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CLASSIFICATION CHANGE

To UNCLASSIFIED

By authority of GDS-GP-4 Date 7-9-74

Changed by L. Shirley

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Scientific and Technical Information Facility

MARTIN

X65 13356

X65 13356 (ACCESSION NUMBER)	(THRU)
93 (PAGES)	2D (CODE)
CR 57240 (NASA CR OR TMX OR AD NUMBER)	31 (CATEGORY)

FACILITY FORM 602

POST SATURN

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SUMMARY REVIEW(U)

Launch Vehicle Study

Oct. 1964

"Available to U.S. Government Agencies and U.S. Government Contractors Only."

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(NASA-CR-57240) SUMMARY REVIEW, LAUNCH VEHICLE STUDY (Martin Co.) 93 p

N75-72112

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ENGINEERING REPORT NO. 12604

CONTRACT NO. NAS 8-11123

~~GROUP 4~~
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MARTIN
BALTIMORE DIVISION
BALTIMORE, MARYLAND 21203

C65-1127

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FOREWORD

This booklet contains copies of the visual aids presented by Martin Company to NASA Marshall Space Flight Center during the Post Saturn Part III Final Summary Review on October 27, 1964. The work was conducted under Contract NAS 8-11123 between NASA and Martin Marietta Corporation.

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CONTENTS

I. INTRODUCTION

PARTS I AND II--CONFIGURATION EMPHASIS

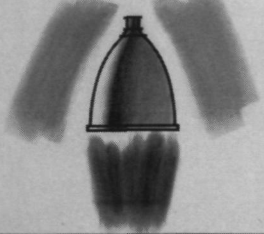
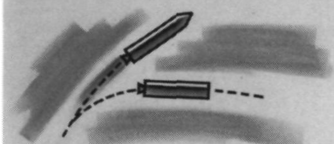
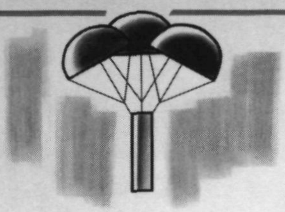
The launch vehicle configurations developed during Parts I and II of the Post Saturn Study reflect the emphasis shown on this chart. (Bold type lettering is used to indicate greater emphasis.) For example, with regard to staging modes considered, the greatest emphasis during Part I was placed on designs utilizing tandem staging.

PART III OBJECTIVES

The three basic objectives of Part III are shown on this chart.

- Design studies were to concentrate in areas that were least defined and which had the greatest effect on the overall system.
- Role of Post Saturn was to be determined by comparing it with potential mission requirements and existing launch vehicle capabilities.
- Items of required advanced technology were to be identified.

PARTS I & II -- CONFIGURATION EMPHASIS

	PART I	*PART II*
	**F-1** **M-1** **SOLID** HP HT RENE	**HP** **RENE** F-1 M-1 SOLID
	**TANDEM** PARALLEL PARTIAL SINGLE	**TANDEM** **SINGLE**
	**EXPENDABLE**	**REUSE**

POST SATURN MARTIN

PART III -- OBJECTIVES

- DESIGN STUDIES OF P.S.L.V. CONCEPTS
- ROLE OF POST SATURN
- ADVANCED TECHNOLOGY RQMTS

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DISTRIBUTION OF EFFORT

Approximately 60% of the effort expended during Part III was addressed to Launch Vehicle Studies, the other 40% to Mission and Operations Analysis. The various activities associated with each of these categories are listed on this chart in order of decreasing effort.

PART III--CONFIGURATION EMPHASIS

The launch vehicle configurations developed during Part III of the Post Saturn Study reflect the emphasis shown on this chart. (Bold type lettering is used to indicate greater emphasis.) For example, with regard to propulsion systems considered, the greatest emphasis during the initial portion of Part III was placed on designs utilizing either a high chamber pressure engine or the M-1 engine. Lesser emphasis was placed on designs utilizing solid motors or F-1 engines.

PART III -- DISTRIBUTION OF EFFORT

LAUNCH VEHICLE STUDIES

DESIGN STUDIES

- CONFIG DEFINITION
- TEST AND LAUNCH OPS
- MANUFACTURING PLANNING
- RELIABILITY
- DEVELOPMENT PLANS
- QUALITY ASSURANCE
- GROUND SUPPORT EQUIPMENT
- ADVANCED TECHNOLOGY REQUIREMENTS

MISSION AND OPERATIONS ANALYSIS

MODEL DEVELOPMENT

- MISSION ANALYSIS
- PILOT RUNS
- MODEL APPLICATION
- DEVELOPMENT PLANS

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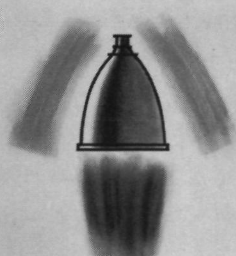
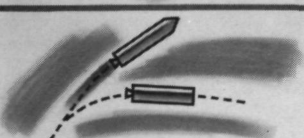
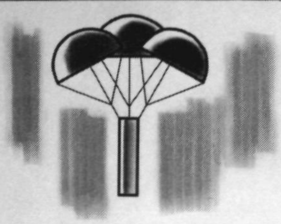
PART III -- CONFIGURATION EMPHASIS

	PRE MID-TERM	*POST MID-TERM*
	HP M-1 SOLID F-1	M-1 300K HP HP F-1
	TANDEM	TANDEM
	PARTIAL REUSE	TOTAL REUSE

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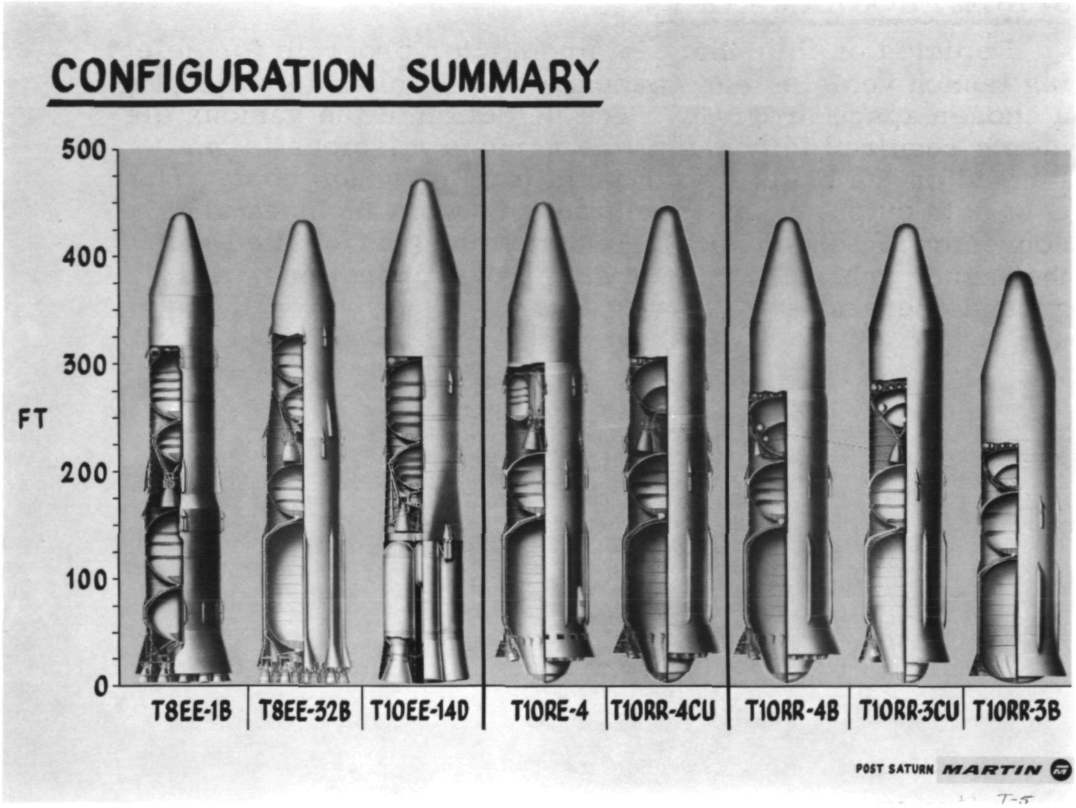
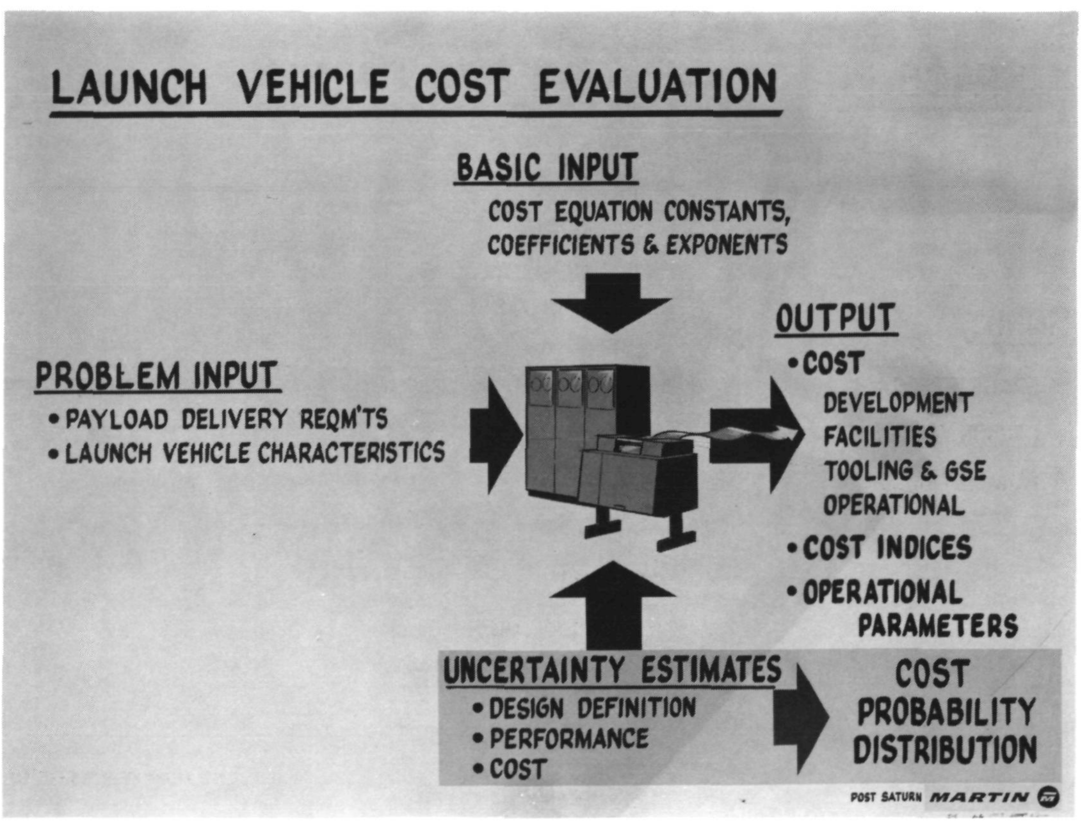
CONFIGURATION SUMMARY

Three categories of candidate configurations are shown. The first category includes expendable configurations utilizing propulsion technology currently under development. The second category incorporates vehicle recovery and the application of advanced engine nozzle technology. The third category includes recovery and the application of both advanced engine and engine nozzle technology. The currently recommended configuration is T10RR-4B. This all-recoverable configuration incorporates a plug cluster of 18 M-1 engines in the first stage and a plug cluster of 16 "300K" engines in the second stage.

LAUNCH VEHICLE COST EVALUATION

The cost evaluation model represented on this chart is a computer program for costing large launch vehicles. This program was initially developed during Part I of our Post Saturn study and since then has been modified on essentially a continuing basis. Throughout this period of time, it has been used extensively to evaluate the cost of the various Post Saturn configurations that have been considered. The program can be exercised with or without a Monte Carlo feature that allows the determination of cost projections as a function of probability of occurrence.


CONFIGURATION SUMMARY
500
400
300
FT
200
100
0
T8EE-1B
T8EE-32B
T10EE-14D
T10RE-4
T10RR-4CU
T10RR-4B
T10RR-3CU
T10RR-3B
POST SATURN MARTIN


LAUNCH VEHICLE COST EVALUATION
BASIC INPUT
COST EQUATION CONSTANTS,
COEFFICIENTS & EXPONENTS
PROBLEM INPUT
• PAYLOAD DELIVERY REQM'TS
• LAUNCH VEHICLE CHARACTERISTICS
OUTPUT
• COST
DEVELOPMENT
FACILITIES
TOOLING & GSE
OPERATIONAL
• COST INDICES
• OPERATIONAL
PARAMETERS
UNCERTAINTY ESTIMATES
• DESIGN DEFINITION
• PERFORMANCE
• COST
COST
PROBABILITY
DISTRIBUTION
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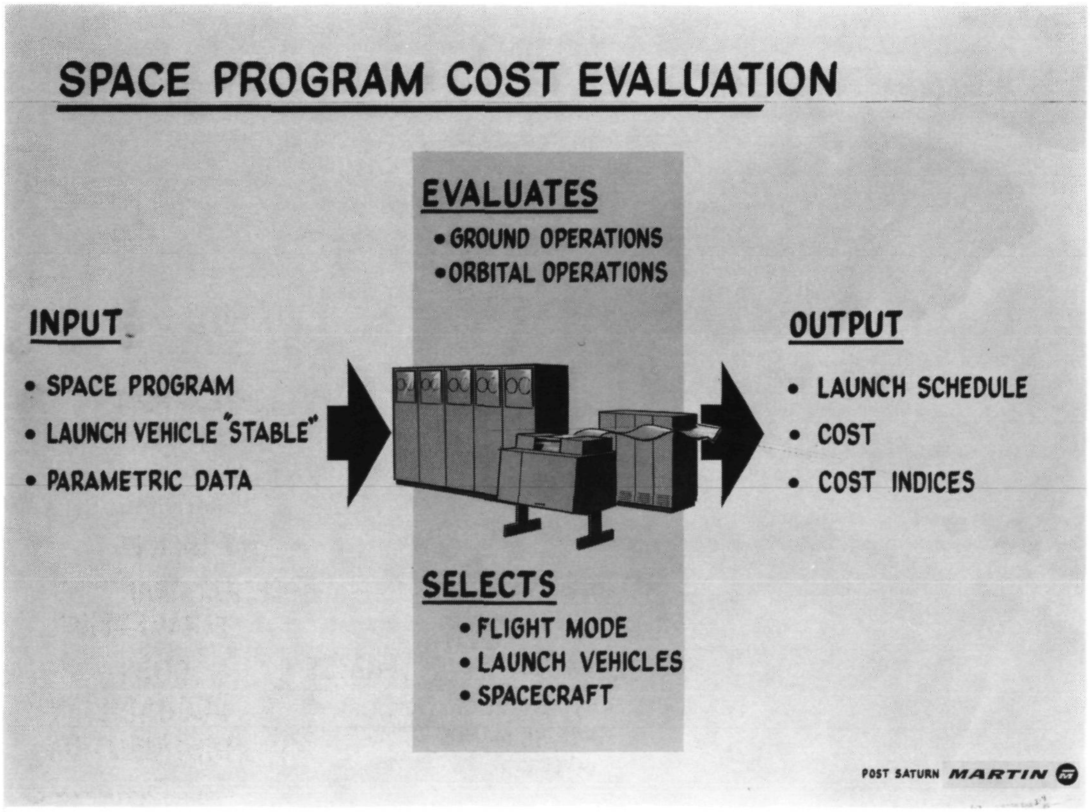
SPACE PROGRAM COST

Depicted on this chart is a computer program for selecting launch vehicles and spacecraft best suited to satisfying a chosen space program. The selection of the various elements required for each of the various missions is performed on the basis of minimum total program cost. The cost data output of this computer program includes all nonrecurring and recurring expenditures associated with the launch vehicle, spacecraft, orbital operations, and ground operations.



II. LAUNCH VEHICLE STUDIES

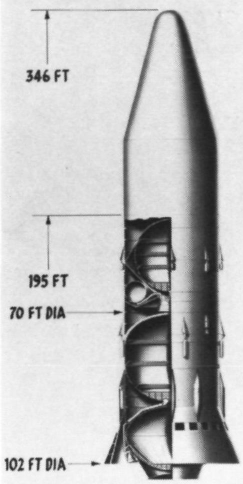
RP VERSUS LH_2--HIGH P_c

- LH_2 vehicle is larger--1.25% length of RP configuration.
- LH_2 vehicle is lighter--70% of liftoff weight of RP configuration.
- LH_2 vehicle is more economical--slight operational and larger total cost advantage.
- LH_2 vehicle has a small advantage over RP on the basis of a qualitative evaluation of 140 factors such as engine development risk, operational flexibility, schedule risk, and explosion hazards. Maximum possible score was 1000 points.
- Recommend the pursuit of high pressure LH_2 propulsion technology.

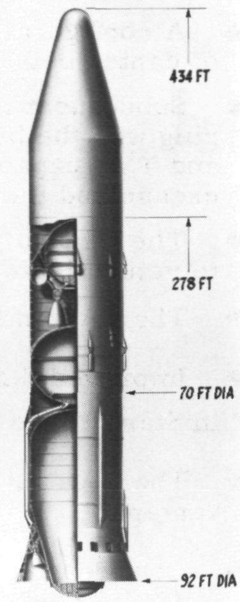
ALTITUDE COMPENSATION EFFECT

Stage I of a two stage vehicle utilizing high performance propulsion systems burns for a period of approximately 200 sec. The time average I_{sp} for a bell nozzle ϵ = 60 and for an altitude-compensated 10% plug nozzle shows a clear advantage for the plug for burning times beyond 130 sec. At the end of 200 sec, the plug nozzle has averaged 8 sec more I_{sp} than the bell nozzle.

RP vs LH_2--HIGH P_C

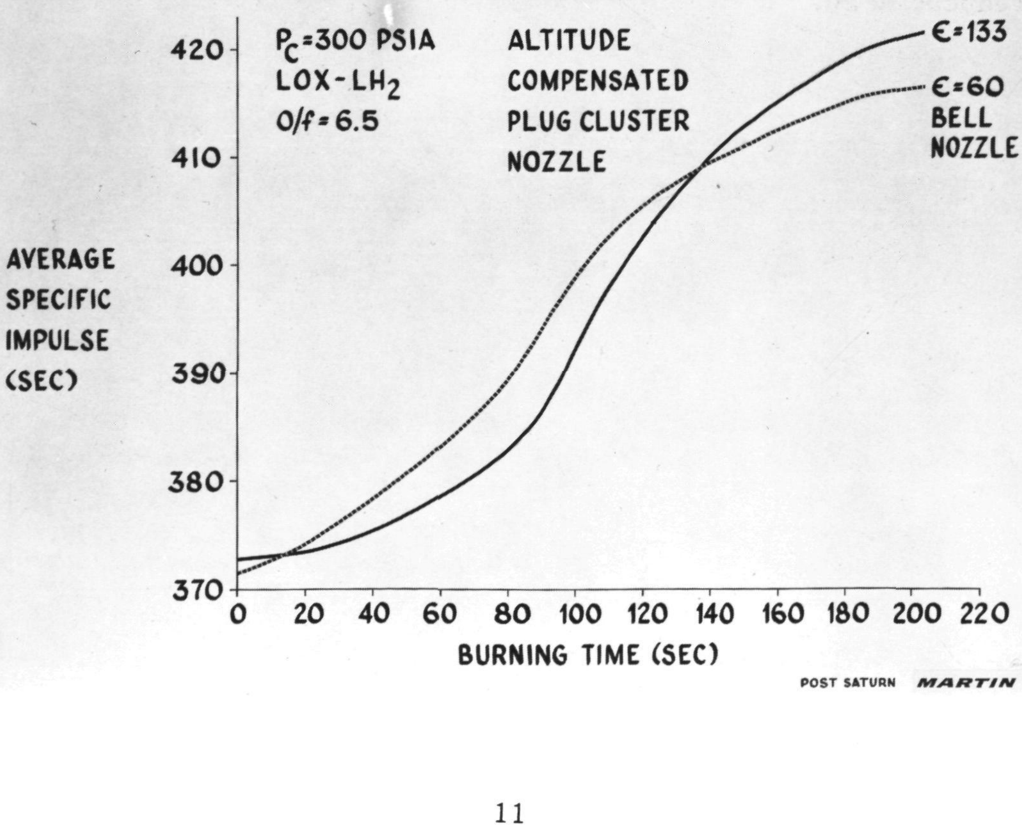


T10RE-2A RP		T10RE-3A LH_2
24/1	NUMBER OF ENGINES NI/NII	18/2
954	PAYLOAD K LB MASS	995
22.35	MASS-LIFT OFF, M LB	15.8
62.4	OCE \$/LB MASS	58.8
145.3	TCE \$/LB MASS	133.9
835	QUALITATIVE EVAL	872



POST SATURN MARTIN

ALTITUDE COMPENSATION EFFECTS



POST SATURN MARTIN

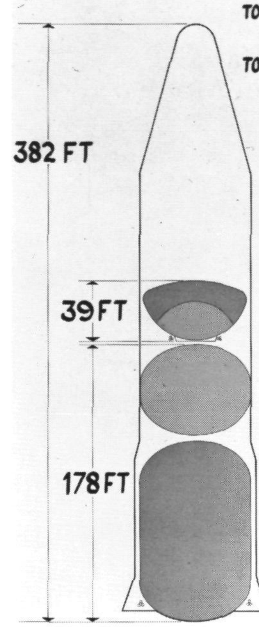
NOZZLE COMPARISON-HI PRESSURE CONFIGURATIONS

- Part II studies indicated the advantage of altitude compensation.
- A configuration utilizing a zero length plug with a cluster of canted bell nozzles was designed.
- Subsequent test development results indicated that a 10% plug was the best compromise between high performance and TVC capability; this requires the vehicle length to increase and the propellant fraction to decrease.
- The toroidal aerospike engine presents an opportunity to return to zero nozzle length.
- The TA configuration is 54 ft shorter than the PN.
- Improved Stage II packaging and a 10-sec I_{sp} advantage in Stage I give 79,000 lb more payload for the TA.
- The TA configuration enjoys a cost effectiveness advantage.
- Confidence in TA performance has yet to be validated. A 3% I_{sp} decrease would cancel the TA case advantage.
- Until TA performance is confirmed by test, the clustered bell plug nozzle approach is recommended.

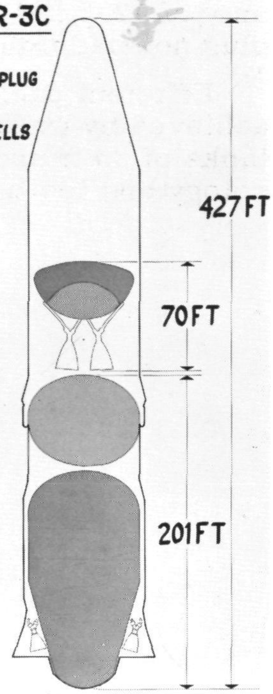
CLASS I M-1/M-1 CONFIGURATION

This configuration utilizes the M-1 engine unmodified except for the Stage I module expansion ratio which has been reduced to 20.

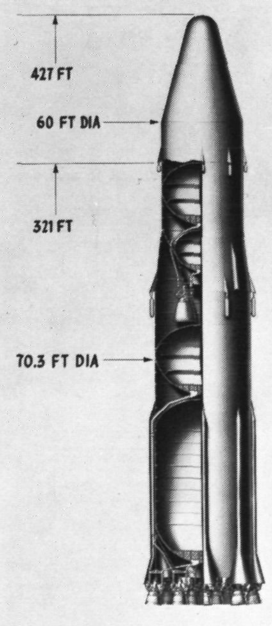
NOZZLE COMPARISON--HIGH-PRESSURE CONFIGURATIONS



T10RR-3B	CONFIGURATION	T10RR-3C
TOROIDAL AEROSPIKE	ENGINE NOZZLE, N-I	BELLS ON 10% PLUG
TOROIDAL AEROSPIKE	ENGINE NOZZLE, N-II	SEPARATE BELLS
1.16	EQUIV SL THRUST/MODULE (LB×10^6)	1.20
1880	EQUIV DEV'T TESTS REQ'D	1450
0.904	PROP MASS FRACTION, N-I	0.891
0.871	PROP MASS FRACTION, N-II	0.871
1044	MINIMUM PAYLOAD (LB×10^3)	965
15.1	LAUNCH WT/PAYLOAD RATIO	16.4
127.0	TOTAL COST-EFFECTIVENESS ($/LB)	137.0
49.0	CUM OPER COST-EFF ($/LB)	54.2
HIGH	RELATIVE ENGINE DEV'T RISK	MODERATE



CLASS I M-1/M-1 CONFIGURATION--T8EE-32B



• ENGINES NI/NII	14/2
• PROPELLANT FRACTION NI/NII	0.884/0.896
• PAYLOAD (LB)/(GRAMS)	712K/323G
• LAUNCH MASS (LB)/(GRAMS)	13.58M/6,150G
• LAUNCH MASS/PAYLOAD MASS	19.2
• RELIABILITY	
LAUNCH TO 225-KM	0.920
LAUNCH TO RECOVERY NI	--
• COST EFFECTIVENESS	
OCE ($/LB)/($/Kg)	110.99/244
TCE ($/LB)/($/Kg)	179.2/397

POST SATURN MARTIN

CLASS II M-1/M-1 CONFIGURATION--T10RE-4

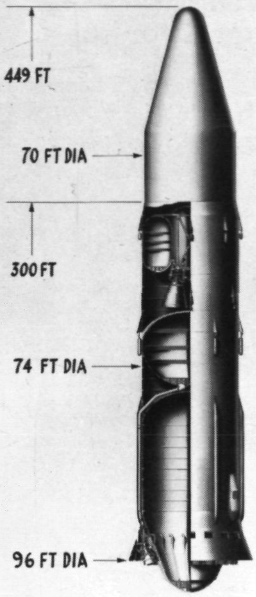
This partially reusable configuration utilizes an advanced application of the M-1 engine. Stage I engines have an ϵ of 12.5 and are installed in a 10% altitude compensating plug nozzle. Stage II utilizes an ϵ of 55 for the bell nozzles.

Efficient packaging of the expendable Stage II has been achieved by utilizing a cluster of cylindrical propellant tanks of 33 ft and 22 ft in diameter, and thus utilizing technology and tooling developed for Saturn V.

REUSABLE M-1/M-1 CONFIGURATION--T10RR-4CU

Recovery and reuse of Stage II requires the addition of solid propellant deorbit rockets to provide a ΔV of 500 fps. Selection of forward end re-entry also requires the addition of long aft skirts with drag flaps to provide static stability. A common dome tank configuration is utilized to minimize weight.

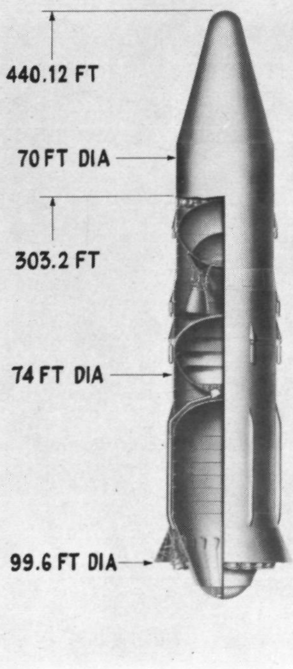
CLASS II M-1/M-1 CONFIGURATION--T10RE-4



• NUMBER OF ENGINES NI/NII	18/2
• PROPELLANT FRACTION NI/NII	0.880/0.877
• PAYLOAD MASS (LB)/(GRAMS)	939K/426G
• LAUNCH MASS (LB)/(GRAMS)	18.70M/8,480G
• LAUNCH/PAYLOAD	20.0
• DRY MASS (LB)/(GRAMS)	1.83M/830G
• DRY/PAYLOAD	1.95
• RELIABILITY	
LAUNCH TO 225 KM	0.920
LAUNCH TO RECOVERY N-I	0.961
• COST EFFECTIVENESS	
OCE ($/LB)/($/Kg)	68.2/150
TCE ($/LB)/($/Kg)	147.3/325

POST SATURN MARTIN

REUSABLE M-1/M-1 CONFIGURATION--T10RR-4CU



• NUMBER OF ENGINES NI/NII	18/2
• PROPELLANT FRACTION NI/NII	0.870/0.842
• PAYLOAD MASS (LB/GRAMS)	820K/372G
• LAUNCH MASS (LB/GRAMS)	18.4M/8,350G
• LAUNCH / PAYLOAD	22.5
• RELIABILITY	0.92
• COST EFFECTIVENESS	
OCE ($/LB)/($/Kg)	71.8/158
TCE ($/LB)/($/Kg)	163.3/360

POST SATURN MARTIN

EFFECT OF RECOVERY--ALL M-1 CONFIGURATION--T10RR-4CU

A cost evaluation to determine the influence of recovery upon this configuration was conducted by stripping the recovery provision from Stage II for the RE cost, and by further stripping of recovery provisions from Stage I for the EE cost.

The data shows a modest improvement--10% in TCE and 31% OCE--for recovery of Stage II over the completely expendable version.

A further look at the data shows that recovery and reuse of Stage II is not profitable. This is due to the relatively small size of the stage, and to the large weight investment in stabilization skirts, flaps, and solid propellant deorbit rockets.

Realizing that recovery of Stage II is not profitable, a re-examination of the use of clustered tanks or common dome tanks for Stage II is in order. Clustered tanks are recommended because of technology, tooling, and hydrostatic test advantage even though the common dome configuration enjoys a 26,000-lb payload and slightly better cost effectiveness.

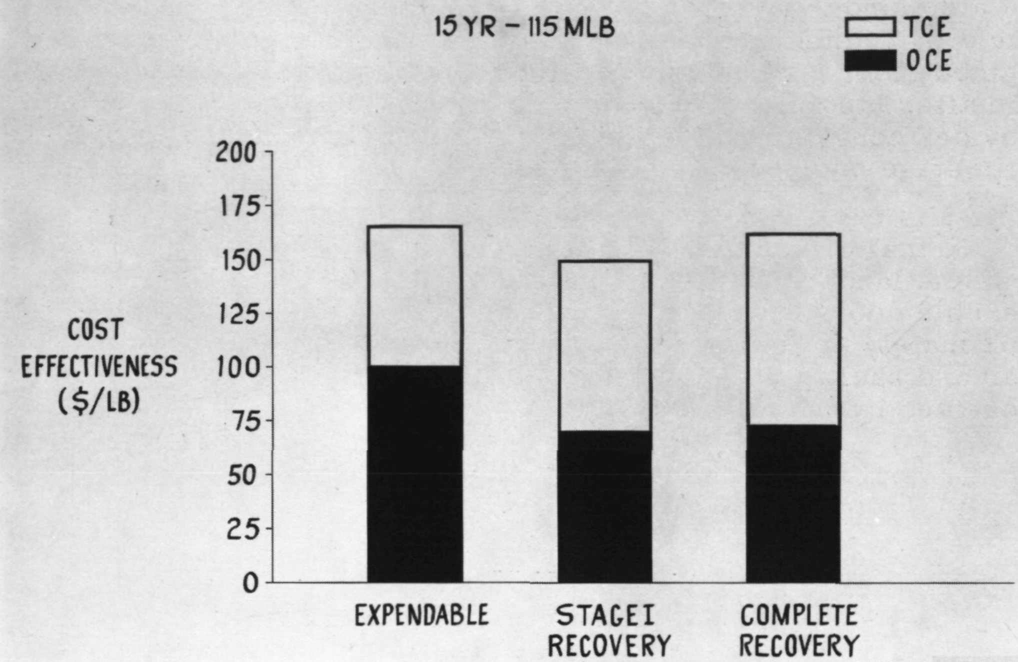
M-1/300K REUSABLE CONFIGURATION--T10RR-4A4

Stage II of this configuration utilizes a shape similar to Mercury and does not require auxiliary drag stabilization devices for supersonic re-entry speeds.

The 16 toroidal aerospike engines in Stage II exhaust through doors in the dual purpose, base-re-entry heat shield. These engines being small in thrust and throttleable are also utilized to provide the deorbit velocity increment of 500 fps.

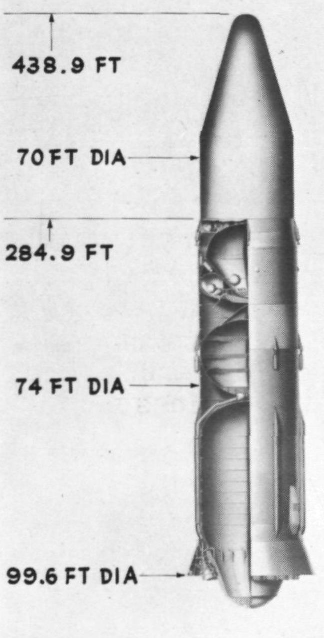
The conical nose on the forward end of Stage II is for water landing load impact attenuation.

EFFECT OF RECOVERY-ALL M-1 CONFIGURATION-T10RR-4CU



POST SATURN MARTIN

M-1/300K REUSABLE CONFIGURATION--T10RR-4A4



- NUMBER OF ENGINES NI/NII 18/16
- PROPELLANT FRACTION NI/NII 0.874/0.838
- PAYLOAD MASS (LB)/(GRAMS) 1.067M/483G
- LAUNCH MASS (LB)/(GRAMS) 18.4M/8,340G
- LAUNCH/PAYLOAD 17.2
- RELIABILITY 0.92
- COST EFFECTIVENESS
  - OCE ($/LB)/($/Kg) 58.4/128.5
  - TCE ($/LB)/($/Kg) 144.2/318

POST SATURN MARTIN

N-II OPTIMIZATION--CONFIGURATIONS

These Stage II designs utilize the 300 K toroidal aerospike engines. 4A4 re-enters the earth's atmosphere engines first and also lands engine first. This requires doors in the heat shield that must be closed during re-entry to protect the engines from aerodynamic heating, and from water immersion after landing. The area between the separate tanks provides convenient stowage for the recovery parachutes and main stage engines.

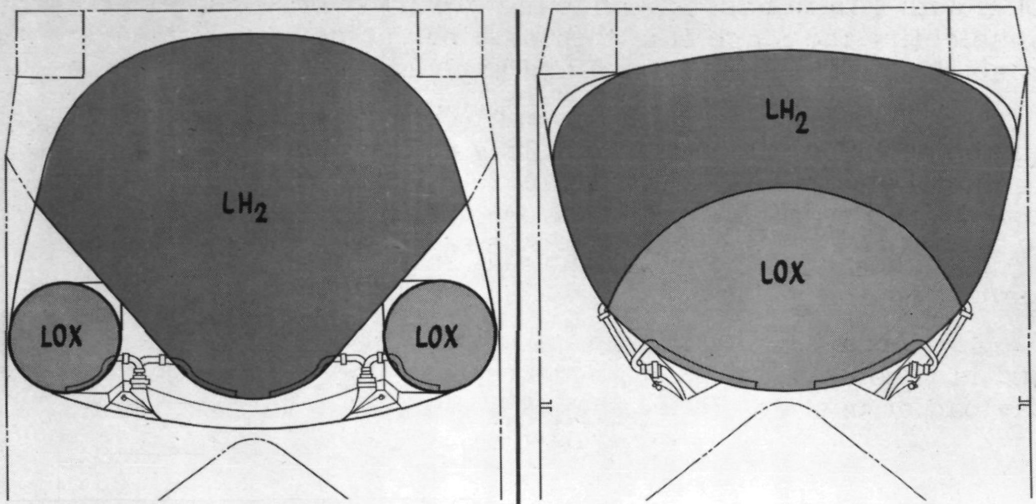
4A5 is packaged around common dome tanks to relieve the two major problems of 4A4. The stage re-enters nose first and lands in this same position thus there are no operable doors in the heat shield and the engines float approximately 45 ft above the water. Stowage of recovery gear and packaging for static stability are the challenges presented by this configuration.

N-II OPTIMIZATION--CONFIGURATIONS

These Stage II designs utilize the "300 K" bell nozzle engines. Each re-enters nose first. Configuration 4A1 utilizes high expansion ratio bell nozzles and an efficient thrust structure that carries into the juncture of the common dome with the tank side wall. The cg of this arrangement is well aft and drag area in the form of operable flaps supported by a cylindrical skirt are required for stability. The aft cg of this configuration also requires auxiliary protection from water for the engines since it floats in a canted position.

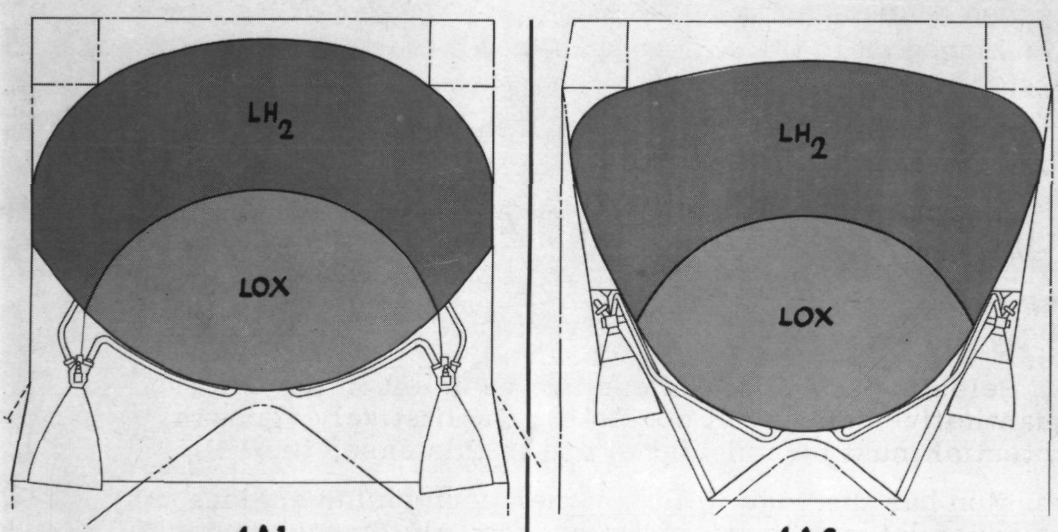
Configuration 4A6 has a less efficient thrust structure and employs a plug nozzle arrangement. The latter utilizing lower expansion ratio nozzles was chosen because of packaging convenience of locating the engines and cg forward. This stage is statically stable and floats with the engines 45 ft above the water.

N-II OPTIMIZATION - CONFIGURATIONS



-4A4 (PRELIM BASELINE)		-4A5	
ENGINE NOZZLE	TOROIDAL AEROSPIKE	ENGINE NOZZLE	TOROIDAL AEROSPIKE
TANKAGE	SEPARATE	TANKAGE	COMMON DOME
RE-ENTRY ATTITUDE	AFT	RE-ENTRY ATTITUDE	FWD

N-II OPTIMIZATION - CONFIGURATIONS



-4A1		-4A6	
ENGINE NOZZLE	SEPARATE BELLS	ENGINE NOZZLE	BELLS ON PLUG
TANKAGE	COMMON DOME	TANKAGE	COMMON DOME
RE-ENTRY ATTITUDE	FWD	RE-ENTRY ATTITUDE	FWD

N-II OPTIMIZATION-WEIGHT AND COST

Maximum variation in payload is 3.2% with 4A5 being the highest. Maximum variation in cost effectiveness is 4% OCE with 4A5 the best. Other factors must be considered in selecting the Stage II of the baseline vehicle since there is so little difference in cost and payload.

Low confidence in achievement of engine performance quoted at 95% of theoretical shifting equilibrium is a factor against 4A4 and 4A5. In addition to this, the heat shield doors are a factor against 4A4.

Edge heating of flaps and engine water protection are factors against 4A1.

Configuration 4A6 has a minimum of undesirable factors and is recommended even though it does not yield the highest payload or best cost effectiveness.

F-1A/300K REUSABLE CONFIGURATION--T10RR-4D

Stage I utilizes uprated F-1 engines in a 10% plug cluster. The engines are hinged off the plug, 0° cant angle, up to 25,000 ft altitude. From this altitude to burnout, the engines are hinged on to the plug to achieve altitude compensation performance benefits.

Stage II is similar to that utilized on Configuration T10RR-4A4.

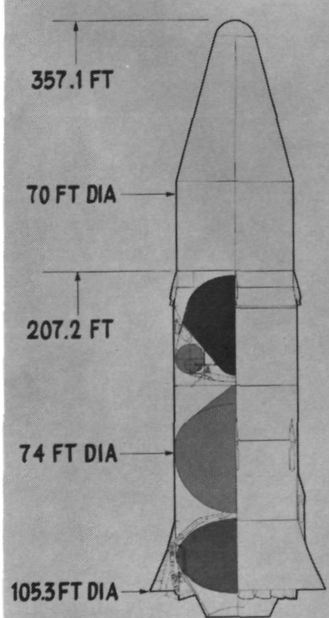
A comparison with T10RR-4A4 utilizing M-1 engines in Stage I shows the F-1 version to be 20% shorter in length, to have 25% heavier launch mass, and 10% less payload than the M-1 version. The total and operating cost are essentially the same for both vehicles.

Selecting the best of these two vehicles on the basis of quantitative data is not possible. Qualitatively, growth potential must be considered and in this case, the LH_2 version has the edge in that higher performing engines may be added later with minimum change; also more successful reuse of LH_2 engines, due to clean burning characteristics is expected. Upon these factors the M-1 vehicle is recommended over the F-1A vehicle.

N-II OPTIMIZATION - WEIGHTS & COSTS

CONFIGURATION	-4A4	-4A5	-4A1	-4A6
•PROPELLANT MASS FRACTION				
N-II	0.888	0.896	0.870	0.891
N-I	0.874	0.874	0.880	0.873
•PAYLOAD (MIN, ENGINE OUT) (LB X10^3)	1067	1096	1068	1064
•COST EFFECTIVENESS ($/LB)				
TOTAL	144.5	140.0	141.5	143.5
CUMULATIVE OPERATING	58.4	55.7	55.9	57.8

F-1A/300K REUSABLE CONFIGURATION--T10RR-4D



- NUMBER OF ENGINES NI/NII — 18/16
- PROPELLANT FRACTION NI/NII — 0.904/0.883
- PAYLOAD MASS (LB)/(GRAMS) — 972K/440G
- LAUNCH MASS (LB)/(GRAMS) — 24.7M/11,200G
- LAUNCH / PAYLOAD — 25.4
- RELIABILITY — 0.92
- COST EFFECTIVENESS
  - OCE ($/LB)/($/Kg) — 60.6/133.5
  - TCE ($/LB)/($/Kg) — 144.0/317.0

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M-1/"300K" BASELINE VEHICLE--T10RR4-B

Incorporated into this configuration are the results of the design studies. Main stage engines provide the de-orbit impulse for Stage II which is statically stable during re-entry without auxiliary drag flaps. Water landing by parachute is accomplished without landing rockets.

Landing of Stage I in the horizontal position is by solid propellant retrorockets after release of the (4) 200-ft diameter main parachutes.

EFFECT OF RECOVERY--BASELINE VEHICLE

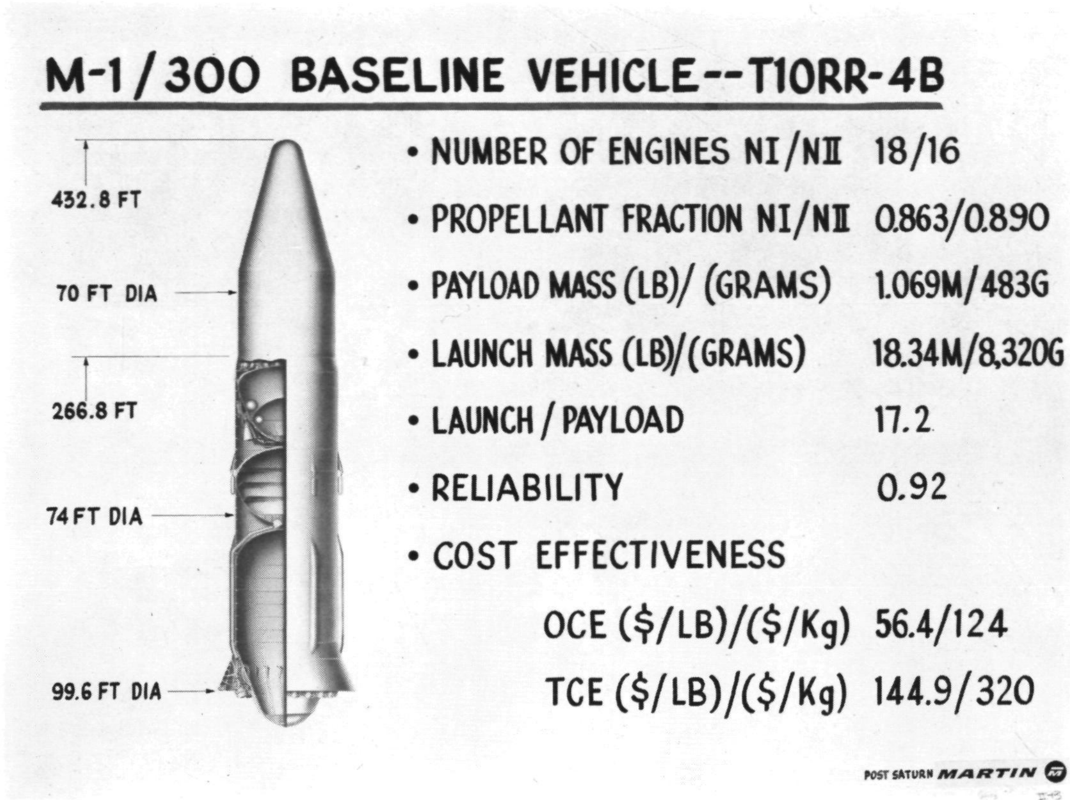
Data shown are for vehicles with stages stripped of easily accessible recovery provisions to form the non-recovered version of that stage. Also there are 7 Stage I's and 5 Stage II's left in inventory at the end of the program on this basis. Percent improvement in cost effectiveness for varying degrees of recovery when compared to the expendable version is:

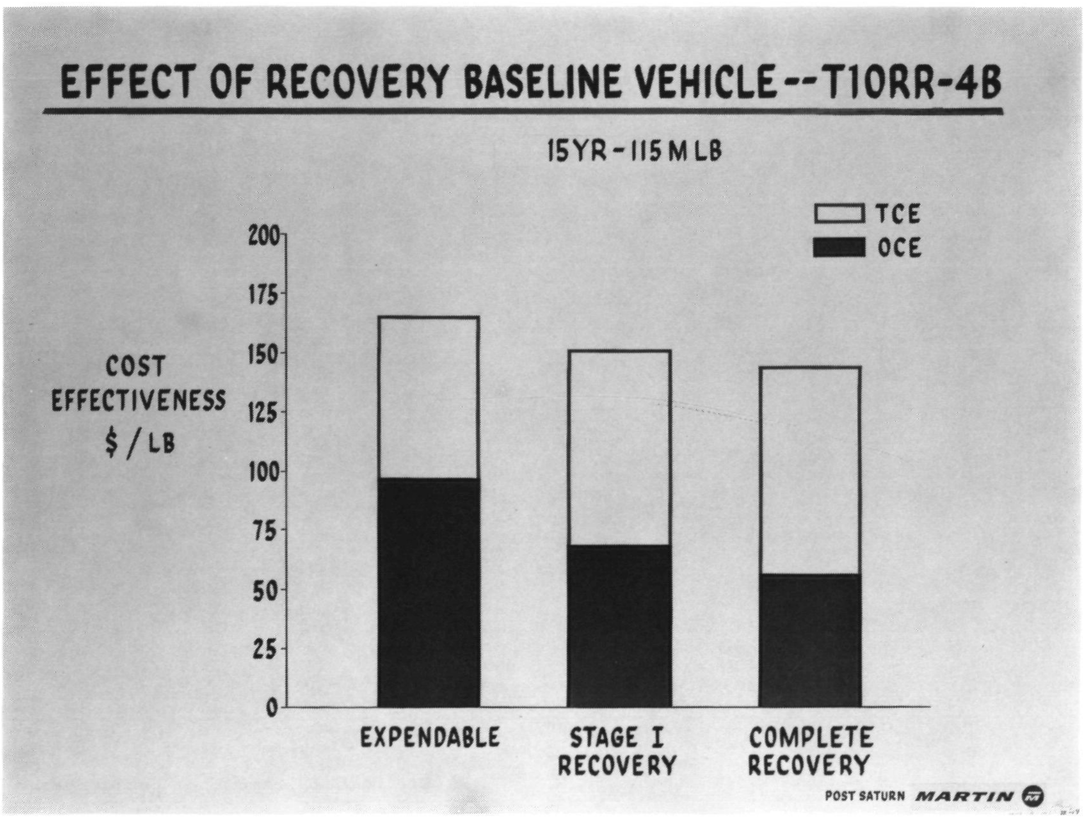
	RE (Recoverable N-I)	RR (Both Stages Rec)
Δ + TCE%	9	13
Δ + OCE%	29	42

Optimizing the stages for expendable versions improves N-I cost effectiveness \$7/lb; selling inventory improves RE \$4.35/lb and RR \$6/lb. The net result in % gains then is:

	RE	RR
Δ + TCE%	8	13
Δ + OCE%	31	46

Complete recovery and reuse of this vehicle is therefore recommended.


M-1/300 BASELINE VEHICLE--T1ORR-4B
432.8 FT
70 FT DIA
266.8 FT
74 FT DIA
99.6 FT DIA
• NUMBER OF ENGINES NI/NII 18/16
• PROPELLANT FRACTION NI/NII 0.863/0.890
• PAYLOAD MASS (LB)/(GRAMS) 1.069M/483G
• LAUNCH MASS (LB)/(GRAMS) 18.34M/8,320G
• LAUNCH/PAYLOAD 17.2
• RELIABILITY 0.92
• COST EFFECTIVENESS
OCE ($/LB)/($/Kg) 56.4/124
TCE ($/LB)/($/Kg) 144.9/320
POST SATURN MARTIN


EFFECT OF RECOVERY BASELINE VEHICLE--T1ORR-4B
15YR-115 M LB
TCE
OCE
COST EFFECTIVENESS $/LB
200
175
150
125
100
75
50
25
0
EXPENDABLE
STAGE I RECOVERY
COMPLETE RECOVERY
POST SATURN MARTIN

DESIGN STUDY SUMMARY

- 29 TECHNICAL & OPERATIONAL STUDIES
- MAJOR CONFIGURATION INFLUENCES
  - T/W 1.25 WITH ENGINE OUT
  - HINGED ENGINES - 3% INCREASED PAYLOAD
  - OPTIMIZED VELOCITY SPLIT
  - STAGE II TANK SELECTION
  - CONTROL ANALYSIS
  - RECOVERY & REUSE

POST SATURN MARTIN

REUSABLE LV STUDIES--RECOVERY DEVICES

STABILIZATION	DECELERATION	IMPACT ATTENUATION
REACTION CONTROL	RETROROCKETS	AIR MATS
PARACHUTE	ROTOR BLADES	STRUCTURAL SHAPE
BALLOON	PARACHUTE	LANDING LEGS
BALLUTE	BALLOON	CRUSHABLE STRUCTURE
CONICAL SKIRT	BALLUTE	
AERO FINS	DRAG CONE	
LV SHAPE		

POST SATURN MARTIN

NOSE LANDING CONCEPT

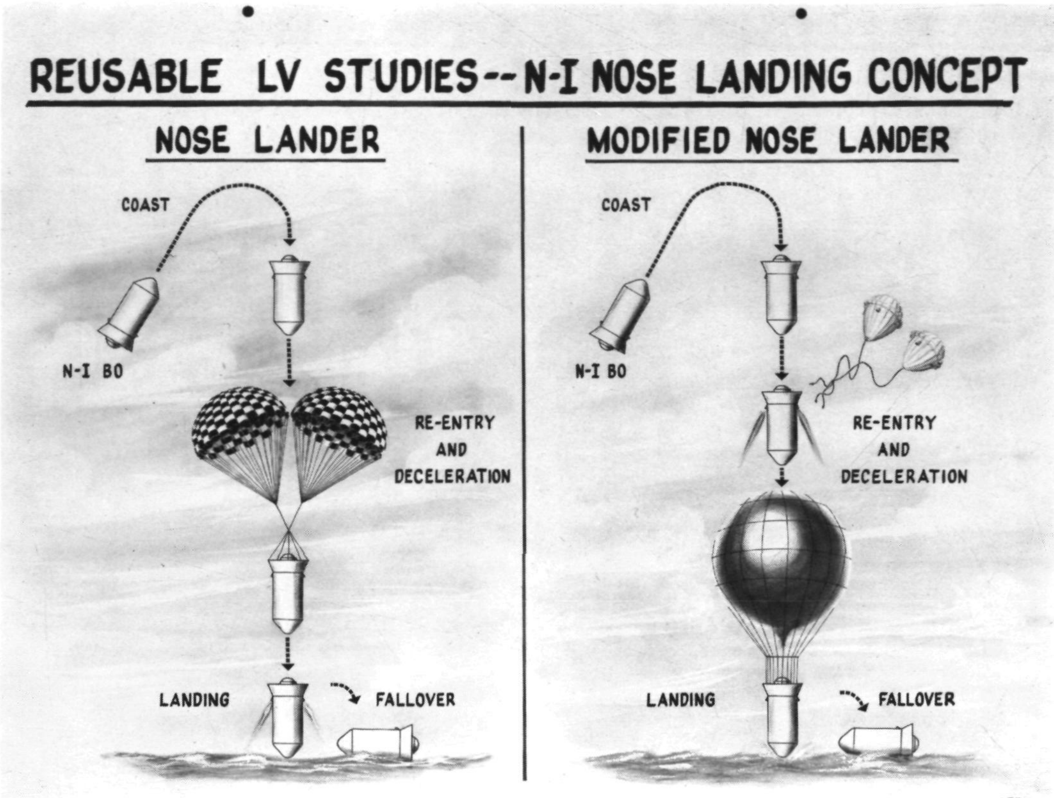
Two methods of nose landing were studied. One utilizes a series of parachutes; the last of which is released just prior to landing. Landing retrorockets further decelerate the stage to a theoretical zero velocity at touchdown; then, as the stage tips over, additional retrorockets are fired to attenuate loads when the side of the vehicle hits the water.

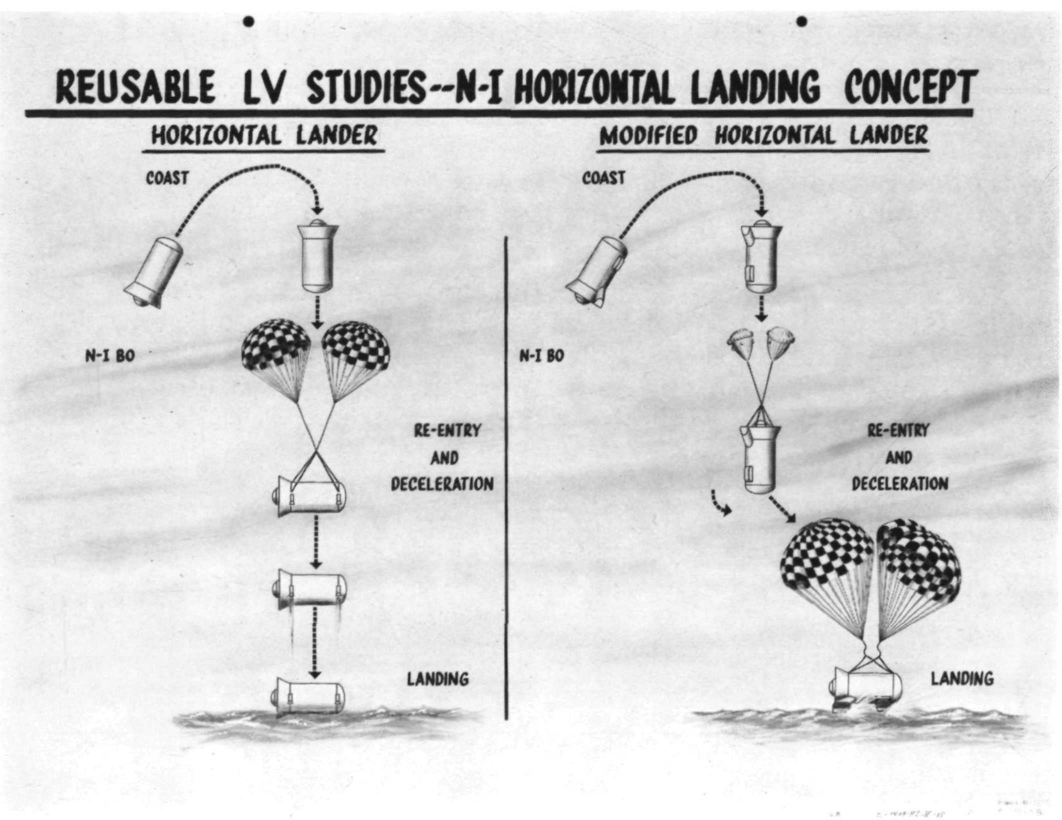
The modified nose lander utilizes an inflatable drag skirt, ballutes and a 540-ft diameter hot gas (250° F) balloon. Rate of descent is controlled by releasing gas from the balloon as it tips over to the horizontal position after nose impact.

REUSABLE LV STUDIES--N-I HORIZONTAL LANDING CONCEPT

Initial phases of recovery for Stage I are as previously described; however, the main parachutes maintain the stage in a horizontal position as it approaches the water. The parachutes are released and terminal landing is controlled by retrorockets at each end of the stage. The stage frames are reinforced to take the line contact landing loads.

The modified horizontal lander is similar to the horizontal except 6 ring sail parachutes, 200-ft in diameter, provide terminal deceleration to 55 fps vertical and 65 fps horizontal velocity components. Impact with the water is through 4 inflated "V" shaped landing legs without the aid of retrorockets. The external asymmetric shape of this version is a primary disadvantage.


REUSABLE LV STUDIES--N-I NOSE LANDING CONCEPT
NOSE LANDER
MODIFIED NOSE LANDER
COAST
N-I BO
RE-ENTRY
AND
DECELERATION
LANDING
FALLOVER
COAST
N-I BO
RE-ENTRY
AND
DECELERATION
LANDING
FALLOVER


REUSABLE LV STUDIES--N-I HORIZONTAL LANDING CONCEPT
HORIZONTAL LANDER
MODIFIED HORIZONTAL LANDER
COAST
N-I BO
RE-ENTRY
AND
DECELERATION
LANDING
COAST
N-I BO
RE-ENTRY
AND
DECELERATION
LANDING

STRUCTURAL SIZING MODEL--T10RE3--N-I STAGE

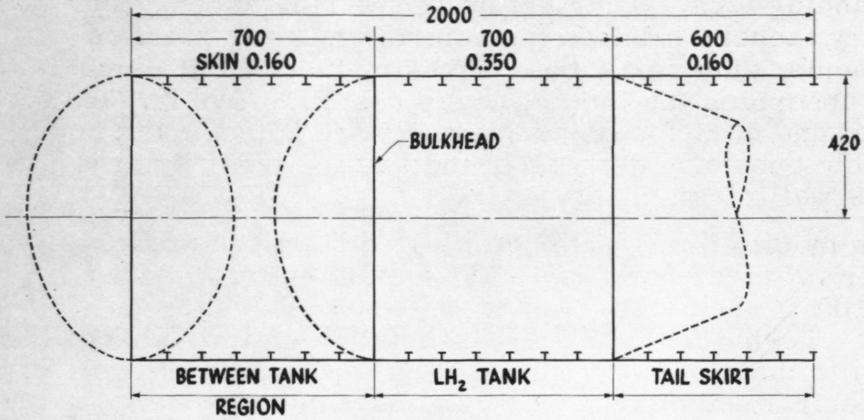
This figure shows the distribution of frame weight to meet flight design load requirements.

FRAME WEIGHT REQUIRED VS IMPACT VELOCITY--T10RE-3

These curves show that the existing tail skirt at 420 lb/ft is good for an impact velocity of 5.5 fps. The hydrogen tank area at 220 lb/ft frame weight is good for 7 fps impact velocity, and the between-tanks section is good for 11 fps.

An increase to 960 lb/ft for the between-bulkhead region and 1260 lb/ft for the tail skirt region is required to raise the landing velocity to 20 fps. This requires a total frame beefup of 140,000 lb for this configuration. This 20 fps velocity has been adapted as a design criterion for the horizontal mode.

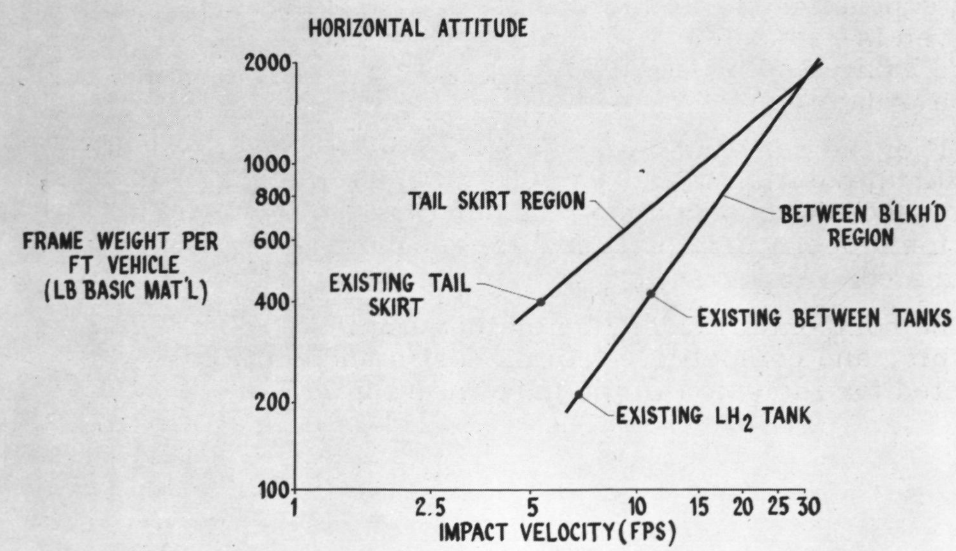
STRUCTURAL SIZING MODEL--T10RE-3 N-I STAGE



FRAMES	12 IN.^{2} @ 90 ($\bar{I}$ = 4.8 IN.^{3})	8 IN.^{2} @ 122 ($\bar{I}$ = 1.57 IN.^{3})	12 IN.^{2} @ 90 ($\bar{I}$ = 4.8 IN.^{3})
WT/FT (LB)	420	210	420
TOTAL/COMPARTMENT (LB)	24,500	12,250	21,000

- ESTIMATED FRAME WEIGHT--N-I STAGE
  58,000 X 1.2 = 70,000 LB
- TOTAL N-I STAGE WEIGHT = 1.043 X 10^6 LB

FRAME WEIGHT REQUIRED VS IMPACT VELOCITY--T10RE-3



POST SATURN MARTIN

REUSABLE LV STUDIES--TAIL LANDING CONCEPT, N-I

The tail sitter recovery version utilizes four dual purpose landing legs. They act as fins during ascent and re-entry, thereby minimizing control and drag skirt requirements; they serve their primary function at touchdown, absorbing the landing loads resulting from a 50-fps vertical and a 55-fps horizontal velocity component. Inflatable buoyancy devices in the landing legs float the engines well clear of the water.

The modified tail sitter employs a 600-ft diameter ram air inflated, hot (250° F) gas balloon to float the stage 100 ft above the surface until retrieval forces arrive. Technical problems associated with the energy system to maintain the 250° F hot gas and control system requirements are factors against this approach.

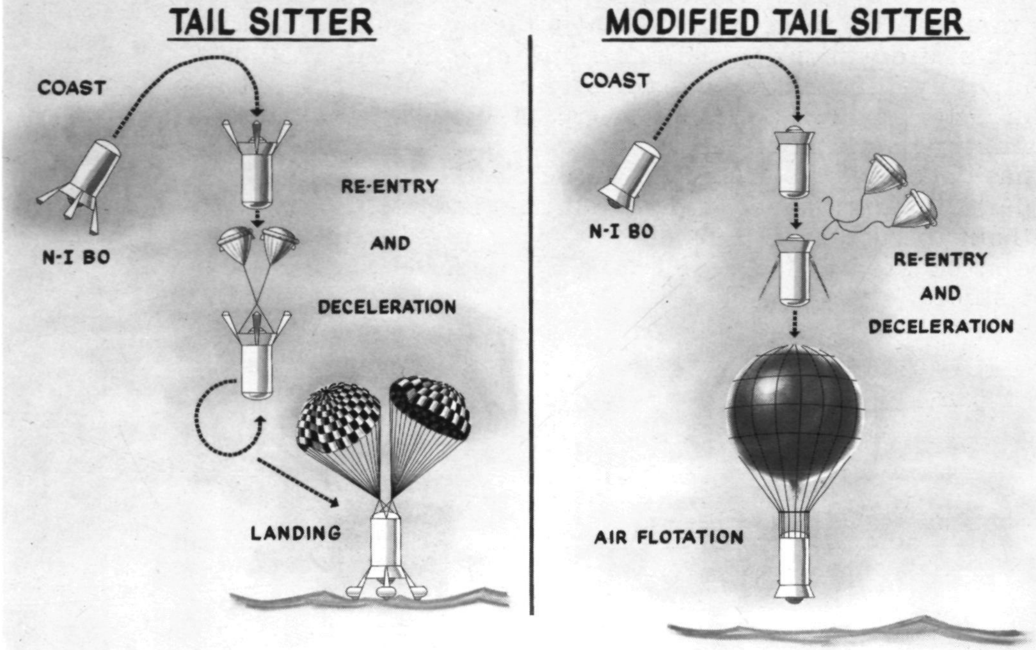
REUSABLE LV STUDIES--RECOVERY CONCEPT COMPARISON

A small TCE spread over the several landing concepts is apparent. The heaviest tail sitter is the least expensive and this is due to the fact that the weight involved is reusable. It, however, cannot be selected as the basic landing mode without further detailed analysis on dynamic stability and system weight.

All of the remaining 3 versions come to rest in a horizontal position in the water, and since the nose lander and the modified nose lander must go through an additional operational sequence to get there, they are eliminated.

The horizontal lander, being the simplest, most reliable, and competitive from a cost standpoint, is selected for inclusion in the baseline vehicle.

REUSABLE LV STUDIES -- N-I TAIL LANDING CONCEPT



REUSABLE LV STUDIES--RECOVERY CONCEPT COMPARISON

CONCEPT	WEIGHT		Δ TOTAL COST-EFF	
	LB	RANK	$/LB	RANK
NOSE LANDER	414,360	(1)	3.5	(3)
MODIFIED NOSE LANDER	417,241	(2)	6.5	(4)
HORIZONTAL LANDER	422,940	(3)	3.3	(2)
TAIL SITTER	464,506	(4)	0	(1)

REUSABLE LV STUDIES--N-II RECOVERY CONCEPT

An active reaction attitude control system orients the stage for the deorbit maneuver. Impulse for the maneuver is provided by main stage engines which utilize propellants from a special tank. Reorientation for re-entry into the earth's atmosphere is then accomplished by the attitude control system. The vehicle is statically stable supersonically. This stability is augmented by three 20-ft diameter parachutes at low supersonic velocities. Final deceleration is accomplished by the use of three 210-ft diameter parachutes. Landing is accomplished without the use of retrorockets. Water impact loads are attenuated with an air bag stowed behind the re-entry hot shield.

ADDITIONAL RECOVERY ASPECTS

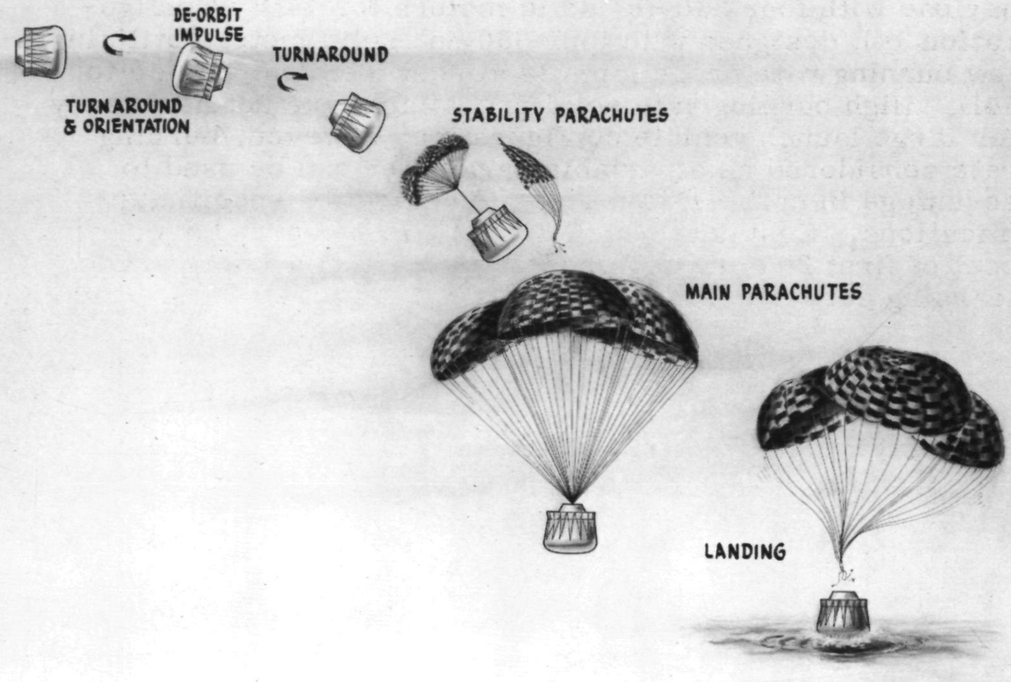
Studies to establish feasibility and stage inventory requirements were conducted. Cycle time was found to be 23 weeks for stages going through periodic refurbishment and 60 weeks for major refurbishment.

Assessment of the recovered stage will begin on the retrieval vessel following dearming of pyrotechnics and purging of tanks. Eddy current techniques for determination of exposure to excessive temperature are being investigated.

Re-entry control analysis for N-I of T10RE-3A utilizing an attitude/altitude rate shows that the vehicle is stable at high supersonic velocities, but at low supersonic velocities augmented control is required to damp oscillations of 1.46 rad/sec.

Retrieval of spent stages by towed barge, self-propelled barges, and self-propelled Catamarans (constructed by bridging 2 destroyer escort vessels) has been investigated as well as the Materials and Manufacturing processes required for the refurbishment of expendable items. Of particular interest is the time required to strip 1.4 acres of ablator and insulation from Stage I, clean the surface, and bond the new ablator. The use of fiber glass cloth as a rip strip between the ablator and structure shows promise.

REUSABLE LV STUDIES--N-II RECOVERY CONCEPT



ADDITIONAL RECOVERY ASPECTS

- LOGISTICS & RECYCLE TIME
- METHODS OF VEHICLE ASSESSMENT
- RE-ENTRY CONTROL ANALYSIS
- MATERIALS RESEARCH
- RETRIEVAL

POST SATURN MARTIN

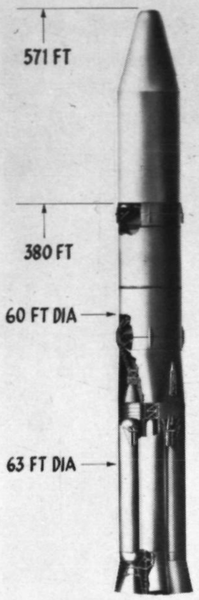
CONFIGURATIONS 14A AND 14D CHARACTERISTICS

Height difference due to basic 60-ft diameter payload and four 300-in. solid motors for 14A; versus 70-ft diameter payload with four 360-in. solid motors for 14D. Configuration 14E designed with four 360-in. solid motors utilizing low burning rate propellants is almost identical in size to 14D. High burning rate solid propellants are not necessary for large launch vehicle configuration; however, burning rate considered as a variable parameter can be used to advantage in optimization of grain design for specific applications. Cost data based upon \$1.75 per pound average cost of first 24 solid motors with 95% cumulative average learning curve.

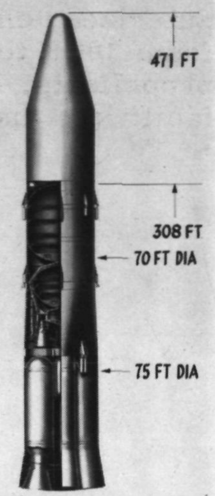
LIQUID ROCKET ENGINE RELIABILITY

The engine reliability test data represents an average of opinion among the engine contractors. It is based in part on past engine development programs, and in the area of most interest, between 0.980 and 0.995, agreement among contractors was fairly good. The purpose is to provide a consistent basis for evaluating various propulsion systems in launch vehicles and to relate development program cost to engine module reliability.

SOLID PROPELLANT CONFIGURATIONS - 14A & 14D

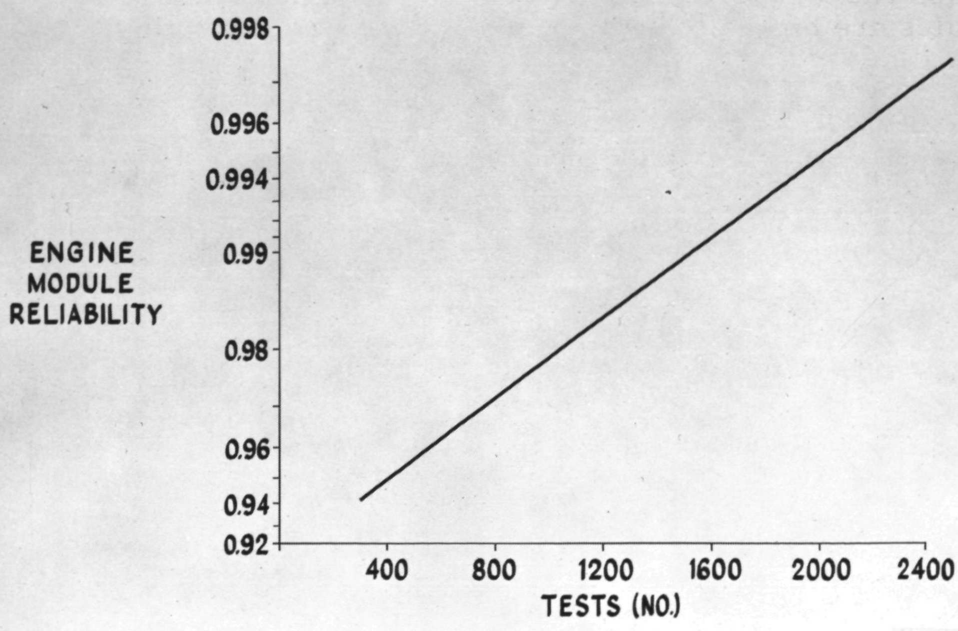


T10EE-14A N-I	T10EE-14A N-II		T10EE-14D N-I	T10EE-14D N-II
4	5	NUMBER OF ENGINES	4	5
0.878	0.903	PROPELLANT FRACTION	0.892	0.895
1.058M		PAYLOAD (LB)	1.094M	
33.67M		LAUNCH WEIGHT (LB)	33.67M	
31.8		LAUNCH WEIGHT/ PAYLOAD WEIGHT	30.8	
49.2M		TOTAL SL THRUST N-I (LB)	49.2M	
7.5M		TOTAL VAC THRUST N-II (LB)	7.5M	
0.92		RELIABILITY 225 KM	0.92	
0.57		SOLID BURNING RATE IN./SEC	1.03	
		COST EFFECTIVENESS		
98 EST		OCE ($/LB)	96	
159 EST		TCE ($/LB)	156	



POST SATURN MARTIN

LIQUID ROCKET ENGINE RELIABILITY



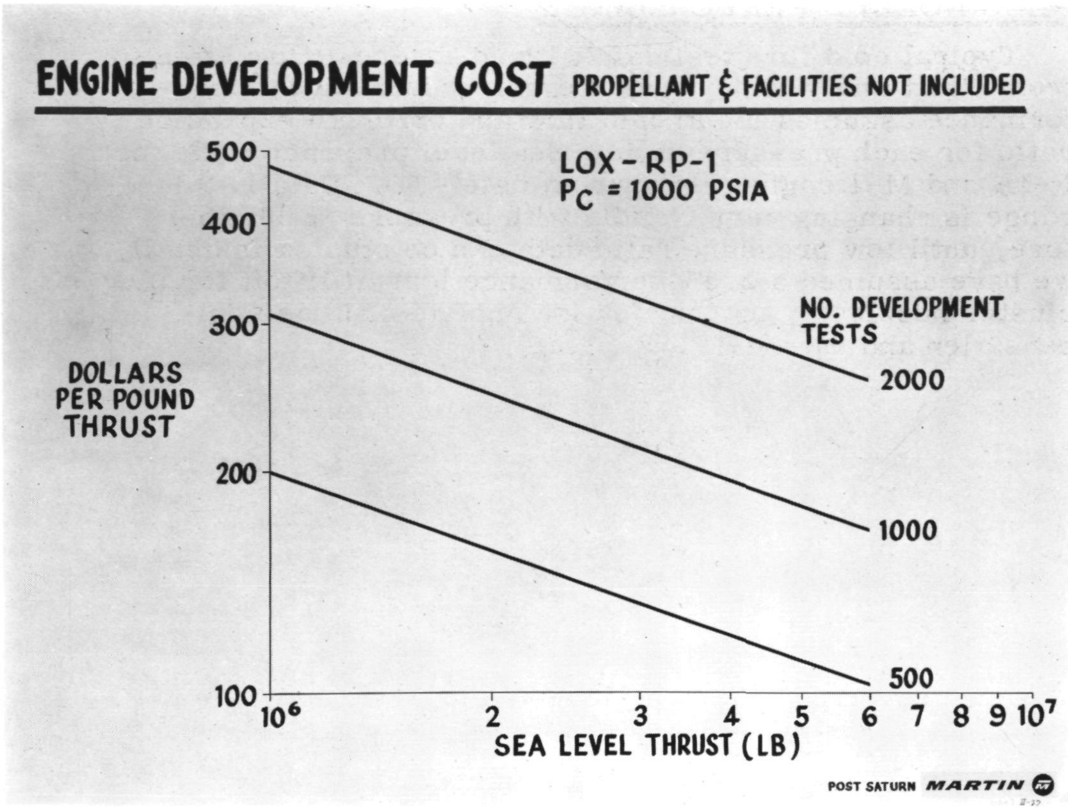
POST SATURN MARTIN

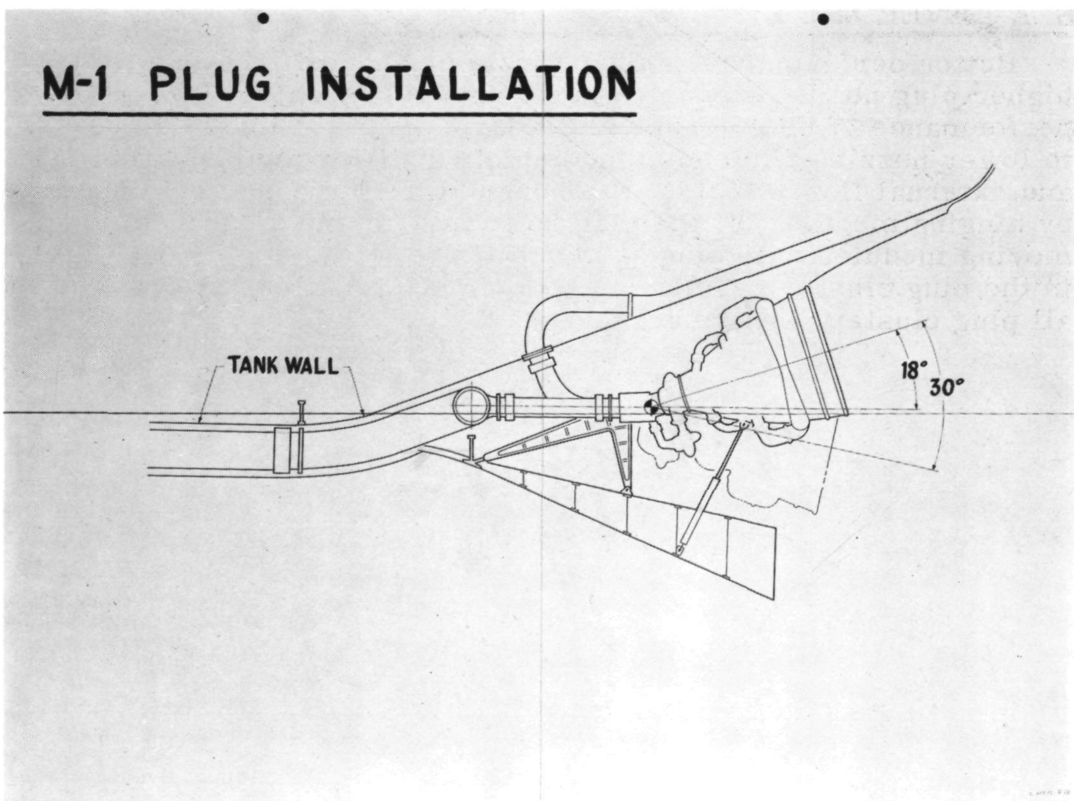
ENGINE DEVELOPMENT COSTS

Data represents a high-side average of engine contractor supplied information, and is expressed in 1963 dollars. Does not include propellants or facilities. Allowance for program growth and contract redefinition is included. Sea level thrust is with fully expanded nozzle. Straight line plots can be expressed in simple equation form for computerized vehicle evaluation model. Cost data available from 300 K to 6000 K lb thrust level for engine development, propellants, engine test facilities, engine manufacturing facilities, and production engines.

M-1 PLUG INSTALLATION

Typical plug cluster installation with 18 engine modules arranged in an annular ring around a 10% plug. Module expansion ratio is 12.5 and plug cluster expansion ratio is 38. Modules are hinged to swing radially away from the plug.


ENGINE DEVELOPMENT COST PROPELLANT & FACILITIES NOT INCLUDED
LOX - RP-1
P_C = 1000 PSIA
NO. DEVELOPMENT TESTS
2000
1000
500
DOLLARS PER POUND THRUST
500
400
300
200
100
10^6
2
3
4
5
6
7
8
9
10^7
SEA LEVEL THRUST (LB)
POST SATURN MARTIN

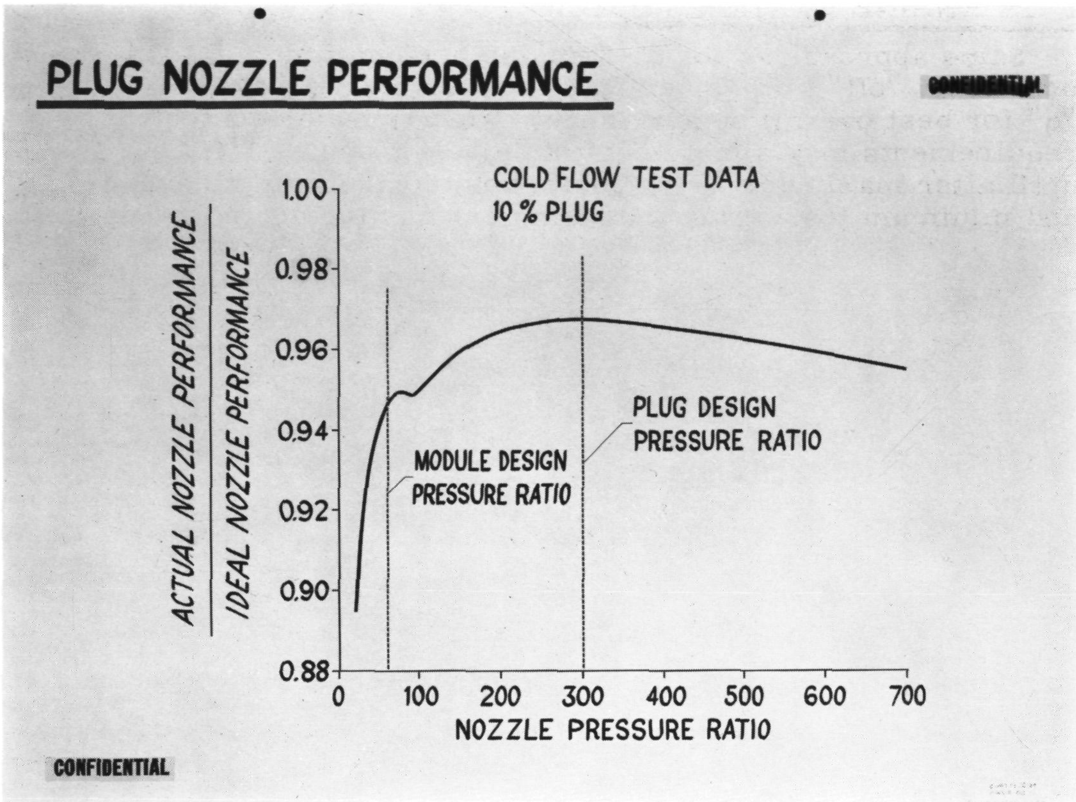

M-1 PLUG INSTALLATION
TANK WALL
18°
30°

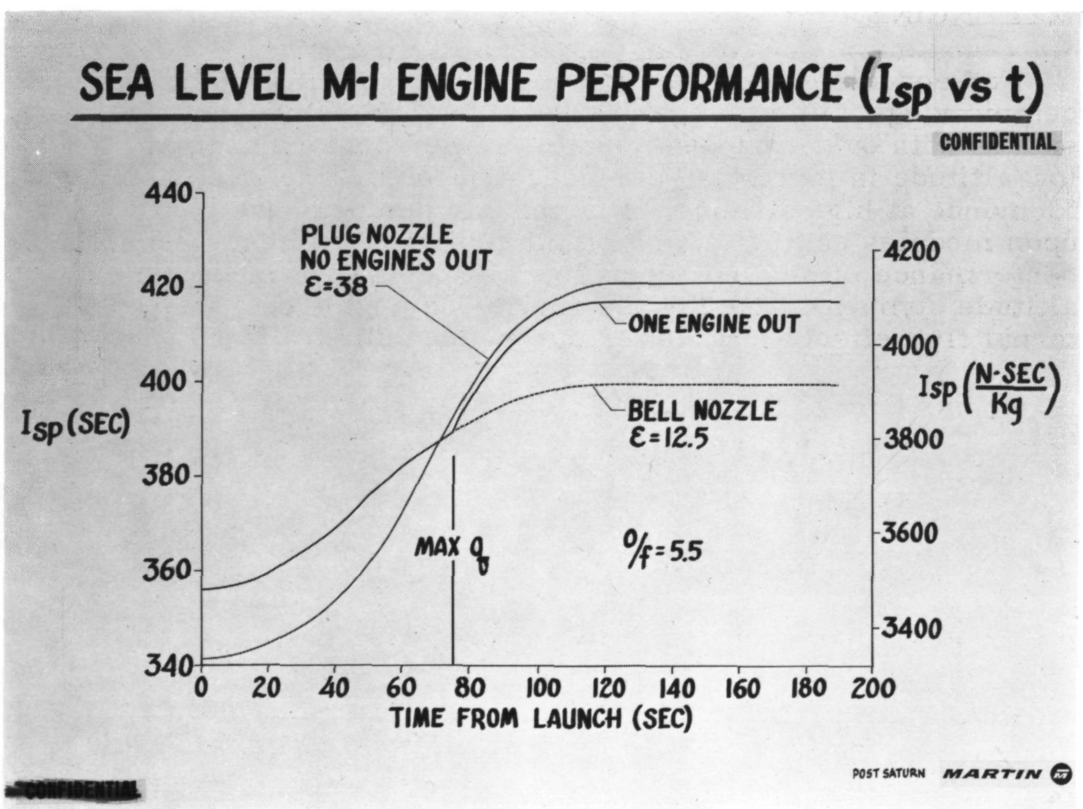
PLUG NOZZLE PERFORMANCE

Typical cold flow test data without external flow effects from Pratt and Whitney test program. Ideal nozzle performance assumes isentropic flow and optimum expansion ratio for each pressure ratio. Sea level pressure ratio for F-1A and M-1 engines is approximately 70. Data in this range is changing very rapidly with pressure ratio; therefore, until low pressure ratio data can be studied in detail, we have assumed a 2.5% performance loss at liftoff for plug cluster nozzles to account for incomplete altitude compensation and external flow effects.

SEA LEVEL M-1 ENGINE PERFORMANCE

Better performance of plug nozzle at high altitude due to higher plug nozzle expansion ratio than bell nozzle. Lower performance of plug versus bell nozzle at low altitude due to lower nozzle efficiency, incomplete altitude compensation, and external flow effects. Best overall performance obtained by hinging modules off the plug up to maximum "q" and moving modules onto plug at high altitude. One engine out in the plug cluster results in approximately 1% loss in overall plug cluster performance.


PLUG NOZZLE PERFORMANCE
COLD FLOW TEST DATA
10 % PLUG
ACTUAL NOZZLE PERFORMANCE / IDEAL NOZZLE PERFORMANCE
1.00
0.98
0.96
0.94
0.92
0.90
0.88
0
100
200
300
400
500
600
700
NOZZLE PRESSURE RATIO
MODULE DESIGN PRESSURE RATIO
PLUG DESIGN PRESSURE RATIO
CONFIDENTIAL

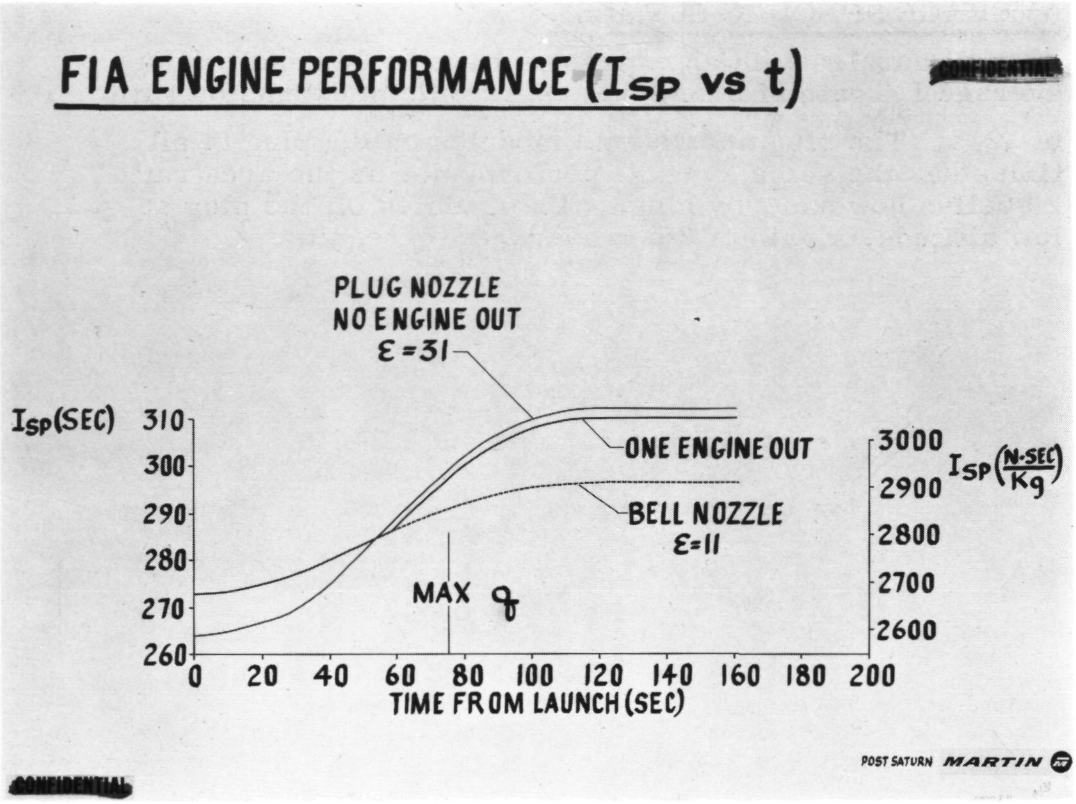

SEA LEVEL M-1 ENGINE PERFORMANCE (I_{sp} vs t)
CONFIDENTIAL
440
420
400
380
360
340
I_{sp} (SEC)
4200
4000
3800
3600
3400
I_{sp} $\left(\frac{N\text{-}SEC}{Kg}\right)$
PLUG NOZZLE
NO ENGINES OUT
ε=38
ONE ENGINE OUT
BELL NOZZLE
ε=12.5
MAX q
O/f = 5.5
0
20
40
60
80
100
120
140
160
180
200
TIME FROM LAUNCH (SEC)
POST SATURN
MARTIN

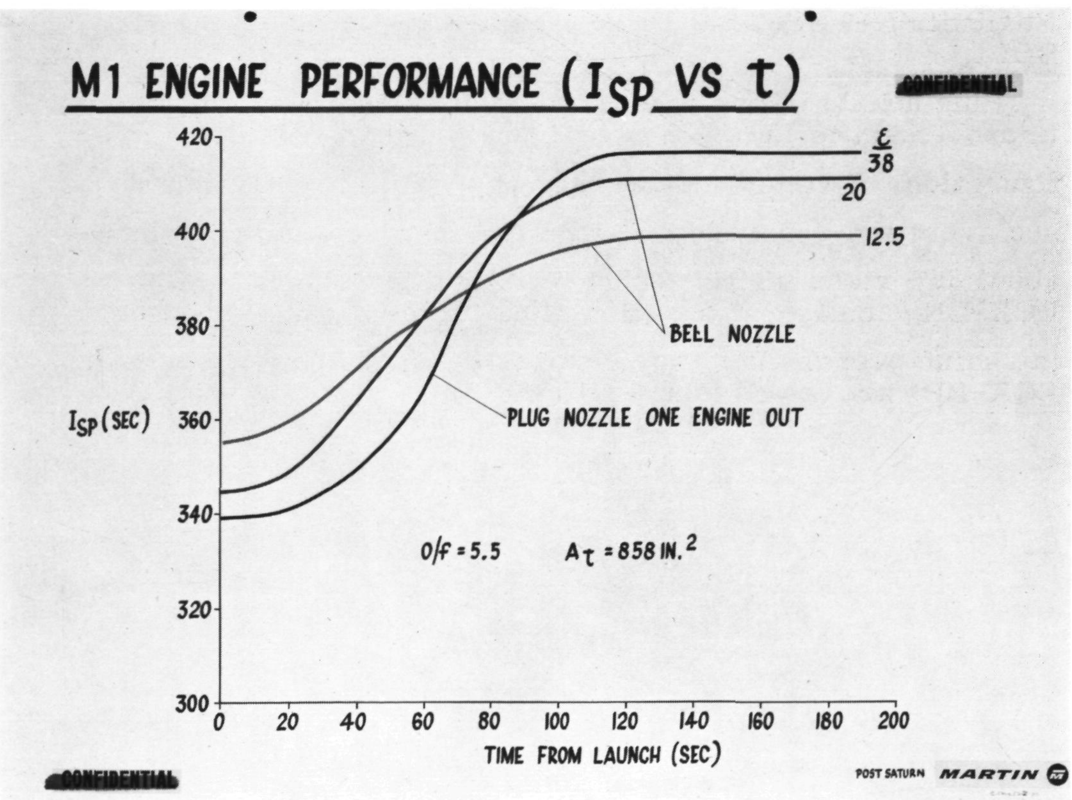
F-1A ENGINE PERFORMANCE

Same approach as for M-1 engine performance. Cross-over from "off" plug to "on" plug occurs before maximum "q" for best overall performance. Thrust vector control requirements may dictate positioning the modules "off" plug until after maximum "q" to obtain best control effectiveness and minimum losses due to hinging engines for control.

M-1 ENGINE PERFORMANCE (I_{sp}) VERSUS t

For conventional engine module arrangement, two concentric rings with bell nozzles, best overall performance occurs with overexpanded nozzle. Lower performance at low altitude is more than compensated for by improved performance at high altitude. Plug nozzle performance based upon modules canted inward on the plug at all times. Poor performance of plug nozzle at low altitude due to incomplete altitude compensation, losses due to one engine out, external flow effects, and lower overall nozzle efficiency.


FIA ENGINE PERFORMANCE (I_{SP} vs t)
CONFIDENTIAL
PLUG NOZZLE
NO ENGINE OUT
ε = 31
ONE ENGINE OUT
BELL NOZZLE
ε = 11
MAX q
I_{SP}(SEC)
310
300
290
280
270
260
3000
2900
2800
2700
2600
I_{SP} ($\frac{N \cdot SEC}{Kg}$)
0 20 40 60 80 100 120 140 160 180 200
TIME FROM LAUNCH (SEC)
POST SATURN MARTIN
CONFIDENTIAL


M1 ENGINE PERFORMANCE (I_{SP} VS t)
CONFIDENTIAL
ε
38
20
12.5
BELL NOZZLE
PLUG NOZZLE ONE ENGINE OUT
I_{SP} (SEC)
420
400
380
360
340
320
300
O/F = 5.5
A_t = 858 IN.^{2}
0 20 40 60 80 100 120 140 160 180 200
TIME FROM LAUNCH (SEC)
CONFIDENTIAL
POST SATURN MARTIN

AVERAGE SPECIFIC IMPULSE

Bell nozzles with an expansion ratio of 20 result in an average I_{sp} gain of 2 sec over bells with an expansion ratio of 12.5. The plug nozzle with modules on the plug at all times has the same average performance as the area ratio 20 bells; however, by hinging the modules off the plug at low altitude, a gain of 8 sec average I_{sp} results.

PROPULSION SYSTEM PERFORMANCE--PLUG VERSUS BELL

Plug nozzle offers less propulsion system performance improvement to LOX-RP1 configuration than LOX-LH_2 configuration. Two seconds average I_{sp} gain for F-1A and 8 sec average I_{sp} gain for M-1. LOX-RP1 configuration has about 35% more installed thrust in the first stage than the LOX-LH_2 configuration. Since the two configurations have the same overall diameter, expansion ratios obtainable with LOX-RP1 are lower than with LOX-LH_2.

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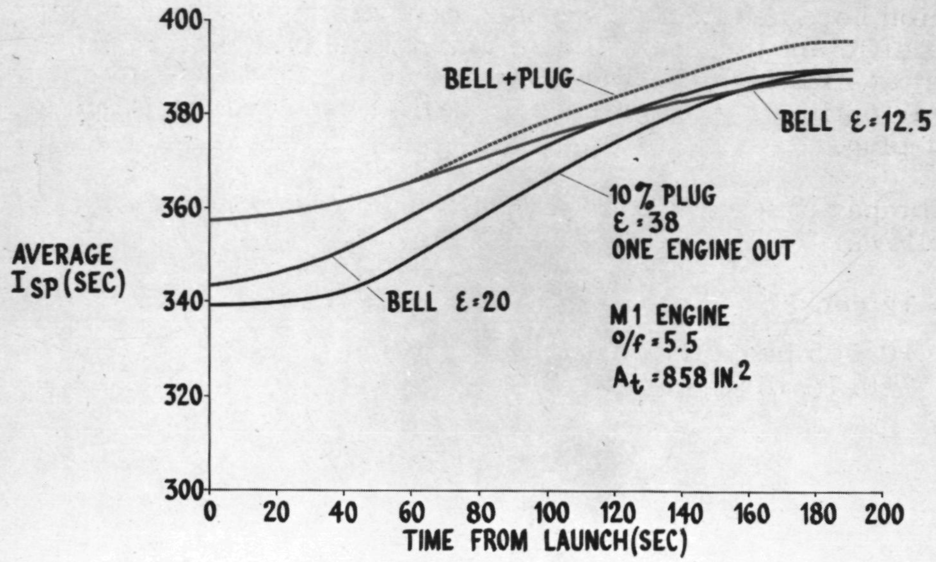
AVERAGE SPECIFIC IMPULSE

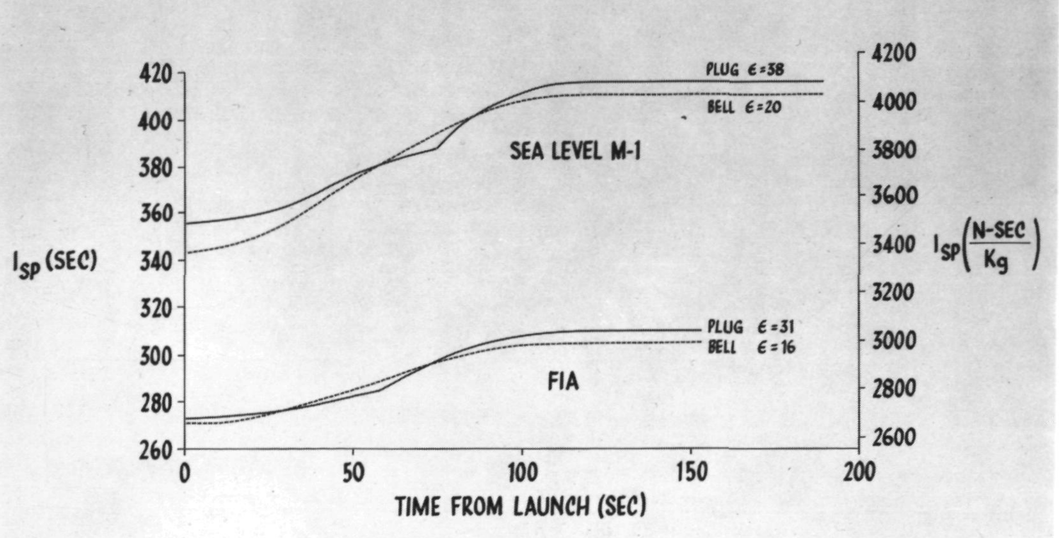


CONFIDENTIAL

POST SATURN MARTIN

PROPULSION SYSTEM PERFORMANCE - PLUG VS BELL

CONFIDENTIAL



POST SATURN MARTIN

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PAYLOAD AND COST DUE TO PLUG--STAGE I

Plug nozzle offers improvement in: Higher average I_{sp} due to improved high altitude performance at high expansion ratio without excessive low altitude losses; higher propellant fraction because annular arrangement of engines results in more efficient thrust structure and heat shield design; higher thrust at liftoff which allows larger vehicle and payload due to near optimum expansion ratio bell at sea level by hinging off of plug.

Comparative values for LOX-RP1 configuration are as follows:

+2 sec I_{sp}	+0.8%	PL
+0.005 propellant fraction	+3.5%	
+0.7% liftoff thrust	+0.7%	PL

EQUIVALENT GIMBAL ANGLE

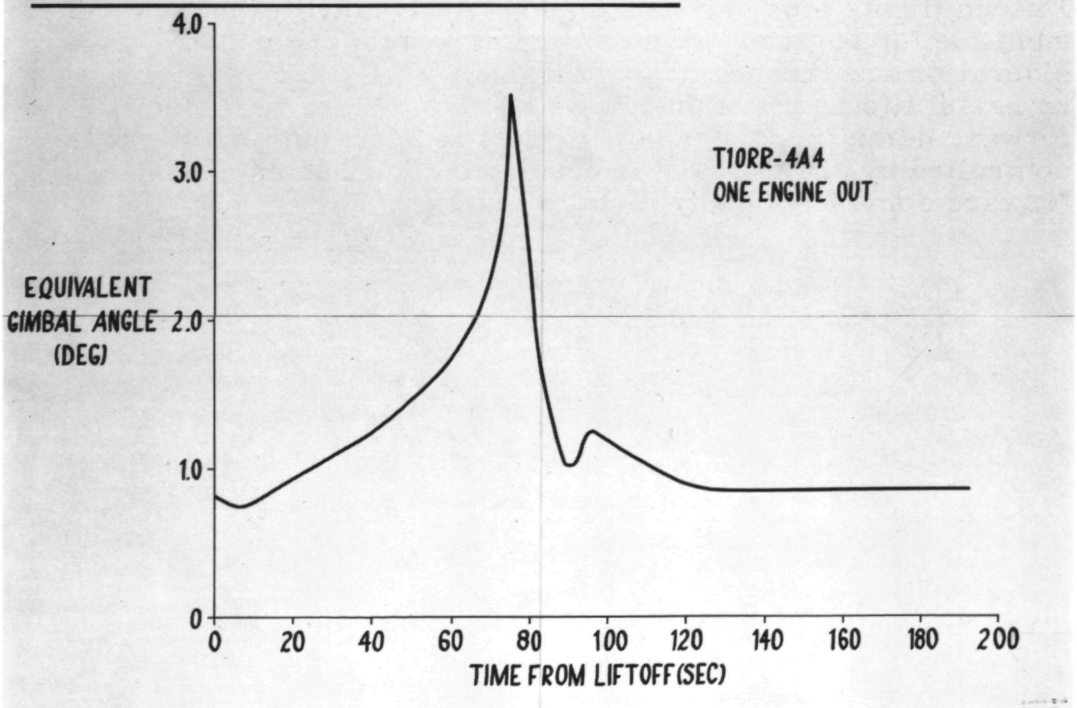
Moment due to one engine out and 0.5° thrust misalignment included. Maximum jet thrust stream deflection of 3.5° occurs at maximum "q".

PAYLOAD & COST DUE TO PLUG--SEA LEVEL M-1

	Δ PAYLOAD (%)	Δ COST	
		TCE (%)	OCE (%)
+8 SECONDS I_{sp}	+2.9	-2.9	-4.2
+0.005 PROPELLANT FRACTION	+2.7	-2.4	-3.1
+3.4% LIFTOFF THRUST	+3.4	0	0
TOTAL	+9.0	-5.3	-7.3

POST SATURN MARTIN

EQUIVALENT GIMBAL ANGLE



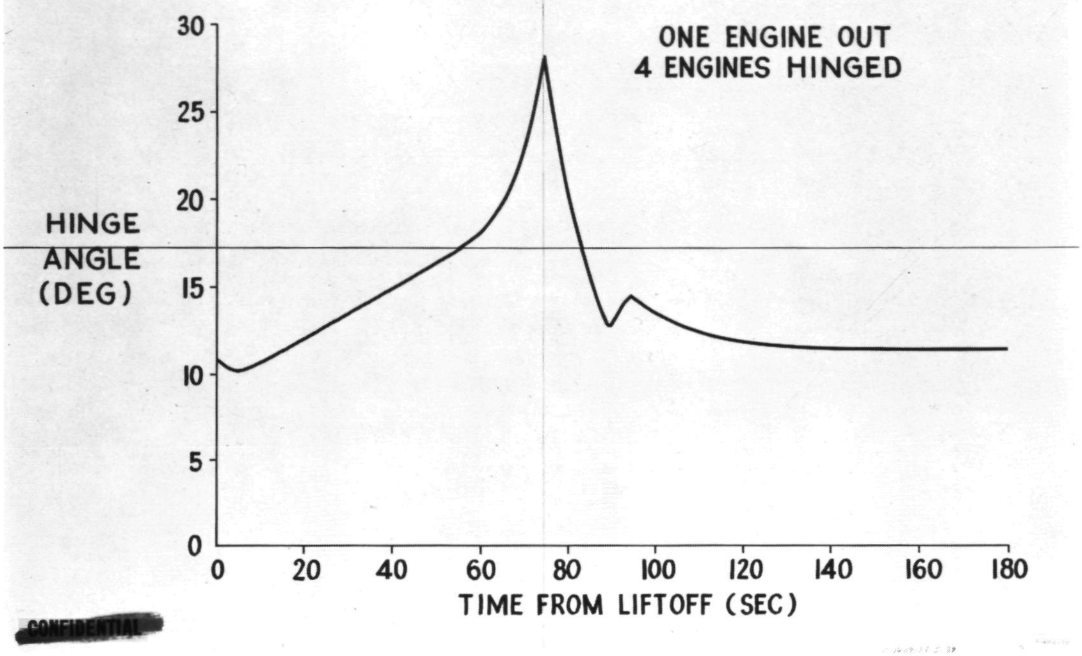
HINGE ANGLE--T10RR-4A4

Plug cluster nozzle installation with 18 sea level M-1 engine modules. Hinge four engines in each quadrant radially away from plug. Required hinge angle computed for duty cycle on previous chart. Since proximity to plug reduces hinging effectiveness, more detailed study may show that modules should be hinged off plug until after max "q".

REUSABLE ENGINE REQUIREMENTS

Approximately 125 launches required for 50/4 model. Fifteen flights plus static firings give maximum expected run time for engines. Some mission models under consideration are three times as large as 50/4; therefore, 10 hr useful life is a reasonable goal. Both stages re-enter forward dome first. Stage I impacts in horizontal position controlled by chutes and retrorockets. Stage II impacts forward dome first controlled by chutes.

HINGE ANGLE-- TIORR-4A4



REUSABLE ENGINE REQUIREMENTS

	N-I	STAGE	N-II
50/4 MISSION MODEL 15 FLIGHTS PER VEHICLE	2.5 70	HOURS STARTS	3.5 100
BALLISTIC RE-ENTRY	15 3 300	AXIAL g LATERAL g ENGINE TEMPERATURE (°F)	9 2 ~
WATER IMPACT	HORIZONTAL 15 5g WET	ATTITUDE VELOCITY (FPS) LOAD FACTOR ENVIRONMENT	ENGINES UP 30 3.5g DRY

POST SATURN MARTIN

III. LAUNCH VEHICLE COST EVALUATION

COST MODEL--CATEGORIES

The cost evaluation model is programmed on the 7094 digital computer and covers all of the launch vehicle costs-- indirect as well as operational. Denoted are the number of separate subcategories into which each of the headings is divided. There are 164 such categories; each is costed separately.

COST MODEL--TYPICAL INPUTS

Typical inputs to the program include the payload weight delivery requirements by year, the type and weight of materials in each stage, number and type of engines per stage, engine thrust, stage ascent and recovery reliability, stage recycle time, type and quantity of propellants and other related information. In addition, coefficients, constants and exponents are input for each of the 164 costing equations in the model. In all there are approximately 2000 separate external inputs per run.

COST MODEL--CATEGORIES

INDIRECT COSTS		82
• DEVELOPMENT	(45)	
• FACILITIES	(17)	
• TOOLING AND GSE	(20)	
OPERATIONAL COSTS		82
• PROCUREMENT	(34)	
• REFURBISH AND REUSE	(40)	
• LAUNCH OPERATIONS	(8)	

COST MODEL -- TYPICAL INPUTS

- PAYLOAD TO BE DELIVERED -- BY YEAR
- VEHICLE PAYLOAD CAPABILITY
- RELIABILITIES
  - LAUNCH
  - RECOVERY
- NUMBER OF REFURBISHMENT CAPTIVE FIRINGS
- STAGE RECYCLE TIME
- DEMONSTRATION PROGRAM
  - FLIGHTS
  - REUSES
- STAGE WEIGHT
  - RAW, PURCHASED, ASTRIONICS, HEAT SHIELD — EXPENDED & REUSED
  - OXIDIZER
  - FUEL
- VEHICLE
  - VOLUME
  - HEIGHT
- TYPE & NUMBER OF ENGINES PER STAGE
- THRUST PER ENGINE
- COEFFICIENTS, CONSTANTS & EXPONENTS FOR 164 COST EQUATIONS

TOTAL OF ~2000 SEPARATE INPUTS

POST SATURN MARTIN

COST MODEL--TYPICAL OUTPUTS

The output includes a tabulation of indirect costs (82), a tabulation by year of the cumulative operational costs (82), the yearly and cumulative operational and total program cost effectiveness and a variety of mission and vehicle parameters, such as the number of launches, the number of stage reuses, the number of new stage procurements and the number of stages in inventory at the beginning of each year in the operational program.

COMPARISON MATRIX

This study was an examination of issues rather than a free for all evaluation of launch vehicles. For example, should the post Saturn launch vehicle be powered by high pressure engines in both stages (HP/HP), by low pressure engines in both stages (M-1/M-1, solid/M-1 or F-1A/M-1), or by high pressure engines in the second stage only (M-1/300K and F-1A/300K)? Furthermore, should the vehicle be completely expendable (EE), partially recoverable (RE), or completely recoverable (RR)? These and corollary issues were raised; in each case a "standard-bearer" configuration was evolved as indicated on the chart. These configurations were then evaluated and compared. The mission used for the initial comparisons involved the delivery of approximately 115,000,000 lb of payload to low earth orbit over a period of 15 years.

COST MODEL -- TYPICAL OUTPUTS

- COST OF EACH OF THE 82 INDIRECT ITEMS
- CUMULATIVE COST OF EACH OF THE 82 OPERATIONAL ITEMS -- BY YEAR
- NUMBER OF STAGE PROCUREMENTS
- NUMBER OF STAGE REUSES
- NUMBER OF LAUNCHES

  YEARLY AND CUMULATIVE (for the three items above)

- STAGE AVERAGE RECYCLE TIME -- BY YEAR
- CUMULATIVE WEIGHT OF PAYLOAD DELIVERED TO ORBIT BY YEAR
- TOTAL INDIRECT COST
- OPERATIONAL COST
- OPERATIONAL COST EFFECTIVENESS

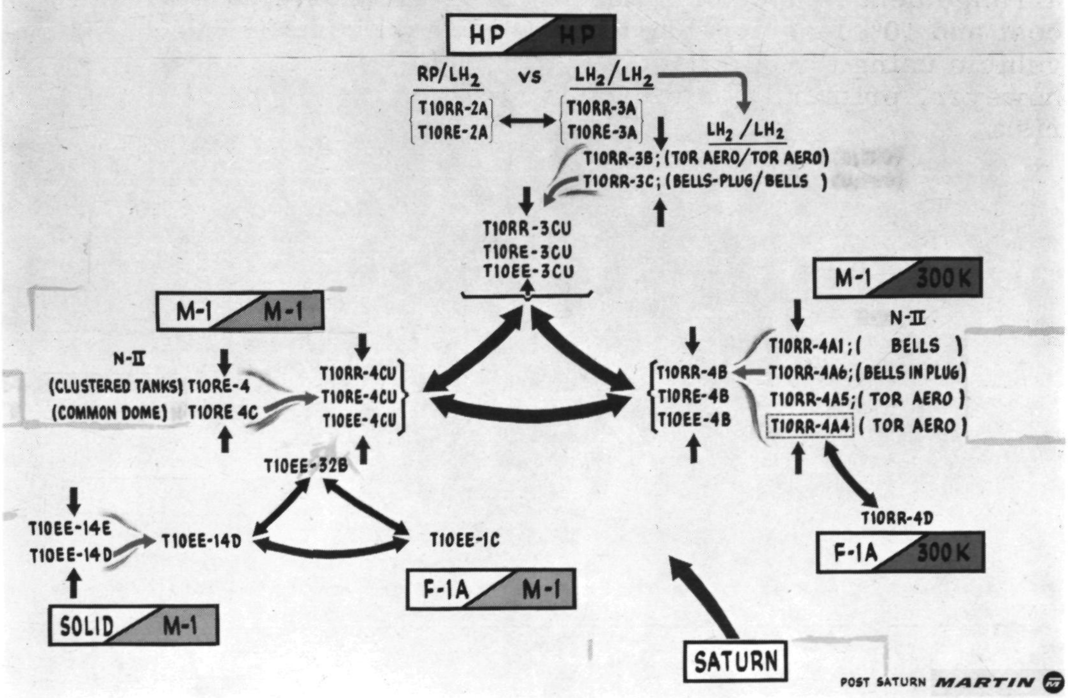
  YEARLY AND CUMULATIVE (for the two items above)

- TOTAL COST
- TOTAL COST EFFECTIVENESS

  CUMULATIVE (for the two items above)

POST SATURN MARTIN

COMPARISON MATRIX



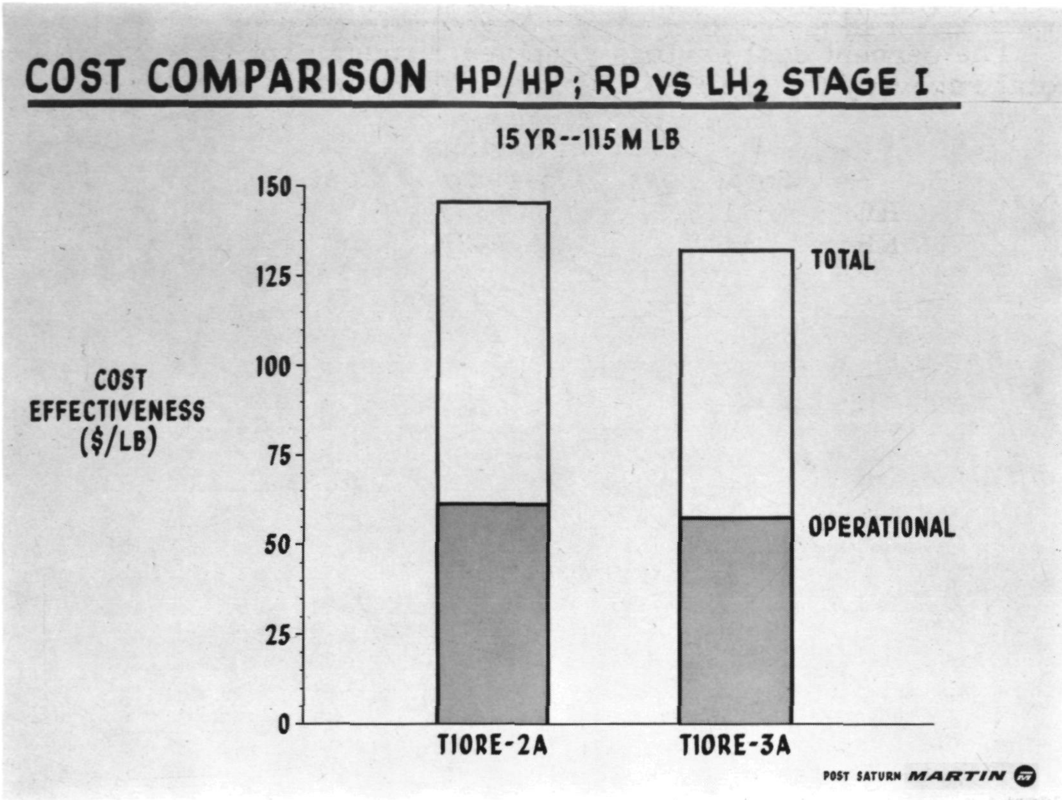
POST SATURN MARTIN

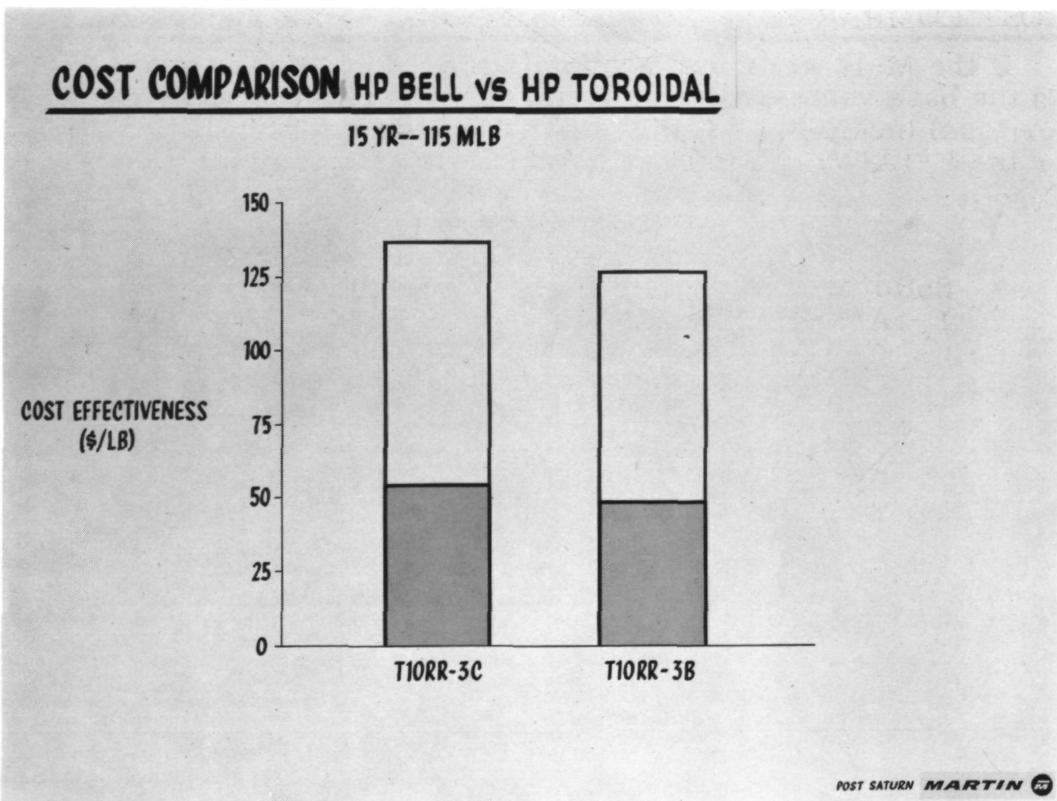
COST COMPARISON--HP/HP; RP versus LH_2 Stage I

The use of hydrogen instead of RP in the first stage results in a savings of approximately 9% in total costs and 6% in operational costs. The saving in total cost is attributable primarily to the fact that only one engine design and development program need be undertaken in the case of the LH_2/LH_2 vehicle.

COST COMPARISON--HP Bell versus HP Toroidal

The vehicle using the toroidal aerospike engine nozzle arrangement is approximately 8% less expensive in total cost and 10% less expensive in operational costs. The vehicle using the bell nozzles (T10RR-36) was chosen, however, primarily on the basis of lower development risk.


COST COMPARISON HP/HP; RP vs LH2 STAGE I
15 YR--115 M LB
150
125
100
75
50
25
0
COST
EFFECTIVENESS
($/LB)
TOTAL
OPERATIONAL
T10RE-2A
T10RE-3A
POST SATURN MARTIN


COST COMPARISON HP BELL vs HP TOROIDAL
15 YR--115 MLB
150
125
100
75
50
25
0
COST EFFECTIVENESS
($/LB)
T10RR-3C
T10RR-3B
POST SATURN MARTIN

COST COMPARISON--HP/HP; Degrees of Recovery

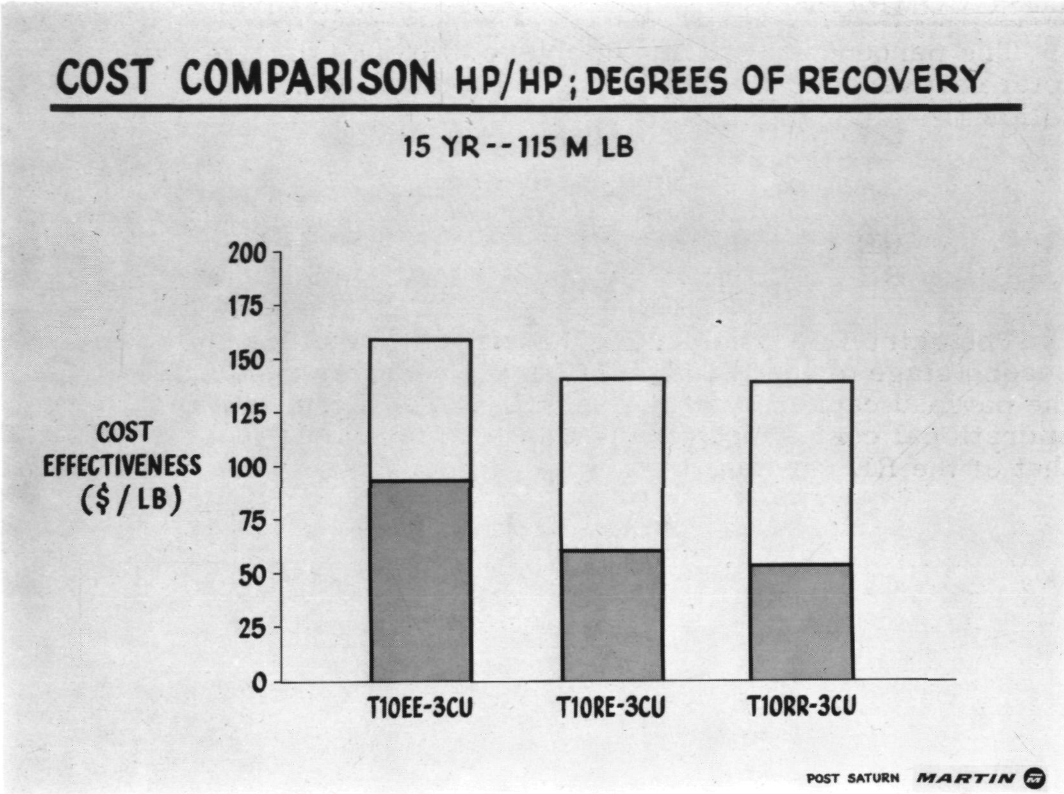
The percent cost savings achieved through partial and total recovery for the HP/HP class of vehicles is as follows:

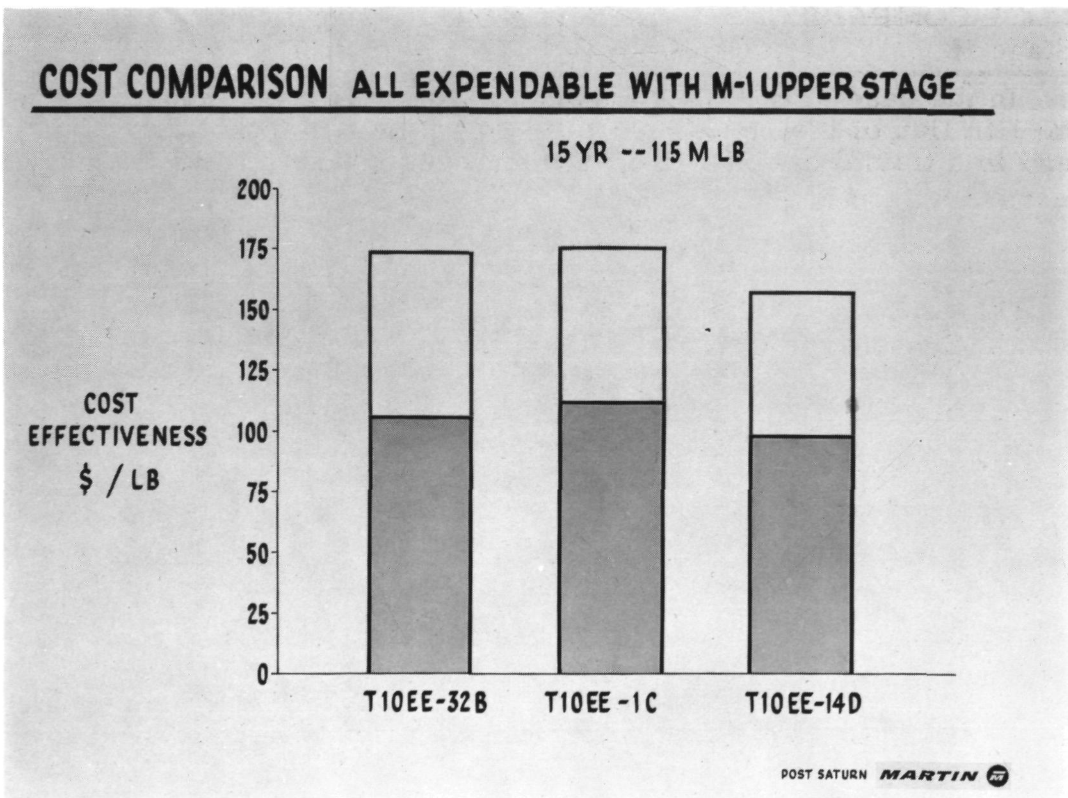
	Percent Savings	
	Total Cost	Operational Cost
RE	+11%	+36%
RR	+11%	+43%

COST COMPARISON--All expendable with M-1 Upper Stage

If the M-1/M-1 configuration (T10EE-32B) is selected as the base vehicle for comparison, the percent savings (or loss) incurred by using either the solid/M-1 (T10EE-14D) or the F-1A/M-1 (T10EE-1C) vehicle is as indicated:

	Percent Savings (or loss)	
	Total Cost	Operational Cost
Solid/M-1	+11%	+10%
F-1A/M-1	-2%	-4%


COST COMPARISON HP/HP; DEGREES OF RECOVERY
15 YR--115 M LB
COST EFFECTIVENESS ($/LB)
200
175
150
125
100
75
50
25
0
T10EE-3CU
T10RE-3CU
T10RR-3CU
POST SATURN MARTIN


COST COMPARISON ALL EXPENDABLE WITH M-1 UPPER STAGE
15 YR --115 M LB
COST EFFECTIVENESS $/LB
200
175
150
125
100
75
50
25
0
T10EE-32B
T10EE-1C
T10EE-14D
POST SATURN MARTIN

COST COMPARISON--M-1/M-1; Degrees of Recovery

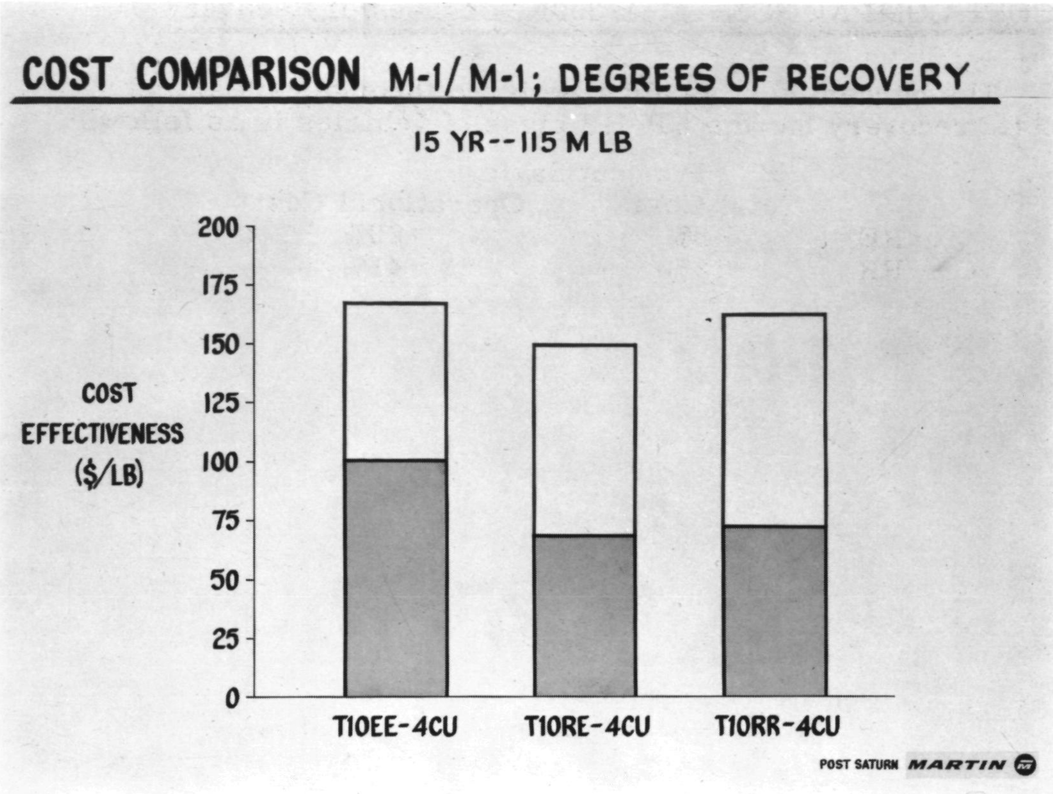
The percent cost savings achieved through partial and total recovery for the M-1/M-1 class of vehicle is as follows:

	Percent Savings	
	Total Cost	Operational Cost
RE	+9%	+31%
RR	+1%	+27%

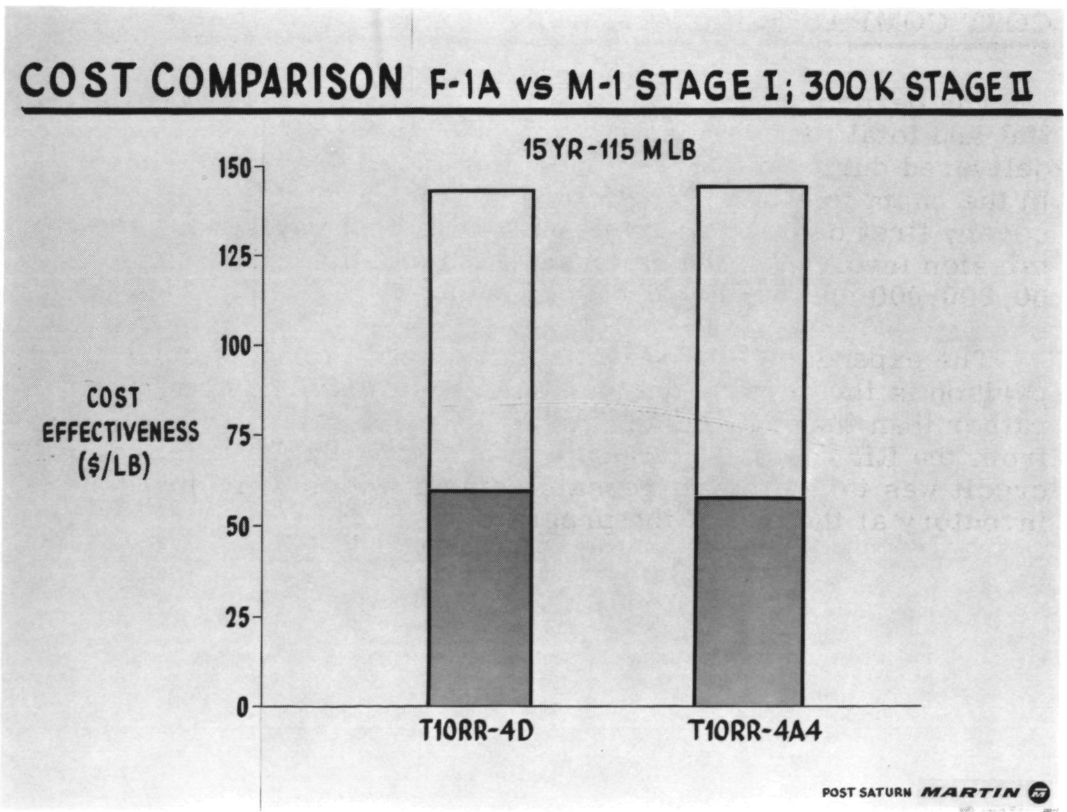
The skirt-flap combination required to stabilize the second stage of the RR vehicle during re-entry reduced the payload capability of the vehicle to the extent that the operational cost effectiveness was less favorable than that of the RE version.

COST COMPARISON--F-1A Versus M-1 Stage I; 300K Stage II

In the case of the LP/HP class of vehicles, use of the M-1 in lieu of the F-1A results in no savings in total cost and in a trivial savings in operational costs (i.e., 4%).



COST COMPARISON M-1/M-1; DEGREES OF RECOVERY
15 YR--115 M LB
COST EFFECTIVENESS ($/LB)
200
175
150
125
100
75
50
25
0
T10EE-4CU
T10RE-4CU
T10RR-4CU
POST SATURN MARTIN



COST COMPARISON F-1A vs M-1 STAGE I; 300K STAGE II
15 YR-115 M LB
COST EFFECTIVENESS ($/LB)
150
125
100
75
50
25
0
T10RR-4D
T10RR-4A4
POST SATURN MARTIN

COST COMPARISON--M-1/300K; Degrees of Recovery

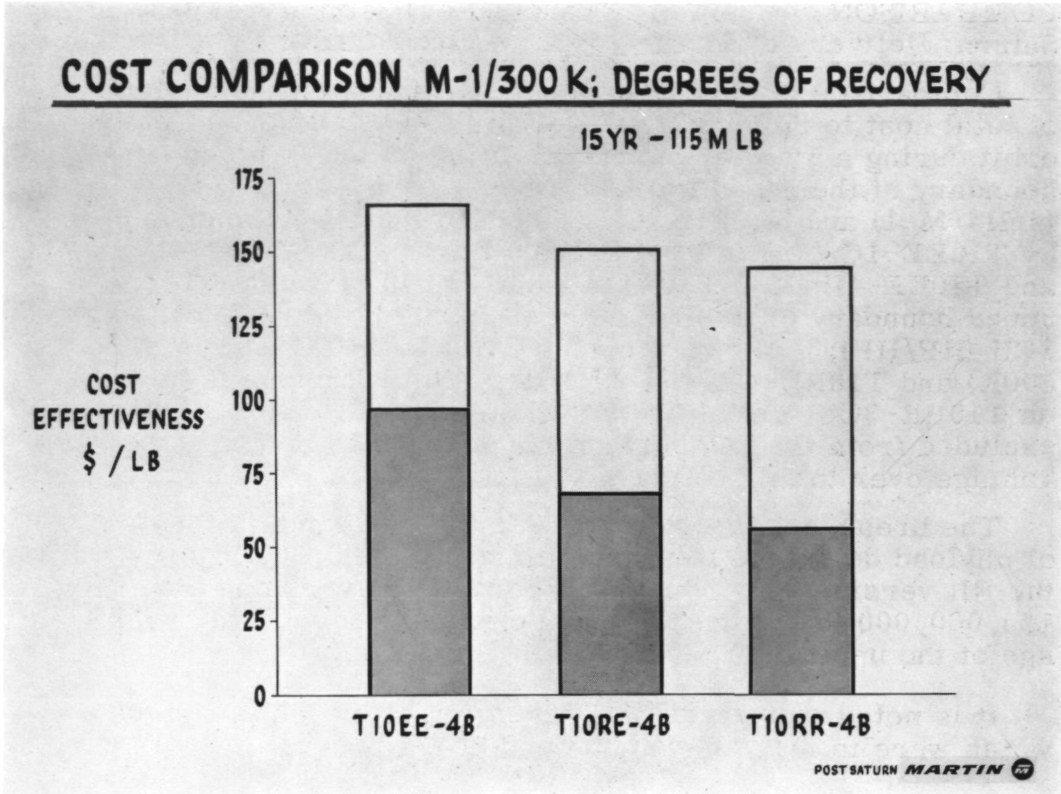
The percent cost savings achieved through partial and total recovery for the LP/HP class of vehicles is as follows:

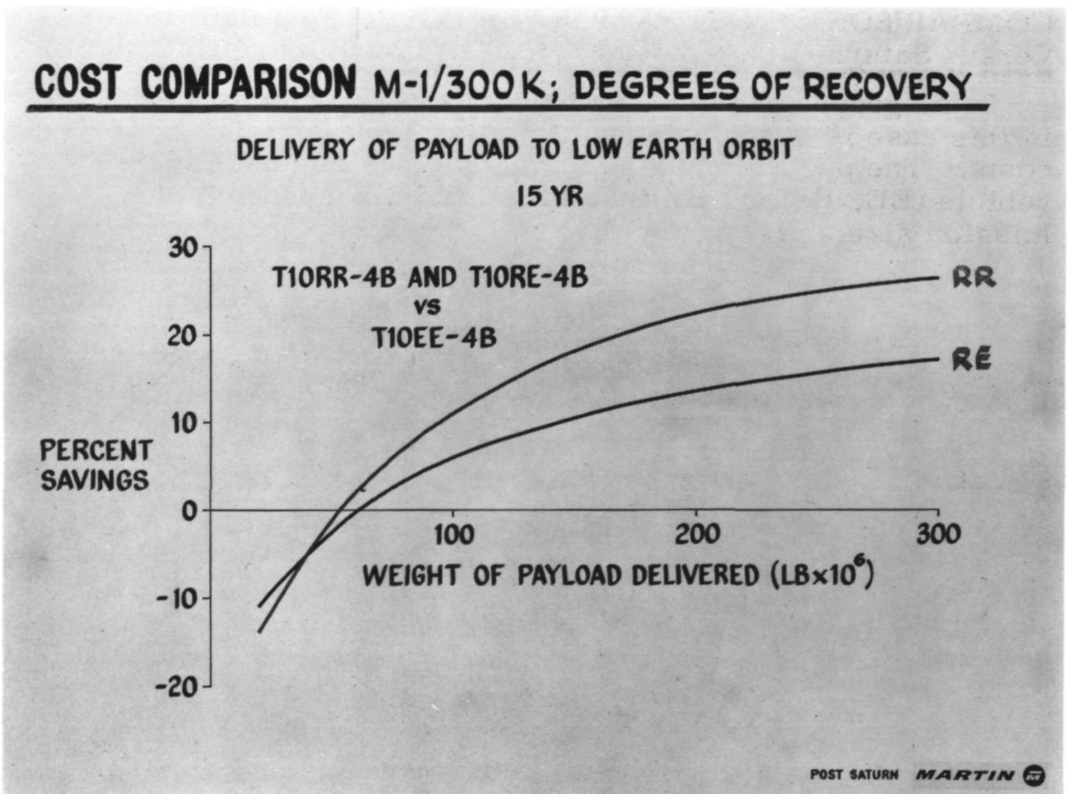
	Percent Savings: Total Cost	Percent Savings: Operational Cost
RE	+8%	+29%
RR	+12%	+41%

COST COMPARISON--M-1/300K; Degrees of Recovery

The percent savings in total cost achieved through partial and total recovery as a function of the weight of payload delivered during a 15-year operational time span are shown in the chart for the LP/HP class of launch vehicles. Recovery first begins to pay off on a total cost basis when the mission involves delivery to low earth orbit of between 50,000,000 and 60,000,000 lb of payload.

The expendable vehicle used as the base for this comparison is the version designed and "optimized" as an EE rather than the version derived by deleting recovery gear from the RR vehicle. Regarding the recoverable vehicles, credit was taken for all reusable stages which were in inventory at the end of the program.


COST COMPARISON M-1/300 K; DEGREES OF RECOVERY
15 YR - 115 M LB
COST EFFECTIVENESS $ / LB
175
150
125
100
75
50
25
0
T10EE-4B
T10RE-4B
T10RR-4B
POST SATURN MARTIN


COST COMPARISON M-1/300 K; DEGREES OF RECOVERY
DELIVERY OF PAYLOAD TO LOW EARTH ORBIT
15 YR
T10RR-4B AND T10RE-4B
VS
T10EE-4B
PERCENT SAVINGS
30
20
10
0
-10
-20
100
200
300
WEIGHT OF PAYLOAD DELIVERED (LB×10⁶)
RR
RE
POST SATURN MARTIN

COMPARISON OF TOTAL COSTS--Post Saturn Versus Saturn; Delivery of Payload to Low Earth Orbit.

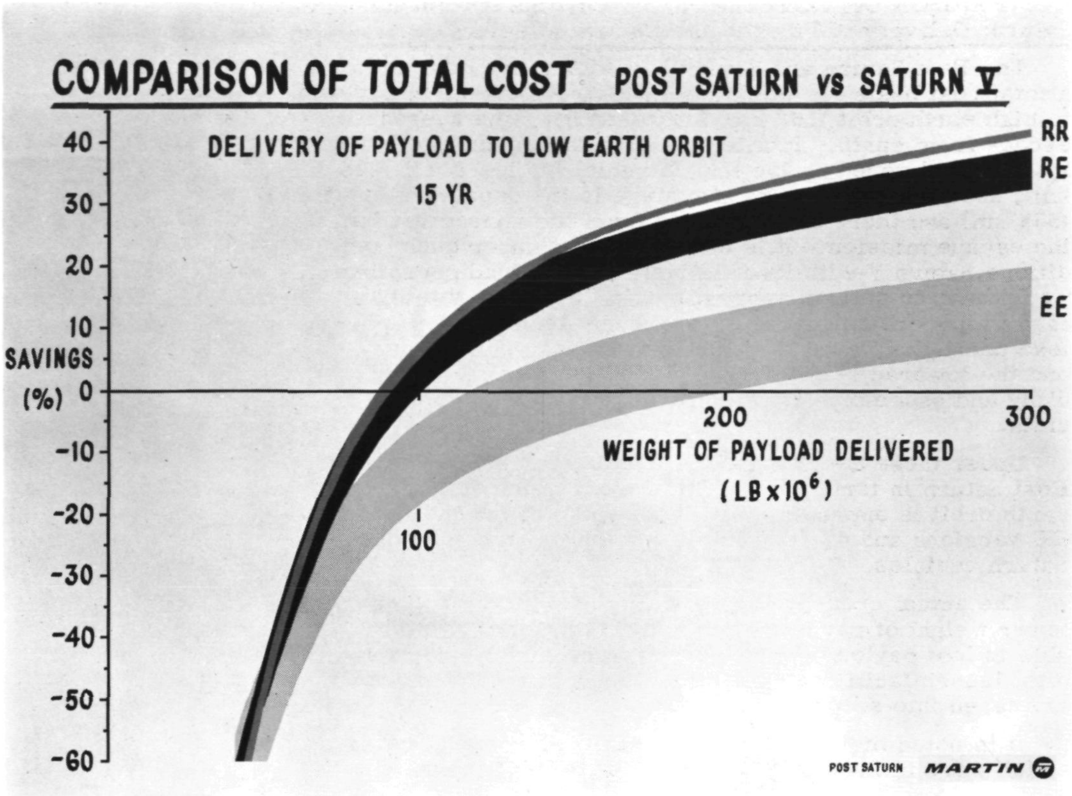
Comparison of Post Saturn with Saturn V on the basis of total cost to deliver equal weights of payload to low earth orbit during a 15-year operational time span. The upper boundary of the EE curve is formed by the T10EE-14D (solid/M-1) and T10EE-3CU (HP/HP); the lower boundary by T10EE-1C (F-1A/M-1). The T10EE-4CU (M-1/M-1) and T10EE-4B (M-1/300K) fall halfway in between. The upper boundary of the RE curve is formed by the T10RE-3CU (HP/HP); the lower boundary by the T10RE-4B (M-1/300K) and T10RE-4CU (M-1/M-1). The RR curve is based on T10RR-3CU and T10RR-4B data. The T10RR-4CU was excluded from the comparison since it offers no cost advantage over the RE version.

The break-even mission size for Post Saturn is terms of payload delivered is approximately 90,000,000 lb for the RR versions; 95,000,000 lb for the RE versions and 135,000,000 lb for the EE versions (using for EE the average of the upper half portion of the curve only).

It is noted that credit was taken for all reusable stages which were in inventory at the end of the program.

COMPARISON OF OPERATIONAL COSTS--Post Saturn Versus Saturn; Delivery of Payload to Low Earth Orbit

Same ground rules as for previous chart; comparison in this case is based on operational rather than total costs. The percent savings for each class of Post Saturn vehicle (EE, RE and RR) are essentially independent of mission size.



PERCENT SAVINGS IN OPERATIONAL COSTS

POST SATURN VS SATURN V
DELIVERY OF PAYLOAD TO LOW EARTH ORBIT

TYPE OF POST SATURN VEHICLE	% SAVINGS
EE	24 - 36
RE	56 - 62
RR	65 - 67

POST SATURN MARTIN

COMPARISON OF TOTAL COSTS--Post Saturn Versus Saturn; Delivery of Payload for Escape Missions

The Post Saturn and Saturn V vehicles are compared on a total cost basis for missions involving delivery of payload to high earth orbit (568 km) for assembly, and subsequent escape from earth. In this type of mission, units of payload are delivered by the launch vehicle to low orbit (225 km), then transferred via transtage to the departure orbit (568 km) and there assembled to form the spacecraft for the escape mission. It is estimated that under such conditions,Saturn V with its relatively low payload capability would have to deliver approximately 25% more total payload to low orbit in order to overcome less efficient payload packaging, a higher spares factor for rendezvous, and the lowered reliability resulting from in-orbit handling and assembly of a significantly larger number of units.

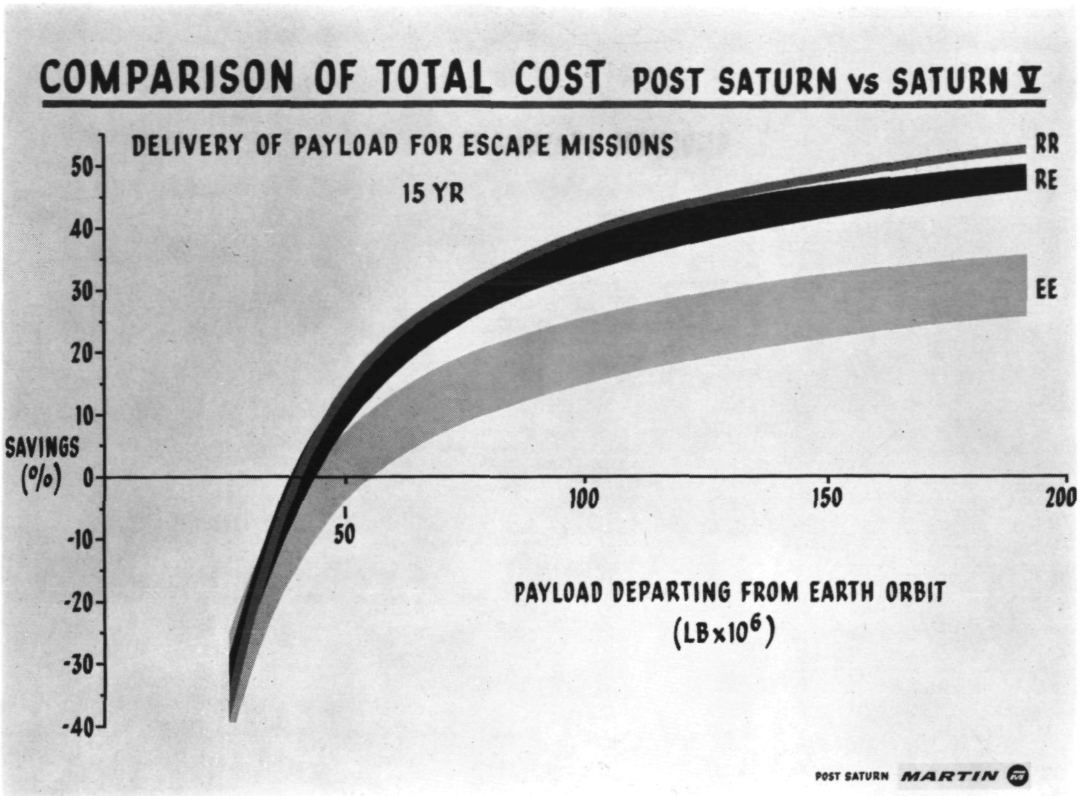
Under these conditions, the break-even mission size for Post Saturn in terms of weight of payload departing from earth orbit is approximately 40,000,000 lb for the RR and RE versions and 47,000,000 lb for the EE type of Post Saturn vehicles.

The actual crossover may well occur at a considerably lower weight of payload when nonvehicle costs attributable to lost payloads, differences in required size of orbital launch facilities, and other items of a related nature are taken into account.

It is noted that credit was taken for all reusable stages which were in inventory at the end of each program.

COMPARISON OF OPERATIONAL COSTS--Post Saturn Versus Saturn; Delivery of Payload for Escape Missions

Same ground rules as for previous chart; comparison in this case is based on operational rather than total costs. The percent savings for each class of Post Saturn vehicle (EE, RE and RR) are essentially independent of mission size.



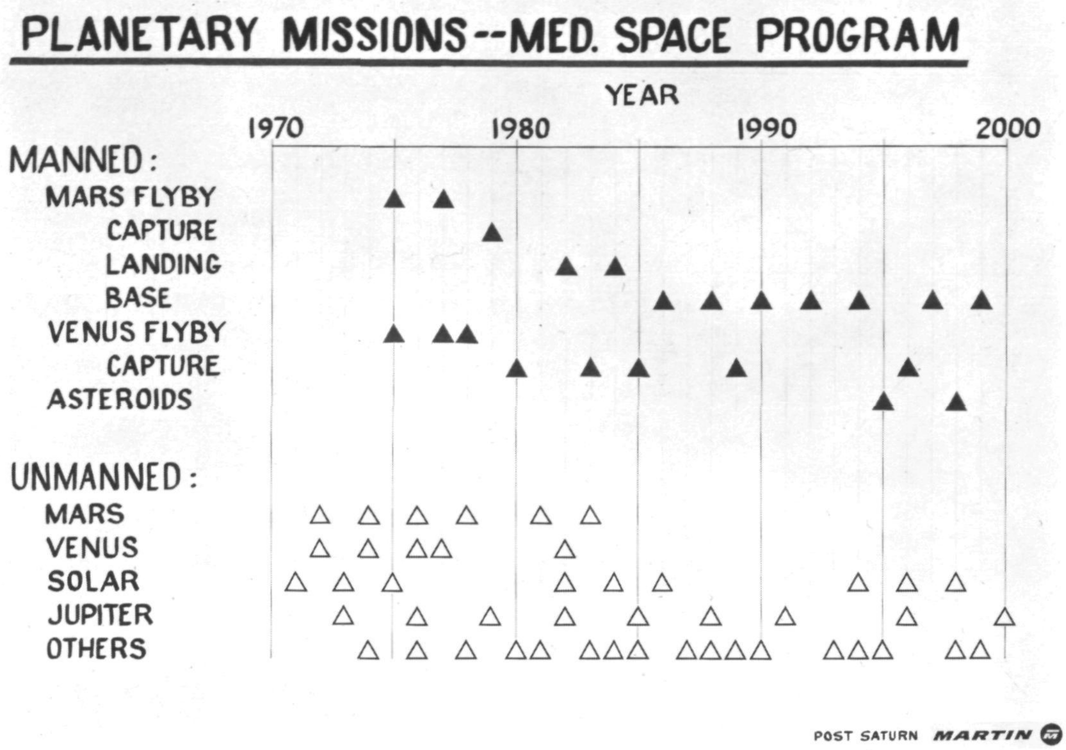
PERCENT SAVINGS IN OPERATIONAL COSTS

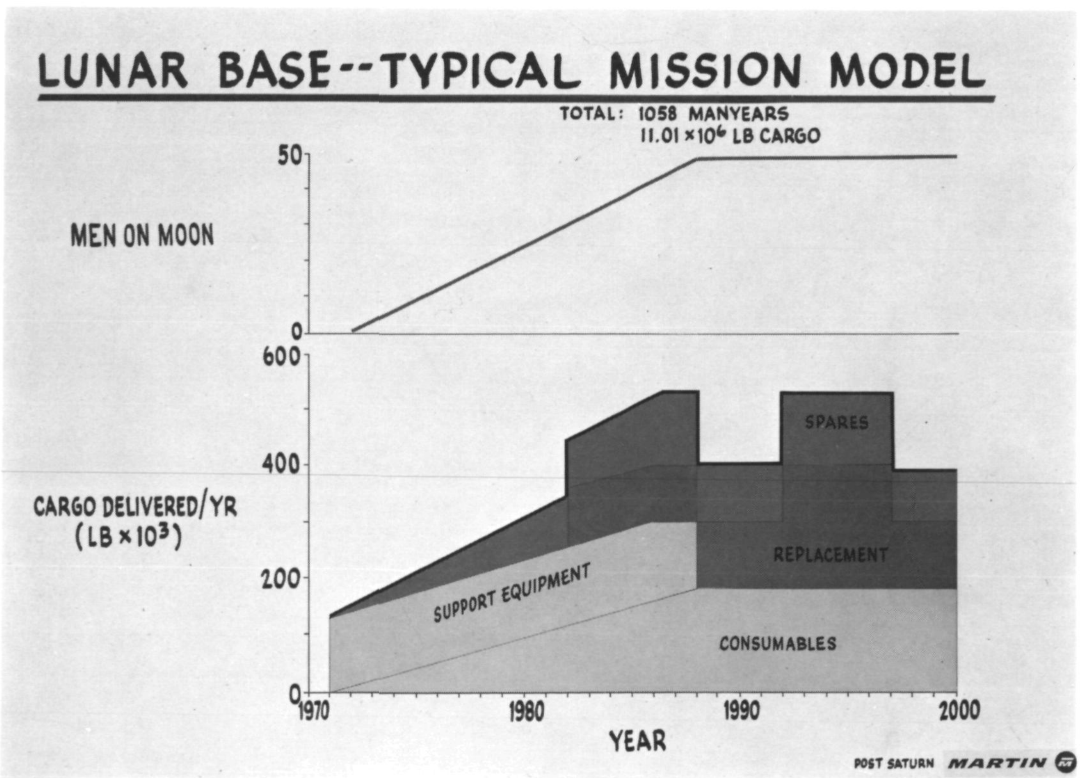
POST SATURN VS SATURN V
DELIVERY OF PAYLOAD FOR ESCAPE MISSIONS

TYPE OF POST SATURN VEHICLE	% SAVINGS
EE	39 - 48
RE	65 - 69
RR	71 - 73

POST SATURN MARTIN

IV. MISSION AND OPERATIONS ANALYSIS


PLANETARY MISSIONS--MED. SPACE PROGRAM
YEAR
1970
1980
1990
2000
MANNED:
MARS FLYBY
CAPTURE
LANDING
BASE
VENUS FLYBY
CAPTURE
ASTEROIDS
UNMANNED:
MARS
VENUS
SOLAR
JUPITER
OTHERS
POST SATURN MARTIN


LUNAR BASE--TYPICAL MISSION MODEL
TOTAL: 1058 MANYEARS
11.01 ×10⁶ LB CARGO
MEN ON MOON
50
0
CARGO DELIVERED/YR
(LB × 10³)
600
400
200
0
SPARES
REPLACEMENT
SUPPORT EQUIPMENT
CONSUMABLES
1970
1980
1990
2000
YEAR
POST SATURN MARTIN

SPACE PROGRAM YIELDS

SMALL PROGRAM

	MAN TRIPS	MANYEARS	CARGO DELIVERED (K LB)	LB IN. VICINITY OF DESTINATION
GLOBAL	3,760	—	—	—
ORBITAL	732	354	364	—
LUNAR	612	306	3112	~ 23,400 K
UNMANNED PLANETARY	—	—	—	485 K
MANNED PLANETARY				
MARS	77	39.4	230	3128 K
VENUS	44	5.4	0	3510K

POST SATURN MARTIN

SPACE PROGRAM YIELDS (MEDIUM PROGRAM)

	MAN TRIPS	MAN YEARS	CARGO DELIVERED (K LB)	KLB IN VICINITY
GLOBAL		-	-	-
ORBITAL	864	432	460	-
LUNAR	1806	903	8,995	~70,000
U. PLANETARY	-	-	-	1,105
MANNED PLANETARY				
MARS	188.0	133.0	1036	8,260
VENUS	83.8	3.87	0.0	2,590
ASTEROIDS	56.0	0.0	0.0	761

POST SATURN MARTIN

SPACE PROGRAM YIELDS (LARGE PROGRAM)

	MAN TRIPS	MAN YEARS	CARGO DEL. (K LB)	K LB IN VICINITY
GLOBAL	8720	—	—	—
ORBITAL	6432	2922	3000	
LUNAR	3273	1636	17,270	~140,000
UNMANNED PLANETARY	—	—	—	1280
MANNED PLANETARY				
• MARS	313	448	1948	15,510
• VENUS	165	232	0	11,580
• ASTEROIDS	96	0	0	1245

POST SATURN MARTIN

MANNED PLANETARY FLIGHT MODES

	FLY-BY	CAPTURE	LANDING	BASE	
START EARTH ORBIT	×	×	×	1	2
ESCAPE EARTH	▲	△	△	△1	△2
BRAKE, PLANET		▲	▲	▲1	▲2
DESCEND, PLANET SURFACE			▲	▲1	
ASCEND, RENDEZVOUS			▲		
ESCAPE PLANET		△	△		△2
BRAKE, EARTH	▲				
RE-ENTER EARTH	▲	▲	▲		▲2

POST SATURN MARTIN

MISSION MODES, LUNAR BASE

MODE:

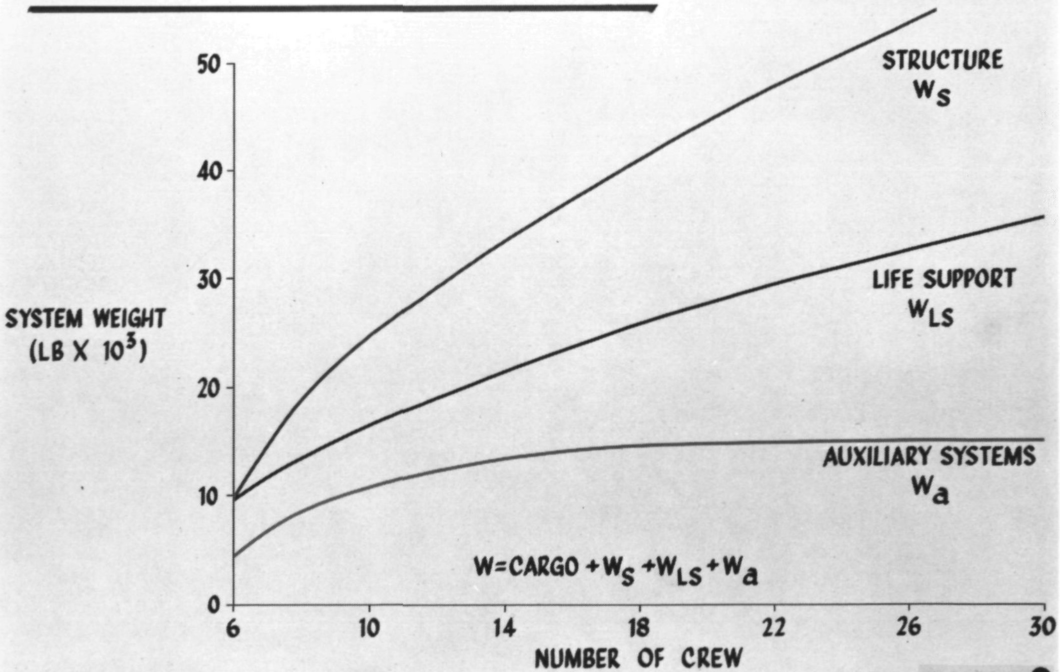
- DIRECT EARTH ESCAPE, CRYOGENIC PROPELLANTS
- DIRECT LUNAR LANDING, CRYOGENIC PROPELLANTS
- DIRECT LUNAR ESCAPE, CRYOGENIC PROPELLANTS
- AERODYNAMIC BRAKING, EARTH

MAKEUP:

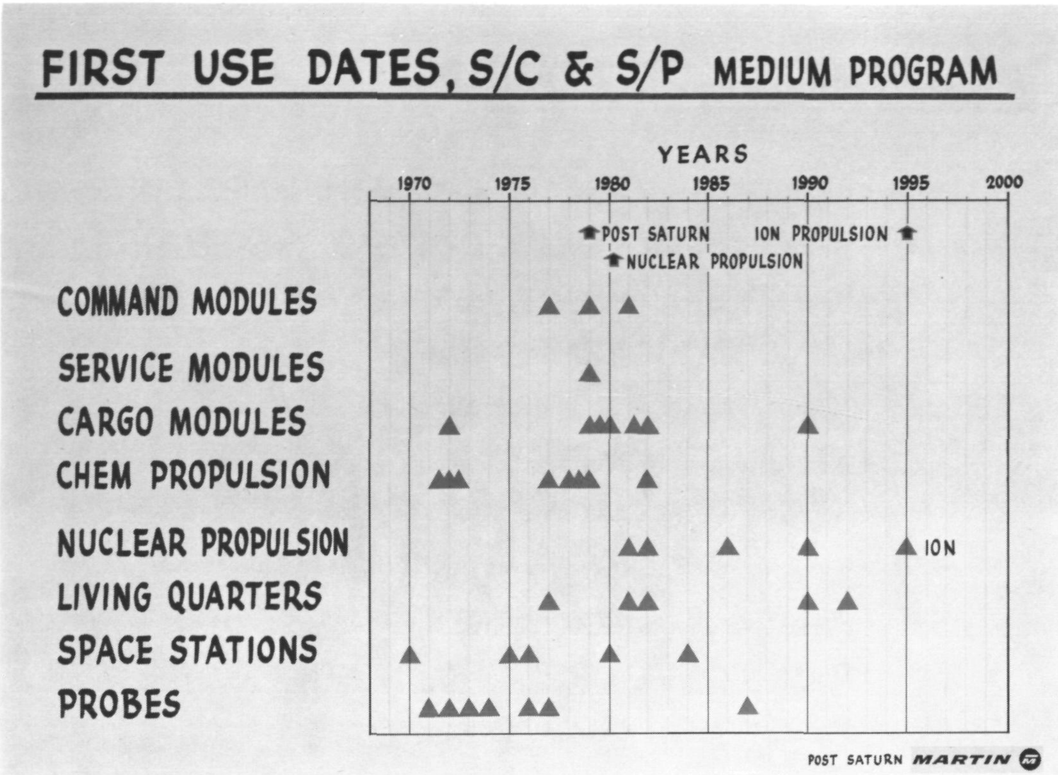
- SV MEN, SV CARGO
- SVN MEN, SVN CARGO, SV MEN
- SVN MEN & CARGO, SVN CARGO, SV MEN
- PS MEN, PS CARGO, SV MEN
- PS MEN & CARGO, PS CARGO, SV MEN
- PS TSE MEN, PS TSE CARGO

POST SATURN MARTIN

SPACE STATION PARAMETERS



POST SATURN MARTIN



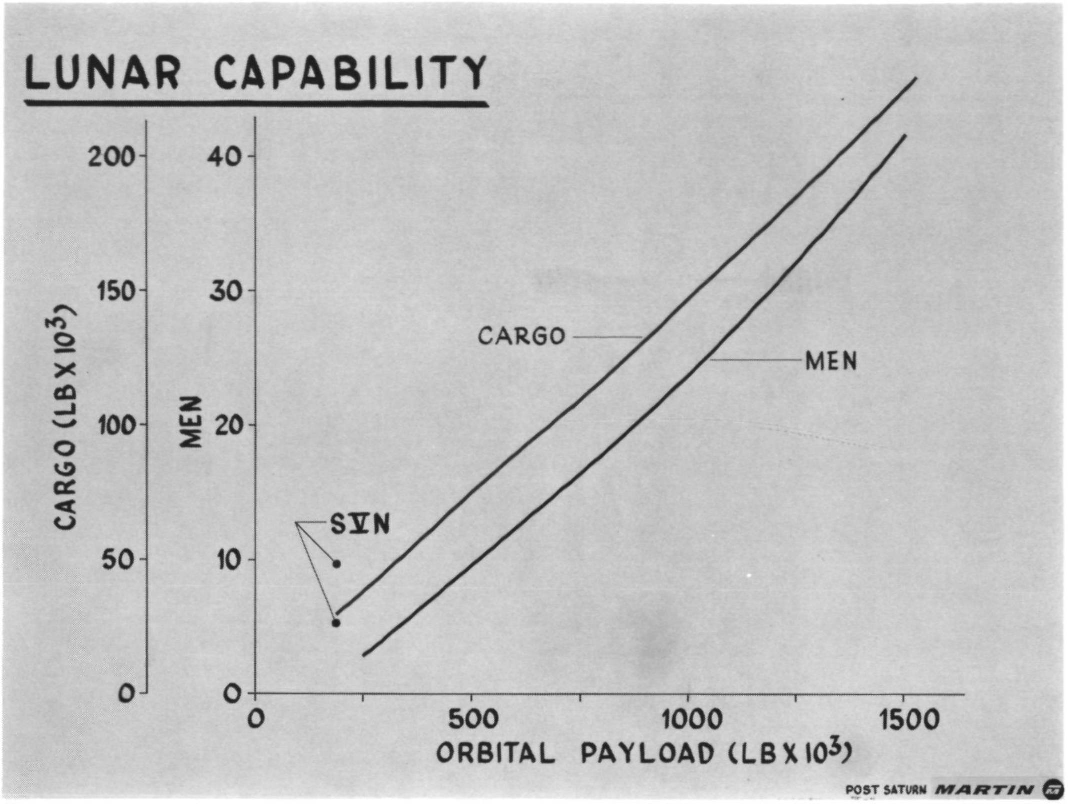
COST ANALYSIS, S/C & S/P

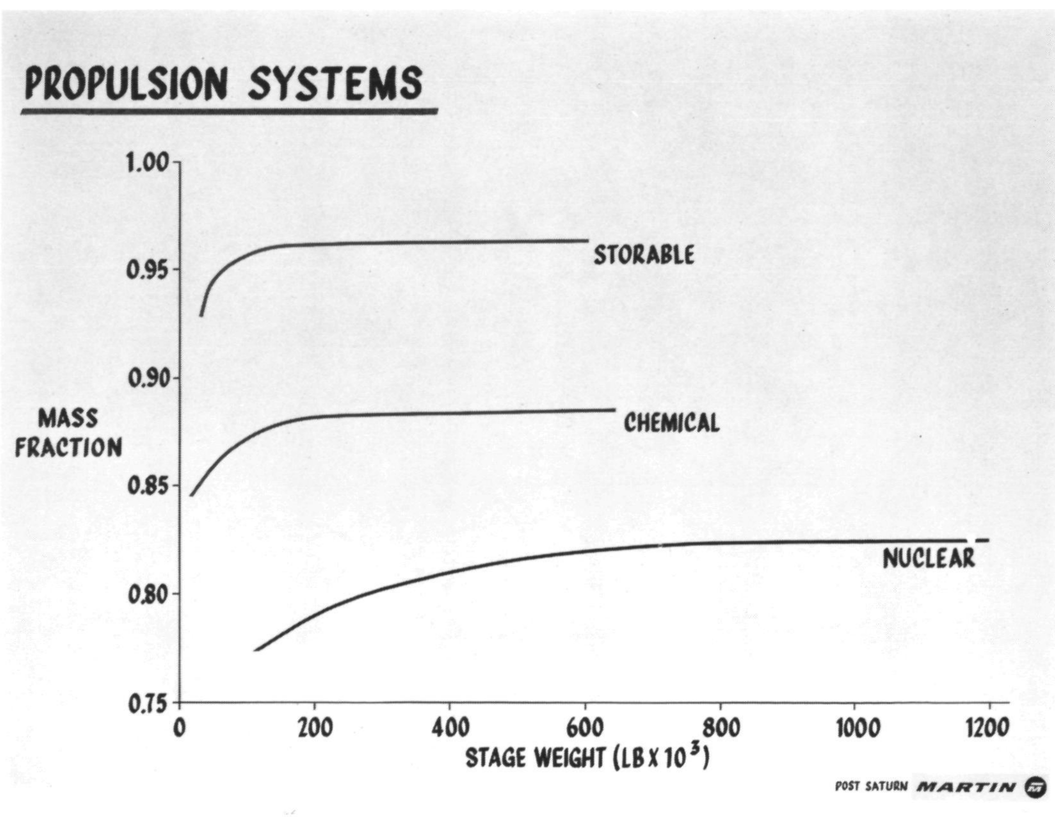
$C \times (D)^{\phi_1} \times (W)^{\phi_2} \times$ PROCUREMENT

$C \times (D)^{\phi_1} \times (W)^{\phi_2} \times g$ DEVELOPMENT

	PROCUREMENT			DEVELOPMENT		
	C	ϕ_1	ϕ_2	C	ϕ_1	ϕ_2
STRUCTURE	179×10^3	0.81	0.3785	1.37×10^6	0.81	0.585
PROPULSION						
THRUST, CHEMICAL	23.4×10^3	0	0.3785	N × PROC COST		
THRUST, STORABLE	23.4×10^3	0	0.3785	N × PROC COST		
THRUST, NUCLEAR	10.3×10^3	0	0.594	N × PROC COST	+	1.8×10^9 (FIRST ONE)
AUX SYSTEMS	444×10^3	0.81	0.3785	3.36×10^6	0.81	0.585
LIFE SUPPORT	65×10^3	0.81	0.3785	1.52×10^6	0.81	0.585

POST SATURN MARTIN


LUNAR CAPABILITY
200
150
100
50
0
CARGO (LB X 10³)
40
30
20
10
0
MEN
S V N
CARGO
MEN
0
500
1000
1500
ORBITAL PAYLOAD (LB X 10³)
POST SATURN MARTIN

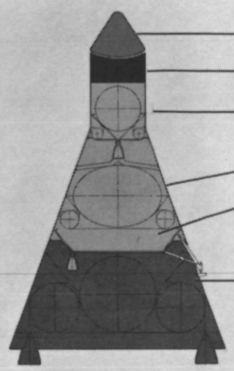

PROPULSION SYSTEMS
1.00
0.95
0.90
0.85
0.80
0.75
MASS FRACTION
STORABLE
CHEMICAL
NUCLEAR
0
200
400
600
800
1000
1200
STAGE WEIGHT (LB X 10³)
POST SATURN MARTIN

SPACECRAFT ANALYSIS

- REQUIREMENTS
- CATEGORIES
- SUBSYSTEMS
- COMPATIBLE SIZING
- COMPATIBLE COSTS

POST SATURN MARTIN

POST SATURN LUNAR PAYLOAD

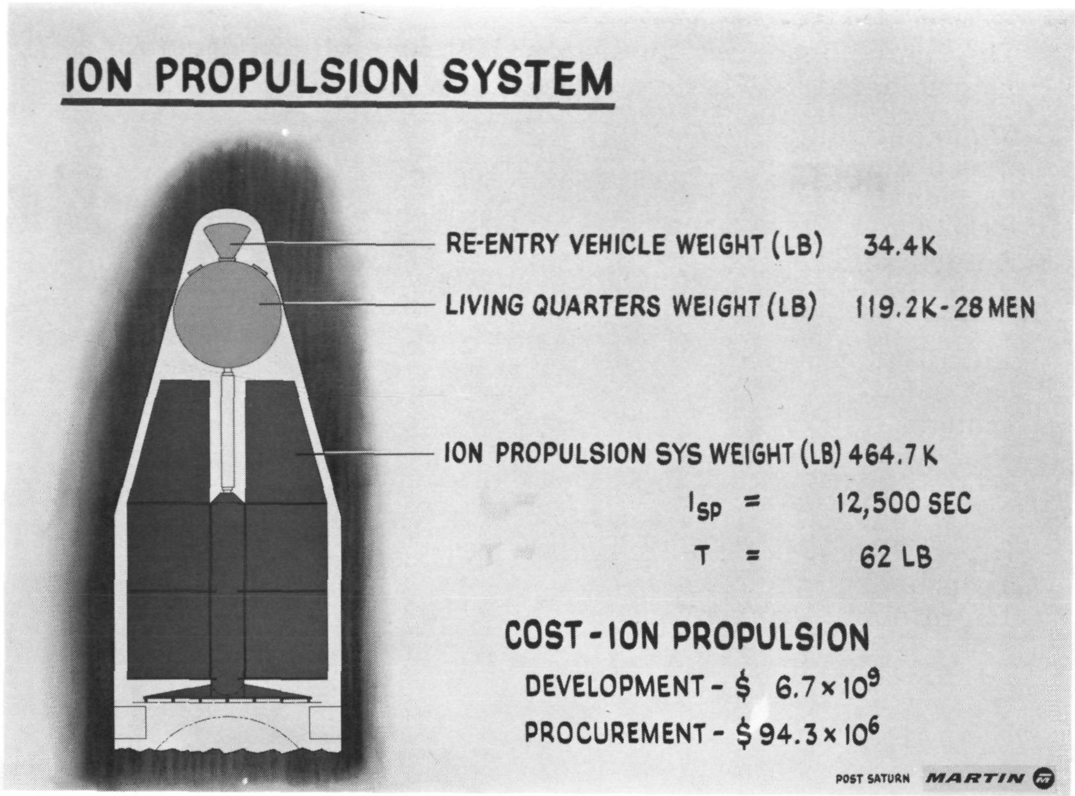


SYSTEM	WEIGHT (LB)
COMMAND MODULE	31.7K
SERVICE MODULE	17.6K
LUNAR ESCAPE	61.6K
LUNAR LANDING	223.6K
CARGO	59.7K
EARTH ESCAPE	605.6K

AVG COST
$\$72 \times 10^6$

YIELD
13 MEN 52.4 K CARGO
OR 148 K CARGO

POST SATURN MARTIN


ION PROPULSION SYSTEM
RE-ENTRY VEHICLE WEIGHT (LB) 34.4K
LIVING QUARTERS WEIGHT (LB) 119.2K-28 MEN
ION PROPULSION SYS WEIGHT (LB) 464.7K
I_{sp} = 12,500 SEC
T = 62 LB
COST-ION PROPULSION
DEVELOPMENT - $ 6.7×10^9
PROCUREMENT - $ 94.3×10^6
POST SATURN
MARTIN

SPACE PROGRAM ACTIVITIES

The figure opposite illustrates the areas of activities for a Space Program. The ground base operational activities are:

Manufacturing
Testing
Launch
Retrieval
Refurbishing.

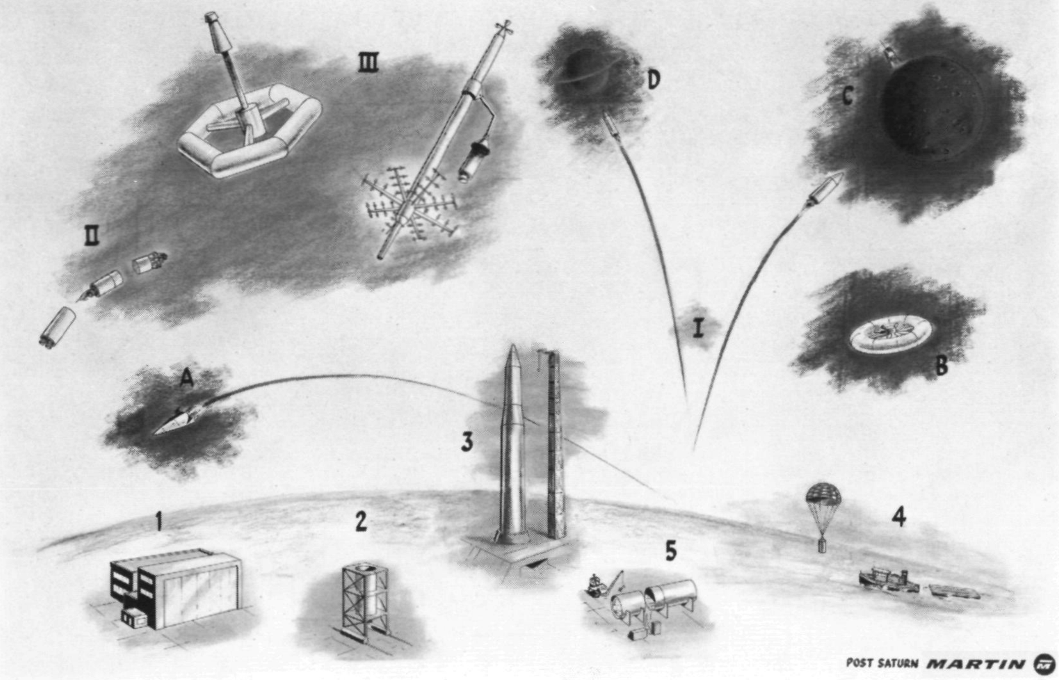
The Missions are:

Global
Orbital
Lunar
Planetary (unmanned and manned).

The Flight Modes are:

Direct Flight Launch
Temporary Mode Orbital Launch
Permanent Orbital Facility Launch.

SPACE PROGRAM ACTIVITIES

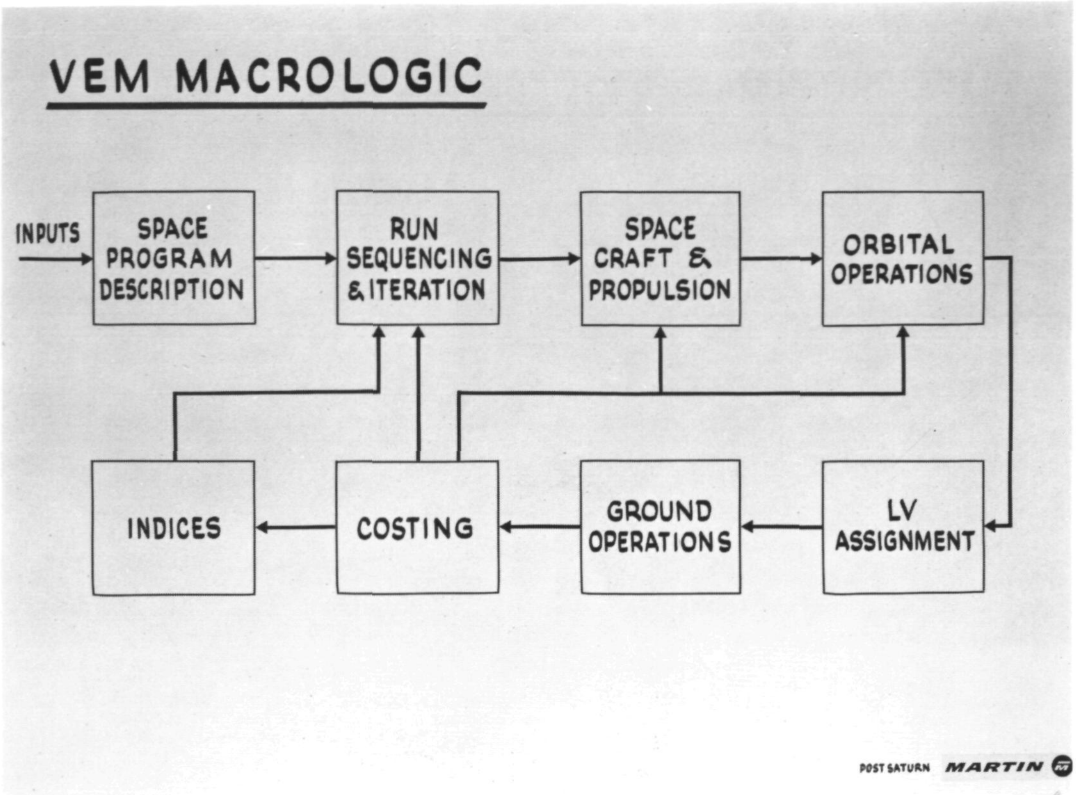


POST SATURN MARTIN

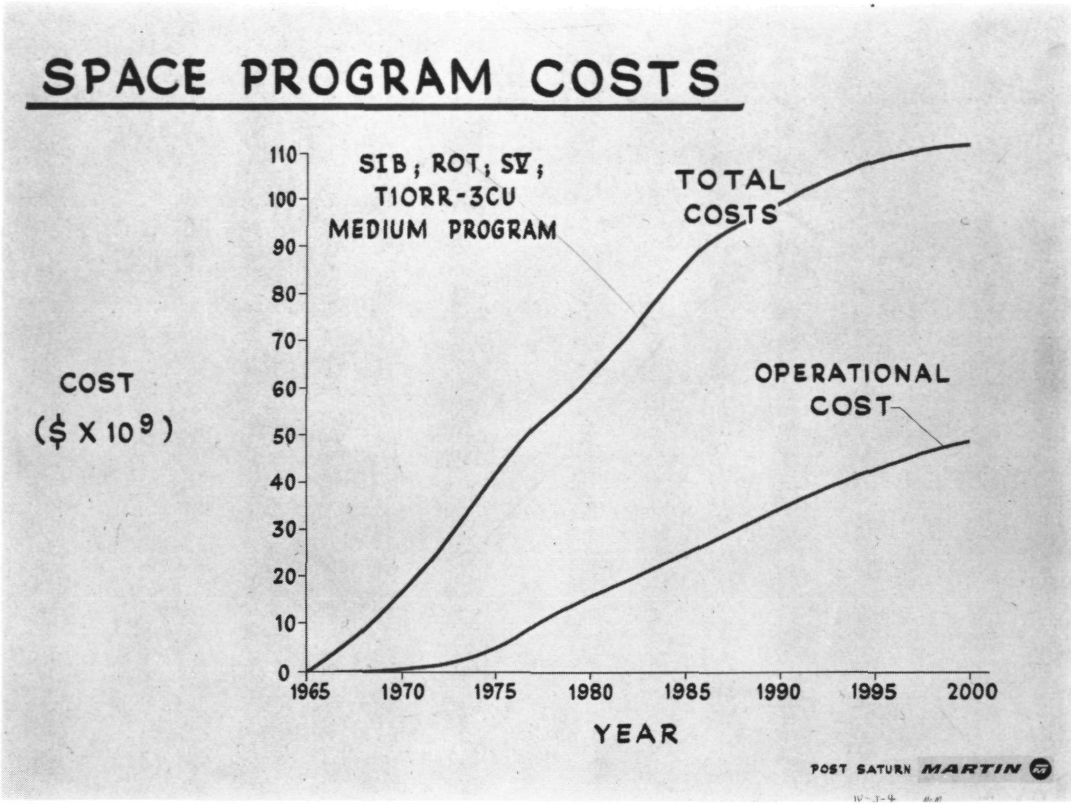
REQUIRED MODEL OPERATIONS

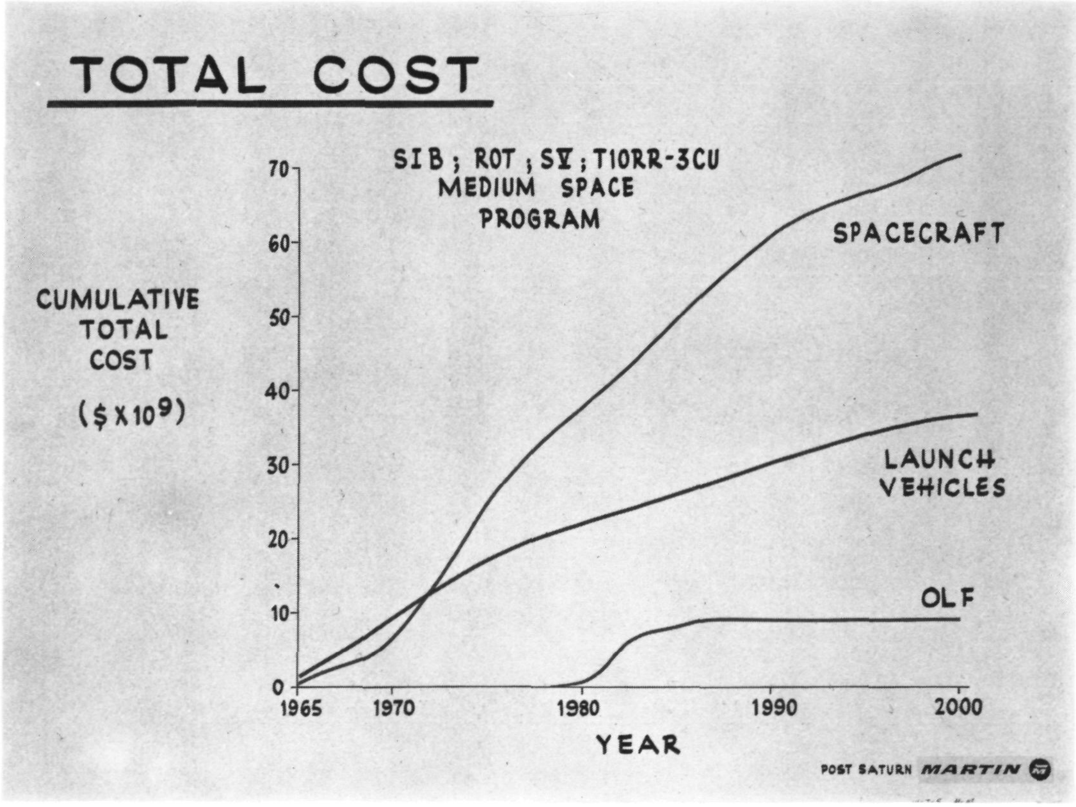
- PROCESS SPACE OBJECTIVES
- SELECT & SIZE SPACECRAFT & PROPULSION
- SELECT FLIGHT MODES
- PROCESS ORBITAL OPERATIONS
- SELECT LAUNCH VEHICLES
- SCHEDULE GROUND OPERATIONS
- SIZE FACILITIES
- COMPUTE & DISTRIBUTE COST
- FEED BACK UPDATED COST INTO SELECTION PROCESS
- COMPUTE PERFORMANCE INDICES

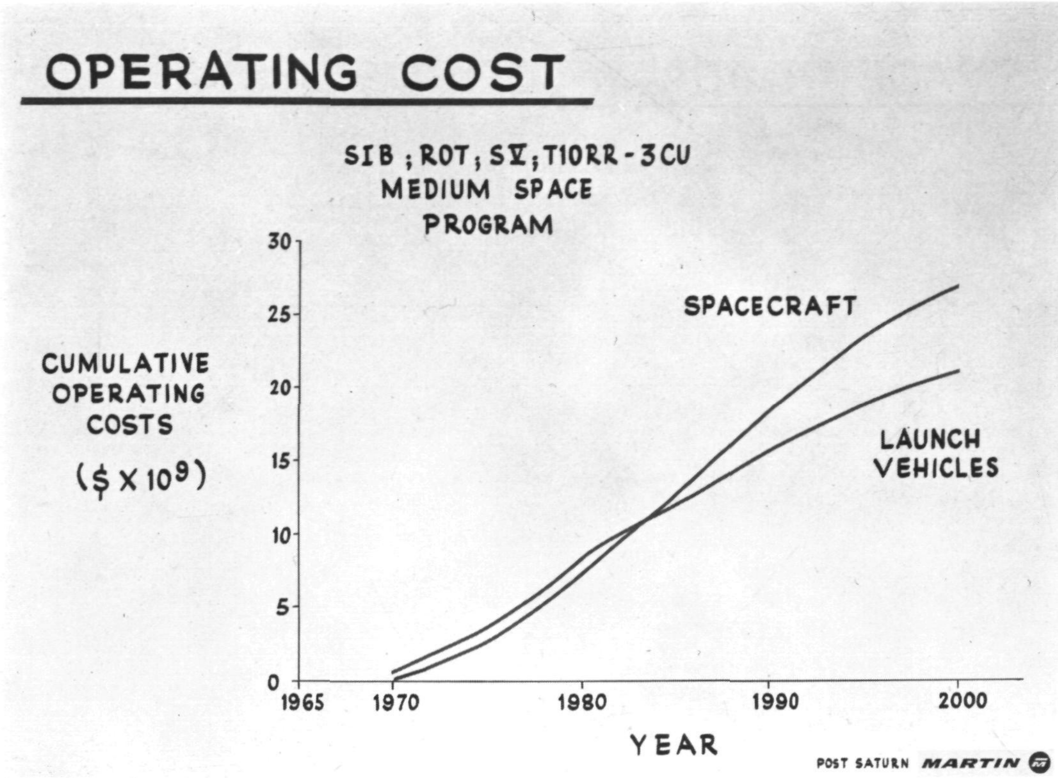
POST SATURN MARTIN

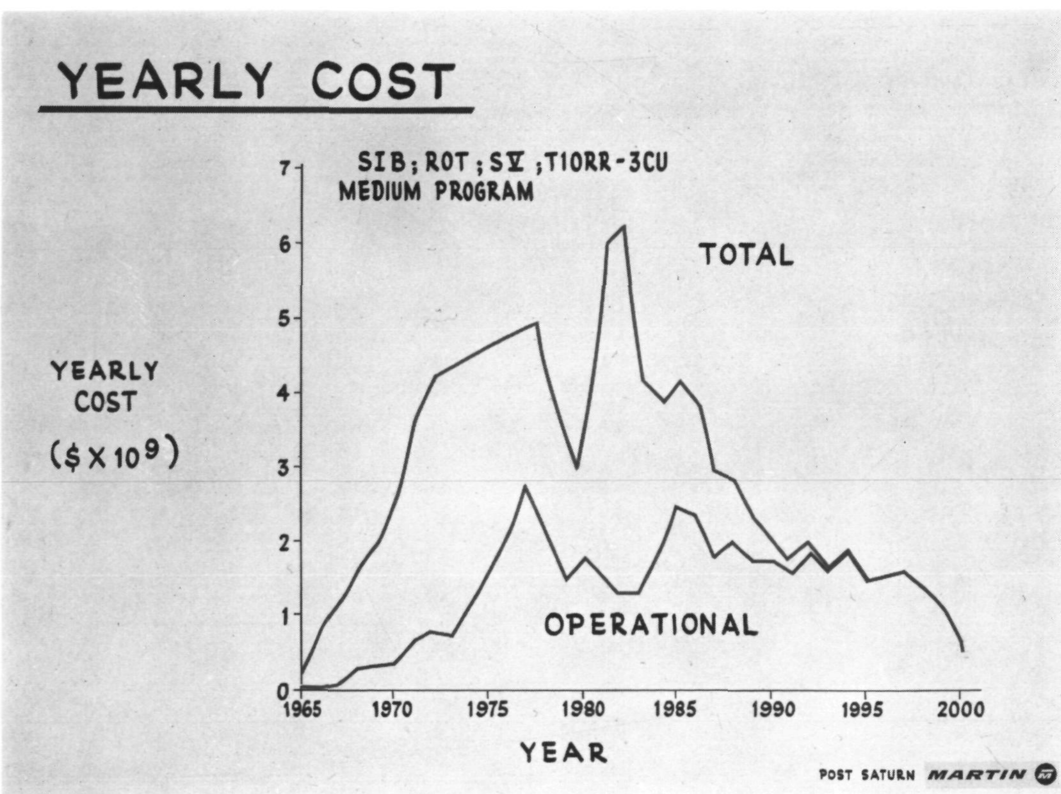


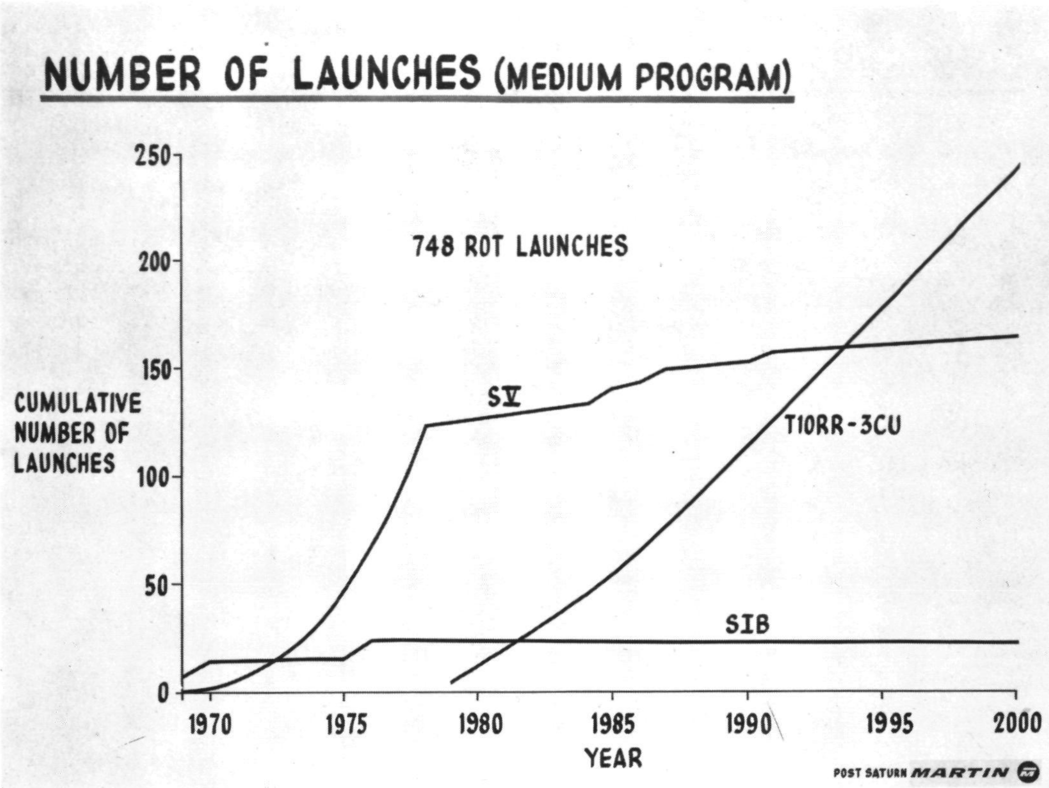
The following ten illustrations present the results of the VEM evaluation.

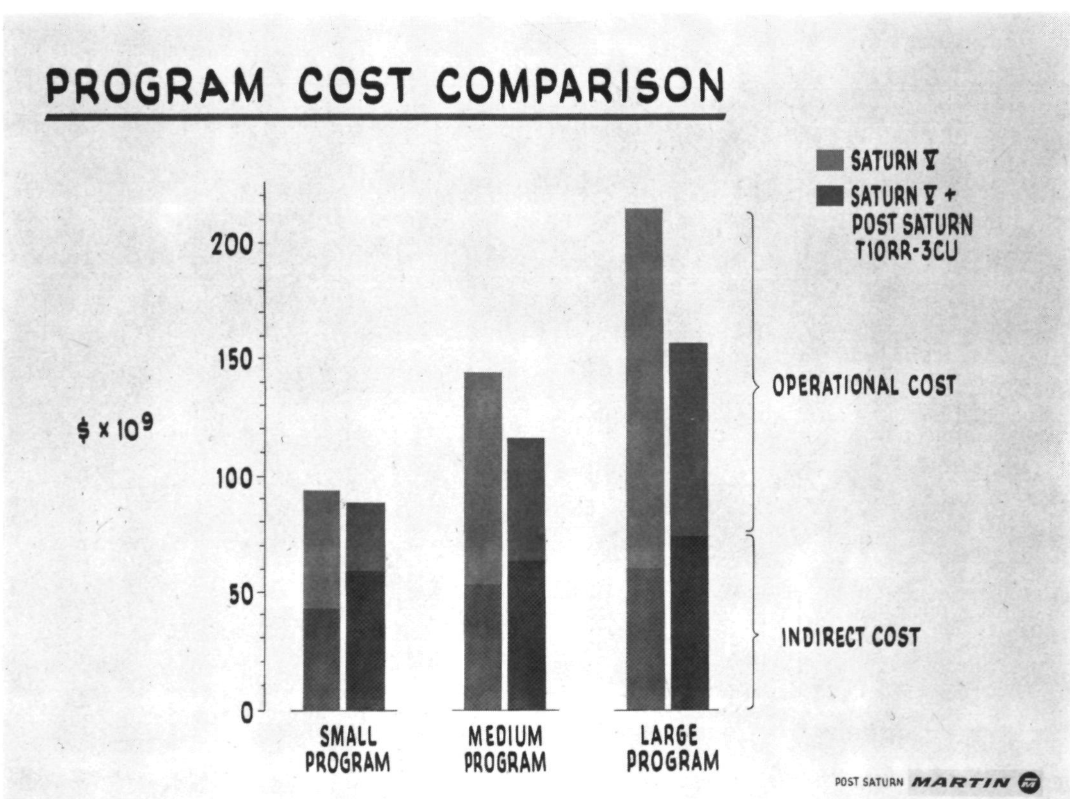

SPACE PROGRAM COSTS
SIB; ROT; SV;
T1ORR-3CU
MEDIUM PROGRAM
TOTAL
COSTS
OPERATIONAL
COST
COST
($ X 10⁹)
110
100
90
80
70
60
50
40
30
20
10
0
1965
1970
1975
1980
1985
1990
1995
2000
YEAR
POST SATURN MARTIN

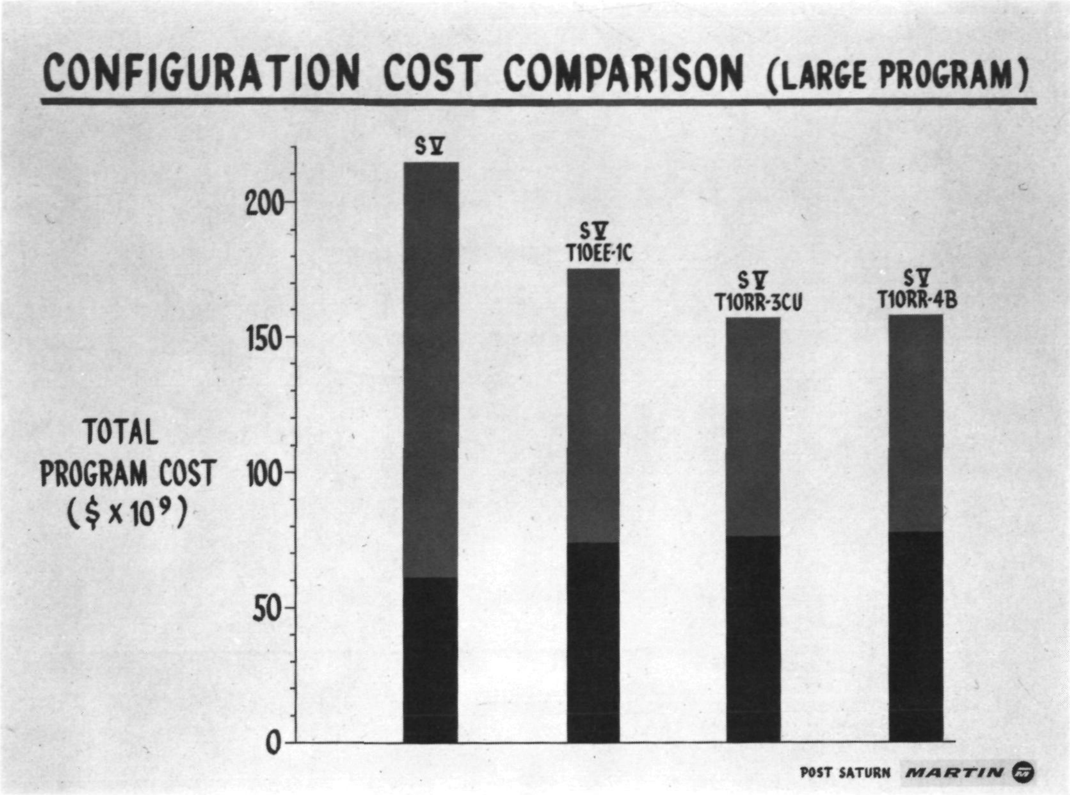

TOTAL COST
SIB; ROT; SV; T1ORR-3CU
MEDIUM SPACE
PROGRAM
SPACECRAFT
LAUNCH
VEHICLES
OLF
CUMULATIVE
TOTAL
COST
($X10⁹)
70
60
50
40
30
20
10
0
1965
1970
1980
1990
2000
YEAR
POST SATURN MARTIN

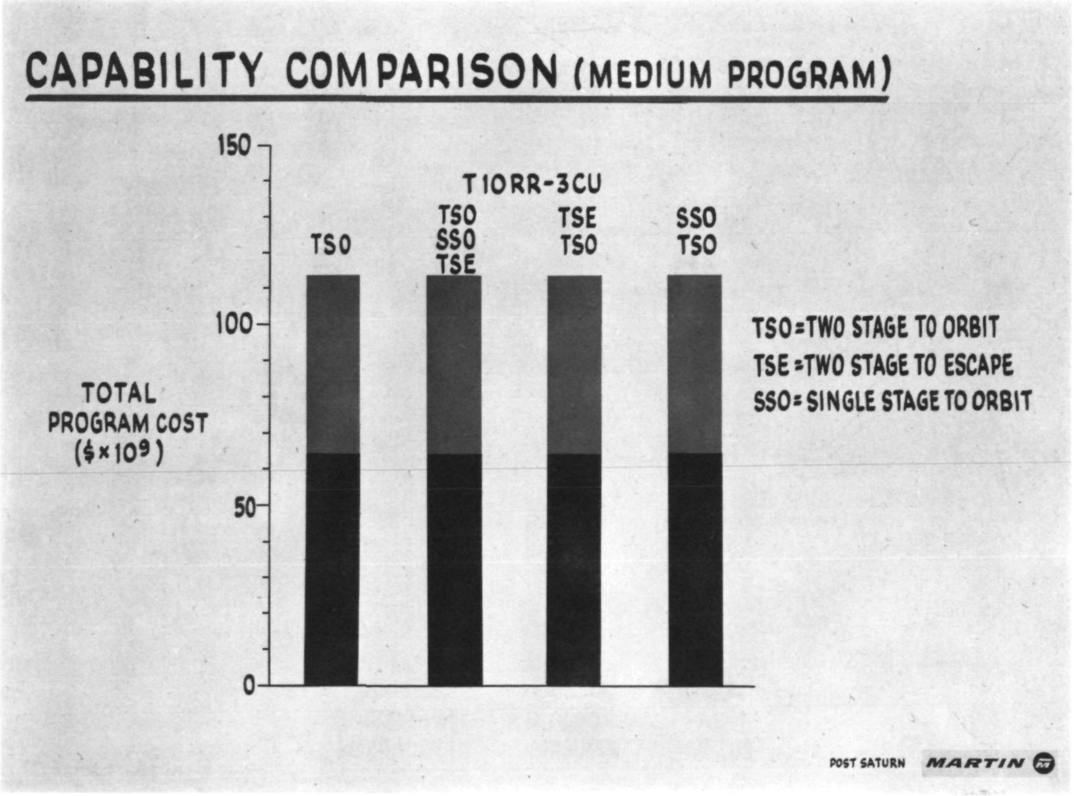

OPERATING COST
SIB ; ROT ; SV ; T10RR - 3CU
MEDIUM SPACE
PROGRAM
CUMULATIVE
OPERATING
COSTS
($ X 10⁹)
SPACECRAFT
LAUNCH
VEHICLES
30
25
20
15
10
5
0
1965
1970
1980
1990
2000
YEAR
POST SATURN
MARTIN

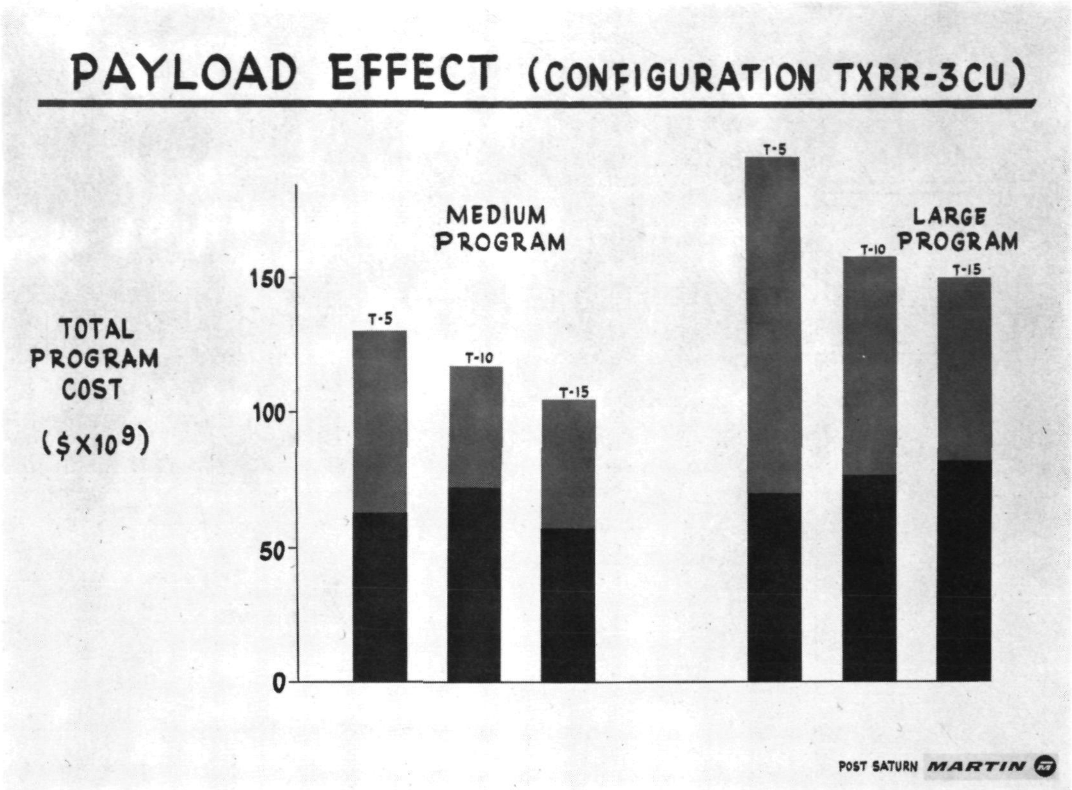

YEARLY COST
SIB; ROT; SV ; T10RR-3CU
MEDIUM PROGRAM
TOTAL
YEARLY
COST
($ X 10⁹)
OPERATIONAL
7
6
5
4
3
2
1
0
1965
1970
1975
1980
1985
1990
1995
2000
YEAR
POST SATURN
MARTIN

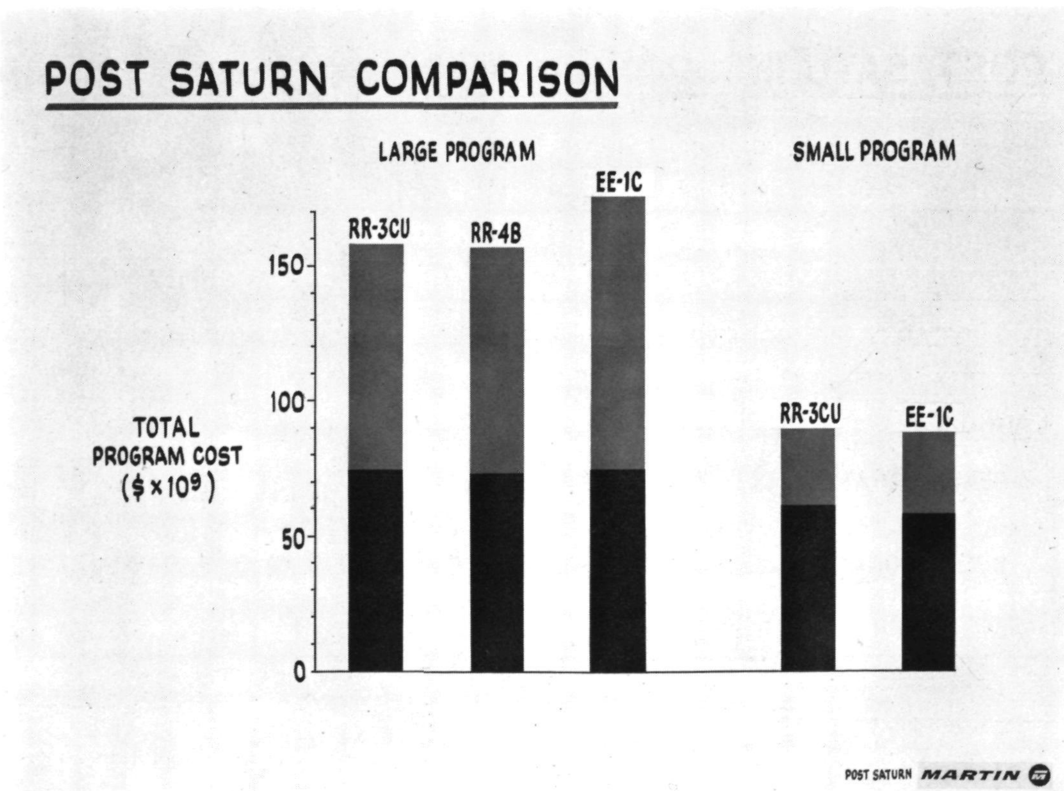

NUMBER OF LAUNCHES (MEDIUM PROGRAM)
748 ROT LAUNCHES
CUMULATIVE NUMBER OF LAUNCHES
250
200
150
100
50
0
SV
T10RR-3CU
SIB
1970
1975
1980
1985
1990
1995
2000
YEAR
POST SATURN MARTIN


PROGRAM COST COMPARISON
SATURN V
SATURN V + POST SATURN T10RR-3CU
$ \times 10^9$
200
150
100
50
0
OPERATIONAL COST
INDIRECT COST
SMALL PROGRAM
MEDIUM PROGRAM
LARGE PROGRAM
POST SATURN MARTIN

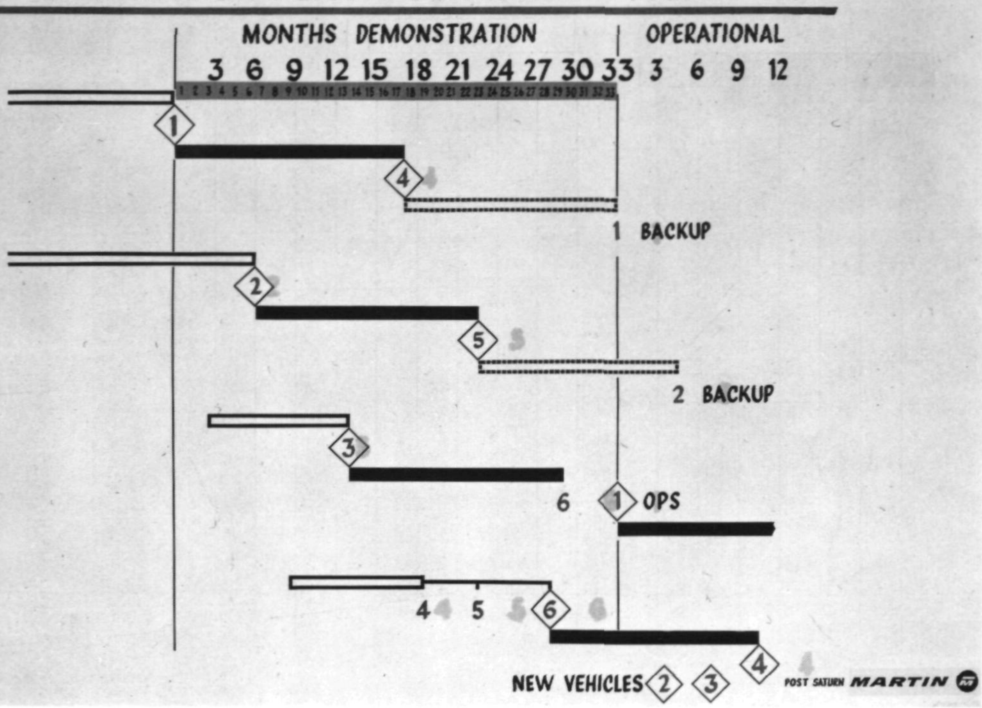

CONFIGURATION COST COMPARISON (LARGE PROGRAM)
TOTAL PROGRAM COST ($ x 10⁹)
200
150
100
50
0
S V
S V T10EE-1C
S V T10RR-3CU
S V T10RR-4B
POST SATURN
MARTIN


CAPABILITY COMPARISON (MEDIUM PROGRAM)
TOTAL PROGRAM COST ($×10⁹)
150
100
50
0
T10RR-3CU
TSO
TSO SSO TSE
TSE TSO
SSO TSO
TSO=TWO STAGE TO ORBIT
TSE=TWO STAGE TO ESCAPE
SSO=SINGLE STAGE TO ORBIT
POST SATURN
MARTIN


PAYLOAD EFFECT (CONFIGURATION TXRR-3CU)
MEDIUM PROGRAM
LARGE PROGRAM
TOTAL PROGRAM COST ($X10⁹)
150
100
50
0
T-5
T-10
T-15
T-5
T-10
T-15
POST SATURN MARTIN


POST SATURN COMPARISON
LARGE PROGRAM
SMALL PROGRAM
TOTAL PROGRAM COST ($ x 10⁹)
150
100
50
0
RR-3CU
RR-4B
EE-1C
RR-3CU
EE-1C
POST SATURN MARTIN

DEMONSTRATION FLIGHT SCHEDULE--T10RR-4A4



POST SATURN 50/4 MISSION

	DEMON.				OPERATIONAL															TOTAL
YEARS	-4	-3	-2	-1	1	2	3	4	5	6	7	8	9	10	11	12	13	14	15	
LAUNCHES																				
DEMONSTRATION		2	2	2																6
OPERATIONAL					4	6	7	8	10	8	10	9	10	9	10	9	9	9	9	127

POST SATURN MARTIN

PROPULSION DEVELOPMENT

TEST	ARTICLE	LOCATION	REQUIREMENTS
COLD FLOW	HEAVY GAGE STAGE I & II	MTF	FACILITY CHECKOUT, 10 FLOW TESTS
		(MFG)	SYSTEM INSTALLATION
		KSC	FACILITY CHECKOUT
		MTF	BACKUP FOR CAPTIVE FIRING
CAPTIVE FIRING	HEAVY GAGE STAGE I & II	MTF	DEVELOPMENT: 7 RUNS 15% LOADING
			3 50%
			3 75%
			2 100% (ALL SYSTEMS PROTO. GSE)
			TOTAL 15
			VERIFICATION: 2 RUNS 15% LOADING } ALL SYSTEMS FINAL GSE
			2 50%
			2 75%
			2 100%
			TOTAL 8
FLIGHT RATING TESTS	ENGINE MODULE I & II	MTF	ENGINE MODULE TESTING

POST SATURN MARTIN

PROPULSION ACCEPTANCE

TEST	ARTICLE	LOCATION	REQUIREMENTS
CAPTIVE FIRING	ALL NEW FLT. STAGES I & II	MTF	ACCEPTANCE: 1 SHORT; 1 LONG DURATION (N-I 30 SEC; 194 SEC) (N-II 30 SEC; 300 SEC)
CAPTIVE FIRING	USED STAGES I & II	MTF	ACCEPTANCE: SAME AS NEW

		DEMON.	OPERATIONAL															
			-1	1	2	3	4	5	6	7	8	9	10	11	12	13	14	15
N-I	NEW	(4)	2	2	2	2	2	2										
	USED	(4+2=6)		4	5	5	5	1	1	1	1	1	1	1	1	1	1	1
N-II	NEW	(4)	3	2	3	3	2	2										
	USED	(4+2=6)		4	5	5	5	1	1	1	1	1	1	1	1	1	1	1

POST SATURN MARTIN

STRUCTURES

TEST	ARTICLE	LOCATION	REQUIREMENTS
TANK PROOF	I & II LOX	MTF	1 LN_2 FILL TO 105% 100 H_2O FILLS TO 100% (CYCLING TEST) 1 H_2O FILL TO 140%
	I & II LH_2	MTF	1 LH_2 FILL TO 105% 10 LH_2 FILLS TO 100% (CYCLING TEST) 1 H_2O FILL TO 140% (SILO TEST)
STAGE	I	MTF	GASEOUS NITROGEN (ULLAGE PRESSURE) AT FLIGHT TEMPERATURE
STAGE	II	MTF	LN_2 AND LH_2 FILL WITH ULLAGE PRESSURES
INTERSTAGE	—	MFG	FLIGHT AND SEPARATION LOADS
GVS	I & II	MFG	1/10 SCALE VIBRATION TESTING
WIO	I & II	—	1/40 SCALE FREQUENCY AND BENDING MODE RESPONSE

POST SATURN MARTIN

OTHER MAJOR TEST ARTICLES

TEST	ARTICLE	LOCATION	REQUIREMENTS
GISMO	I & II	MFG	COMPLETE COMPLEMENT OF SYSTEM & GSE PRESSURE TANKS (SMALL) ENGINE MASS COMPUTER STIMULI FOR LAUNCH & FLIGHT SIMULATION
WIND TUNNEL	VEHICLE	—	SCALE MODELS
RECOVERY	I & II	—	MODEL & FULL-SCALE
RETRIEVAL	I & II	—	MODEL & FULL-SCALE
FLIGHT	I & II	MILA	6 FLIGHTS (6 EXPENDABLE VEHICLES) (4 REUSABLE VEHICLES)

POST SATURN MARTIN

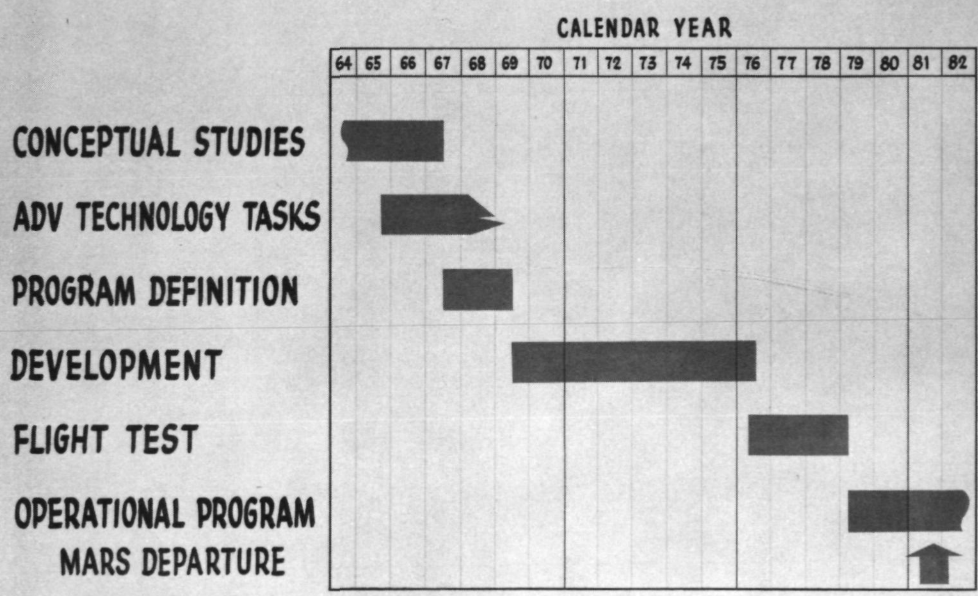
V. CONCLUSIONS AND RECOMMENDATIONS

CONCLUSIONS

- POST SATURN ECONOMICAL FOR "SMALL" SPACE PROGRAM
- "LARGE" PAYLOAD CAPABILITY DESIRED
- M-1/300K CONFIGURATION PREFERRED
- TOTAL RECOVERY / REUSE RECOMMENDED
- ADVANCED TECHNOLOGY EFFORT NEEDED

POST SATURN MARTIN

POST SATURN PROGRAM PLAN



POST SATURN MARTIN

V-2

RECOMMENDATIONS

- CONTINUE SUPPORT OF CURRENT TECHNOLOGY/ DEVELOPMENT PROGRAMS
- FURTHER CONCEPTUAL DESIGN OF BASELINE VEHICLE
- EVALUATE APPLICATION OF NEW CONCEPTS
- DETAILED DEVELOPMENT PLAN
- CONTINUE MISSION AND OPERATIONS ANALYSIS

POST SATURN MARTIN

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