of both propellants throughout flight. The helium sphere is stored in the oxidizer tank to conserve weight and space through increased gas density. The cold gas is heated in the engine heat exchanger before injection into the propellant tanks. The gas accumulator, fill and topping valves, overpressure relief, and shut-off valves perform the same functions as for model 902-1.

8.2.1.3 Models 902-3 and -4 Advanced Engine (LO_2/LH_2-FD)

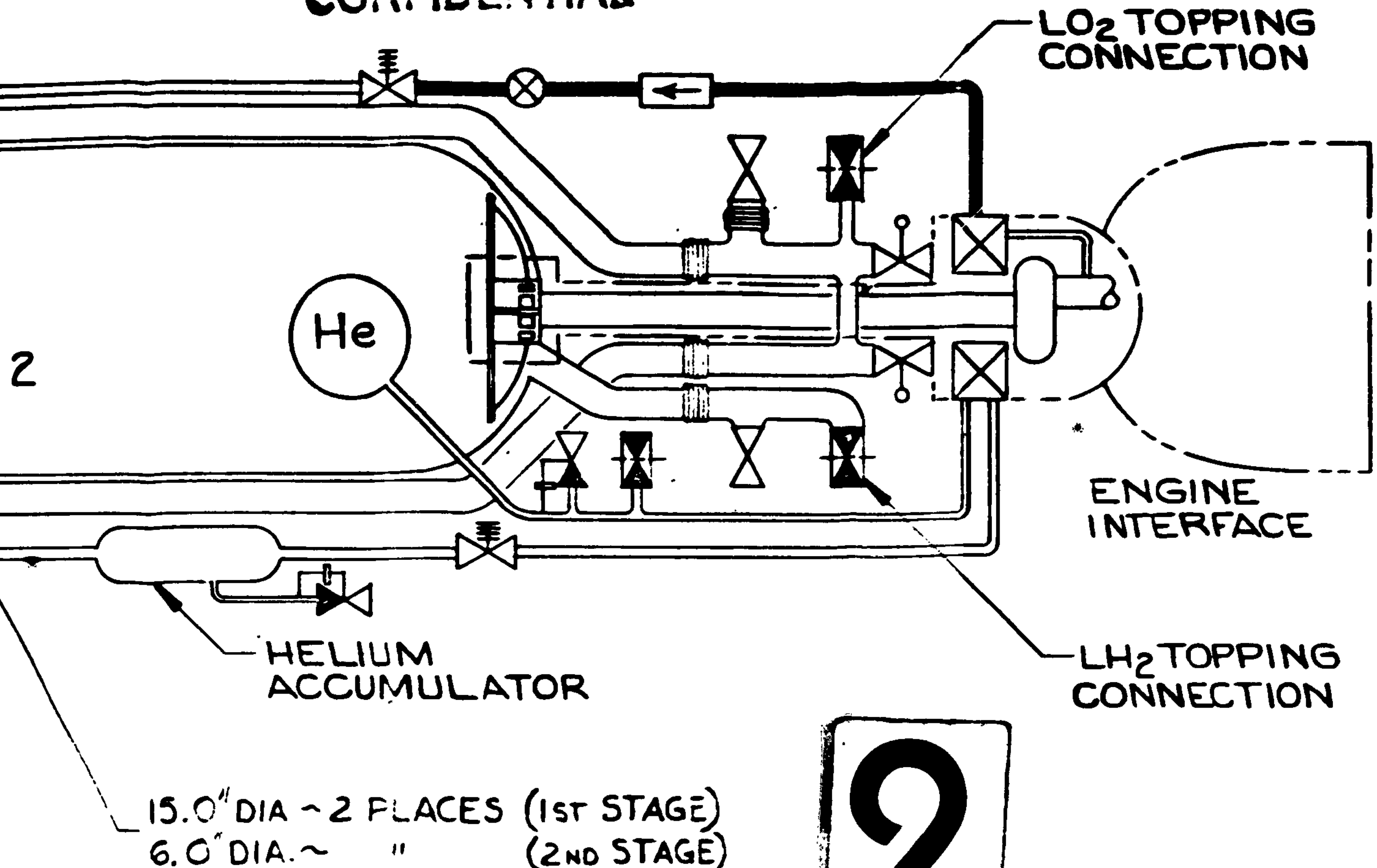
The propellant subsystem diagram is identical for these two models and is shown in Figure 8.4. The system is virtually the same as for model 902-1 except that the bell is replaced by force deflection engine. The diagram is also applicable to the upper stage of the -3 model. The most significant change introduced by the use of the force-deflection engine is the incorporation of the hydrogen turbopump inlet into the tank bottom, thus eliminating the usual fuel line between tank and engine. The two oxidizer feed lines are interconnected immediately upstream of the pre-valves to allow oxidizer circulation, through heat pump action, thereby minimizing chances for geysering.

8.2.2 Tankage Arrangement

From the standpoint of the propellant feed system, the oxidizer tank should be placed forward of the fuel tank. This is true for both the conventional LO_2/RP and the high energy cryogenic propellants. In the latter case the greater density of the LO_2 can be effectively


LO2
LH2
15.0" DIA ~
6.0" DIA. ~
LEGEND
PRE-VALVE
FILL & DRAIN VALVE
RELIEF & VENT VALVE
RELIEF VALVE
FORCE CLOSE REGULATOR VALVE
SELF SEALING QUICK DISCONNECT
CHECK VALVE
SHUT OF VALVE
HEAT EXCHANGER
FLEXIBLE BELLOWS
ANTI-VORTEX BAFFLE
PUMP
GASEOUS HYDROGEN

1



2

NOTE:
FIRST STAGE ONLY SHOWN FOR MODEL 902-3 - SECOND STAGE SCHEMATICALLY IDENTICAL.

DIAGRAM ALSO APPLICABLE FOR 902-4

SCALE: NONE

CALC	REVISED	DATE	SCHEMATIC-PROPELLANT SYSTEM MODELS 902-3 & 902-4	FIG. 8.4
CHECK				D2-12072
APR			BOEING AIRPLANE COMPANY SEATTLE 24, WASHINGTON	PAGE 58
APR				
DWN. K. OSBORNE 8-8-61				

U3 4070 7000 (WAS BAC 1690-R2)

used to achieve a large hydraulic head at the turbopump inlet enabling the LO_2 tank pressure to be reduced and the oxidizer turbo-pump to operate at a relatively large NPSH. In the case of hydrogen, the hydraulic head change is almost negligible. Inasmuch as low values of turbopump NPSH are more easily achieved in hydrogen, its aft position is not seriously penalized. With the LO_2 tank forward, the unavailable oxidizer is contained in the feed lines rather than spread out over the large tank bottom thereby reducing residual propellant weight at burnout.

Though less significant, the LO_2 tank also optimizes in the forward position in a LO_2/RP system. This is due primarily to the much lower vapor pressure of RP-1 and secondarily, to the greater density of LO_2.

8.2.3 Pressurization Systems

A number of potential approaches to the pressurization system for a large vehicle exist. These systems differ from one another on the basis of the pressurizing gas used, the gas source, gas temperature involved, and the venting system characteristics. Stored systems, using either hot or cold nitrogen, helium or combustion products are the accepted state-of-the-art and can be readily adapted to these large vehicles. The inherent advantages in reduced total system weight of the hot gas systems has, however, been long recognized and the current trend is in this direction. This approach offers minimum residual gas weight.

Under adverse environmental conditions, all hot-gas systems are subjected to rapid pressure transients due to gas-liquid heat exchange; and these systems employing propellant vapors may suffer complete pressure collapse since the gas is condensible.

Composite systems, however, where a small amount of cold helium is used for initial pressurization and as a blanket or thermal barrier over the propellant to minimize heat transfer to the pressurizing gas is one approach to an efficient and reliable system.

The pressure systems selected for this study are either composite or simple helium systems which result in system simplicity, minimum residual gas weights with reasonable system reliability. Cost, relatively severe gas containment problems, and possible shortage of helium were not considered in the choice of the pressurizing media.

8.2.4 Development Items

This study has placed primary emphasis of the achievement of good reliability through a simple resizing of current systems. There is undoubtedly considerable development required from the sheer size requirements of the components, piping, and tankage. However, it is believed that size is the main problem and therefore amenable to solution through application of current technologies. The use of the force-deflection engine does not appear to make these problems any more severe.

8.2.5 Tank Baffling

The tank baffles fall into four main types; slosh decoupling, anti-vortex, supporting, and in the case of cryogenics, anti-fountain.

Current and past programs at Boeing indicate that comparatively simple light weight baffle designs are quite affective in the suppression of vortices and tank outlet unporting near burnout. Very simple designs also exist to combat fountain affect during cryogenic propellant loading.

A detailed analyses of the ratio of vehicle rigid pitch frequency and body bending frequency to deep wave slosh frequency is required to establish definite requirements for slosh decoupling baffles. Such a detailed analyses is beyond the scope of this contract and was not conducted. However, past studies at Boeing on similar vehicles indicate that slosh baffles will probably be required in the oxidizer tank and possibly even in the fuel tank for these study vehicles.

8.2.6 Control Valves

Control valves selected for the baseline vehicle propellant systems are of the type presently in use. Propellant fill valves and pre-valves are electrically controlled, hydraulically or pneumatically actuated. This type valve has proven itself in present LO_2 systems. Current design type mechanical quick disconnect couplings are well suited for use in helium fill lines and topping connections required by these vehicles.

Vent valves associated with cryogens should be of the pilot-impulse type to prevent valve freezing. These are presently used with success in LO_2 systems.

8.2.7 Joint Connections

The need to eliminate propellant leakage at permanent and breakable line joints becomes more pronounced for large vehicles employing advanced, high energy propellants. Special joints are required with the cryogens for minimizing heat leak while maintaining line integrity.

Bayonet type joints have proven to be effective against heat leak and cryogen leakage and are proposed for use in the propellant systems where jacketed lines are required.

8.2.8 Line Problems

There are many areas associated with propellant lines which could have serious repercussions from lack of proper design considerations. These include such items as gas traps, contaminant traps, excessive line losses, thermal stresses, and geysering.

The propellant lines of a cryogenic vehicle are likely to geyser if not adequately insulated. Geysering, in this case, refers to a sudden blowing out of the liquid in a line and refilling of the line in a cyclic manner. Heat added to the propellant in a line causes decrease in the local static pressure. This unstable condition produces increased generation and expansion of gas which rapidly expels most of the liquid contained in the line. This causes uneven thrust buildup at engine start.

The heat-leak-to-line and the line-length-to-diameter ratio are the two major parameters controlling the onset of geysering. An increase in either will eventually result in geysering. Line insulation and/or liquid re-circulation are the principal means of controlling this phenomena. Integral line bellows offer the most attractive solution to line distortion associated with cryogens and induced vehicle bending loads.

It does not appear that the above items will present insurmountable problems. They will have to be investigated in detail, for specific configurations, to greater depth than permitted during this study.

8.2.9 Insulation

Use of cryogenic propellants introduces the phenomena of cryopumping, boil-off, and icing which must be controlled. In addition, insulation systems must control structural and propellant temperatures. Insulation to limit boil-off will be required only for hydrogen. Insulation systems, as well as structural materials, must be compatible with propellants.

Vacuum blankets wrapped around external surface of tanks with special formed vacuum pads for tank heads, common tank head included, offers one solution to insulation problem; however, weight and handling problems may overcome the advantages. Another approach is bonded polyurethane foam on internal surface of tanks with bonded layer of mylar separating foam from the cryogen. Lines may be covered with vacuum blankets or bonded polyurethane foam.

8.2.10 Boost Pumps

A potential trade exists between the use of tank mounted boost pump for forcing the propellants into the main turbopumps and the use of tank pressure alone for performing this function. Customer furnished data indicates that reasonably low values of turbopump NPSH can be achieved at a small increase in turbopump weight. In this study NPSH values of 2 PSI for hydrogen, 7.4 psi for lox, and 12 psi for hydrocarbons were used. These values resulted in reasonable tank pressures obviating the need for additional turbo machinery.

8.2.11 Emergency Provisions

For the purposes of this study no special emergency provisions are incorporated in the basic propellant subsystem except for emergency defuel in the event unsafe conditions exist in the area of the loaded vehicle. Emergency defuel is accomplished by the onboard helium system supplemented by additional inert gas from the ground based system. Pressurizing gas is forced into the propellant tank through the flight regulators and liquid is withdrawn through the filling connections. After liquid depletion, inert gas continues to purge the tanks.

8.2.12 Summary

In general, there are no major differences in the propellant subsystem resulting from the use of the force-deflection engine in lieu of the conventional bell engine. Some secondary effects do exist as follows:

(a) The bell engine studied employs engine gimballing for vector control while the F-D engine employs gas injection. This is conducive to an inherently more reliable propellant feed system through elimination of the gimbal bellows.

(b) Of the two basic engine studied, the propellant inlet arrangement on the bell engine is more amenable to a direct feed line routing of the oxidizer lines. This is presumed to be a function of inlet arrangements of the particular engine geometries under study rather than an inherent advantage associated with a particular engine type.

(c) The propellant feed lines to the E-D engine tend to be somewhat smaller than those to the conventional bell engines due to the slightly higher I_{sp} values inherent in an altitude compensating engine for first stage application. This difference disappears on upper stage applications.

9.0 CONTROL SYSTEM

The control problems of the Models 902-1 through -4 configurations are similar in nature and are considered collectively herein. Single and tandem stage configurations carrying non-lifting payloads and flying a zero "g" trajectory to an orbital altitude of 300 nautical miles are involved. Without special provisions, the booster-payload combinations are unstable aero-dynamically and must be both attitude stabilized and guided along the prescribed trajectory by the guidance and control system. In these respects the control system requirements are identical to current operational vehicles.

Consideration of man rating the booster leads to a requirement for provision of aerodynamic stability in the event of engine shut down. This requirement is in addition to those of present operational vehicles. It may be met by the addition of fixed fin area, or by use of a flared skirt, located at the base of the first stage configuration. Both methods have been examined. The skirt method has advantages in providing a mount to support the booster on the pad, in alleviating launch clearance requirements, and in reducing air loads impinging upon the vectored nozzles. It also is simpler to make an attachment to the booster engine. Either stabilization method would be acceptable in fulfillment of the control function.

Since increase of the booster aerodynamic stability is accompanied by a fin weight penalty, a minimal requirement of neutral stability was selected. The effect of neutral aerodynamic stability is to decrease thrust vector control requirements in providing control system "stiffness" when compared to current unstable vehicles. Thrust vector control system damping requirements are almost the same in either case. Because the center of gravity moves forward further than the center of pressure shifts as propellant is consumed, the booster stability increases with time from launch. This is helpful to the stage separation process and further alleviates thrust vector requirements for provision of control stiffness. It does increase thrust vector deflections for accomplishing trajectory maneuvers. Such maneuvers may be expected to be small in this regime and as a consequence, no particular problem is foreseen.

Since the inclusion of neutral aerodynamic stability tends to reduce thrust vector control requirements below that required for less stable boosters, previous studies and experience may be used to provide conservative guidelines in the controls area. Specific solutions to vehicle stability must of course be made by a closed form analysis of the hardware control components, engine and vehicle airframe characteristics. Such analyses are beyond the scope of this study. Detailed slosh and structural coupling stability analyses are, therefore, not included. When such studies are made, their solution may be expected to be eased due to the stable airframe.

Trends of control problems arising as a function of booster size, fuel type, engine type and booster performance for boosters less stable aerodynamically than those considered here are presented in reference 15.2. Preliminary review of this program indicates the control trends presented therein are applicable to the configurations being studied here with equal validity.

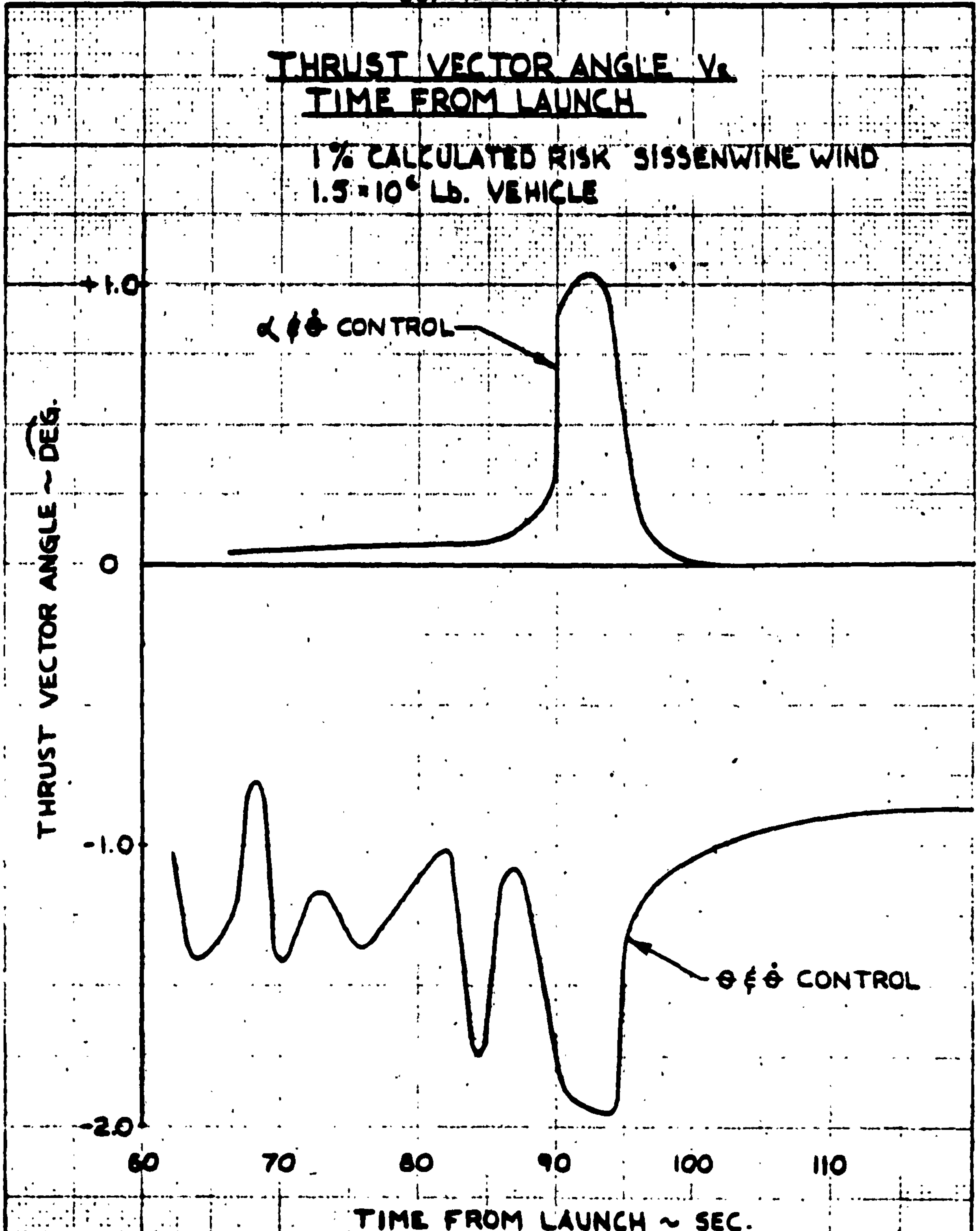
10.0 VEHICLE AUXILIARY SYSTEMS

10.1 GENERAL

Several of the lower weight and cost subsystems are considered briefly below. For the most part these systems will not vary greatly between the configurations considered in this study. This is particularly true for guidance telemetry, destruct and identification provisions.

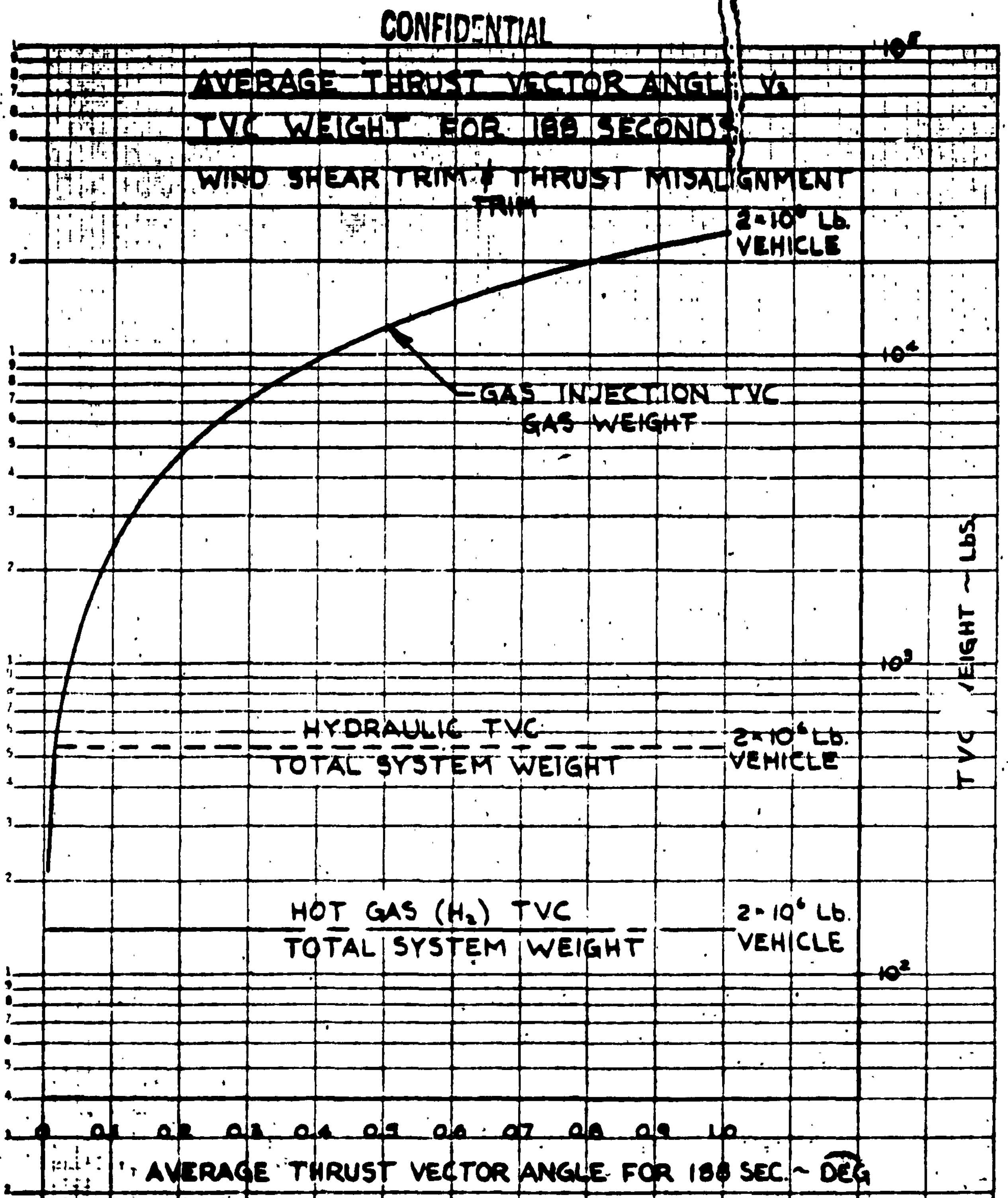
10.2 THRUST VECTOR CONTROL

One possible exception to the above is with regard to provisions for thrust vector control. A continuous thrust misaligment tolerance for the engine is stipulated. Use of gas injection for control may impose a severe weight penalty caused by gas flow to trim out the 1/2° thrust misalignment and to meet the average thrust angle required to overcome wind shear disturbances. Wind shear requirements were estimated by extrapolating the results of a continous digital flight simulation of a 1.5 million pound booster with several control laws being examined. Figure 10.1 shows the thrust vector requirements for two control laws representing the greatest and least average thrust vector angle for the 1.5 million pound thrust vehicles. Fuel weight is such a small portion of the weight of a conventional thrust vectoring system, and such a predominant portion of a fluid injection system where significant trim is required that a comparison is made on that basis. Figure 10.2 shows the effect of thrust vector trim on the weights of the two types of systems for a 2 million pound configuration. The characteristics of the fuel injection system (I_{sp} = 260 sec and magnification factor = 2) were supplied by



CALC		REVISED	DATE	THRUST VECTOR ANGLE REQUIREMENTS	FIG 10.1
CHECK					D2-12072
APPR.				BOEING AIRPLANE COMPANY SEATTLE 24, WASHINGTON	PAGE 69
APPR.					
DRN	B.T. BROWN 8-15-61				

BAL 1003 R 4 2 7000



CALC			REVISED	DATE	AVERAGE THRUST VE[illegible] ANGLE Vs TVC WEIGHT	FIG 10.2
CHECK						D2-[illegible]
APPR					THE BOEING COMPANY SEATTLE 24, WASHINGTON	PAGE 70
APPR						
DRN	B.T. Brown	8-15-61				

Aerojet. Boeing studies tend to confirm these figures. The above would indicate further study is required of methods for obtaining vector control when fixed engines such as the F-D type are involved.

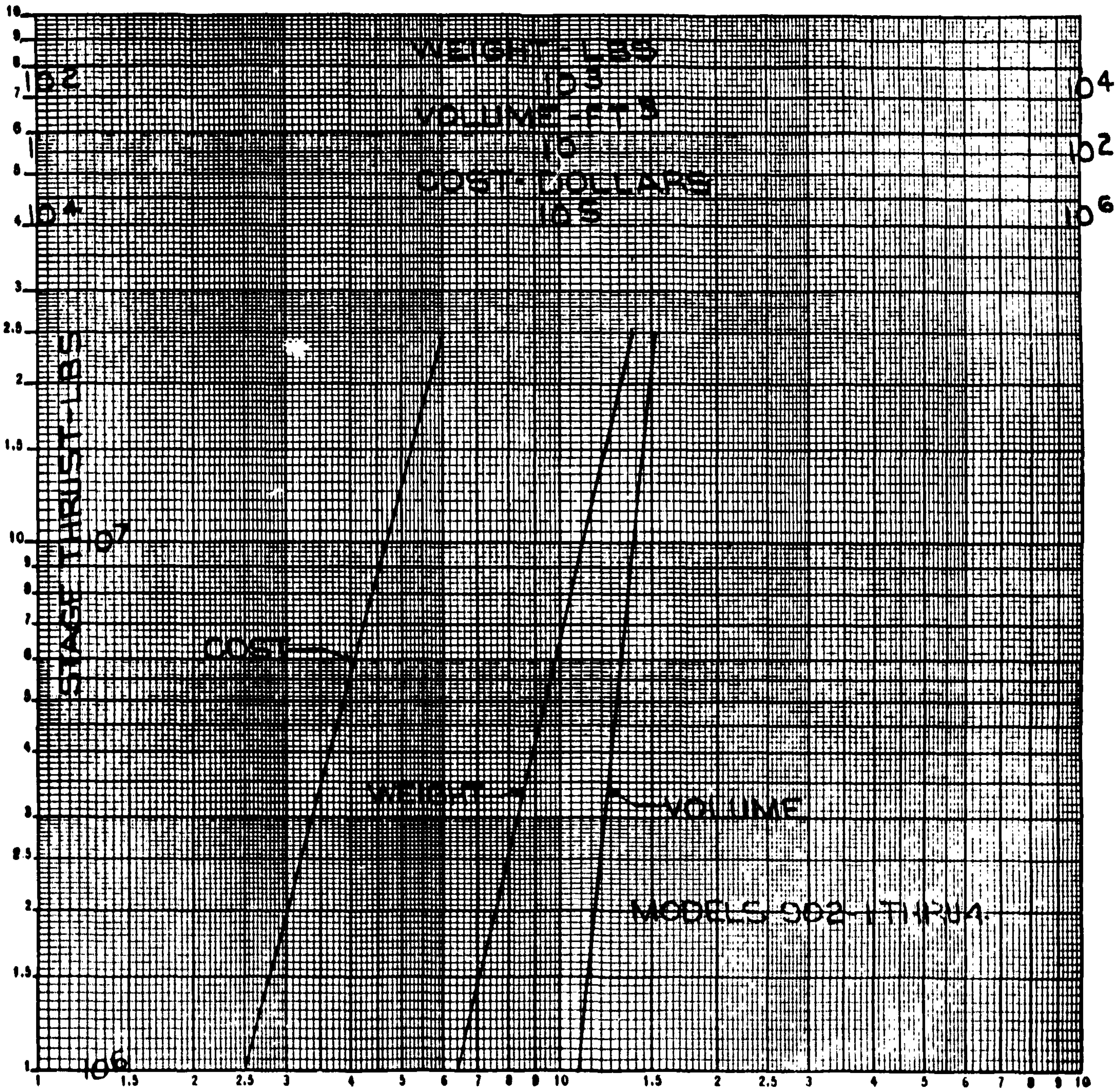
10.3 ELECTRICAL POWER

For purposes of this study a conventional 28 volt DC supply was considered. Batteries are a logical energy source, chosen largely on the basis of extensive operational experience and the related confidence in achieving high reliability. Power level and duty cycle are not expected to vary appreciably with booster thrust in the range of interest, so that source weights may be considered constant. The distribution system, or network, is affected by booster size, but not appreciably by choice of fuel or engine design. The net effects of variations in thrust level on electrical system weight, cost and volume are shown in Figure 10.3. Availability and reliability of components are not expected to be problems, nor are they expected to vary significantly with changes in the key parameters of this study.

11.0 GROUND SUPPORT

In general, ground support provisions will not vary significantly with engine choice per se for similar propellants within the limits of this study. Since all vehicles perform with the same general function, are fabricated to similar manufacturing launch site location and operated in like manner to those systems considered in reference 15.2, the costing criteria for ground support used in reference 15.2 were followed in this study.

ELECTRICAL SYSTEM CHARACTERISTICS AS A FUNCTION OF BOOSTER THRUST



CALC.	J.S.M.	8-14-61	REVISED	DATE	ELECTRICAL SYSTEM CHARACTERISTICS	D2-12072
CHECK						FIG. 10.3
APPD.						
APPD.					BOEING AIRPLANE COMPANY	PAGE 72
DWN.	K.OSBORNE	8-15-61				

12.0 ECONOMIC ANALYSIS

12.1 INTRODUCTION

This section presents the numerical results of the cost analysis, a discussion of the cost techniques, and the assumptions and ground rules followed during economic analysis of the four vehicle configurations considered in this study. In addition curves are presented showing the estimated variation of cost for major vehicle components over a first stage vehcile thrust range of $.6 \times 10^6$ to 6.0×10^6 pounds.

12.2 SYSTEM COSTS

Figure 12.1 shows estimated costs applicable to the number one vehicle for the Model 902-1 thru 902-4 vehicles. Figure 12.2 shows estimated total system costs including Research and Development, production and operating costs for each vehicle for production totals of 25; 100; and 400 vehicles. It is seen that the airborne vehicles account for the major portion of the recurring costs throughout the vehicle R&D and production quantity spectrum.

Figure 12.3 indicates the relative cost performance for the four basic vehicles considered in this study. These curves reflect the estimated performance of each vehicle as discussed in section 5.0 and the predicted cumulative system reliability discussed separately in section 13.0.

Reference to figure 12.2 indicates the Models 902-3 and 902-4 advanced vehicles to show 4% and 37% respectively less cost than the model 902-1 LO_2/LH_2 baseline vehicle. The 113,800 pound payload capability of the

FIGURE 12.1

VEHICLE COST STATEMENTS

(Dollars in Millions)

NUMBER 1 VEHICLE

Boeing Model	Baseline 902-1	Baseline 902-2	Advanced 902-3	Advanced 902-4
Propellant	LO_2/LH_2	LO_2/RP-1		
Interstage	$ .8	$ .7	$ .7	$ .1
Tankage, Plumbing, and Structure	8.0	5.7	7.8	5.4
Subsystem Equipment	3.1	3.5	2.8	2.8
Subtotal (Less engines)	$11.9	$ 9.9	$11.3	$ 8.3
Engines	1.8	1.7	1.7	1.6
Total Number 1 Vehicle Costs	$13.7	$11.6	$13.0	$ 9.9

Figure 12.1

U3 4070 7000 (WAS BAC 1846 F-R3)

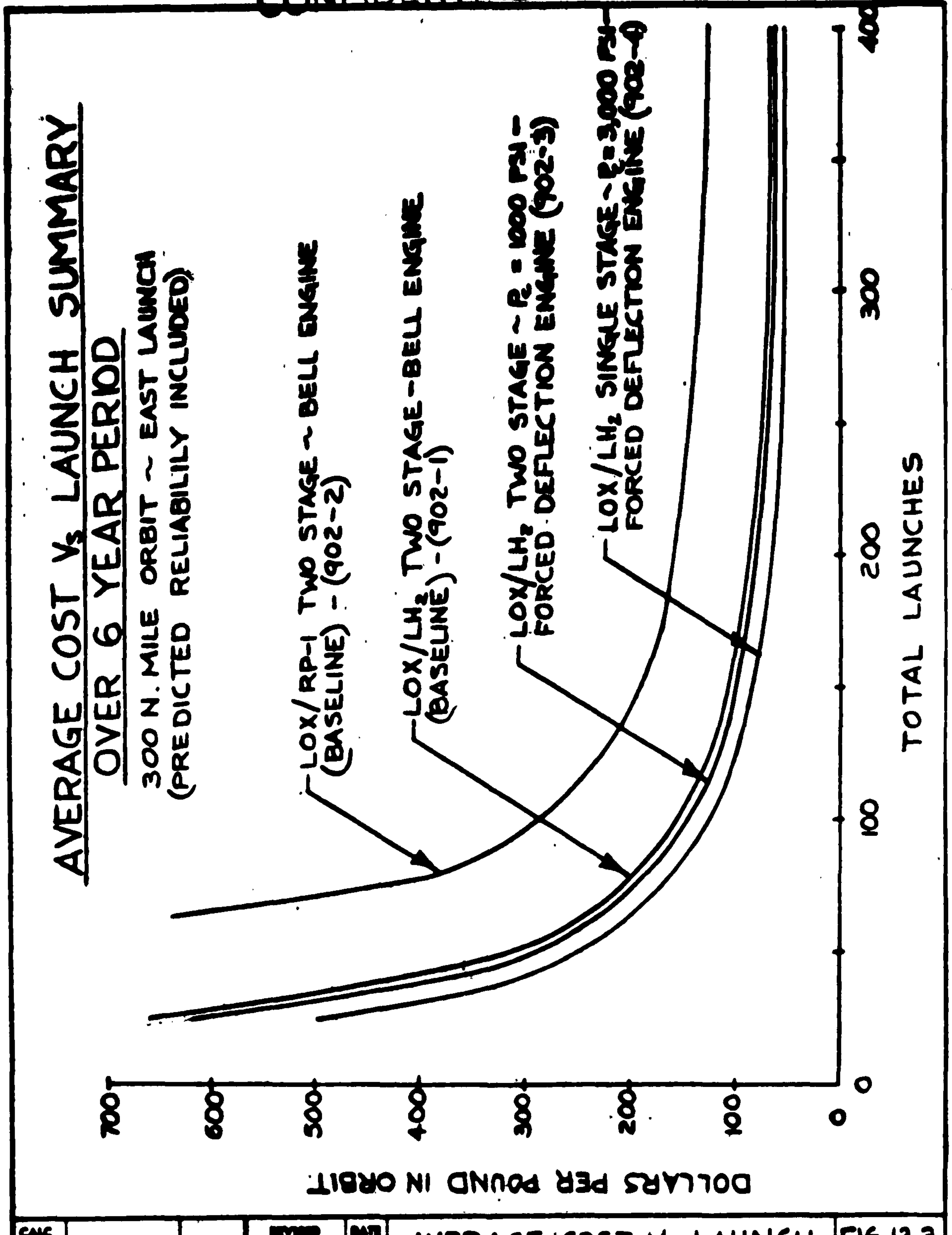
FIGURE 12.2

SYSTEM COSTS
(Dollars in Millions)

Vehicles	902-1			902-2			902-3			902-4		
Launches	25	100	400	25	100	400	25	100	400	25	100	400
Research & Development Program	$582	$ 583	$ 584	$486	$ 486	$ 487	$563	$ 564	$ 564	$496	$496	$ 496
Production Program												
Airborne Vehicles	229	661	1,995	197	559	1,697	219	632	1,864	163	466	1,369
Ground Systems	44	93	253	45	96	261	44	93	253	48	108	240
Operating Costs	57	127	318	49	102	239	56	122	308	45	98	241
Total System Costs	$912	$1,464	$3,110	$777	$1,243	$2,684	$882	$1,411	$2,989	$[illegible]	$[illegible]	$2,[illegible]
Dollars per Pound Payload In Orbit	$661	$ 155	$ 67	$1,224	$ 287	$ 126	$618	$ 145	$ 62	$990	$129	$ 54

US 4070 7000 (WAS BAC 1000 F-R3)

CONFIDENTIAL



CALC			REVISED	DATE	AVERAGE COST Vs LAUNCH SUMMARY	FIG. 12.3
CHECK						D2-12072
APPD						
APPD					BOEING AIRPLANE COMPANY	PAGE 76
DRN	G.T. BROWN	8-15-6				

CONFIDENTIAL

single stage Model 902-4, however, is 22,200 pounds or 18.7% less than the Model 902-3. The predicted reliability of the single stage vehicle is in its favor. All factors combined results in a small advantage to the single stage Model 902-4.

12.3 COST VARIATIONS

The estimated variation of costs for three major vehicle categories as a function of booster thrust level (.6 x 10^6 to 6 x 10^6 pounds) is shown by Figure 12.4. The weight variation over the same thrust range is evaluated in Section 6.0.

12.3.1 Single Stage to Orbit Cost Results

Figure 12.5 shows the results of a cost analysis made to determine the optimum value of T/Wo for the single stage to orbit vehicle (Model 902-4. Minimum costs are obtained at T/Wo of 1.3 to 1.4 depending on the total quantity of launches. The actual optimum value may be influenced by the desirability of making this vehicle capable of also operating with upper stages to achieve versatility. The study schedule did not allow this possibility to be analyzed in detail.

12.4 COSTING GROUND RULES AND TECHNIQUE

This section presents the cost estimating and cost analysis methodology utilized during the study.

12.4.1 Cost Estimating

System cost data presented in this document were founded on parametric values taken from The Boeing Company related contract experience and detailed estimates. The absence of detail design data precluded the

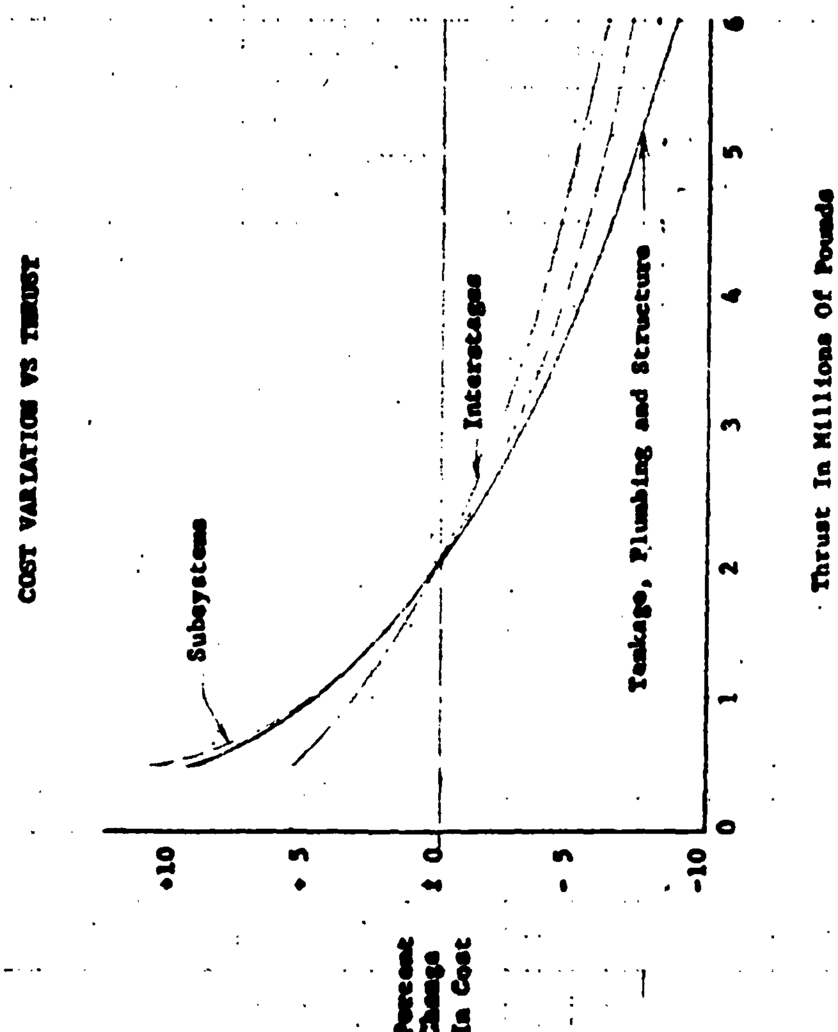
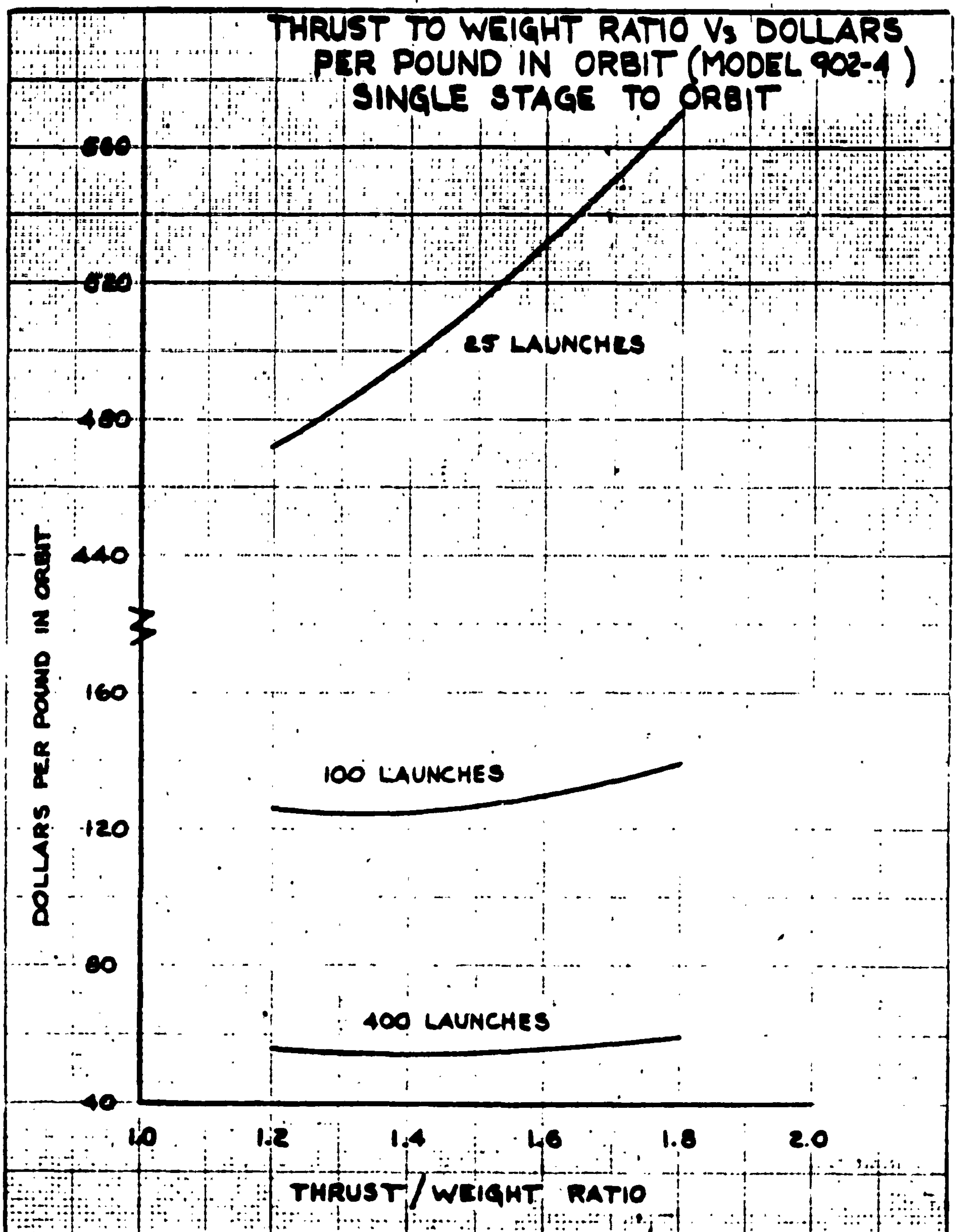


FIGURE 12.4

CALC			REVISED	DATE		D2-12072
CHECK						
APPR						
APPR					THE BOEING COMPANY SEATTLE 24, WASHINGTON	PAGE 78



CALC		REVISED	DATE	THRUST TO WEIGHT RATIO Vs DOLLARS PER POUND IN ORBIT MODEL 902-4	FIG. 12.5
CHECK					D2-12072
APPD				BOEING AIRPLANE COMPANY SEATTLE 24, WASHINGTON	PAGE 79
APPD					
DRN	B.T. BROWN 6-21-61				

use of alternate estimating techniques. The consistency of the cost base was stressed throughout the study to prevent an unwarranted economic advantage being awarded to any of the design concepts evaluated.

12.4.2 Cost Techniques

The research and development costs were estimated by relating the task required to a similar known task containing actual costs, considering such factors as complexity, reasonable level of manpower and the state-of-the-art.

Manning was estimated as the cost of maintaining work crews required at the launch base, and it was assumed that government personnel would be used. This cost was based on user taking delivery of major assemblies and system components upon arrival at the launch site. Manning costs also included the labor required to maintain the base facilities and ground equipment.

Airborne vehicle follow-on production costs were estimated using number one unit cost per pound parameters for the items listed on the weight statements. The Boeing Company experience curve formulas were utilized to compute costs. This formula is defined as follows:

Unit values = ax^{-n}, where a = #1 unit value,

x = unit number,

n = slope constant = $\frac{2 - \log(\%\ \text{slope})}{.30103}$

Cumulative values equal the summation of the unit values.

The costing of base facilities and ground equipment was performed by interpolating from known costs.

The operating costs were computed as a function of propellant weight, launch schedule, manpower and spares provisioning requirements. The costs were estimated by an examination of each of these subcategories and an analysis of the associated costs such as: cost per pound of propellant, average annual salaries, annual spares requirements, and maintenance and repair as a percentage factor to the total facilities value.

12.5 RESEARCH AND DEVELOPMENT PROGRAM

Total R&D costs were composed of engineering, development, and test of the airborne vehicle and ground systems, and also included R&D tooling and flight test program. This program assumes no major state-of-the-art advances.

Booster development associated with the airborne system includes the costs of: structural components, such as interstage and tankage, and subsystems equipment such as secondary power, controls, and pressurization equipment. Engine development costs were taken from information furnished in chart form by Aerojet-General Corporation.

Ground systems development costs were composed of the estimated design and evaluation effort for barges, transporters, slings, launch complexes, checkout and launch equipment, assembly and test equipment, propellant storage and loading facilities, and utilities.

The estimated construction and production costs for major segments of the ground system, such as test base facilities and transportation and handling equipment, were based on the assumption that the test base would be located within an existing Air Force Base complex. However, all launch facilities and equipment were assumed to be significantly different in capacity and design than existing test sites, thereby requiring procurement of ground systems unique to the system evaluated.

Estimated costs for providing a basic set of contract tools to be utilized in the fabrication and assembly of test vehicles and limited quantities of follow-on production vehicles were included in R&D costs. These costs associated with further duplication of tools to sustain a high rate of production were included in the follow-on production costs as were all recurring tool maintenance and repair costs.

A flight test program utilizing the equivalent of fourteen airborne vehicles was costed. The cost for the static test, dynamic test and a battleship unit was included with the flight test units. Flight test operations consisting of test site manning, data acquisition, range usage, propellants, utilities, test site maintenance, test vehicle spares, etc., were included.

12.6 FOLLOW-ON PRODUCTION COST

An analysis of recurring production effort was made to derive the costs of airborne vehicles, tooling, operating base facilities and equipment, and training of base operating personnel. Engine costs were segregated in accordance with the terms of the contract.

12.6.1 Airborne Vehicle

Production costs for the airborne vehicle number one were based on parameters developed by the Boeing Company yielding cost per pound for items listed on the weight statement.

Production engine costs were taken from information furnished by Aerojet General Corporation. In order to use these charts for all stage engines, vacuum thrust was converted to sea level thrust per direction of an Aerojet-General representative.

Airborne vehicle production costs included production tooling for engines derived from information furnished in graph form by Aerojet-General Corporation. The balance of the vehicle tooling costs included estimated costs for labor and materials to fabricate duplicate tools and to sustain production tools.

12.6.2 Ground System

The ground system included costs supplemental to the test base facilities and ground equipment. It was assumed that a portion of the test complex would be retained for the follow-on production program. Costed as part of the ground system was transportation equipment, transportation costs and handling equipment. Training of operating personnel and other initial manning costs were also included.

12.7 OPERATING COSTS

Operating costs were estimated to sustain a launch program over a six year period. A major cost item was the airborne vehicle spare components required during the pre-launch checkout phase. The estimated cost of these spares for all except engines was based on a Boeing estimate of replenishment requirements. Engine spares requirements were based on information supplied by Aerojet-General Corporation.

All propellant required to load the liquid propellant boosters over the six year operational phase was costed to include an allowance for boil-off and other losses.

The cost of maintenance, operation, and replacement of facilities and ground equipment required for airborne vehicle assembly, stage mating, propellant loading, pre-launch check-out and launching was included in operating expenses.

13.0 RELIABILITY

13.1 INTENT

Since reliability characteristics are a function of specific hardware and many of the subsystems comprising these vehicles are still in the concept stage, the reliability numbers shown here should be considered as comparative from vehicle to vehicle rather than as indicating absolute levels of reliability.

13.2 SCOPE

The analysis presented below is concerned primarily with the comparison of the four vehicles studied in the regime between lift-off and final stage burn-out. If the vehicle stands in the ready condition for substantial lengths of time, those components in active service, such as gas pressure regulator, which cannot be checked out immediately prior to lift-off must be considered to be operating for the ready time. If the item can be checked out and proved to be operating immediately prior to lift-off it is assumed that its likelihood of failure is no different for subsequent time intervals than it was for the previous intervals of the same length.

13.3 SIGNIFICANT FACTORS

The variation in operating time from vehicle to vehicle appears to have the greatest effect on reliability. Next comes the difference between liquid hydrogen and RP-1 fuels, the latter being less active, easier to handle, thus less likely to cause a hardware failure. The differences in engines affect this evaluation on the order of 4%.

Figure 13.1 shows a reliability comparison of the vehicles which indicates a definite advantage for the Model 902-4, Single Stage to Orbit. This is the result of the shorter operating time and greater simplicity applicable to the single stage vehicle.

13.4 RELIABILITY GROWTH

Figure 13.2 shows the predicted increase in reliability with successive launches. The two top curves represent the "instantaneous reliability" or probability of anyone vehicle performing satisfactorily. The two lower curves represent the "cumulative reliability" or a measure of success of any total number of launchings. The cumulative reliability forms the basis for development of predicted system cost performance. The inherent higher reliability of the single stage vehicle noted previously is evident over the total launch spectrum.

13.5 ASSUMPTIONS

1. Failures of subsystems and components are exponentially distributed.
2. Stages of the various vehicles are similar enough to warrant using one failure rate (adjusted for propellants used) with appropriate time of operation for all stages.
3. The conventional bell nozzle engine with gimbal thrust vectoring and the forced deflection engine with throttled gas thrust vectoring are of equal complexity within the present limits of evaluation.

RELIABILITY COMPARISON (MISSION) -MODEL 902-1 thru -4

Vehicle Model	Propellants & Engine	First Stage: Operating Time, sec.	First Stage: Reliability, %	Second Stage: Operating Time, sec.	Second Stage: Reliability, %	Instantaneous Over-all Reliability, (2) %
902-1	LO_2/LH_2 Bell	219	99.22	264	99.16	96.42
902-2	$LO_2/RP\text{-}1$ Bell	195	99.42	223	99.29	96.73
902-3	LO_2/LH_2 Forced Deflection	246	99.12	232	99.17	96.33
902-4	LO_2/LH_2 Forced Deflection					
(1) t/w=1.1		310	98.91			98.91
t/w=1.4		243	99.13			99.13
t/w=1.8		185	99.34			99.34
t/w=2.0		130	99.54			99.54

Table

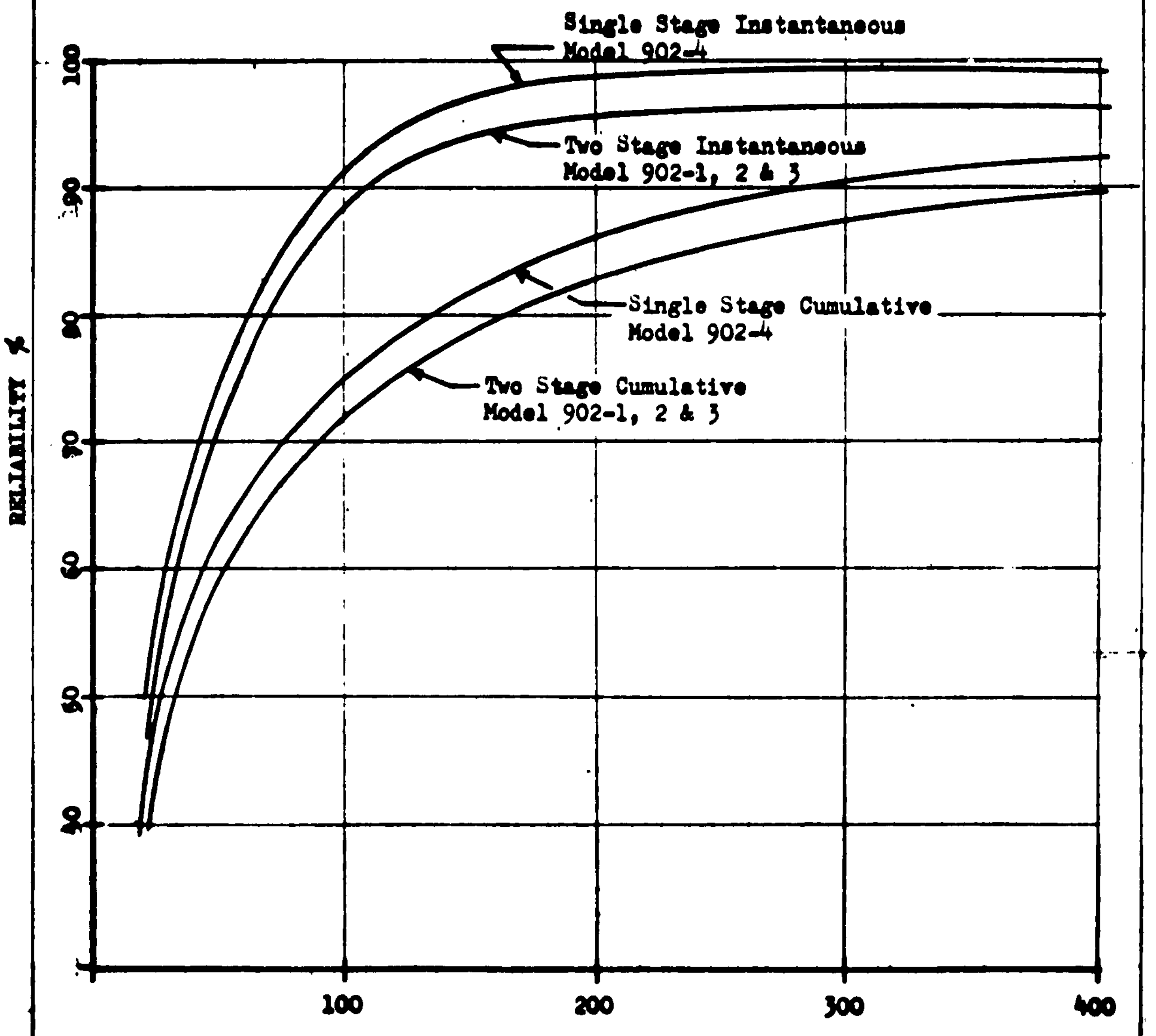
(1) t/w = thrust to weight ratio. This range of values is given to show the effect of t/w on reliability through operating (burning) time variation. It is assumed that changing the t/w does not change the basic failure rate.

(2) $R_{OA} = R_1 \times R_2 \times R_{sep}$ where R_1 = First Stage Rel., R_2 = Second Stage Rel., R_{sep} = Reliability of first stage separation and second stage initiation and equals 98%.

FIGURE 13.1

FIGURE 13.2

RELIABILITY GROWTH



14.0 UNCONVENTIONAL ARRANGEMENTS

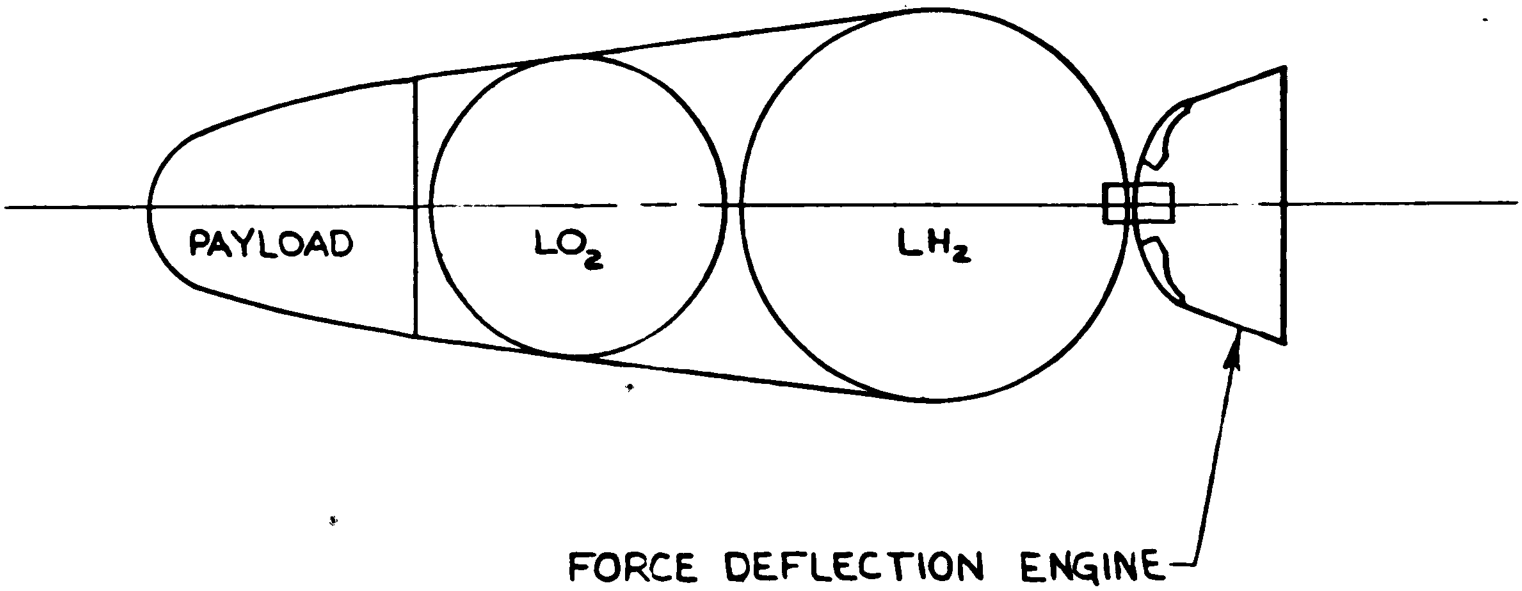
14.1 INTRODUCTION

The purpose of this section is to qualitatively present unconventional booster concepts to which the application of the unconventional engines may be particularly advantageous. This is offered primarily as an aid in evaluation and choice of possible configurations for further study.

14.2 SUMMARY

Provided an engine of adequate performance, the principal opportunity for optimization of a booster system lies in the arrangement of the propellant provisions with respect to other design requirements. Included in propellant provisions are tankage, pressurization, and induction systems. Of these, tankage is by far the most significant item. For conventional applications, the familiar tandem cylinder, relatively slender arrangements of Models 902-1, -2, -3 and -4 fulfills most compromise requirements. However, from a container standpoint minimum surface is achieved by spherical tankage. One such arrangement is represented by Models 902-5A, as shown in fig 14.1. From the standpoint of stability during boost, the tankage is best towed as in the original Goddard models and illustrated by Model 902-5B in fig 14.2. On a tandem tankage vehicle, interstage structure and one tank end might be eliminated by immersing the second stage engine in the first stage tank as shown on Model 902-5C, fig 14.3.

CONF



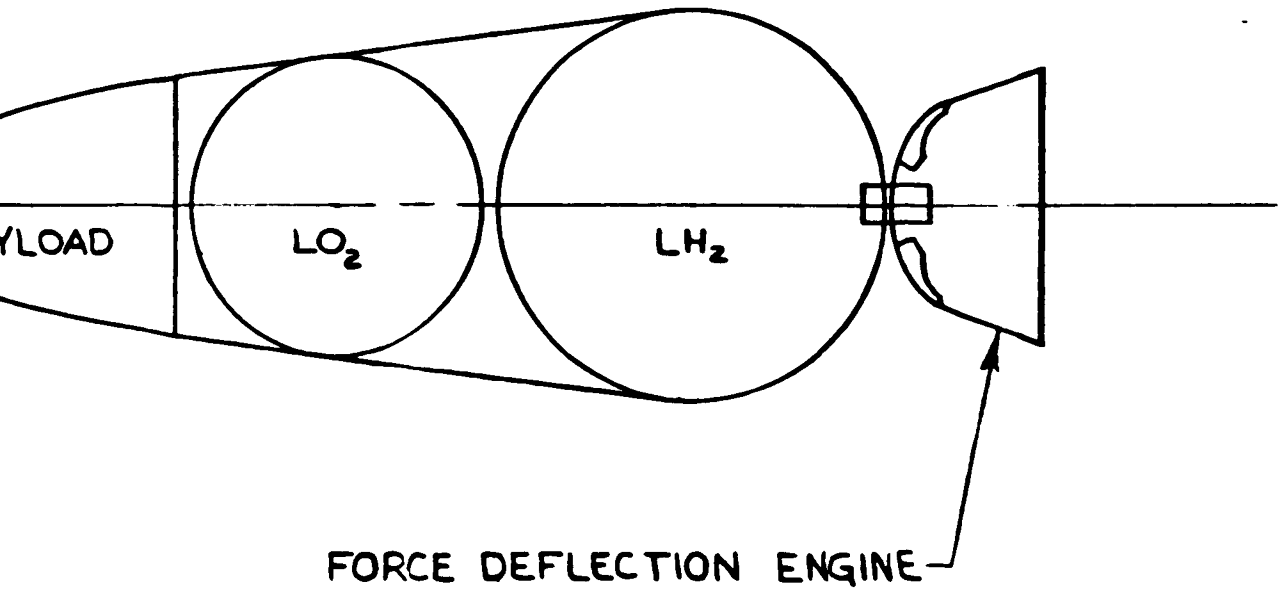
1

CALC			REVISED	DATE	MODEL 902-5a
CHECK					
APR.					
APR					BOEING AIRPLANE COMPAN
DRN	B.T. BROWN	8-14-61			SEATTLE 24, WASHINGTON

U3 4070 7000 (WAS BAC 1680-R2)

CONF

CONFIDENTIAL

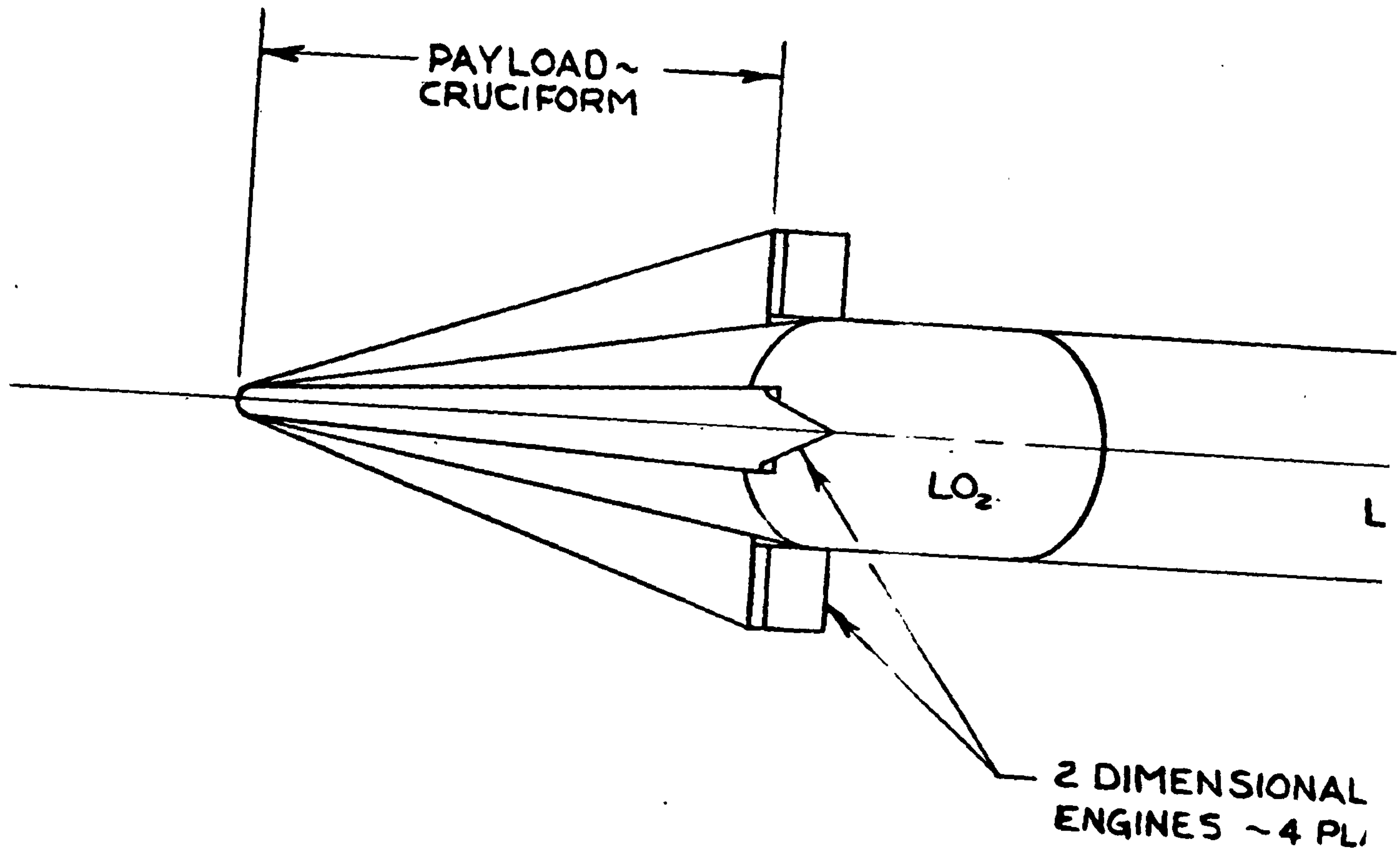


2

			REVISED	DATE		
CALC					MODEL 902-5a	FIG. 14.1
CHECK						D2-12072
APR.					BOEING AIRPLANE COMPANY SEATTLE 24, WASHINGTON	PAGE 90
APR						
DRN.	B.T. BROWN 8-14-61					

US 4076 7000 (WAS BAC 1600-R2)

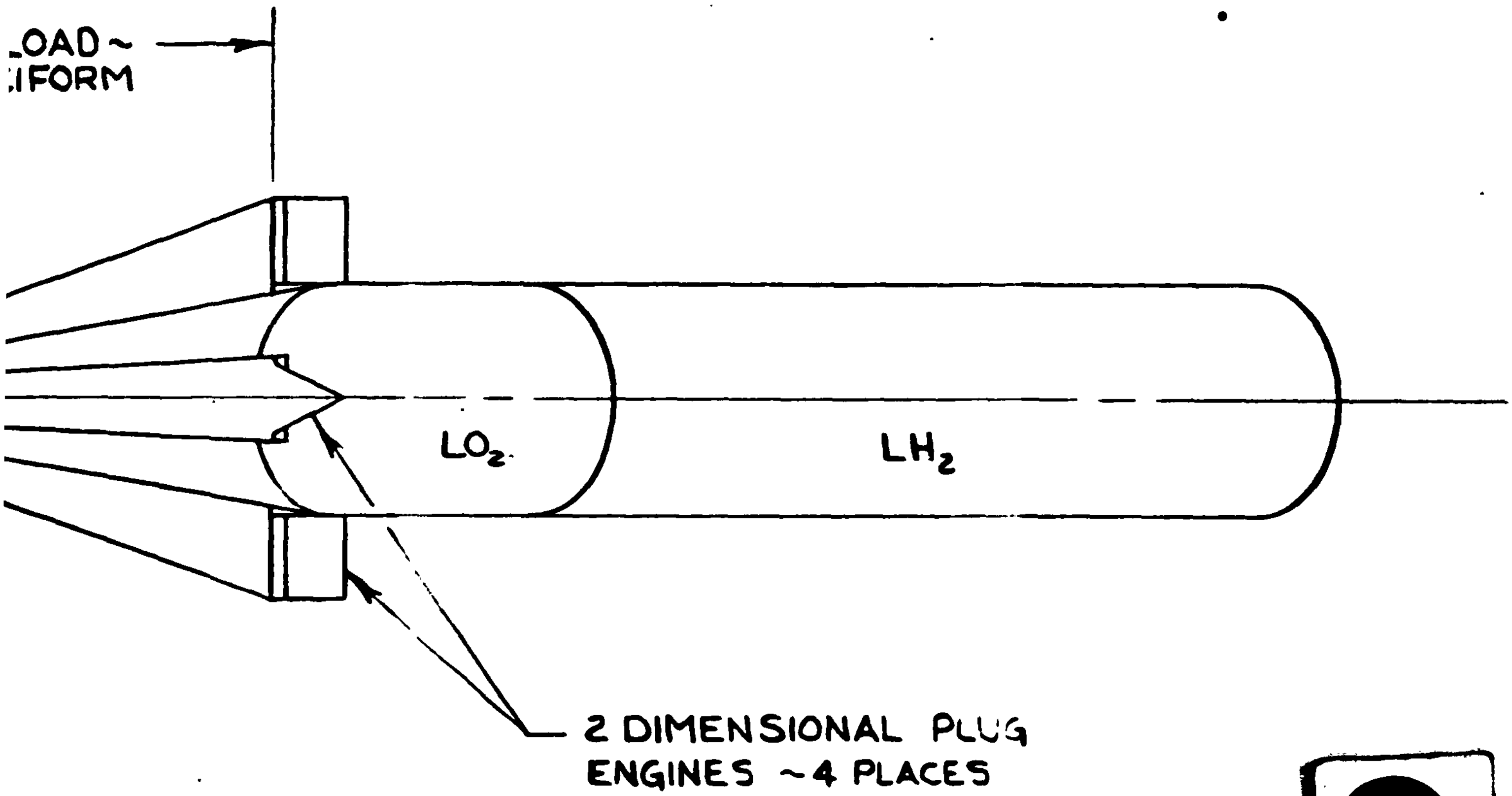
CONFIDENTIAL



1

CALC			REVISED	DATE
CHECK				
APR.				
APR.				
DRN	B.T. BROWN	[illegible]		

U3 4075 7000 (WAS BAC 1000-R2)



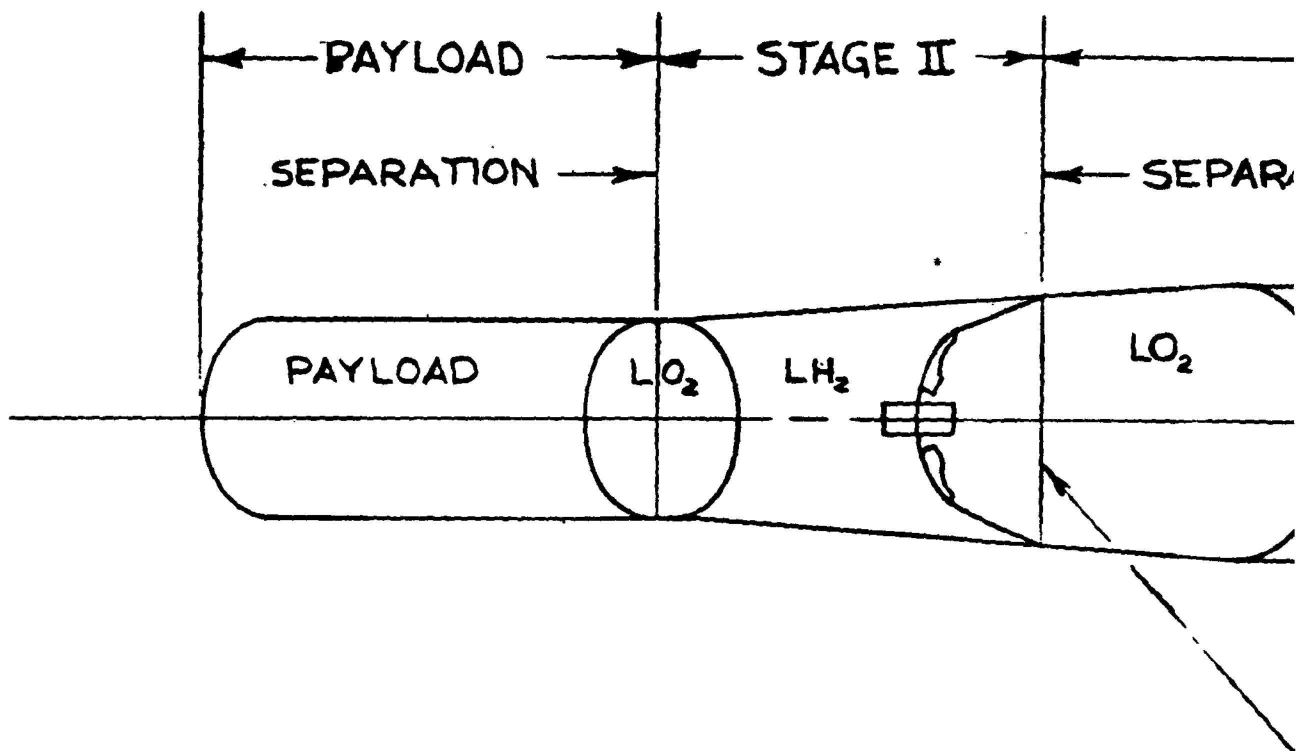
2

CONFIDENTIAL

			REVISED	DATE		
CALC					MODEL 902-5b	FIG. 14.2
CHECK						D2-12072
APR.					BOEING AIRPLANE COMPANY	PAGE
APR.					SEATTLE 24, WASHINGTON	91
DRN	B.T. BROWN	8-14-61				

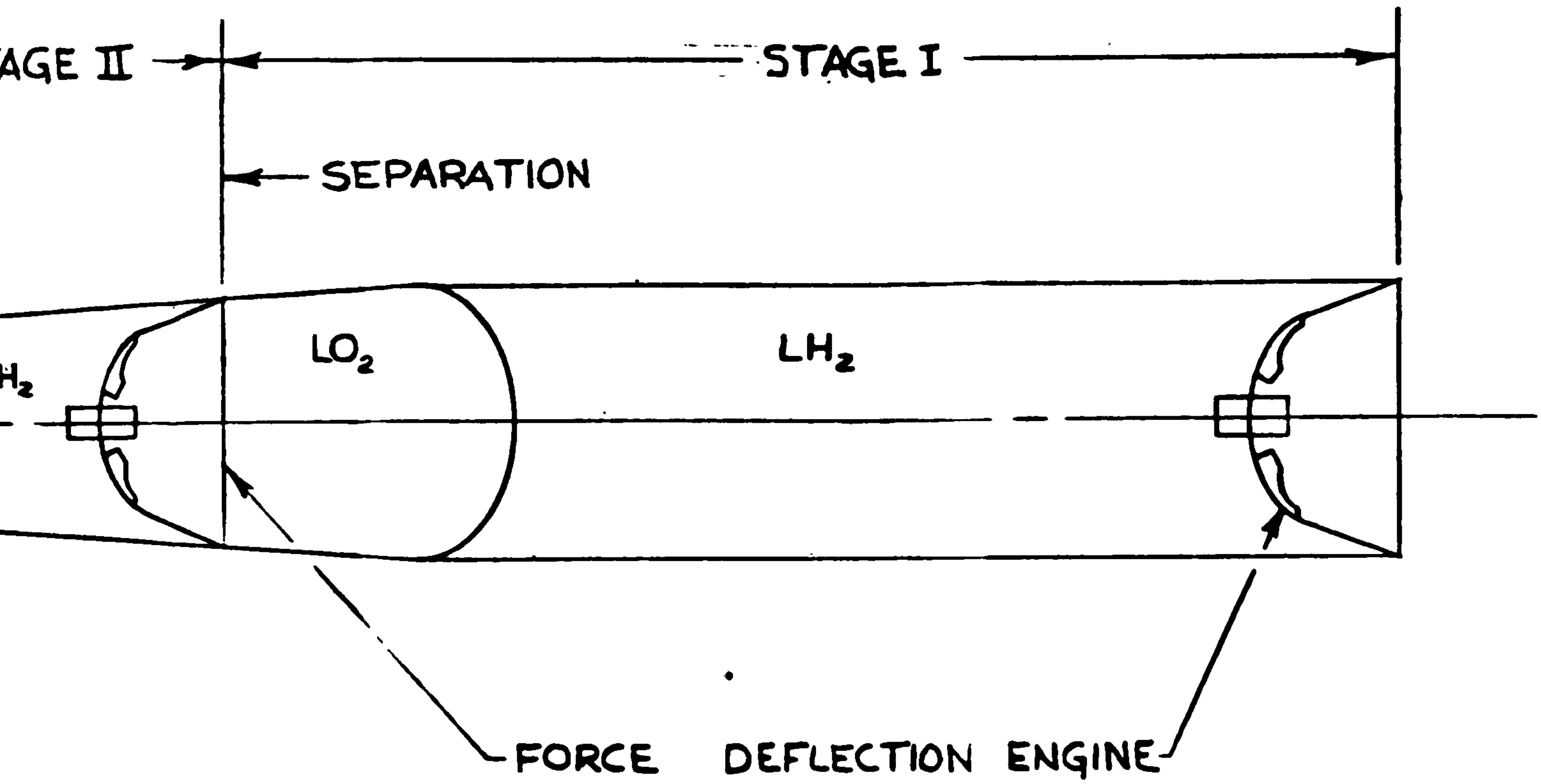
U3 4070 7000 (WAS BAC 1000-R2)

CONFIDENTIAL



1

CALC
CHECK
APR.
APR
DRN



2

			REVISED	DATE		
CALC					MODEL 902-5c	FIG. 14.3
CHECK						D2-12072
APR.						
APR					BOEING AIRPLANE COMPANY SEATTLE 24, WASHINGTON	PAGE 92
DRN	B.T. BROWN	6-14-61				

14.2 Cont.

These configurations represent somewhat idealistic approaches to the tankage problem. It is recognized that previous studies on similar arrangements have revealed undesirable characteristics. However, because of the potential gains, it is believed that further work in the areas represented by Models 902-5A, -5B and -5C is justified and should be undertaken before a final recommendation is made.

In addition to the unconventional Model 902-5 arrangements sketched, other varied concepts were examined, including some suggested by the Aerojet-General Corporation at the onset of this study. The more pertinent of the latter are briefly commented on in the paragraphs following the Model 902-5 series descriptions. No evaluation has been made of the lifting body vehicles or air breathing engine applications suggested because of time limitation.

14.3 <u>MODEL 902-5A</u>

This is a single stage vehicle employing the volumetric criteria of model 902-4. See fig. 14.1. Spherical tankage has been used in an effort to reduce tankage weight. It will be noted that vehicle length is also reduced. It is recognized that, while excellent as pressure vessels, the spherical tanks will present support problems due to mass effects of propellant and structure when subjected to accelerations. Preliminary work indicates that a structural system might be devised which could result in a significant weight saving. As is possible in other applications of the F-D engine, advantage is taken of the possibility of allowing the ground support structure to extend through the air vent ports of the engine and engage the thrust structure, thereby

14.3 Cont.

eliminating the requirement for special ground support structure on the vehicle base. The use of bungee retractable compression props might be considered for high launch wind conditions.

14.4 MODEL 902-5B

This vehicle is arranged with engines forward as shown in fig. 14.2. The concept was used by Goddard in his early models and minimizes stability and control problems. Such configurations largely eliminate the need for an elaborate gantry, since the tankage can extend below the surface of the launch area without the requirement for exhaust disposal as in conventional types. The configuration shown has a two dimensional plug nozzle engine mounted in the trailing edge of each of the cruciform arranged wings of a re-entry vehicle. Light weight tankage is assured since all members are primarily in tension and stabilized by tank pressure. The details of propellant delivery and engine exhaust impingement must be worked out and trade studies made before the advantages can be confirmed. It will be recognized that other engines may also be employed on tractor configurations.

14.5 MODEL 902-5C

The configuration shown in fig 14.3 represents a two stage tandem "tankage" vehicle with the second stage engine immersed in the first stage tank. This essentially eliminates one tank head and the usual interstage structure. However, the resulting inverted tank head will not be as efficient structurally and will increase unusable propellants. It appears that this approach is particularly suited to fixed engine

14.5 Cont.

installations having a minimum requirement for mechanisms. Full advantage of this concept requires that the engine components be incorporated into the second stage tank bottom. Positive shut offs would be required for upper stage propellant control. Developmental work would be required to accommodate the environment created by intimate contact of engine and accessory components with the propellants. In order to further reduce length, the first stage engine could likewise be incorporated in the tank bottom as shown. It is anticipated that propellant delivery to engine may be somewhat complicated by this arrangement due to internal connection requirements.

14.6 AGC UNCONVENTIONAL CONCEPTS

14.6.1 Standard Vehicle, Mod. I (No Gimbal, thrust vector control by secondary injection)

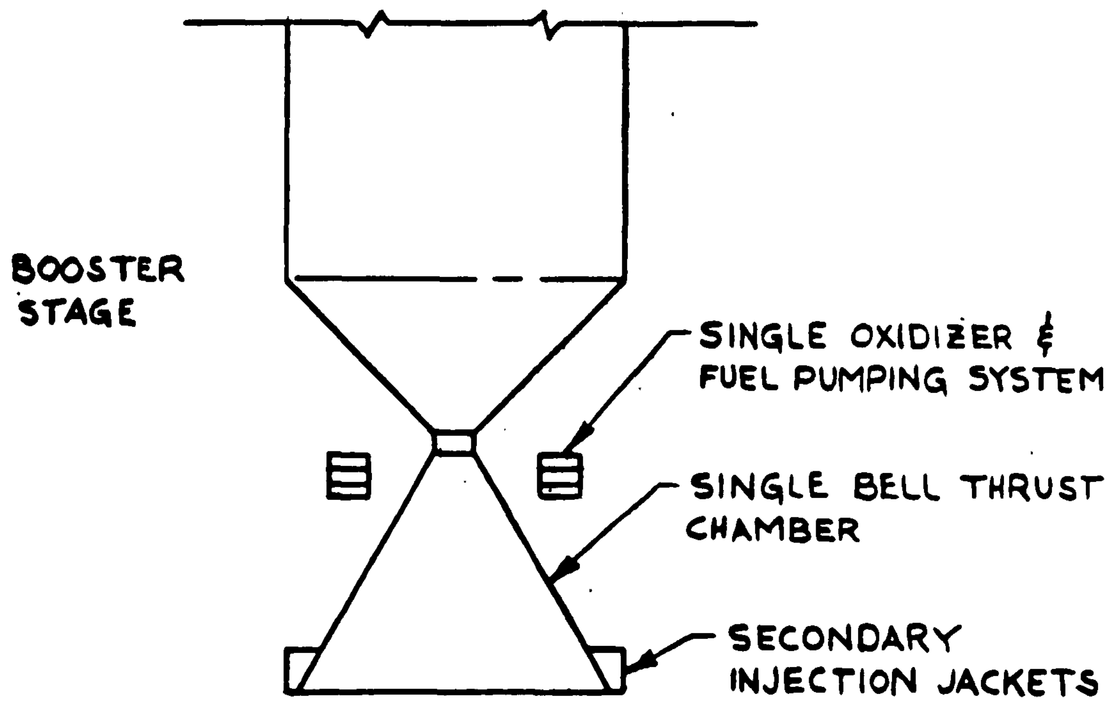
Refer to Sketch Fig 14.4. As discussed in Section 10.2, it appears that, because of continuous demand to correct vehicular thrust alignment discrepancies, secondary injection for thrust vector control would require analysis for each application in order to establish desirability from a propellant requirement standpoint.

14.6.2 Standard Vehicle, Mod. II (Tank embedded engine, thrust vector control by secondary injection)

Same comment as for Mod. I above, In addition, significant increase in structural weight required to stabilize the inverted tank ends, may approach weight savings due to stage shortening achieved by this design. See Model 908-5C, paragraph 14.5.

STANDARD VEHICLE MOD. I

NO GIMBAL
THRUST VECTOR CONTROL BY
SECONDARY INJECTION



			REVISED	DATE		
CALC					STANDARD VEHICLE MOD. I	FIG. 14.4
CHECK						D2-12072
APPD						
APPD					BOEING AIRPLANE COMPANY	PAGE
DRN	B.T.BROWN	8-17-6[illegible]				96

14.6.3 Standard Vehicle, Mods. III, IV and V (Engine and pump clustering concepts)

These were not considered in order to concentrate the limited time available on applications for the basic F-D engine concept.

14.6.4 Standard Vehicle Mod. VI (Submerged 2nd stage engine in first stage tanks)

See comments on Model 902-5C, paragraph 14.5.

14.6.5 Standard Vehicle Mods. VII and VIII (Clustered booster units)

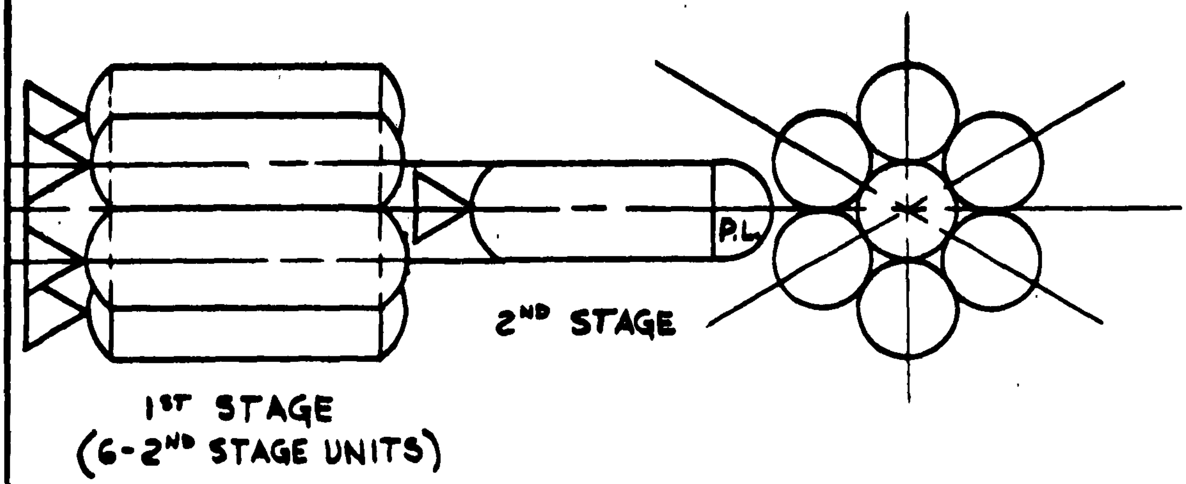
These were not considered in order to concentrate the limited time available on applications for the basic F-D engine concept. Refer to 14.5 for conceptual sketches.

14.6.6 Unconventional Tankage

Refer to fig 14.6. For toroidal, clustered spherical, clustered cylindrical, spherical and disk tankage configuration concepts have in common the structural problem of engine thrust transfer to the propellant contained. Distribution of the thrust load by a multiplicity of engines renders the control problem critical as well and imposes further structural penalties if engine out conditions are considered. It was considered beyond the scope of this study to investigate these areas sufficiently to permit valid conclusions to be drawn.

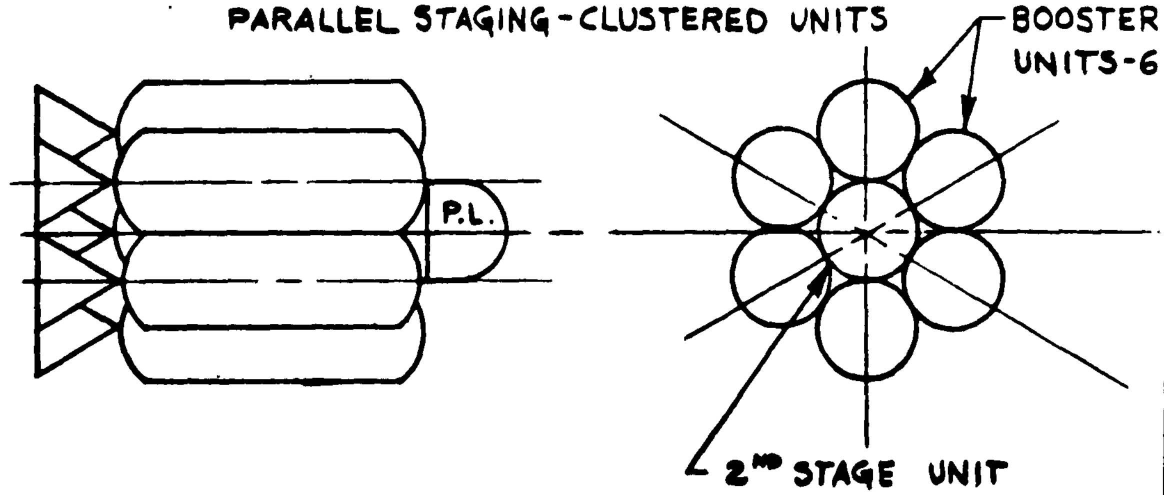
STANDARD VEHICLE MOD. VII

CLUSTERED BOOSTER UNITS



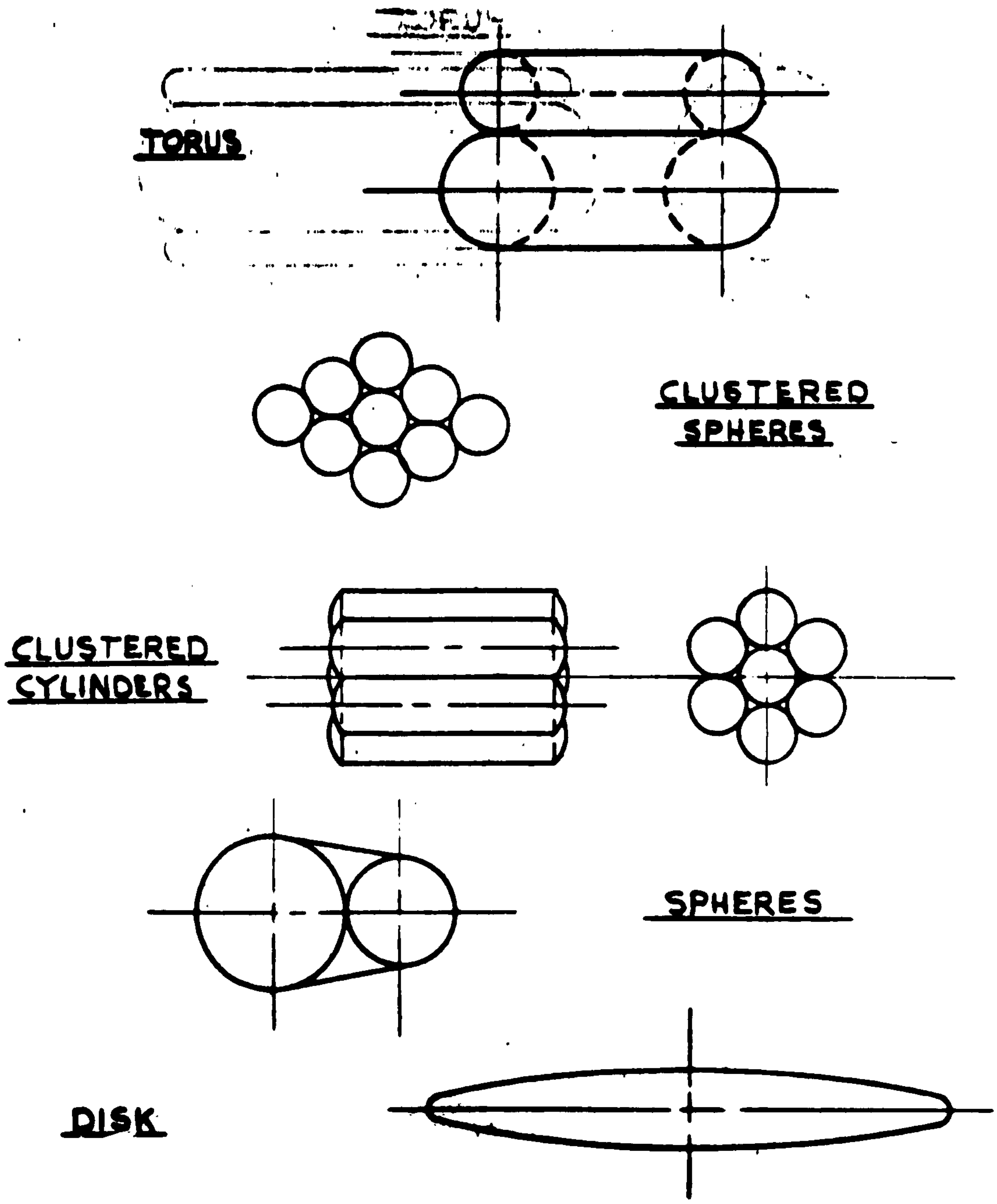
STANDARD VEHICLE MOD. VIII

PARALLEL STAGING-CLUSTERED UNITS



			REVISED	DATE		
CALC					STANDARD VEHICLE MODS. VII & VIII	FIG 14.5
CHECK						D2-12072
APPD						
APPD					BOEING AIRPLANE COMPANY	PAGE
DRN	B.T. BROWN	8-17-61				98

UNCONVENTIONAL TANKAGE



CALC			REVISED	DATE	UNCONVENTIONAL TANKAGE	FIG. 14.6
CHECK						D2-12072
APPD						
APPD					BOEING AIRPLANE COMPANY	PAGE
DRN	B.T. BROWN	8-21-61				99

15.0 REFERENCES

15.1 Boeing Document D2-9145, Phase I Report, "Advanced Propulsion System Study", Contract AF 04(611)-5970, dated October, 1960.

15.2 Boeing Document D2-10696, Final Report, "Advanced Propulsion System Study", Contract AF 04(611)-5970, dated December, 1960.

15.3 Aerojet General Report, "A Study of Unconventional Rocket Engines, Task I", Contract NAS 51025.

15.4 Aerojet General Report LRP 125 Special, Revision B, "Liquid Rocket Engine Parameter Study," dated July 15, 1959.

15.5 Boeing Document D2-12154, "Advanced Propulsion System Study, Phase III", Contract AF04(611)-7029, dated August 1961.