CR-66352

STUDY OF THE INFLUENCE OF SIZE OF A MANNED LIFTING BODY ENTRY VEHICLE ON RESEARCH POTENTIAL AND COST

FINAL REPORT

Part I. Summary

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Prepared by

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ABSTRACT

This study presents data—based upon a developed logic, task definitions, vehicle criteria, system analyses and design, and concepts of operation and implementation—with which the usefulness and cost of an entry flight research program can be evaluated.

The study defines 52 specific research tasks of value in developing operational lifting body systems, primarily for near-earth missions. Parametric design and performance data are evolved within a matrix of 5 vehicle sizes (with 1, 2, 4, 6 and 8 men) and 4 boosters (GLV, Titan III–2, Titan III–5 and Saturn IB) for all flight phases, from launch to landing. The design studies include vehicle arrangements, weight, aerodynamic heating and subsystem details. Systems integration analyses yield both design data, subsystem tradeoffs, and development and operations plans; and they lead, in turn, to cost effectiveness analyses which become the primary basis for vehicle and program selection.

A 25-foot long, 3-man vehicle weighing 12 342 pounds is selected for a research program of 9 manned (plus 2 unmanned) flights. This vehicle performs the maximum number of tasks and affords the highest research value per unit cost and the lowest cost per unit of payload in orbit; the estimated program cost is $1 billion. A detailed preliminary design of this vehicle is accomplished, including layout drawings and descriptions of each subsystem to identify available hardware as well as future options. Modifications for secondary research objectives—rendezvous and docking and supercircular entry—are considered.

The study also includes a brief examination of 2 smaller unmanned vehicles as alternate approaches to reduce cost.

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May 1967

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MARTIN MARIETTA CORPORATION
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FOREWORD

This document is a part of the final report on a "Study of the Influence of Size of a Manned Lifting Body Entry Vehicle on Research Potential and Cost," conducted by the Martin Marietta Corporation, Baltimore Division, for the National Aeronautics and Space Administration, Langley Research Center, under Contract NAS 1–6209 dated April 1966. The final report is presented in eight parts:

- I Summary
- II Research Program Experiments
- III Flight Performance
- IV Candidate Entry Vehicle Designs
- V Systems Integration
- VI Research Vehicle Size Selection and Program Definition
- VII Selected Entry Vehicle Design
- VIH Alternate Approaches

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CONTENTS

INTRODUCTION

Lifting Entry Vehicles are now being considered for NASA and military missions. This interest arises from the unique maneuvering capabilities which lifting entry affords. In particular, the significant Glide Path Control of the Lifting Body yields a large landing footprint and, in turn, permits more frequent return from earth orbit to a precise landing point. The lifting body class of entry vehicles has already demonstrated its ability to land horizontally like a conventional aircraft.

Current flight research programs are addressing some of the fundamental technology questions associated with lifting entry. They are limited in scope, however, and will not provide all of the important research and operational information and experience required to optimize future mission systems. This study is intended to provide planning data from which the utility and cost of a Flight Research Program, using a possible manned Lifting Body Entry Vehicle, can be assessed.

Among recent analytical investigations, the NASA Flight Research Center sponsored two studies on a "Minimum Manned Lifting Body Entry Vehicle." These studies, which examined a low-cost vehicle for conducting limited manned entry research, were completed in January 1966. Subsequently, the NASA Langley Research Center initiated the broader study presented in this report.

OBJECTIVES

The present study defines specific entry research tasks and then, after comparing various approaches, identifies the best vehicle and crew size for conducting the research. While the two earlier studies (NAS 4–839 and NAS 4–840) were based on the M2–F2 configuration, this study concentrates on the HL–10 (fig. 1). The general conclusions of the present study are applicable to either configuration.

Table 1 paraphrases the objectives from the contract Statement of Work: (1) to identify technology problems in relation to future missions, with primary emphasis on the entry phase in returning from near-earth orbits, (2) to

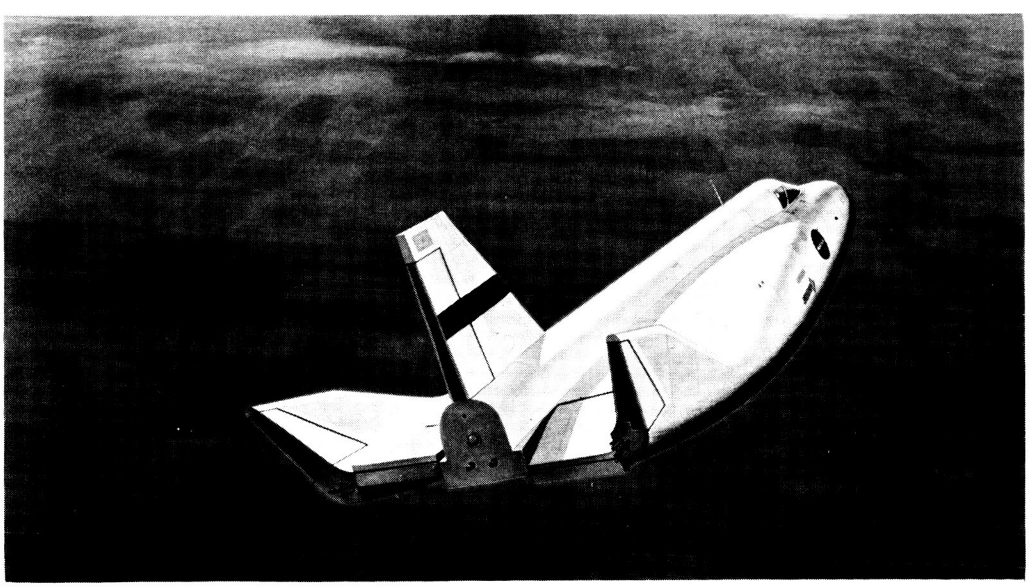


Figure 1. HL–10 Lifting Body Entry Vehicle

develop accurate design, development program, and cost information for five entry vehicles designed to carry 1, 2, 4, 6, and 8 men, respectively, (3) to select for each entry vehicle the most suitable launch vehicle from the specified Gemini Launch Vehicle, versions of the Titan III, and the Saturn IB, (4) to determine research accomplishment as a function of vehicle size, crew size, and flight program size, (5) to select the minimum entry vehicle size and launch vehicle combination that maximizes research value in relation to total program cost (note that only entry research at near-orbital speeds influences the entry vehicle selection), (6) to establish an optimum research flight test plan for the selected system, and (7) to identify modifications to the selected system that are necessary to conduct orbital research on rendezvous, docking, and crew and cargo transfer and entry research at supercircular velocities.

An important ground rule of the study is that the research vehicle, although not selected or outfitted as a mission vehicle, shall attempt to provide information normally acquired with a full-scale prototype.

TABLE 1.—STUDY OBJECTIVES

- Definition of Technology Problems (Near-Earth Missions)
- Accurate Weight, Cost, Development Schedule Estimates (f: Size)
- Launch Vehicle Selection (f: Size, Research Potential)
- Research Accomplishment (f: Size)
- Best Research Vehicle System
- Research Plan for Recommended System
- Growth Potential

STUDY PLAN

The study logic and phasing are shown in figure 2. During the first quarter, the primary task was to define the necessary research experiments. Design criteria were established, subsystem tradeoffs were conducted, basic performance data were generated and the Martin Marietta cost models were revised and tested. In the second quarter, the primary tasks were the design of the five candidate vehicles and a cost-effectiveness comparison of the vehicles in various flight programs. This quarter ended with selection of the best vehicle for conducting entry research. In the final quarter, the recommended vehicle was designed in more depth, and the associated development, flight program and cost were defined in detail.

DISCUSSION OF RESULTS

Experiment Identification

Figure 3 indicates the logic and steps for identifying and then defining flight research tasks necessary to optimize future mission systems.

The first step is to identify potential operational missions for manned lifting body entry vehicles in the HL–10 performance class. Eight different missions have been defined in sufficient detail to establish the key entry performance and environment parameters. Table 2 shows some of these parameters for five of the missions. The missions involving higher entry speeds are not shown because research in this regime is only a secondary objective. The principal point of table 2 is that very little variation occurs in the entry parameters for missions involving near-earth operations (excluding return from a synchronous orbit, shown for satellite repair and maintenance).

The next step in the initial study phase is to review current technology in relation to that required for future missions. This review, accomplished jointly by NASA Langley Research Center and Martin Marietta personnel, has revealed no crucial problems that would prevent the successful development of a manned, medium lift-to-drag vehicle using present technology. On the other hand, the conduct of a flight research program will provide useful information in areas normally investigated during a prototype phase. These areas include the full-scale aerodynamic

TABLE 2.—FUTURE MISSION CHARACTERISTICS

Mission	Entry conditions 400 000 ft (122 km): Velocity, V, fps (km/sec)	Flight path angle, γ, deg	Acceleration, g	Entry environment: Heat rate, $\dot{q}$, Btu/ft²-sec (kW/m²)	Total heat, q, Btu/ft² $\times 10^3$ (MJ/m²)	Dynamic pressure, Q, lb/ft² (kN/m²)
Space station support	$\approx$25 000 ($\approx$7.62)	-1 to -3	1 to 4	80 to 270 (908 to 3064)	70 to 150 (794 to 1702)	350 to 400 (16.8 to 19.2)
Satellite inspection	$\approx$25 000 ($\approx$7.62)	-1 to -3	1 to 4	80 to 270 (908 to 3064)	70 to 150 (794 to 1702)	350 to 400 (16.8 to 19.2)
Space observation	$\approx$25 000 ($\approx$7.62)	-1 to -3	1 to 4	70 to 270 (794 to 3064)	60 to 150 (681 to 1702)	350 to 400 (16.8 to 19.2)
Search and rescue	$\approx$25 000 ($\approx$7.62)	-1 to -3	1 to 4	70 to 300 (794 to 3405)	70 to 170 (794 to 1929)	350 to 400 (16.8 to 19.2)
Satellite repair and maintenance	25/35 000 (10.67)	-1 to -5	1 to 5	80 to 380 (908 to 4313)	70 to 200 (794 to 2270)	400 to 600 (19.2 to 28.7)

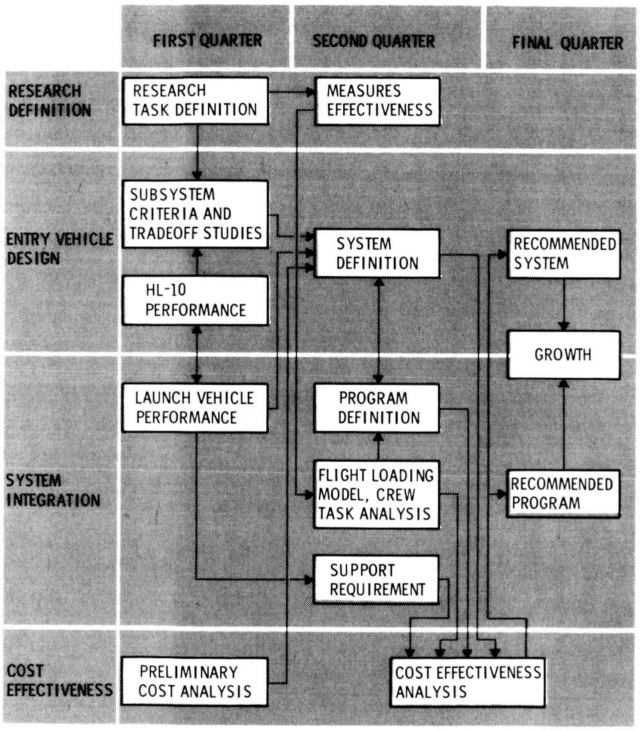


Figure 2. Study Flow

environment, operating procedures, and state-of-the-art advancement in subsystem development, all vital to obtaining efficient mission systems.

Fifty-two experiments have been defined in detail, including a technology assessment, justification for flight testing, the crew involvement, test procedure (with desired trajectory conditions), and vehicle resources requirements. The most appropriate entry flight conditions, when all experiments are considered, are shown in figure 4. These data have been generated by a new analog trajectory analysis technique.

Research task designations are indicated in table 3 along with the principal categories and example experiments. Twenty-seven of the 52 experiments are classified as confirmation or verification tests, 22 as state-of-the-art advancement tests, and three as pure research.

Candidate Vehicle Designs

The five candidate entry vehicles are configured to the minimum size capable of carrying 95th percentile crewmen in the quantities specified. The design criteria (table 4) reflect a conservative design approach which will ensure a flexible and safe research tool. For example, the heat shield design is based on all-turbulent

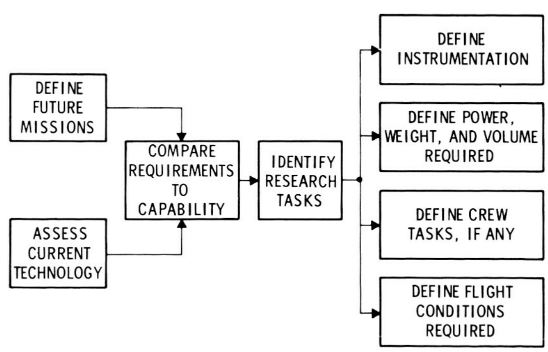


Figure 3. Research Task Definition Logic

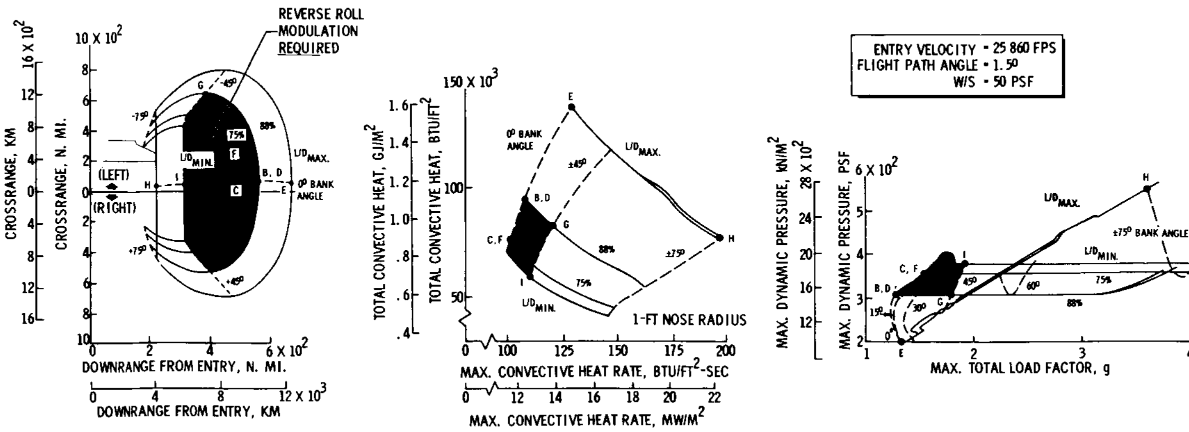


Figure 4. Entry Conditions

TABLE 3.—EXPERIMENT CATEGORIES

	Number
AV—Avionics	2
FC—Flight Controls	4
FM—Flight Mechanics	16
SM—Structures/Materials	18
GN—Guidance/Navigation	7
HF—Human Factors	2
PP—Propulsion	3
	52

Verification

AV–1 Antenna window material test
FM–4 Measure control effectiveness
SM–7 Ablator ascent heating—cold soak

Technology Advance

GN–5 Hypersonic entry guidance techniques
SM–8 Refurbishable heat shield demonstration
FC–2 Adaptive flight control system

Pure Research

FM–12 Boundary-layer survey
SM–16 Catalytic wall experiments

boundary layer heating conditions for the full maneuver capability of the HL–10. Launch abort capability throughout ascent, using full vehicle recovery by parachutes where vehicle size and volume permit (i.e., the vehicles sized for 4, 6 and 8 men), is a design requirement. The criteria also require redundant subsystems not only for crew safety but also for research mission success.

The resulting entry vehicle designs and the five crew arrangements are summarized in figure 5. Note the vehicle designation terminology

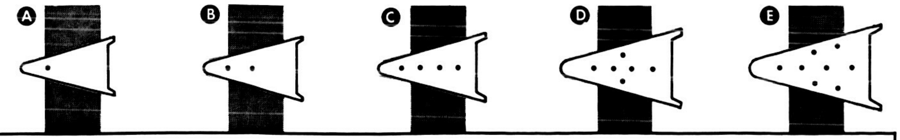
TABLE 4.—DESIGN CRITERIA

- Heat shield design—turbulent heating and complete footprint
- Max q = 1200 psf, 6-g pull-up
- 5-orbit system design capability—3 orbit mission
- Solid rocket motors—abort and deorbit
- Crew ejection seats or vehicle recovery chutes
- Horizontal landing capability—any prepared site
- Redundant systems—safety and mission success

introduced on this figure. The designation C/4 stands for a C-sized vehicle, 23.4 feet (7.13 m) long, with a four-man crew. On research missions, smaller crew complements have been considered and, therefore, a designation such as C/2 refers to a C-sized vehicle carrying only two crewmen.

Vehicle lengths vary from 20 to 26.4 feet (6.1 to 8 m) and the weights from 6943 to 13 754 pounds (3149 to 6239 kg). The wing loading is constrained on the D and E vehicles to 55 psi (81.8 kg/m²) to limit touchdown speeds to no more than 230 knots (118 m/sec). The total launch weight includes a simple conical adapter joined to the entry vehicle base and containing the abort and deorbit solid rockets.

The weight and arrangement information is based on (1) detailed heat shield and structural analyses, (2) layouts of point designs with high-degree optimization for the specific criteria, and (3) subsystem tradeoffs and performance evalu-



	A		B		C		D		E	
LENGTH, FEET (M)	20.0	(6.1)	21.3	(6.48)	23.4	(7.13)	25.0	(7.62)	26.4	(8.05)
SPAN, FEET (M)	14.27	(4.35)	15.17	(4.62)	16.7	(5.09)	17.84	(5.44)	18.84	(5.74)
AREA, SQUARE FEET (M^2)	143	(13.285)	162	(15.050)	196	(18.208)	224	(20.810)	249	(23.132)
ENTRY VEHICLE WEIGHT, LBM (KG)	6943	(3149)	8181	(3710)	10 283	(4662)	12 001	(5442)	13 754	(6238)
W/S, LBM/FT^2 (KG/M^2)	48.5	(236.0)	50.5	(246.5)	52.5	(256.0)	53.5	(261.5)	55.2	(269.7)
TOTAL WEIGHT, LBM (KG)	8096	(3672)	9479	(4299)	11 820	(5359)	13 727	(6225)	15 664	(7104)

Figure 5. Candidate Entry Vehicles

ations of flight-proven equipment, wherever possible.

The structural analyses covered six different structural concepts, using several combinations of materials, as well as the structural implications of launch dynamics.

Figure 6 shows the results of one of the analyses on the selected primary heat shield concept. The concept employs three elastomeric silicone ablators developed by Martin Marietta and tested on PRIME and the X–15. This reusable heat shield concept (fig. 6) has also been evaluated under another NASA contract (NAS 1–5253). In the concept, the ablator is contained in a honeycomb matrix bonded to a fiber glass substrate, which in turn is bolted to the pressure shell through a standoff area filled with microquartz insulation. The total heat shield weights are given for turbulent and laminar boundary layers as a function of vehicle size.

Typical of the subsystem tradeoffs performed during the parametric design study is that for controls actuation. In this evaluation, four different aerodynamic flap actuation and power system combinations were considered. Their total weights varied by as much as a factor of five. The selected approach, based on state of the art, reliability, and systems integration effects, is a dual-active hydraulic system. Design hinge moments and subsystem weights as a function of vehicle size are shown in figure 7.

Vehicle weight breakdowns are summarized in figure 8. More than 40 percent of the entry

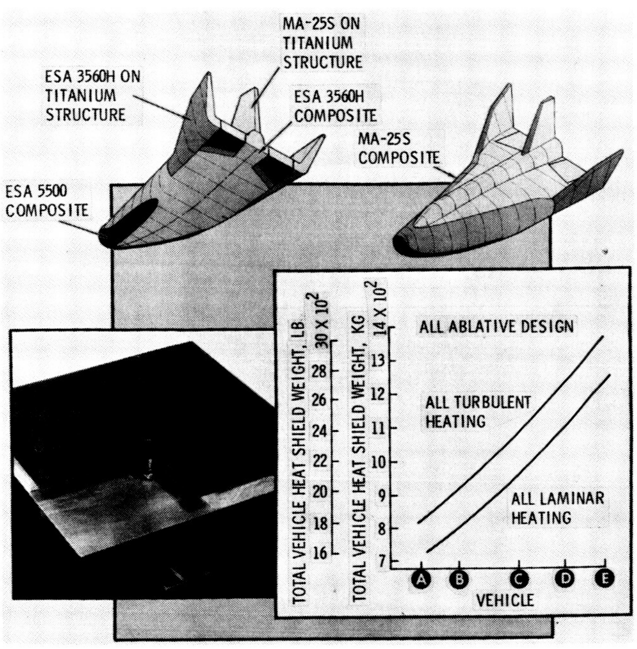


Figure 6. Heat Shield Weights and Materials

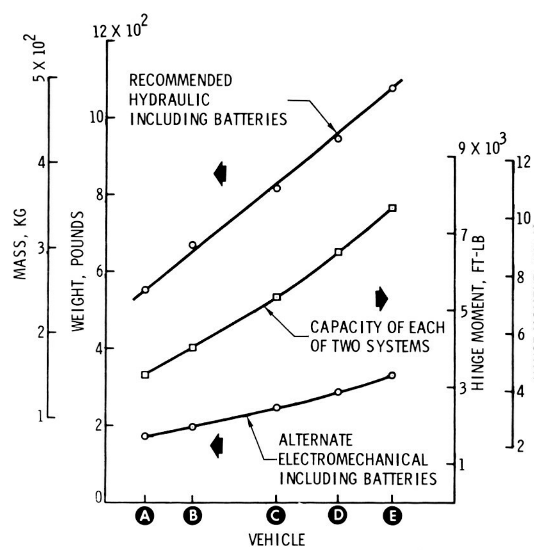


Figure 7. Controls Actuation Subsystem

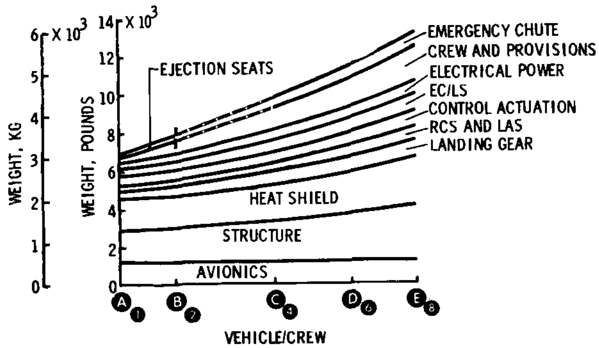


Figure 8. Weight Breakdown Summary

vehicle weight occurs in the heat shield and structure. The hydraulic actuator systems and their associated electrical power supply account for eight percent of the entry vehicle weight.

One important design feature which shows a significant variation with vehicle size is the pilot visibility through the canopy (fig. 9). All five vehicle designs have a single canopy shape for which wind tunnel data are currently available (although modified by a conical windshield). Satisfactory direct visibility during approach and touchdown is not available in the A and B vehicles with this canopy design.

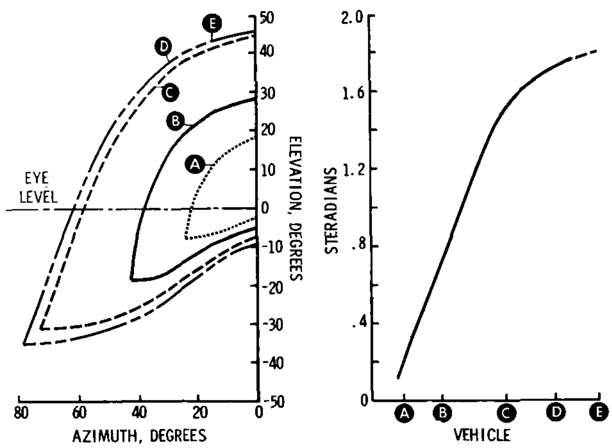


Figure 9. Effect of Size on Visibility

The weight and volume available for experimental equipment (fig. 10) are resources of fundamental importance to the cost effectiveness analyses. While the A and B vehicles have a small amount of excess volume, the larger vehicles can provide volume for experiments only by off-loading crew members. The analysis uses

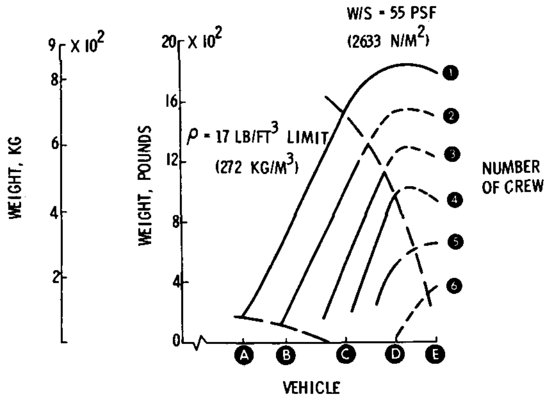


Figure 10. Weight Available for Experiments

a practical equipment packaging density of 17 pounds per cubic foot (272 kg/m³). The flattening of the curves for the D and E vehicles arises from the wing loading constraint referred to previously.

Finally, launch vehicle payload capabilities are plotted against entry vehicle size in figure 11. The analysis shows that the Titan III with two-segment solid rockets is more than adequate for all of the entry vehicles. However, Air Force plans no longer include this version. Consequently, the Titan III with five-segment solid rockets has been selected as the most suitable launch vehicle for this entry research program.

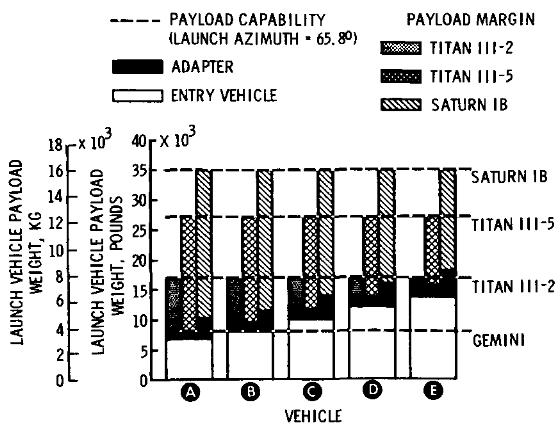


Figure 11. Launch Vehicle Suitability

Research Flight Program Formulation

One of the major objectives of this study is to compare the research potential of the candidate entry vehicles. Such a comparison requires the measurement of research potential as a function not only of vehicle size but also of crew size and the number and types of flights in addition to the particular experiments loaded on any given flight.

A numerical measure of research value for each of the 52 experiments was derived using the definition shown in table 5. The intrinsic values result from an application of the Law of Comparative Judgment in which 11 engineers, including both NASA and Martin Marietta participants, compared each of the 52 experiments to every other experiment. Table 6 shows the ranking for all of the experiments in a relative value scale ranging from 1 to 237; it is the result of a statistical analysis of the paired comparisons. Seven tasks require crew participation to yield any value; 17 can be performed better with crew participation; and 28 of the experiments can be completely automated.

TABLE 5.—RESEARCH VALUE DEFINITION

$$V_L = V_0 \Sigma [V_I] \times [P_I]$$

Variable	Name	Obtained from
V_0	Intrinsic value	Law of Comparative Judgment
V_I	Informational value	Information theory (flights, entry conditions)
P_I	Expectancy of obtaining informational value	Probability theory

To determine the total research value of a given multiflight program, the delivered value of each experiment is derived from the intrinsic values (table 6) modified for the number of repetitions, the particular test conditions and the estimated reliability of the data system and mission success (table 5).

The procedure for optimizing research task loading on each of the flights requires inputs for: (1) the kinds of entry trajectories (fig. 4), (2) the vehicle payload resources (fig. 10), (3) the experiment values (table 6), (4) the crew time available for conducting experiments, and (5) the experiment loading constraints. The two typical experiments summarized in table 7 are representative results of the task analyses conducted to establish crew requirements during each phase of flight. Typical examples of the constraints established for order of precedence in loading experiments are shown in table 8.

The methodology used to assign entry research tasks to a given flight plan is designed

TABLE 6.—RESEARCH TASK RANKING AND INTRINSIC VALUE

Rank	Code	Task description	Value	Crew
1	SM–1	Ablative heat shield performance and analysis correlation	237	U
2	FM–8	Measure heat rate distribution	216	U
3	FM–3	Evaluate flying qualities	213	N
4	FM–2	Evaluate aerodynamic characteristics	213	B
5	FM–7	Measure pressure distribution	184	U
6	FM–4	Measure control effectiveness	145	B
7	GN–4	Inertial navigation error propagation	145	U
8	GN–5	Hypersonic entry guidance techniques	143	N
9	FM–13	Ablation effects on hypersonic aerodynamics	139	U
10	GN–1	Primary navigation and guidance performance	134	B
11	EV–2	Evaluate reuse capability and refurbishment	138	U
12	FC–1	Flight control system evaluation	133	N
13	FM–5	Measure elevon shock interaction	139	U
14	SM–6	Movable surface heat shield design problems	120	B
15	SM–2	Ablative heat shield joints	120	U
16	SM–8	Refurbishable heat shield demonstration	114	U
17	FM–17	Hypersonic boundary layer transition	115	B
18	GN–6	Terminal navigation and guidance techniques	99	N
19	FM–14	Viscous effects on lift and drag	97	U
20	GN–2	Backup guidance performance	84	U
21	SM–17	Ascent static and dynamic response criteria	80	U
22	SM–7	Ablator ascent heating, cold soak, and entry effects	79	U
23	SM–5	Insulation cavity pressure	79	U
24	SM–9	Radiation heat shields	77	B
25	SM–3	Ablator materials comparison	75	U
26	GN–3	Autonomous orbital navigation	74	B
27	FM–6	Measure entry stability and control at various cg locations	75	B
28	FC–2	Adaptive flight control system	71	B
29	FM–12	Boundary layer survey	71	U
30	FC–3	Digital flight control mechanization	67	B
31	GN–7	Air data measurements	62	U
32	SM–14	Afterheat effects	59	U
33	FC–4	Flight control actuation experiment	60	B
34	FM–15	Measure plasma thermodynamics	59	U
35	PP–3	Landing assist propulsion	60	N
36	HF–2	Crew biomedical and performance monitoring	55	B
37	SM–10	Radiative and radiative-to-ablative heat shield joints	52	U
38	SM–12	Ablator overcoat on radiative heat shields	52	U
39	PP–2	Jet exhaust/vehicle boundary layer interactions	53	B
40	SM–13	Heat shield instrumentation sensor studies	43	U
41	PP–1	Jet impingement effects and analytical correlation	41	B
42	SM–11	Active and passive structural cooling	38	B
43	SM–16	Catalytic wall experiments	35	U
44	AV–2	Satellite communication experiment	32	B
45	HF–1	Pilot control and vehicle landing after zero g	31	N
46	FM–16	Effects of electrophilic fluid injection	25	U
47	SM–15	Transpiration cooling system	22	B
48	FM–9	Measure gas cap radiation heat transfer	19	U
49	AV–1	Antenna window material test	13	U
50	FM–18	Use of ventral antenna to alleviate RF blackout	13	B
51	SM–18	In-flight heat shield repair	5	N
52	FM–19	Synergetic maneuver simulation without thrust	1	B

N—Necessary; B—Beneficial; U—Unnecessary

TABLE 7.—CREW TASK LOADING

Mission phase	Time, min	Crew utilization ratio			
		Basic	GN–5	FM–6	Total
Ascent	0-8	0.5	0	0	0.5
First orbit	8-98	0.5	0.1	0	0.6
Second orbit	98-188	0.5	0.1	0	0.6
Third orbit	188-229	0.4	0.1	0.2	0.7
Deorbit	229-257	0.6	0.2	0	0.8
400-200k feet	257-260	0.8	1.0	0.2	2.0
280k to pull-out	260-262	0.9	1.0	0.2	2.1
Pullout to 200k	262-279	0.7	0.8	0.4	1.9
200k to Mach 6	279-282	0.8	0.5	0.3	1.6
Mach 6 to Mach 2	282-285	0.8	0	0.2	1.0
Mach 2 to Mach 0.8	285-287	1.0	0	0	1.0
Approach & land	287-289	1.0	0	0	1.0

to obtain maximum total value from the research accomplished. The task assignment model is a development and implementation of linear programming techniques. It contains constraint equations which permit consideration of such conditions as research task precedences, concurrences, exclusions and resource sharing. In this study, the solution to problems with up to 15 flights and 52 research tasks has involved an input matrix of 6005 variables, 350 constraint equations, and 59 326 entries.

TABLE 8.—TASK LOADING CONSTRAINTS

Task	Loading constraints				
	Must go with	Do not do with		Do first	
GN–1	—	—	—	—	—
GN–2	—	—	—	(1) GN–1	—
GN–3	GN–4	—	—	(1) GN–1,	(1) GN–2
GN–4	—	—	—	(1) GN–1,	(1) GN–2
GN–5	GN–4	GN–1,	GN–2	(1) GN–1,	(1) GN–2

The results of the flight loading analysis are shown in figure 12. The value of research performed in programs varyings from 5 to 11 flights is plotted against vehicle size. Every point on these curves has been optimized by trading off the number of crewmen against the amount of experimental equipment carried. The circled numbers shown in figure 12 by each break in the curves represent the optimum number of crewmen on research flights for an entry vehicle of given size. The limit line at 3400 is the theoretical maximum value of research if all tasks could be loaded and tested under optimum entry conditions and if no failures were experienced.

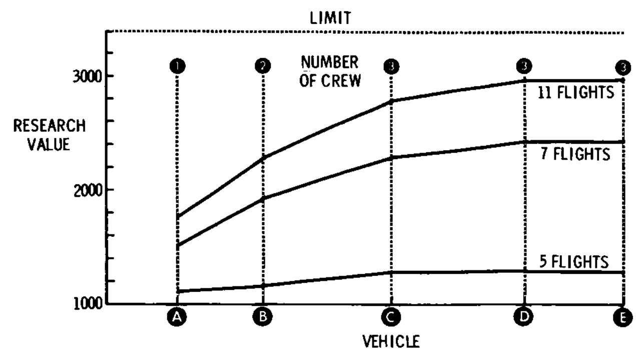


Figure 12. Program Value Versus Vehicle Size

Figure 13 shows that some experiments will be omitted from programs of 9 or less flights with, for example, a D-size vehicle. With the smaller vehicles, this omission is more severe.

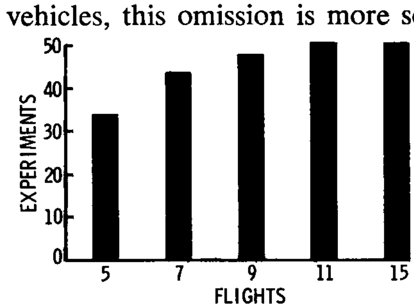


Figure 13. Number of Experiments Versus Program Size

Parametric Cost Estimates

To generate the cost data required in this study, a technique utilizing cost models has been employed. In general, costs produced with this technique are substantially higher than (approximately two times) those developed with conventional pricing techniques. The reason is that the mathematical models contain cost estimating relationships based on costs reported after program completion. These costs include allowances for all program changes, stretchouts, and related sources of increased cost, normally experienced on real programs, that are difficult if not impossible to predict prior to program go-ahead.

Two cost models, both based on historical program data, have been utilized. Their general characteristics are summarized in table 9. SSCOM (Space System Cost Model) was used during the first two quarters of the study to establish program cost tradeoffs with entry vehicle size. In the final quarter, COCOM (Coincident Cost Model) was used to generate detailed costing information on the selected system. The latter model requires much more detail in the input data. Total program costs estimated by these two models agree within four percent.

Inputs to the cost models include vehicle subsystem weights and complexity, number of flight vehicles and ground test articles, number of launch vehicles, number and scope of refurbishment cycles, and the program span. Typical input data are shown in figure 14.

TABLE 9.—COST MODELS

Characteristic	SSCOM	COCOM
Input hardware entries	18	61
Input program entries	10	179
Detail output costs	60	318
Summary output costs	9	4
Fiscal funding output costs	No	Yes
Number of cost equations	60	318
Historical data sources	Gemini, Apollo, Mercury	PRIME, Gemini, X–15, Apollo, Mercury, Cost Studies
Cost equation form	$Wc_i\left(\frac{W_i}{W_s}\right)^{\beta}$	$N^L\left(1+\alpha\right)^{\Delta y}$

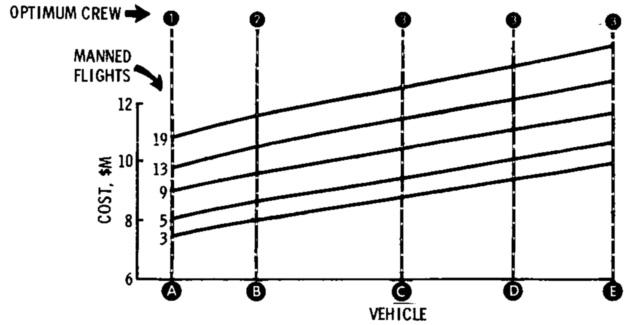


Figure 15. Effect of Entry Vehicle and Program Size on Cost

The costing results of the parametric phase of the study are shown in figure 15. As expected, the larger vehicles cost more than the smaller ones; however, for a given number of flights, an increase in vehicle size from A to E increases program cost by only 30 percent. No real discontinuities appear in these curves because all of the entry vehicles employ the same Titan III launch vehicle with five-segment solids.

Vehicle and Crew Size Selection

Selection of the best vehicle and crew size for conducting the lifting body research program would be relatively straightforward if specific program budgetary constraints or total program research value goals were established. Such conditions, however, do not exist at this time. Selection also would be simple if a definitive operational requirement for a mission system existed. Again, this is not currently the

RESEARCH FLIGHTS	NEW ENTRY VEHICLES		
	DEVELOPMENT TEST	UNMANNED FLIGHT	MANNED FLIGHT
5	5	2	2
7	5	2	3
9	5	2	4
11	5	2	4
15	5	2	5

NOTE: DEVELOPMENT TEST VEHICLES INCLUDE: (1) PAD ABORT, (2) q_{MAX} ABORT, (3) STRUCTURE, (4) DYNAMIC, (5) SYSTEMS

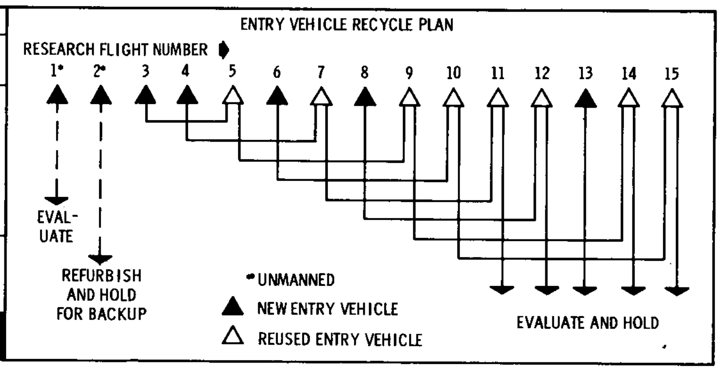


Figure 14. New Vehicle Requirements Versus Program Size

case. Therefore, a number of different selection criteria must be considered.

The selection criteria actually used include the system and program combinations which yield the least cost, the most research value, the most research value per dollar, and the most margin for incorporating experiments not yet identified. The criteria also include other considerations related to design, operation, and system growth. This selection approach identifies the smallest entry vehicle which is cost effective, practical, and flexible in its research applications, without totally disregarding its capability in a mission application. If, at a later date, the ground rules for vehicle selection should change, this study provides the basic data for determining the cost and program effects of changing vehicle size.

Figure 16 summarizes cost effectiveness as a function of vehicle size. These data are presented for an 11-flight program (two unmanned and nine manned), which has proven to be the most cost effective number of flights. The research value increases with increased vehicle size, up to the D vehicle, where it then holds a constant level. The corresponding optimum number of crewmen is indicated. Program cost increases linearly with vehicle size. Value per dollar reaches its maximum level for the C vehicle, holds constant for the D vehicle, and then declines. The traditional parameter of "dollars per useful pound in orbit" is minimum for the D vehicle. Finally, the average weight margin for unidentified experiments is maximum for the D vehicle.

Table 10 summarizes the analysis of other influences upon vehicle selection. In the chart, "U" is unsatisfactory and "M" is marginal; uncoded areas are satisfactory. Among essential considerations, the table shows that a one-man crew in any of the vehicles is not capable of participating in important experiments. Direct visibility available to the pilot in the A and B vehicles during approach and landing is unacceptable. Many important experiments cannot be included on an A vehicle, and several cannot be included on the B vehicle (the actual numbers vary with the number of flights).

TABLE 10.—OTHER CONSIDERATIONS

	Essential			Desirable		
Vehicle	Crew loading	Visibility	Experiment coverage	Vehicle abort	Tunnel	Mission application
A	U	U	U	M	U	U
B	U*	U	M*	M	U	U
C	—	—	—	—	U	M
D	—	—	—	—	—	—
E	—	—	—	—	—	—

* With 1-man crew

Among the desirable considerations in table 10, abort recovery of the whole entry vehicle appears to be marginal for the A and B vehicles because of inadequate space for parachute stowage. This means that crew safety is compromised. Also, a tunnel in the vehicle base, sized to accommodate an astronaut even in an unpressurized suit (assuming an airlock in the adapter module), cannot be fitted into the A, B and C vehicles. Lack of a tunnel is a serious

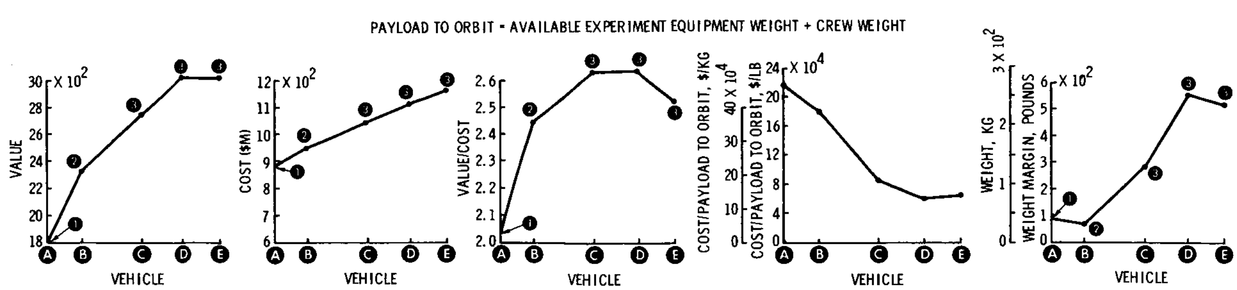


Figure 16. Cost Effectiveness Summary

handicap in extravehicular activity and rendezvous and docking experiments. Finally, the A and B vehicles are too small for application in a logistics ferry operation, and the C vehicle is marginal at best.

This analysis of cost effectiveness and vehicle capabilities, as shown in figure 16 and table 10, is the basis for selecting the D/3 configuration as the best vehicle and crew combination to conduct the entry research program.

Best Research Vehicle

The selected D/3 vehicle is 25 feet (7.62 m) long, 17.8 feet (5.43 m) in span, and 12.4 feet (3.78 m) high (fig. 17). The entry vehicle weighs 12 342 pounds (5598 kg). Its wing loading is 55 pounds per square foot (2.63 kN/m^2). The D/3 vehicle has 75 cubic feet (2.12 m^3) available for experimental equipment and can carry a maximum operational crew of six.

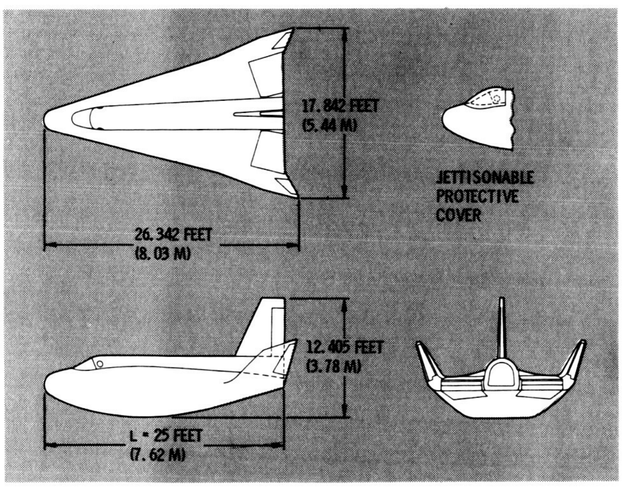


Figure 17. D/3 Vehicle Three View

The cutaway view (fig. 18) and the inboard profile (fig. 19) show the internal arrangement. The primary heat shield is all-ablative (see fig. 6) with removable panels for refurbishment (except on the cooler portions of the fins and base). The panels can be replaced by experimental ablative and radiative panels, including active cooling and plumbing. Space has also been provided for the much thicker heat shields associated with supercircular entry. The body shell structure is made of welded 2219–T6 aluminum alloy. The pressurized compartment starts at the nose gear bulkhead and ends just forward of the elevons. The main entrance hatch is immediately aft of the canopy bulkhead. To minimize tail weight, the structure of the center and tip fins and elevons is titanium with an 800° F (700° K) capability.

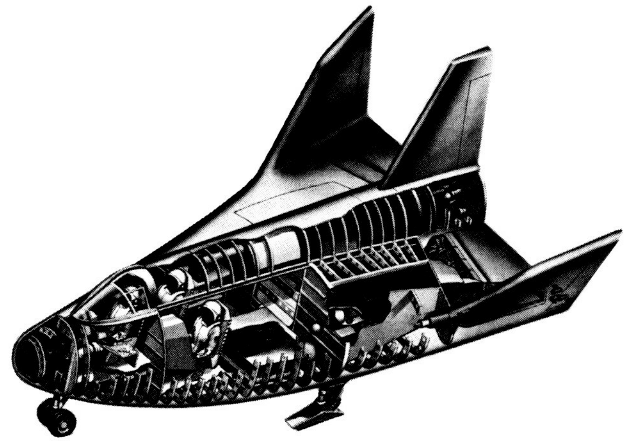


Figure 18. D/3 Vehicle Cutaway

The main landing gear configuration is a wheel embedded in a wire brush skid to minimize pitch acceleration on initial contact and to reduce the resulting body loads. The four emergency recovery parachutes are located above the main gear wells. A drag chute in the base provides both slideout braking and stabilization.

A jettisonable ablative cover protects the pilot's canopy during entry at flight speeds above Mach 2. A side porthole in the cover permits external viewing during ascent and while in orbit.

Development of the crew compartment arrangement is based largely upon the use of the spatial mockup shown in figure 20. A fold-back pilot seat permits access on the ground and crew position exchange in space. Emergency egress is available through both the main hatch and the blowout windshield.

The primary 100-percent oxygen, 5-psia (1.66 N/m^2) life support system is backed with a 3.5-psia (1.16 N/m^2) suit pressurization sys-

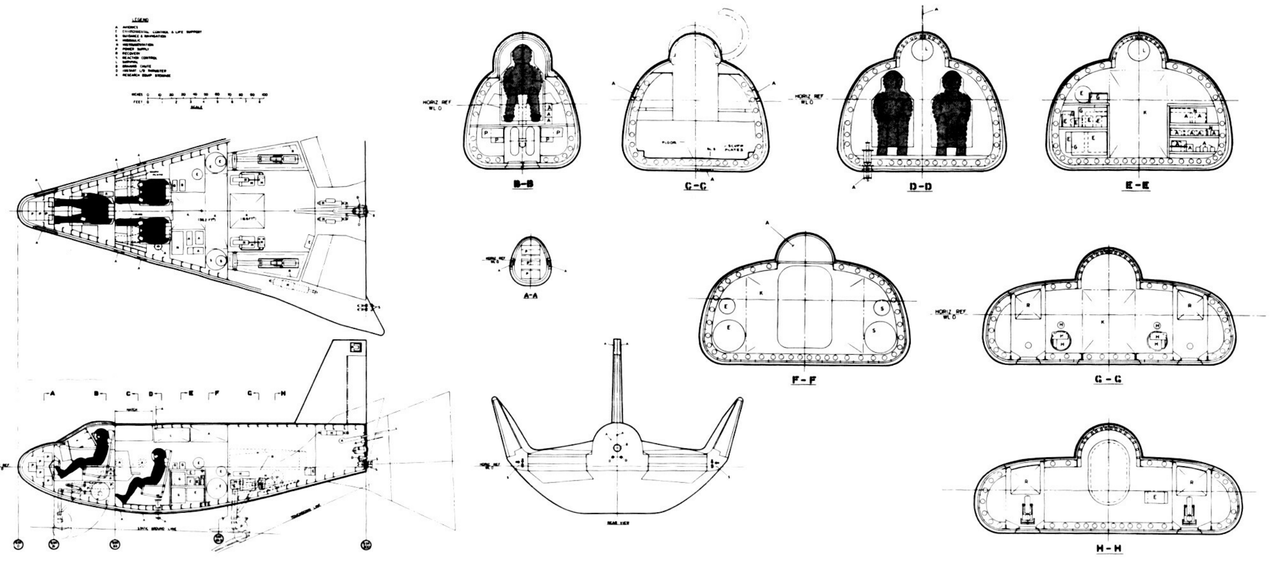


Figure 19. D/3 Vehicle Inboard Profile

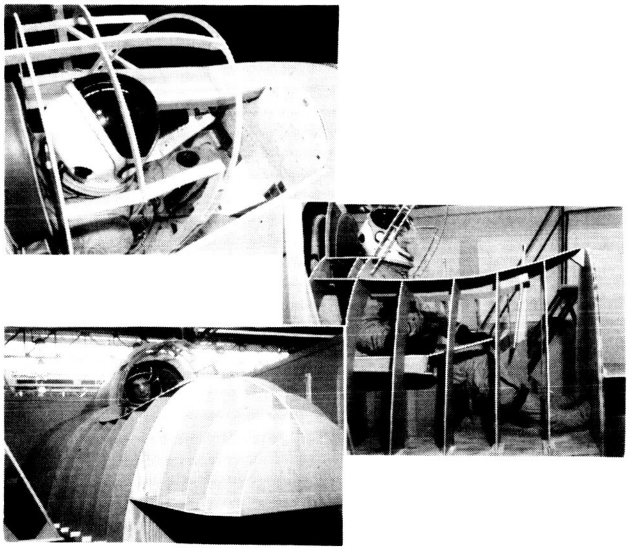


Figure 20. D/3 Vehicle Crew Station Mockup

tem. Cold plates with redundant loops are used for equipment cooling. Heat is rejected from the vehicle by water evaporation.

The primary navigation and guidance subsystem employs an inertial platform, computer, and horizon scanners. Backup guidance is provided by a strapdown inertial system. The simple guidance logic is suitable not only for manual control and pilot monitoring of automatic control but also for switchover from the primary to the backup system if a malfunction occurs during entry. Terminal navigation and guidance is based on ground-generated commands.

The electronic flight control subsystems uses pilot fly-by-wire inputs and three parallel channels per axis, with automatic switching through comparator circuits.

Attitude control in space is maintained by a hydrogen peroxide reaction control system with redundant valving. The landing assist rocket, affording 1600 pounds (7.1 kN) of thrust, is integrated with this system. Aerodynamic surfaces actuated by redundant hydraulics using a silver-zinc battery power supply, provide attitude control in the atmosphere. The system is sized for worst abort dynamic pressures.

The data handling subsystem, designed for 2048 data measurements, uses remote multiplexers, central control units, S-band transmitters, and tape recorders. Very high frequency voice, ultrahigh frequency command, and C-band tracking functions are all provided.

The deorbit and abort propulsion system consists of four spherical solid-rocket motors mounted inside the conical adapter, which supply a total velocity increment of 570 fps (178.7 m/sec). Multiple redundancy is available for normal deorbit.

Table 11 shows the degree to which the selected design complies with the Statement of Work guideline for making maximum use of available components and technology. Table 12 presents a group weight statement. Note that no ballast is required if a one-percent aft movement of the center of gravity proves to be acceptable.

TABLE 11.—SUBSYSTEM STATUS SUMMARY

Subsystem	Existing Hardware, %	Existing Technology, %	Flight experience*
Heat shield	NO	80	PR
Structure	NO	100	PI
Crew system	40	80	Gem, Ap, PI, HL–10
Landing gear	NO	100	M2-F2, HL–10, X–15
Recovery	NO	100	Mer, Gem
Flight control actuation	NO	80	PR
Electronic flight control	50	90	PI, PR
Communication	90	100	Gem, Ap
Guidance & navigation	90	100	PR, F111, OGO
Instrumentation	60	90	Ap, Gem, PR
EC/LS	70	100	Ap
Electrical power	90	100	Ap, PR, Gem
RCS & LAS	90	100	LM sim
Deorbit/abort	NO	100	Mer, Gem

* PR—PRIME, PI—PILOT, Gem—Gemini, Ap—Apollo, Mer—Mercury

TABLE 12.—D/3 WEIGHT STATEMENT

	lbM	kg
Entry vehicle	**12 342**	**5598**
Structure	2735	1241
Heat shield	2710	1229
Crew & provisions	823	373
Display panels	206	93
Electrical system	354	161
Environmental system	498	226
Guidance, navigation & comm.	503	228
Instrumentation	625	283
RCS & LAS	350	159
Surface controls	946	429
Landing gear	555	252
Emergency chutes	678	307
Ballast	308	140
Test equipment	1051	477
Adapter	**1963**	**890**
Structure	510	231
Deorbit—abort propulsion	1065	483
Electrical	288	131
Miscellaneous	100	45
Launch weight	**14 305**	**6488**

Figure 21 shows the D/3 vehicle mounted on the selected Titan III launch vehicle with five-segment solids.

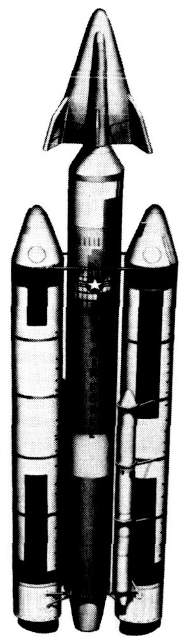


Figure 21. Launch Configuration—D/3 Vehicle and Titan III

Flight Operations

A nominal three-orbit mission has been selected to provide time for conducting such experiments as GN–3 (autonomous orbital navigation), SM–7 (ablator cold soak) and SM–18 (inflight heat shield repair). For safety, all systems are designed for a five-orbit duration. The selected nominal launch azimuth of 65.8 degrees provides good tracking and communications coverage with existing networks. It also permits return to Edwards Air Force Base, the primary landing site from the second through the fifth revolution without large crossrange maneuvers. Return is possible to Eglin Air Force Base, as an alternate landing site, on the first through the fourth revolution. The selected elliptic orbit with a perigee of 80 nautical miles (148.2 km) and an apogee of 200 nautical miles (370.4 km) minimizes ascent abort acceleration and heating conditions and deorbit impulse requirements. The resulting mission profile, including the time coverage of ground stations, is shown in figure 22. The inset shows that for a 09:00 a.m. launch, landing operations are available at both sites in daylight on three orbits and at one site on the first and fifth orbits.

The design environment as well as the operational crossrange available for mission planning is derived from extensive open- and closed-loop trajectory analyses conducted during the study. Data from these analyses also serve in evaluating navigation and guidance accuracy in the presence of simulated three-sigma perturbations. Table 13 shows typical results from the latter. Autonomous inertial navigation from deorbit until acquisition by the landing site tracking radar produces a vehicle position uncertainty that increases with time. The longest duration trajectory (3600 seconds) results in the errors shown for three-sigma initial conditions. The terminal guidance scheme, however, which involves commands generated on the ground using the selected reference trajectory guidance scheme, is able to correct the residual errors existing at the end of the autonomous phase. These errors are reduced to 100 yards (91.4 m) or less at flare initiation. As a result, the selected system and operations provide reasonable crew safety even under poor weather or lighting conditions.

TABLE 13.—GUIDANCE ACCURACY

Autonomous Phase (deorbit to Mach 5)*

Maximum initial error source	Navigation error at Mach 5, n. mi	(km)
Alignment $\Delta\phi_Y = 18$ min	12.1	(22.4)
Velocity $\Delta X = 8$ fps (2.4 m)	14.1	(26.1)
Position $\Delta Y = 0.5$ n. mi. (0.9 km)	6.7	(12.4)
RSS total (all sources)	24.6	(45.6)

Terminal Phase

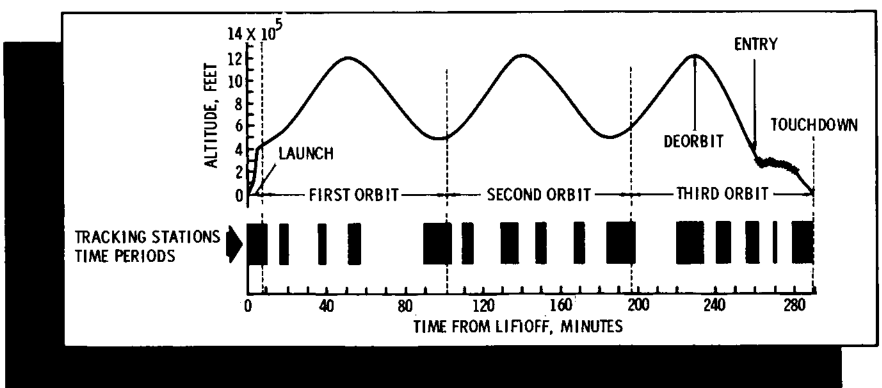
Off-nominal conditions	Error at flare initiation, ft (m) Lateral	Range	
$\Delta C_L = +14\%$	−2 (−0.6)	134	(40.7)
$\Delta\rho = +60\%$ (above 1959 ARDC)	−3 (−0.9)	182	(55.6)
Headwind = 100 fps (30.5 m/sec)	−2 (−0.6)	−304	(−92.6)

* Data supplied by Autonetics

Program Plan and Cost

A key task of the study is to maximize research value per dollar for the selected system. This has been accomplished by continuing the flight loading analysis, after determining that an 11-flight program is the most cost effective, to establish clearly the best sequence of flights. The optimum order of flights (table 14) results in a program value of 3006, a two percent gain over the D/3 value in figure 12.

The refurbishment plan and the overall program plan are fundamental inputs to the cost estimation procedure. A summary of the selected



LANDING SITE	PARAMETER	ORBIT NUMBER 1	2	3	4	5
PRIMARY: EDWARDS AFB	CROSSRANGE, N. MI. LANDING TIME, HR:MIN.		250 9:26 A.M.	10 11:00 A.M.	10 12:34 P.M.	250 2:08 P.M.
ALTERNATE: EGLIN AFB	CROSSRANGE, N. MI. LANDING TIME, HR:MIN.	168 10:06 A.M.	320 11:40 A.M.	192 1:14 P.M.	192 2:48 P.M.	

NOTE: 9:00 A.M. ETR LAUNCH
$i = 35.7°$
LOCAL MEAN SOLAR TIMES

Figure 22. Mission Profile

TABLE 14.—OPTIMUM ORDER OF RESEARCH FLIGHTS

Flight sequence	Condition code	Entry condition
1	A	High altitude abort (unmanned)
2	B	High total heat (unmanned)
3	C	Nominal entry
4	C	Nominal entry
5	C	Nominal entry
6	F	Medium crossrange
7	F	Medium crossrange
8	F	Medium crossrange
9	G	High crossrange
10	G	High crossrange
11	I	High airloads, low downrange

refurbishment approach is presented in table 15. To restrain both time and cost, this approach features the in-place conduct of as much subsystem functional testing as possible. Figure 14, presented earlier, shows that five ground test articles and a total of six flight vehicles are required to support an 11-flight program.

Figure 23 presents the overall program plan, assuming nominal pacing. The program, including a six-month Phase B (which may be unnecessarily long), spans 59 months. The plan includes a Little Joe II flight to demonstrate the abort and emergency recovery systems and an extensive air launch program with a B–52. Both free-glide and liquid rocket-powered flights are employed for crew training and proficiency maintenance and for evaluation of performance and flying qualities at transonic and subsonic speeds. A new pylon is required for this program because of configuration differences with present air-launch vehicles.

The design and program information, including limited new facility requirements for the selected system, is reflected in the COCOM-generated program costs summarized in figure 24.

TABLE 15.—REFURBISHMENT SUMMARY

Subsystem or assembly	Percent of subsystem recycled: Test in place	Remove & bench test	Remove & refurbish	Remove & replace
Structure	97	0	0	3
Heat shield	0	0	0	100
Control surfaces	60	↑	0	40
Landing gear	85	|	0	15
Electronic flight control	100	Bench	0	0
Guidance & navigation	100	tested if	0	0
Hydraulics	95	out	0	5
EC/LS	95	of	3	2
Reaction control	98	toler-	0	2
Electrical power & control	85	ance or malfunc-	9	6
Communications	78	tioning	2	20
Instrumentation & displays	95	|	0	5
Crew support	75	↓	5	20
Emergency recovery	10	0	0	90
Visibility	100	0	0	0

Only the height of the bars has significance in this costogram. The overall cost of approximately one billion dollars is composed of 47 percent nonrecurring and 53 percent recurring. The required fiscal funding allocations are shown in table 16.

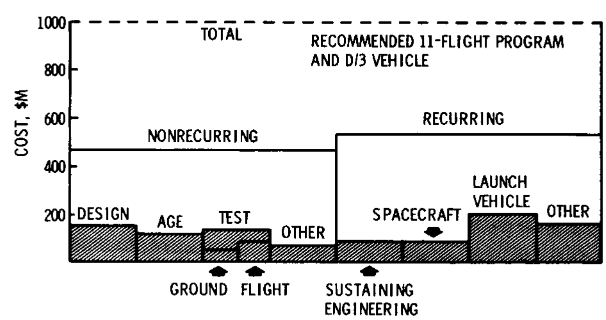


Figure 24. Costogram

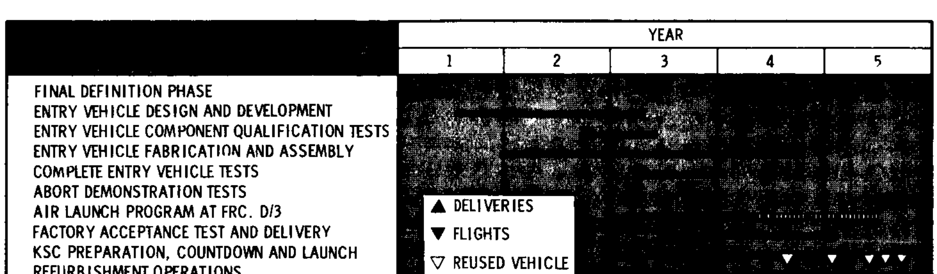


Figure 23. Program Plan

TABLE 16.—FISCAL FUNDING REQUIREMENTS

	1968	1969	1970	1971	1972
Nonrecurring	98	326	46		
Recurring		49	168	197	119
Total ($M)	98	375	214	197	119

Growth for Secondary Objectives

System selection is based principally upon the ability to conduct entry research at or below orbital velocity. However, one of the final study requirements is to determine those modifications necessary to the selected vehicle to enable it to conduct orbital and supercircular entry research.

Orbital research is defined as experiments in rendezvous, docking, and crew and cargo transfer. For such research, the only significant change required in the entry vehicle is the addition of an aft hatch. Other mission provisions are incorporated in a special cargo and docking module and a jettisonable propulsion module for major plane change and phasing corrections (fig. 25). An aft-facing crew station and window are provided for control of the docking phase. The cargo module is sized to use the full payload capacity of the selected Titan III launch vehicle. Mission duration is extended, in this case, to 16 orbits to permit multiple practice operations; necessary additions to the environmental control and power system are contained in the new modules.

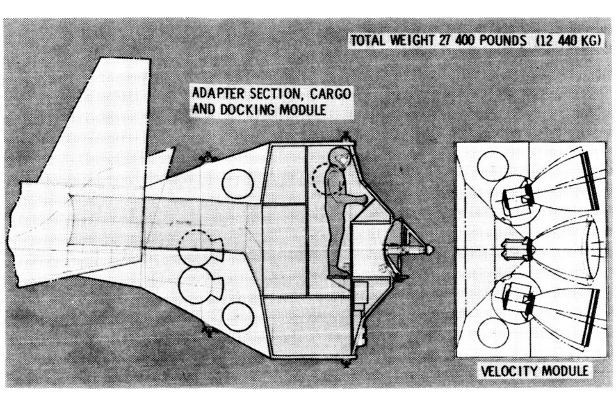


Figure 25. Rendezvous and Docking Experiment

Supercircular entry velocity research is very limited with vehicles in the weight and mission duration class considered in this study and with the available launch vehicles. The conceptual design of a velocity module for the D configuration, using a Saturn IB launch vehicle, is shown in figure 26. The overall propellant mass fraction, including entry vehicle propulsion as well as the velocity module (with the pressure fed, first stage engines from the Lunar Excursion Module), is only 0.47. The maximum entry velocity which can be achieved from near-earth orbits is about 29 000 fps (8.84 km/sec). This speed may be high enough to conduct useful research on convective heating at high Reynolds numbers, but it is marginal for useful experimentation on radiative heating. In any case, changes are required in the entry vehicle: the heat shield must accommodate higher heating rates, the control actuation and electrical power must account for higher hinge moments, and the guidance and control logic must be modified.

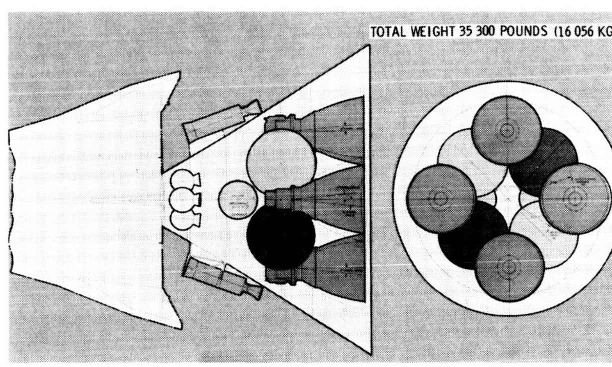


Figure 26. Velocity Module for Saturn IB

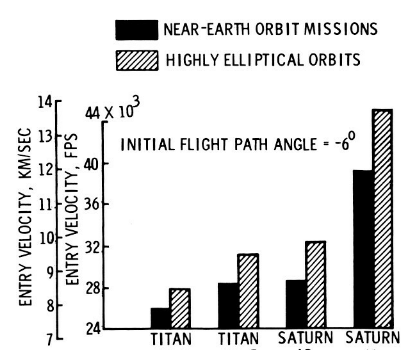


Figure 27. Entry Vehicle Potential

Additional abort separation rockets are needed because of the greater explosion hazard associated with the Saturn IB.

Higher entry velocities would be possible if the Saturn V were used or if highly elliptical orbits were adopted (fig. 27). However, the extension in mission duration associated with the latter is not compatible with the selected system.

ALTERNATE APPROACHES

In a supplementary task, the possibility that approaches other than those specified in the contract Statement of Work may provide significant research potential at reduced total program cost or program cost per year has been investigated. This task is exploratory—in reality, a byproduct of the basic study—and its results are tentative at best.

A first look at this task immediately suggests the use of a mission vehicle prototype, such as an HL–10 logistics ferry vehicle, for entry research. However, an established mission requirement for such a vehicle does not exist, and many of the specified entry research experiments do not require man (shown previously in table 6). Consequently, this special task concentrates on unmanned research vehicles similar to the existing PRIME but larger. Table 17 shows that smaller vehicles have the potential for reducing cost as well as achieving high supercircular entry speeds with available launch vehicles.

Use of an unmanned alternate approach, of course, will make it necessary to investigate the integration of man into future lifting entry vehicles through separate studies. In particular, this can involve further exploitation of the X–15, the M2–F2, the HL–10 and the SV–5P air-launch vehicles. In addition, much more could be accomplished with piloted ground simulation and variable stability aircraft such as the modified F–106 trainer.

TABLE 17.—SUMMARY OF APPROACH

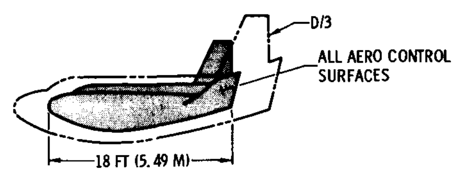
Why?

- Manned research program is expensive
- Many useful experiments do not require man

How?

- Minimize initial research program cost:
  Reduce entry vehicle size & weight
  Avoid man-rating & crew training
- Continue to exploit present flight programs:
  FRC air launched M2–F2, HL–10, SV–5, X–15
  PRIME
- Exploit piloted ground simulation
- Utilize prototype of operational entry vehicle for research

One alternate approach, designated G/0, is the smallest vehicle which maintains the complete HL–10 aerodynamic configuration, including all movable surfaces. Its critical dimensions, when practical allowances are made for structure and thermal protection, are the fin trailing edges. Figure 28 shows this vehicle to be 18 feet (5.49 m) long.



- ENTRY VEHICLE WEIGHT (NEAR-EARTH ORBIT)

	POUNDS	KG
AIRFRAME INCLUDING HEAT SHIELD	2688	1220.4
SUBSYSTEMS	1904	864.4
RESEARCH EQUIPMENT*	105	47.7
TOTAL	4697	2132.4
W/S	40 PSF	194 KG/M^2

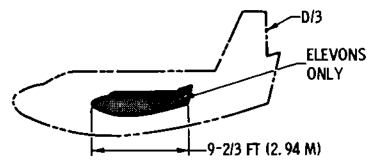
- VOLUME FOR RESEARCH EQUIPMENT .... 42.8 FT^3 1.212 M^3
- RECOVERY TECHNIQUE: NORMAL HORIZONTAL LANDING
- LAUNCH VEHICLE POSSIBILITIES:
  ORBITAL ENTRY: TITAN III, SATURN IB
  SUPERCIRCULAR ENTRY

	ENTRY SPEED, FPS (KM/SEC) NEAR EARTH	ELLIPTICAL
TITAN III-5	33 800 (10.3)	37 800 (11.5)
SATURN IB**	33 150 (10.1)	37 100 (11.2)
SATURN V**	38 700 (11.8)	43 000 (13.1)

* 730 POUNDS (331.4 KG) MAXIMUM
** MAY USE MULTIPLE ENTRY VEHICLES IN LM HANGAR

Figure 28. Alternative I—G/0 Vehicle

The second alternate approach vehicle, designated F/0, is designed for hypersonic and supersonic research only. Like PRIME, the upper control surfaces are simulated in a raised position, and the resulting aft compartment is used to house hydraulic actuators, a drogue chute, and other subsystem equipment. This vehicle, shown in figure 29, has more volume for experimental equipment and thicker heat shields than PRIME. The F/0 vehicle weight for entry at near-orbital speeds is about twice that of PRIME.



- ENTRY VEHICLE WEIGHT (NEAR-EARTH ORBIT)

	POUNDS	KG
AIRFRAME INCLUDING HEAT SHIELD	822	373.2
SUBSYSTEMS	844	383.2
RESEARCH EQUIPMENT	34	15.4
TOTAL	1700	771.8
W/S	50 PSF	243 KG/M^2

- VOLUME FOR RESEARCH EQUIPMENT .... 2 FT^3 0.057 M^3
- RECOVERY TECHNIQUES: AIR SNATCH OR PARAGLIDE
- LAUNCH VEHICLE POSSIBILITIES:

ORBITAL ENTRY: TITAN III, SATURN IB*

SUPERCIRCULAR ENTRY	ENTRY SPEED, FPS (KM/SEC) NEAR EARTH	ELLIPTICAL
TITAN III-5	36 900 (11.2)	41 200 (12.6)
SATURN IB*	35 350 (10.8)	39 500 (12.0)
SATURN V*	39 200 (11.9)	43 900 (13.4)

* MAY USE MULTIPLE ENTRY VEHICLES IN LM HANGAR

Figure 29. Alternative II—F/0 Vehicle

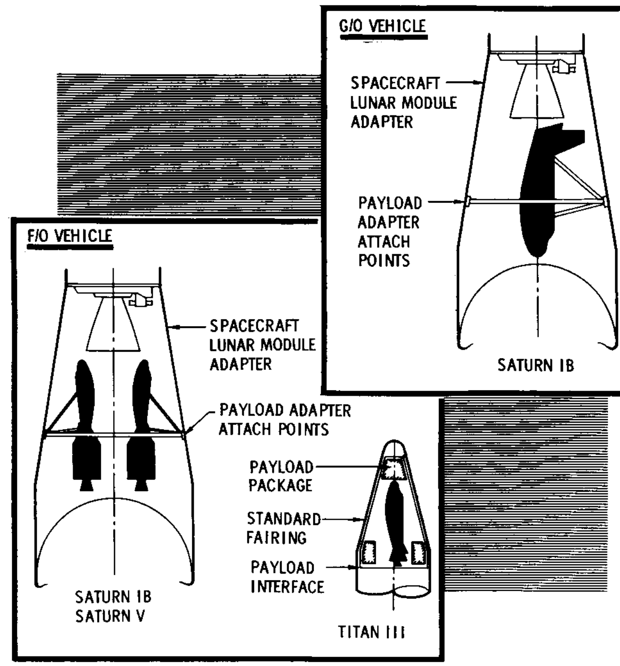


Figure 30. Piggyback Launch Possibilities

Several piggyback configurational concepts are shown in figure 30. This launch method is worth considering because of the possibly significant reduction in cost for entry research which may be achieved by sharing launch operations and expenses with other programs. However, if the entry research is assigned a secondary role on a given launch, the probability of success may be reduced.

The program cost estimates shown in table 18 assume orbital entry speeds only for the alternate approach vehicles, seven flights, and use of the Titan III core launch vehicle. Also shown are estimates of total program value for the D/3 and G/0. Program cost, as noted, is significantly reduced with either of the alternate vehicles, while the total value per dollar improves at least for G/0. For the much smaller F/0 vehicle, the original experiment definitions and relative value are not appropriate. Size does not allow the same amount of instrumentation, and the small scale introduces some uncertainty in interpretation for larger vehicles.

In summary, the results of this task are preliminary, and they do not warrant firm conclusions.

TABLE 18.—PROGRAM COST AND VALUE COMPARISON*

Vehicle	Crewmen	Value	Cost, $ M	Value/cost ratio
D/3	3	2300	850	2.7
G/0	0	1400	400	3.5
F/0	0	?	185	?

* For 7-flight program

CONCLUSIONS

This study has accomplished the specific objective stated in table 1. In many cases, the results of the study are parametric (although based on point designs and program plans). Future trends and the effects of ground rules changes, therefore, can be determined from the data presently generated. The principal study conclusions are summarized in table 19.

TABLE 19.—PRINCIPAL CONCLUSIONS

- Many flight experiments needed to optimize future systems
- Optimum entry vehicle size for research: 25 feet (7.62 m)
- Larger entry vehicle: same research potential and larger program cost
- Optimum crew size for research: 3 men
- Minimum 1-man vehicle has less research potential
- D/3 vehicle has mission capability
- Optimum research program: 11 flights
- Selected launch vehicle: Titan III with 5-segment solids
- Nominal program span: 5 years
- Total program cost: $1 billion

Optimization of future mission systems employing lifting entry will require the conduct of many research experiments like the 52 defined for this study. An HL–10 of optimum size for this flight research is 25 feet (7.62 m) long, carrying a three-man crew. This vehicle, designated D/3, weighs 12 342 pounds (5598 kg). If a larger vehicle were employed, it could provide the same total program research value but it would cost more. On the other hand, a smaller one-man vehicle would have much less research potential and provide substantially less research value per dollar. The selected D/3 vehicle also has an added capability for mission applications in that it can house as many as six men and provide space for a crew transfer tunnel.

An 11-flight program, including two unmanned and nine manned flights, offers a good compromise between cost and research value. For such a program, in which the primary research objective is near-earth orbital entry, the Titan III core with a pair of five-segment solid strap-on rockets is the practical launch vehicle choice. A normally paced program would span five years from go-ahead through evaluation of the eleventh flight.

The total cost of the recommended flight research program, based on a cost model using historical program data, is approximately one billion dollars. Peak fiscal funding ($375 million) is required in the second year.

The strongest justification for pursuing this program is its potential benefit to future mission systems which will employ lifting entry. Benefits accrue from both the individual research experiments and the flights combining many experiments. A realistic appraisal of the recommended experiments and flights shows that many of the experiments and most of the flights will contribute to future system optimization. The major areas of contribution are summarized in table 20.

TABLE 20.—PROGRAM JUSTIFICATION
(for future lifting entry programs)

Potential Improvement	Experiments*	Flights*
• Reliability and safety	39	11
• Knowledge of environment	27	11
• Flight operational techniques	10	11
• Knowledge of spacecraft reuse	7	11
• Knowledge of man in loop	11	9
• Crossranging performance	10	8
Potential Reduction		
• System weight	40	10
• System cost	17	10

* Out of 52 experiments and 11 flights

RECOMMENDATIONS

A sound engineering, cost and planning basis has been established by this study for undertaking a flight research program if NASA had an immediate requirement.

In lieu of a full program go-ahead, several useful and appropriate study areas are identified which can significantly improve the system definition or solve specific problems revealed in the present study. One important example is the crew task loading imposed both by normal and emergency flight operations and by the research tasks. Additional analysis and simulation are highly desirable. Another example is the space allowance at the forward crew station; the mockup evaluation shows that pressurized suit operations could be marginal.

Another potential problem which deserves additional study is the possibility of turbulent separated flow on the bottom surface of the vehicle when elevons are deflected. Such a condition could have a significant impact on elevon dynamic hinge moments and local heating.

Still another important example for useful additional study is the broad area of vehicle reuse. Its design implications involve subsystem qualification levels, environmental isolation, degree of redundancy, maintainability and accessibility, onboard local environment monitoring, and subsystem performance trend monitoring. Its operational implications involve component replacement schedules, levels of postflight inspection and subsystem functional verification testing, and the amount of equipment removal required.

Table 21 lists these and other areas for further study. Of particular interest are an extended investigation of alternate approaches and a detailed investigation of research related to the use of lifting bodies for planetary return.

TABLE 21.—SUBJECTS FOR ADDITIONAL STUDY

Design improvement

- Crew station • Visibility • Base attachment
- Crew transfer • Automatic landing

Crew studies

- Research task analysis • Energy management
- Terminal guidance

Earth return from deep-space missions

Possibility of turbulent boundary layer separation

In-depth study of reuse

Alternate approaches (F/0, G/0, piggyback)

- Technical requirements • Cost and effectiveness
- Experiment evaluation