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APOLLO

Final Report

Configuration

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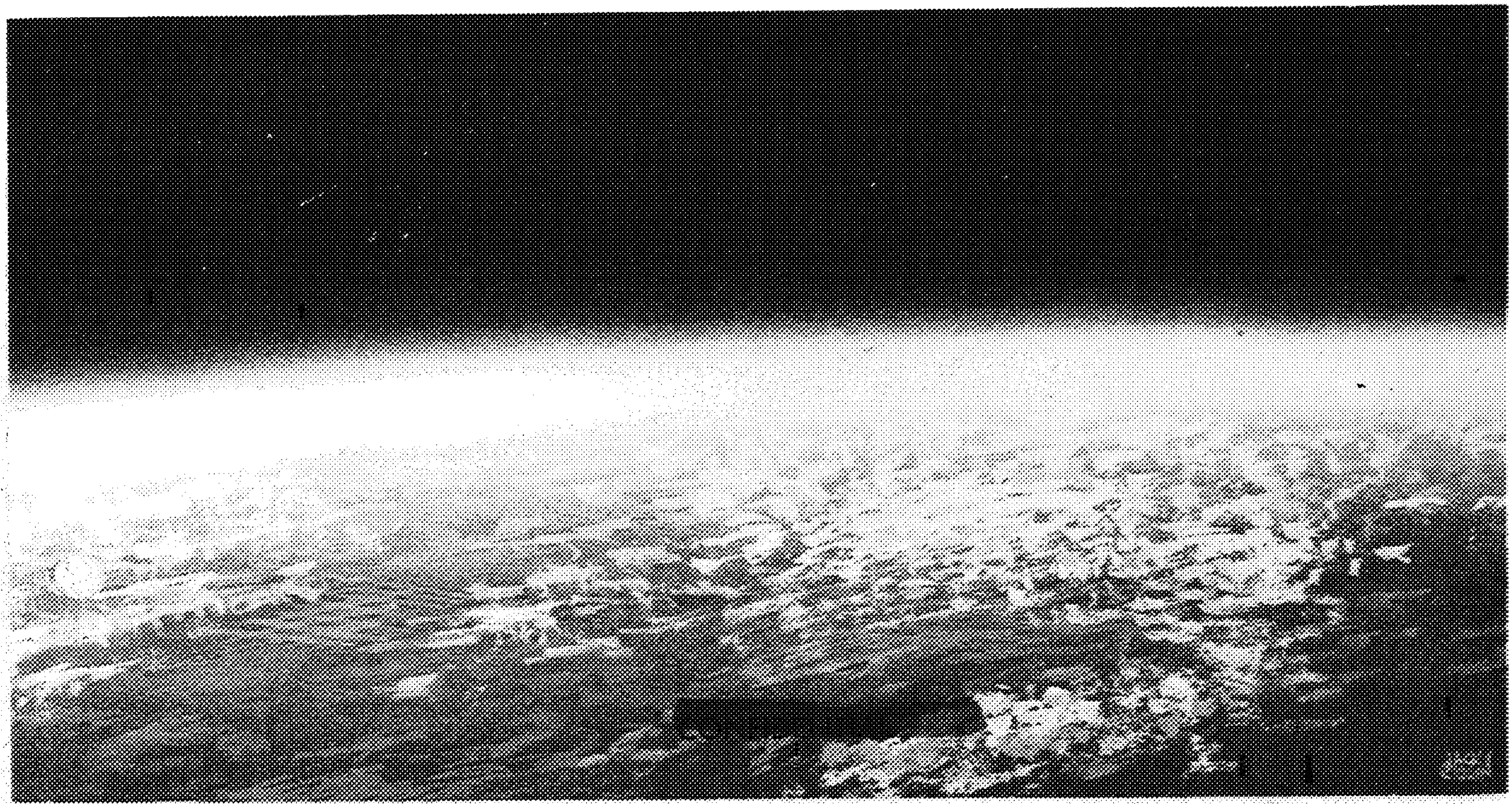

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MODEL 410 — THE SYSTEM AND ITS OPERATION

*A BRIEF DESCRIPTION**

Model 410 is the spacecraft system recommended by Martin for the Apollo mission. Its design satisfies the guidelines stated in NASA RFP–302, as well as a more detailed set of guidelines developed by Martin during the Apollo design feasibility study.

We conceive the ultimate Apollo mission to be a manned journey to the lunar surface, arrived at by the preliminary steps of earth orbit, circumlunar and lunar orbit flights. Operational procedures proved out in the early steps will be carried over into the advanced steps, thus establishing a high level of confidence in the success of the lunar flights. With the recommended system, manned lunar orbit missions can be made as early as 1966.

Operational Features

For a circumlunar flight when the moon is at its most southerly declination (Fig. p–1) the launch operation proceeds southeast from Cape Canaveral and down the Atlantic Missile Range. The Saturn C–2 third stage shuts down when orbital velocity is reached at an altitude of 650,000 feet. What follows is a coasting orbit passing over the southern tip of Africa, the Indian Ocean and up the Pacific Missile Range. In this interval the crew checks out all onboard equipment, which has just passed through the accelerations, noise and vibration of the boost phase. If the pilot-commander is satisfied that all systems are working properly, the third stage is restarted and the spacecraft is injected at parabolic velocity northwest of Hawaii. If the pilot-commander is dissatisfied with the condition of the vehicle or crew, he separates from the Saturn S–IV, starts the mission abort engine, re-enters at the point shown in Fig. p–1 and lands at Edwards AFB.

Continuing translunar flight from the point of injection, the trajectory trace swings down over the Caribbean and then west over South America. This particular trajectory passes within 240 naut mi of the moon, then turns back for a direct re-entry some six days after launch. Re-entry occurs southwest of Hawaii some 3300 naut mi from the Edwards AFB landing site.

Tracking. The range coverage provided by present and planned facilities is shown in Fig. p–1 for this trajectory and for a second return trace representing the case when the moon is at the most northerly declination. This second trajectory establishes the 10000-naut mi re-entry range requirement for Apollo to meet the guidelines of operation on every day of the lunar month and of operation into a single landing site.

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*For more complete descriptions, see ER 12000 or ER 12001.

Abort. During the critical launch and checkout phase, abort will be possible at any time : at the crew's discretion, automatically or by ground command. Up to nine minutes after launch (from Canaveral), the abort landing is restricted to the AMR for a circumlunar flight. Beyond this point the pilot has the option of continuing to any point along the AMR, PMR or into Edwards AFB through the use of the mission abort propulsion system and the inherent downrange maneuverability of the Model–410.

The Selected Spacecraft

The Apollo space vehicle (Model 410 spacecraft plus launching vehicle) is shown in Fig. p–2. The spacecraft—that portion of the space vehicle which makes the flight to the moon—consists of these three modules:

(1) Command module, housing the three crew members during all thrusting periods, e.g., launch from earth, any corrections to the flight path during flight in space, during re-entry and, ultimately, during landing and launch from the moon. It is the operating center from which all control of the flight is made.

(2) Propulsion and equipment module, containing all the propulsion units which operate between the point of final booster separation and re-entry after the lunar flight. It is separated from the command module at 200 naut mi from the earth on the return trip. It is designed with tankage for lunar takeoff and will be offloaded for less ambitious missions.

(3) Mission module—contained within the outer frame of the propulsion and equipment module—providing space during the lunar voyage for scientific observations and crew living functions.

Command Module

With its lifting capability, the Apollo command module represents a step forward in technology over ballistic vehicles, Mercury and (to the best of our knowledge the *Boctók* (*Vostok*). The lift results from the capsule's shape—a blunted cone flattened on the top (see Fig. p–3).

Heating and radiation protection. The Model 410 is shaped conservatively for aerodynamic heating in addition to its relatively high L/D (0.77). By accepting the large convective heat load of a nose radius smaller than that of the Mercury type, the Model 410 shape tends to minimize radiative heat transfer which is less well understood and harder to protect against. The thermal protection system provides excellent protection for the crew from the large aerodynamic heat loads, from space radiation (including solar flares) and from meteorites.

The normal mission radiation dose will not exceed the five rem limit defined by NASA. If the crew should encounter a solar event as severe as that following the May 10, 1959 flare, they would receive a dose of only 67 rem—well within the 100 rem dose limit set by Martin as tolerable during an emergency.

Thermal protection for re-entry is provided by a composite shield of deep charring ablator (nylon phenolic) bonded to superalloy honeycomb panels which are set off and insulated from the water-cooled pressure shell. The control flaps are protected from the high initial heat rate by an ablator bonded directly to the flap. The long-time, lower heating rates are handled by re-radiation from the backside. The aft bulkhead is protected by a fiberglas phenolic honeycomb panel with a foamed polyurethane insulation.

Crew provisions. The crew has access to all electronic and electrical equipment in the command module for maintenance and replacement. Both pilots have two-axis sidestick and foot controllers as well as a manual guidance mode used with the computers inoperative for deep space and re-entry operations.

Cabin pressure is maintained at the equivalent of 5000 feet altitude ("shirt sleeve" environment). Protective suiting is donned only for launching and landing, but need not be inflated except in emergency.

Guidance. The guidance system consists of both automatic and manual star tracking equipment, as well as two inertial platforms and two general purpose digital computers. Two windows, with ablative heat shield covers, are provided for use with tracking instruments.

Flight control. Pitch and yaw attitude control within the atmosphere is provided by flaps driven by hot gas servos. Outside the atmosphere dual reaction controls are used. Roll is controlled at all times by a dual reaction system.

Communications. Communications equipment includes a K_e band for reentry, a C-band for the pre-reentry and both HF and VHF rescue beacons for landing and recovery.

Landing system. The landing system consists of a steerable parachute, retrorocket combination, enabling the M–410 to avoid local obstacles, trim out wind drift and reduce sinking speed to a nominal three feet per second—low enough for safe landing on any kind of terrain or in very rough seas. In the event of retrorocket failure, accelerations on the crew will not exceed 20 G.

Launch escape propulsion system (LEPS). LEPS is a thrust-vector-controlled, solid rocket system which separates the command module from the rest of the space vehicle in the event of an emergency during launch pad operations or during boost through the atmosphere. In an off-the-pad abort, it lifts the command module to an altitude of more than 4000 feet. During a normal boost trajectory, LEPS is jettisoned at 300,000 feet.

Propulsion and Equipment Module

The propulsion and equipment module (shown in Fig. p–3) contains propulsion devices and equipment which are not necessary for re-entry. Its outer skin serves both as a load carrying structure and as a meteorite shield for the propellant tanks, mission module and other equipment.

Propulsion devices. The mission engine, used for trajectory correction and abort, is a high preformance, modified LR–115 (Pratt & Whitney), developing 15,600 pounds of thrust. A total of 10,450 pounds of liquid hydrogen and liquid oxygen propellants may be carried, sufficient for lunar takeoff.

Four vernier engines, with 300 pounds of thrust each, are used for midcourse correction, ullage impulse to settle the mission engine propellants and for thrust vector control during operation of the mission engine. In addition there are two sets of six control jets which provide 30 pounds of thrust for roll, pitch and yaw control.

Power sources. Spacecraft equipment is powered by fuel cells (2 kw) which under normal conditions, use the boiloff from the mission propulsion system. A supply of independent reactants is provided for emergencies. Battery power is used during re-entry.

Communications. Four large antennas fold out to provide S-band communications and X-band radar altimeter information. VHF communications gear is also provided.

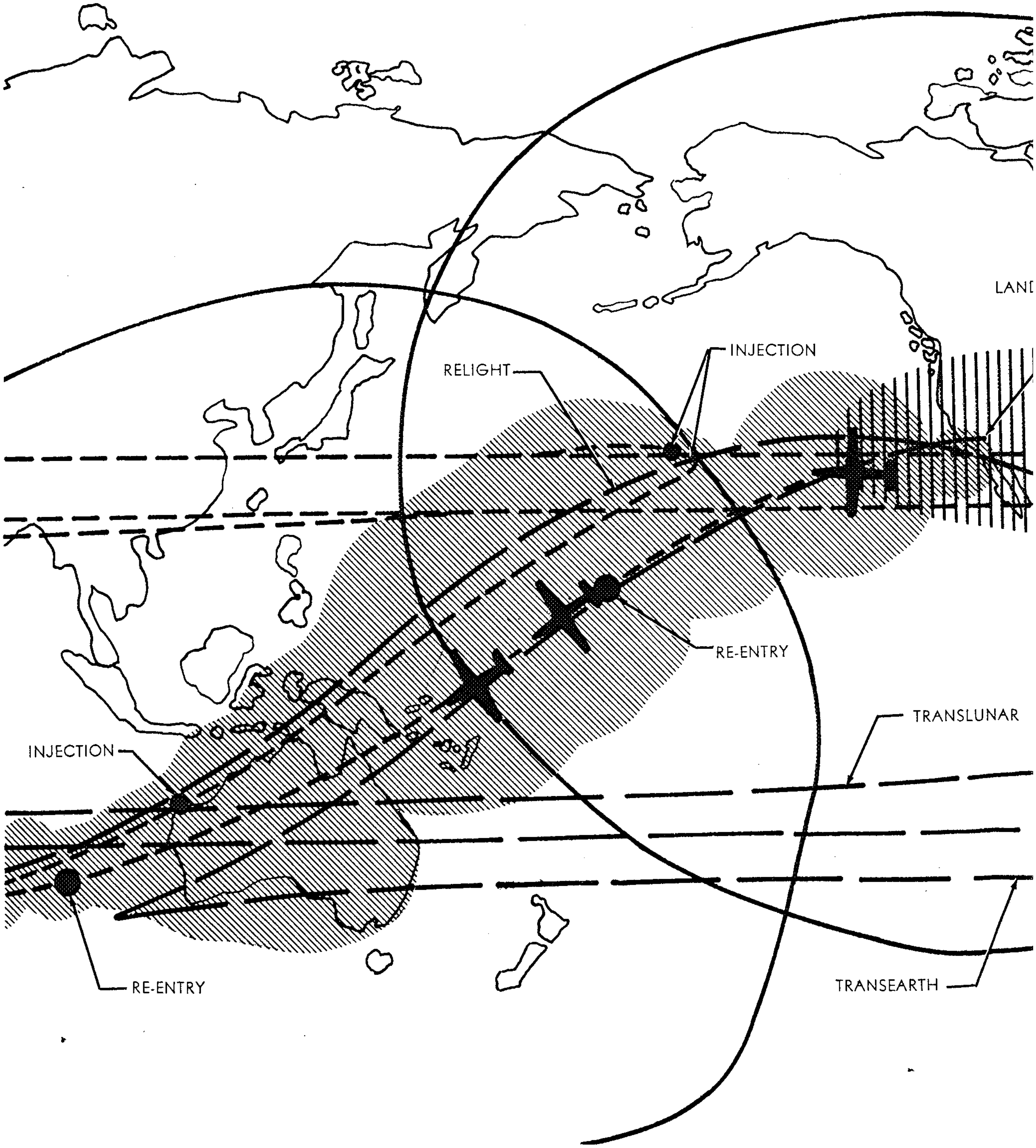
Mission Module

The mission module provides 400 cubic feet of living space during the lunar voyage. It serves as a midcourse work-rest area, providing freedom of movement and privacy. For operations on the lunar surface it will be a base of scientific investigations, and will serve as an airlock. The same "shirt sleeve" environment at 12.2 psi is maintained as in the command module.

The mission module provides the space and flexibility required for effective lunar reconnaissance and scientific experimentation. An Eastman-Kodak camera-telescope has been selected, for example, which has one-meter resolution at lunar orbit altitude of 50 naut mi.

MODEL 410 WEIGHT SUMMARY

MISSION	CIRCUMLUNAR	LUNAR ORBIT	LUNAR TAKEOFF
COMMAND MODULE	6954	6954	6954
PROPULSION AND EQUIPMENT MODULE	7372	13,192	15,618
LAUNCH ESCAPE PROPULSION SYSTEM	185	185	0
ADAPTER	489	489	0
EFFECTIVE LAUNCH WEIGHT	15,000	20,820	22,572



LANI
INJECTION
RELIGHT
RE-ENTRY
TRANSLUNAR
INJECTION
RE-ENTRY
TRANSEARTH

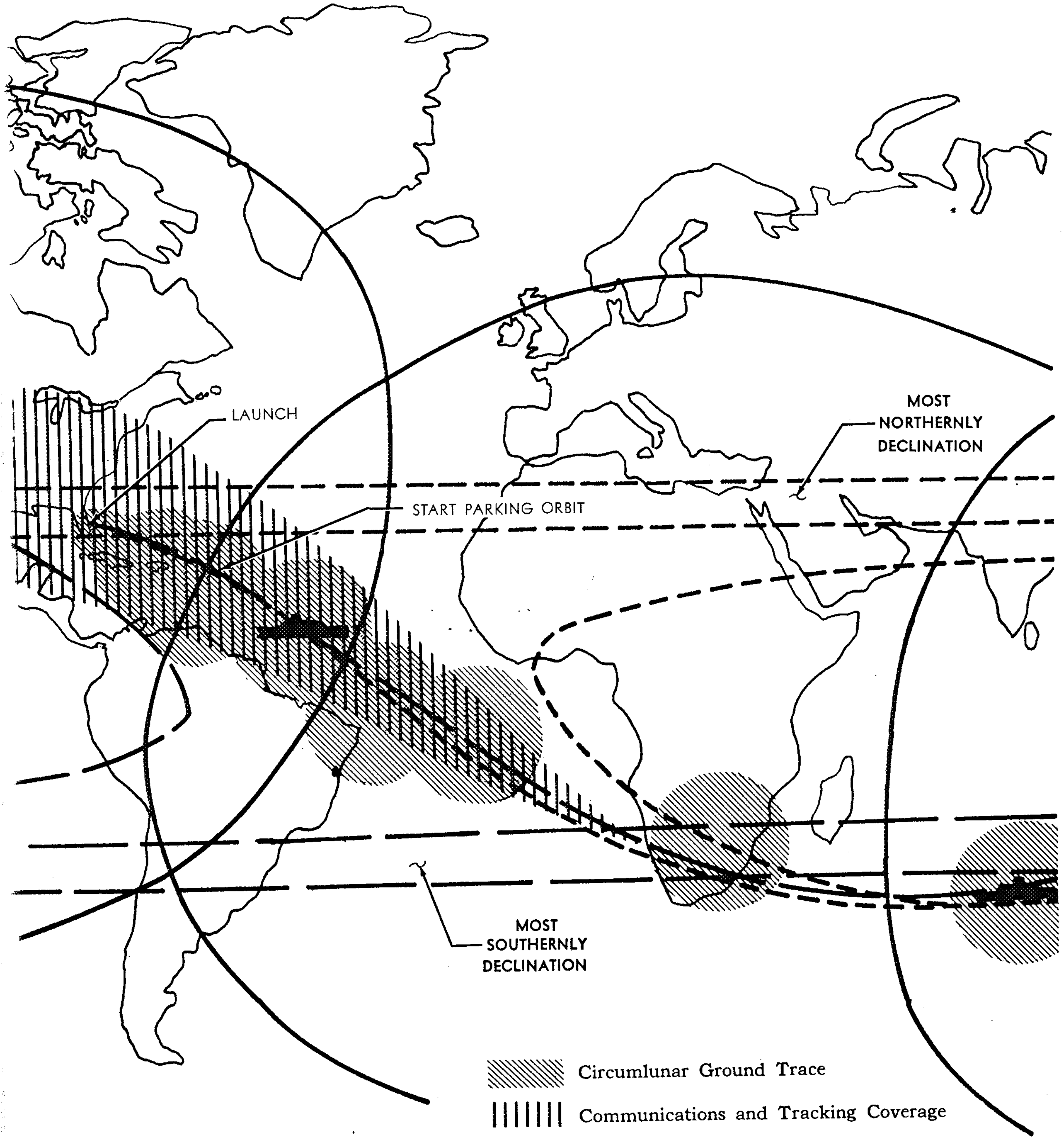

LAUNCH
START PARKING ORBIT
MOST NORTHERNLY DECLINATION
MOST SOUTHERNLY DECLINATION
▨ Circumlunar Ground Trace
||||||| Communications and Tracking Coverage

Fig. p-1 Model 410 Circumlunar Trajectory and Range Coverage

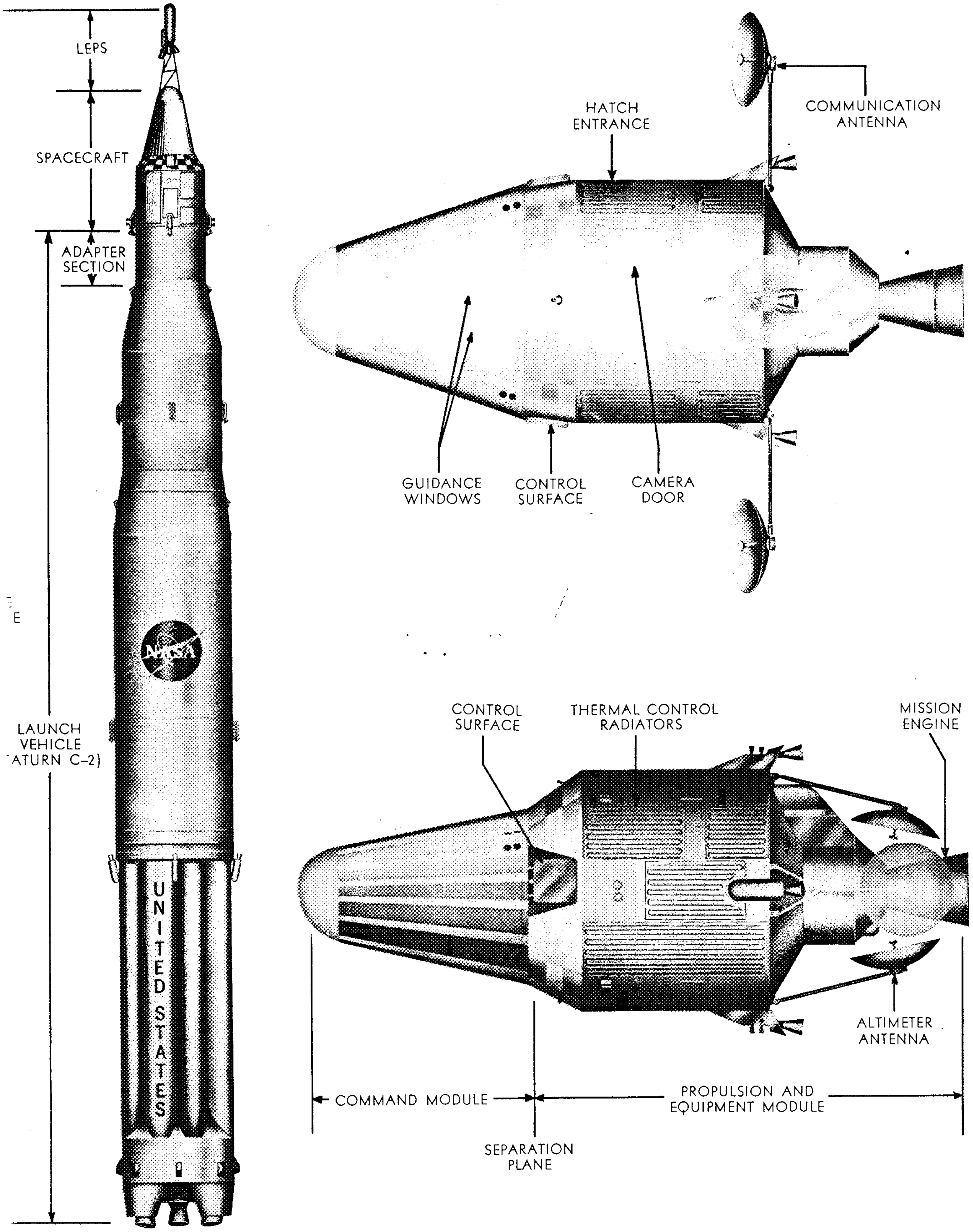

LEPS
SPACECRAFT
ADAPTER SECTION
LAUNCH VEHICLE
ATURN C-2)
NASA
UNITED STATES
HATCH ENTRANCE
COMMUNICATION ANTENNA
GUIDANCE WINDOWS
CONTROL SURFACE
CAMERA DOOR
CONTROL SURFACE
THERMAL CONTROL RADIATORS
MISSION ENGINE
ALTIMETER ANTENNA
COMMAND MODULE
PROPULSION AND EQUIPMENT MODULE
SEPARATION PLANE

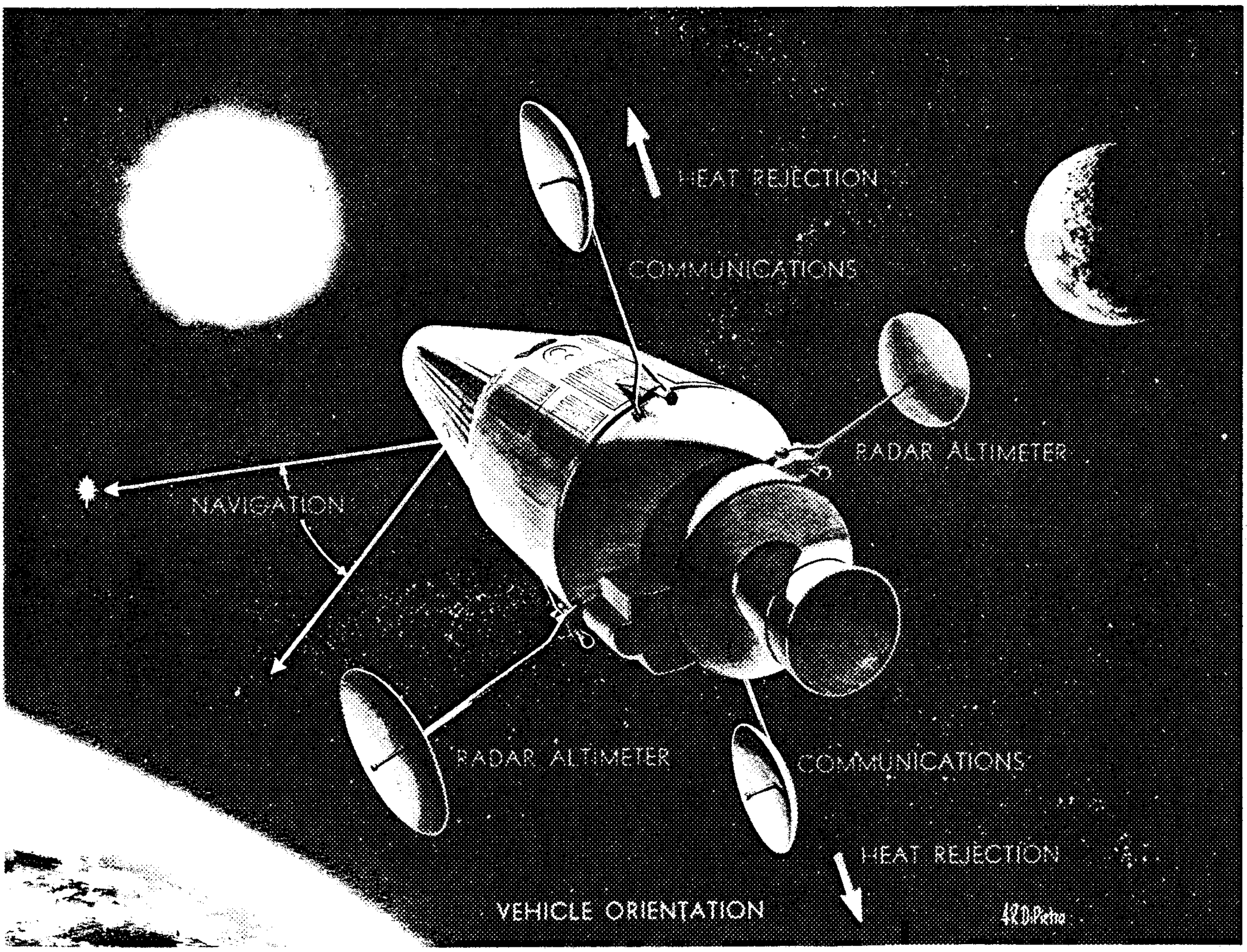
Fig. p-2. Model 410 Apollo Space Vehicle

MISSION	EFFECTIVE GROSS WEIGHT (lb)	PROPULSION ΔV CAPABILITY (fps) MISSION	PROPULSION ΔV CAPABILITY (fps) VERNIER
CIRCUMLUNAR	15000	1830	525
LUNAR ORBIT	20820	6100	525
LUNAR TAKEOFF	22572	8600	200

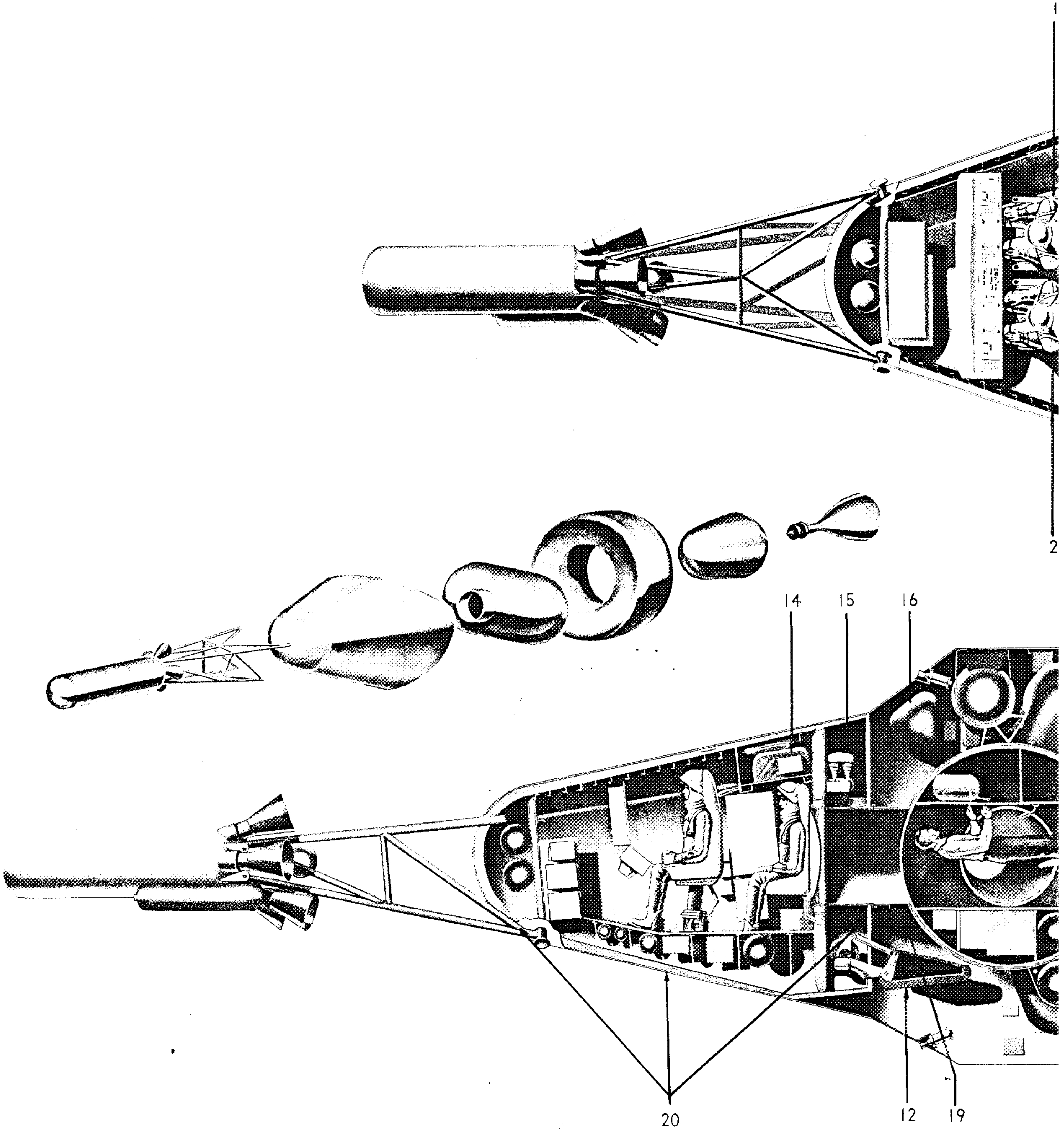
VOLUMES (cu ft)	
COMMAND MODULE	350
MISSION MODULE	400
MISSION H_2 TANK	400
MISSION O_2 TANK	122

PROPULSION SYSTEM DATA

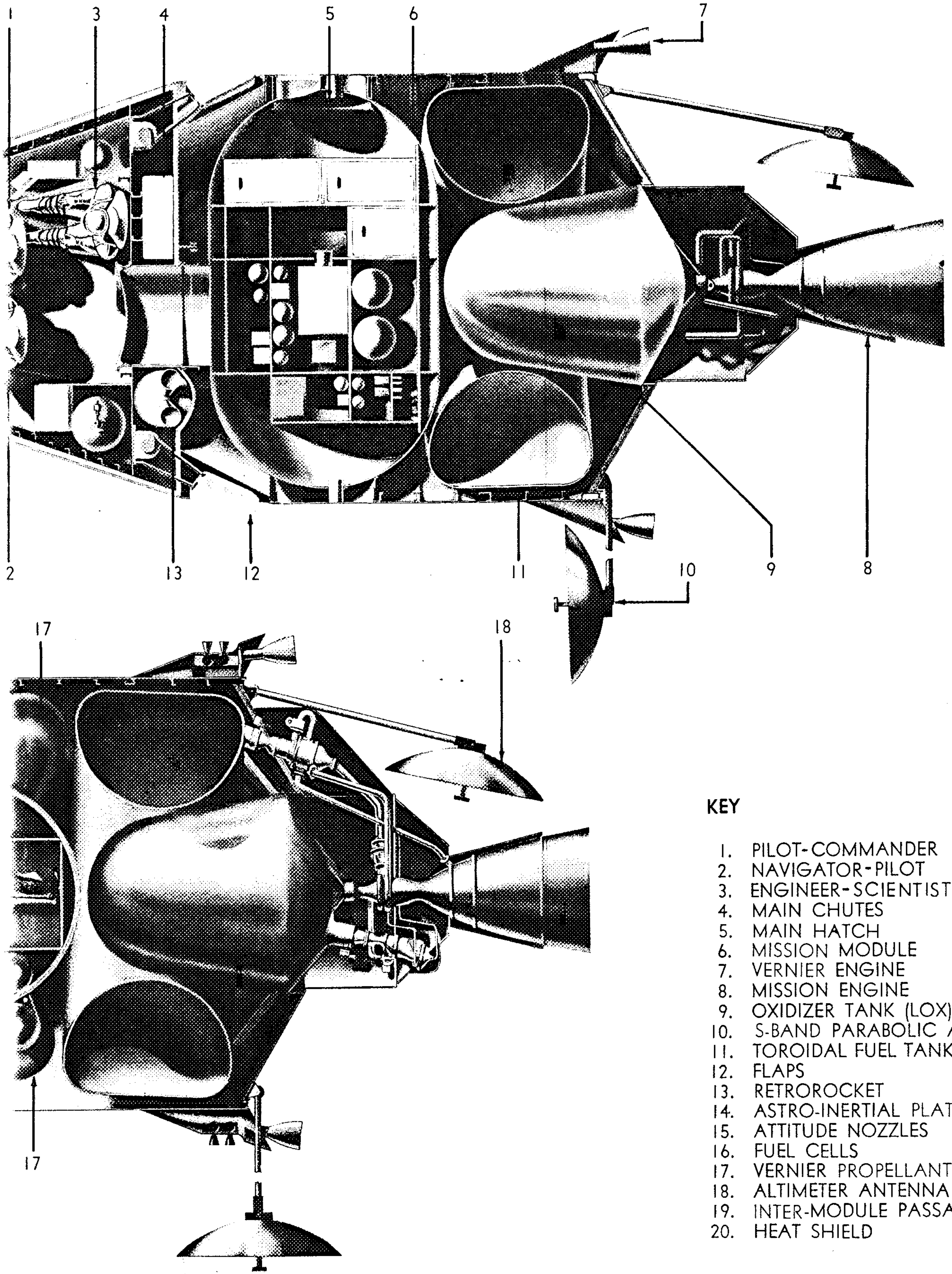
PURPOSE	TYPE	ISP. (sec)	THRUST (lb)
MISSION (1)	H_2—O_2 (ADV. LR115)	427	15600
VERNIER (4)	N_2H_4/UDMH–N_2O_4	315	300 EACH
ATTITUDE CONTROL (14+BACKUP)	N_2H_4/UDMH–N_2O_4	250–315	15 TO 50


HEAT REJECTION
COMMUNICATIONS
RADAR ALTIMETER
NAVIGATION
RADAR ALTIMETER
COMMUNICATIONS
HEAT REJECTION
VEHICLE ORIENTATION

FOLDOUT FRAME 2


1
2
14
15
16
20
12
19

FOLDOUT FRAME 1



KEY

1. PILOT-COMMANDER
2. NAVIGATOR-PILOT
3. ENGINEER-SCIENTIST
4. MAIN CHUTES
5. MAIN HATCH
6. MISSION MODULE
7. VERNIER ENGINE
8. MISSION ENGINE
9. OXIDIZER TANK (LOX)
10. S-BAND PARABOLIC ANTENNA
11. TOROIDAL FUEL TANK (LH)
12. FLAPS
13. RETROROCKET
14. ASTRO-INERTIAL PLATFORM
15. ATTITUDE NOZZLES
16. FUEL CELLS
17. VERNIER PROPELLANT TANK
18. ALTIMETER ANTENNA
19. INTER-MODULE PASSAGE
20. HEAT SHIELD

Fig. p-3. Model 410 Apollo Inboard Profile

FOLDOUT FRAME 2

I. Introduction 1
  A. Design Criteria 1

II. Configuration Considerations 3
  A. Propulsion and Equipment Module Location 3
  B. Launch Escape Propulsion System Location 3
  C. Command Module (Re-Entry Vehicle) 3

III. Command Module 7
  A. Aerodynamic Considerations 7
  B. Thermal Protection System 17
  C. Arrangement Considerations 18
  D. Landing System 20
  E. Crew Safety and Design Flexibility 23
  F. Weight Considerations 23

IV. Mission Module 25
  A. Mission Uses 25
  B. Selection Process 25

V. Propulsion and Equipment Module 27
  A. Requirements 27
  B. Mission Engine Considerations 29

VI. Artificial "G" Studies 35

VII. Alternate Configurations 39

VIII. Lunar Landing Studies 57
  A. Introduction 57
  B. Lunar Landing Module Configurations 57
  C. System Description and Operation 63
  D. Modifications to the Apollo Spacecraft 65
  E. Weight 65

IX. Growth Version 67

I. INTRODUCTION

An important phase of the Apollo study being presented in this series of reports (Ref. ER 12003 through 12011, 12017 and 12018) covers the development, comparison and selection of vehicle configurations appropriate for the Apollo missions. In the NASA Request for Proposal, RFP 302, it was stipulated that the contractor should "recommend, define and substantiate a logical approach to a spacecraft." Of all the areas of design and analysis covered by Project Apollo, none has more influence on the ultimate success of the program than configuration selection, and yet none is less subject to objectivity.

The spacecraft configuration is inseparably involved with vehicle weight and performance and has a major influence on system reliability and crew safety not only from system and component failures but also space environmental hazards. The spacecraft configuration is inseparably tied to the operational concepts. This is a very important point—change the operational concept and the vehicle will change.

This report includes the general considerations of the location of the major components with respect to each other and the order and orientation of the various propulsion systems with respect to the launch vehicle. It also includes the considerations which led to the use of the modular arrangement (Ref. Chapters III and IV) and summarizes the results of studies which were made to establish the shape and arrangement of the various components such as the Command Module (Chapter III), the Mission Module (Chapter IV) and the Propulsion and Equipment Module (Chapter IV). Data regarding subsystem design characteristics for the chosen vehicle and the selection process used, are included in the other reports in this series.

A. DESIGN CRITERIA

The Apollo spacecraft design development has been based on a number of concepts and rules, all of which have major effects on the selection of the various modules of the configuration, the systems and the arrangement of the overall spacecraft. The major rules and concepts which were laid down by NASA for the study were:

(1) *That the vehicle gross weights are to be within booster capabilities.* These capabilities have been established at 15,000 pounds for the early circumlunar missions and at approximately 20,000 pounds for the lunar orbit mission.

(2) *A three-man crew size.* For this study, a 95 percentile man is used and a shirt sleeve environment is furnished.

(3) *A fourteen-day mission.* The distribution of time for various phases of the mission were not established.

(4) *That the vehicle should be designed for the lunar orbit mission.*

These guide lines or ground rules as laid down have been supplemented by other Martin-established criteria which further serve to narrow the possible vehicle choice and arrangements. The major factors were:

(1) *The spacecraft should be designed for lunar takeoff.* Consideration of development time including the tests required to prove out the vehicle in conjunction with the natural future requirement for moon exploration, led to establishment of this guide line. The ΔV established for lunar takeoff is 8600 fps instead of the 6100 fps established as necessary for lunar orbit.

(2) *The spacecraft is to be designed to use systems, materials and processes either available or in the late stages of development.* The use of developed items leads to less program risk, better reliability (proof through previous use) and earlier availability of the spacecraft.

(3) *The basic design should not include artificial g provisions.* Configuration studies (See Chapter VI) have shown that a weight increase of at least 50% would have to be paid for these provisions (Ref. Chapter VI). Further, our studies and tests have indicated that there is no real requirement for g in space missions. These provisions will not be made.

(4) *The design should be arranged to provide escape from the vehicle with a single separation.* Need for simplicity and reliability led to the

adoption of this ground rule. Inherent quick escape is built into the system.

(5) *The design arrangement should provide inherent protection and safety from meteorites and cosmic radiation.* From a weight point of view, utilization of a "bumper" concept as conceived by Whipple is the superior method of protecting against meteoritic penetration. This concept was selected for use in all cases. Radiation protection is to be provided by using the structure and equipment as shielding with a minimum added for specific shielding. The dosage limits were established at 5 rem for a normal mission, 100 rem for an emergency mission.

(6) *The design should allow for landing under a wide range of meteorological conditions and terrain.* Basic design of the vehicle is predicated on landing at Edwards AFB for all missions. If landing at some other spot is necessary provisions are to be made to avoid local obstacles and to give zero ground speed when ground winds are up to 36 fps.

(7) *The design should provide for maximum crew space within the weight limitations.*

(8) *The design should be easily adaptable for use in alternate missions.* Missions considered should include earth orbit, rendezvous and lunar exploration as well as the lunar takeoff.

(9) *The vehicle operations should be based upon concepts which avoid the necessity for using a "parking" orbit during the return to earth.*

(10) *The design should provide for launch on any day during the lunar month.*

(11) *The maximum load factor during reentry is to be less than 6 g for the design conditions.*

II. CONFIGURATION CONSIDERATIONS

The Apollo comprises five basic parts (Ref. Fig. II–1). These are:

(1) Launch escape propulsion system (LEPS)

(2) Command Module or re-entry vehicle (CM)

(3) Mission Module (MM)

(4) Propulsion and Equipment Module (PEM)

(5) Adapter section.

Modules such as the adapter section, LEPS, Command Module and Propulsion and Equipment Module are required for all versions of Apollo though their individual detail geometry and arrangement may differ greatly. The Mission Module, on the other hand, is not absolutely necessary for performing the functions expected of Apollo according to the NASA guide lines. The Model 410 design is based upon the modular approach (incorporation of a Mission Module) for many reasons. The most salient are that the modular approach gives more mission flexibility, leads to a slightly lighter vehicle and gives a capability for the ultimate missions such as lunar landing. Further discussion of the uses of the module is presented in Chapter IV.

The various Apollo modules may be arranged in many ways. The adapter section will always be attached to the launch vehicle. The arrangements will be narrowed by considering the various modules and determining the best location for each.

A. PROPULSION AND EQUIPMENT MODULE LOCATION

The most important item of equipment contained within the PEM is the mission engine. It is, in general, desirable to have this at the aft end of the spacecraft with the engine on the centerline so that the thrust vector will nominally go through the cg of the spacecraft. Configurations with the mission engine facing forward were not seriously considered since the engine would be exposed to areodynamic heating and loads during ascent unless an auxiliary, jettisonable fairing were provided. If two or more engines were provided, placed outboard, fairings would be required and control of the spacecraft would be extremely difficult if one of the engines failed. The reasons for using a single engine in M–410 are shown in Chapter V.

Another factor to be considered is the configuration flexibility. If the propulsion system is confined to the rear part of the spacecraft, it is relatively easy to substitute alternate propulsion systems during the early missions if this is necessary to expedite the program.

For these reasons, the PEM has been arranged with the mission engine at the rear of the spacecraft.

B. LAUNCH ESCAPE PROPULSION SYSTEM LOCATION

The two most practical locations of the LEPS are forward of the escape vehicle or aft of it. The LEPS has been placed forward of the escape vehicle on the Model 410 because this arrangement is stable, the LEPS is more easily jettisonable, it does not require more than one propellant bottle, the thrust loss due to nozzle cant is minimized and it is a system similar to that proven in Project Mercury.

C. COMMAND MODULE (RE-ENTRY VEHICLE)

With the PEM located in back and the LEPS in front, the two locations left for the re-entry vehicle are between the LEPS and the mission module or between the mission module and the PEM. The re-entry vehicle selected for the Model 410 is a forward-facing cone with a flat top. Reasons for its selection are given in Chapter III. Arrangements which include the re-entry vehicle behind the mission module have somewhat better protection against meteorites since the heat shield is then protected by the structural shell but the complexity involved in separation of many parts during launch escape, the lack of direct access to the outside in emergencies and the fact that the re-entry vehicle has to be reoriented after

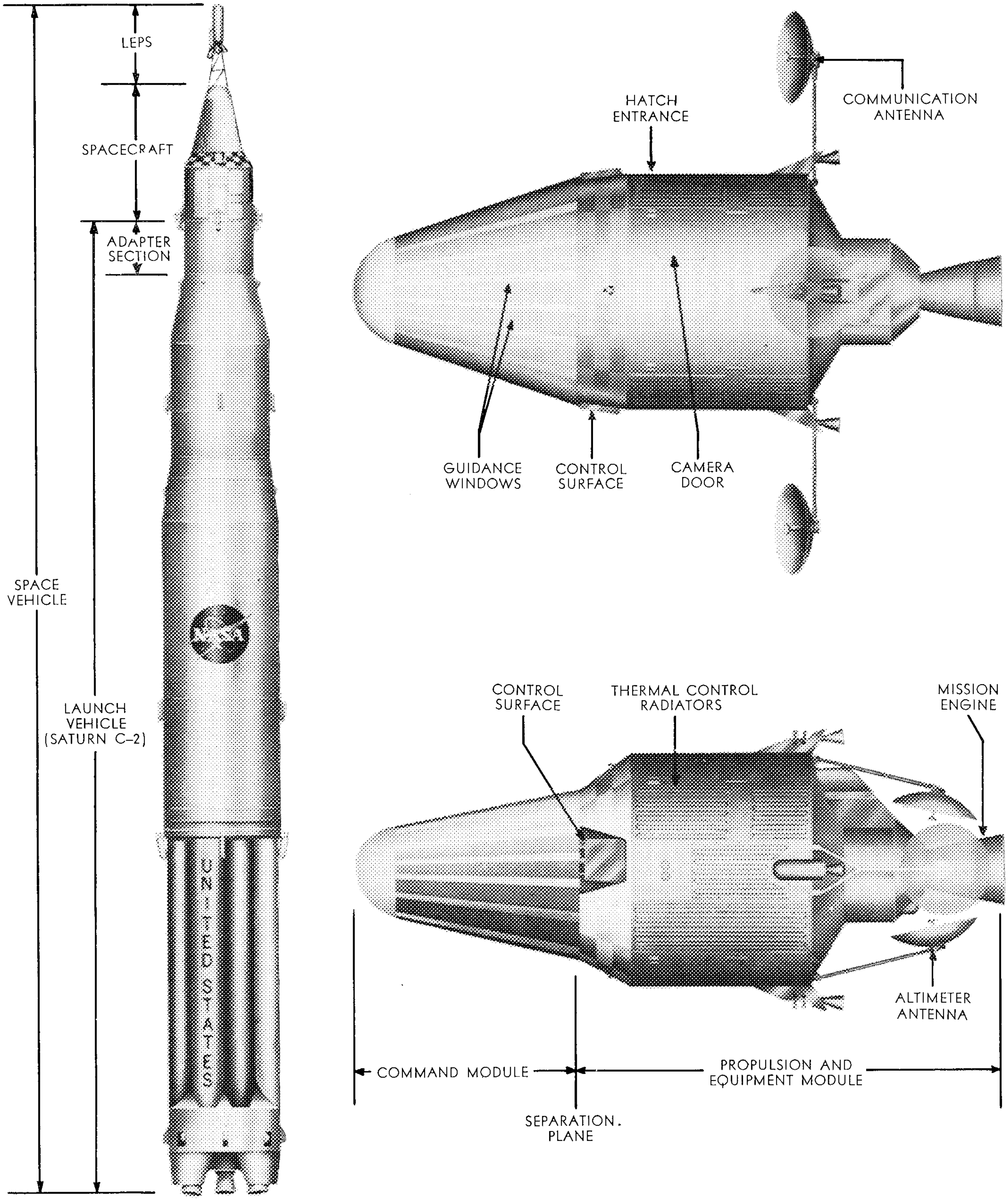

LEPS
SPACECRAFT
ADAPTER SECTION
SPACE VEHICLE
LAUNCH VEHICLE (SATURN C-2)
UNITED STATES
HATCH ENTRANCE
COMMUNICATION ANTENNA
GUIDANCE WINDOWS
CONTROL SURFACE
CAMERA DOOR
CONTROL SURFACE
THERMAL CONTROL RADIATORS
MISSION ENGINE
ALTIMETER ANTENNA
COMMAND MODULE
PROPULSION AND EQUIPMENT MODULE
SEPARATION. PLANE

FIG. II-1. MODEL 410, APOLLO SPACE VEHICLE

separation for re-entry are reasons why this approach was not selected.

The Model 410 has been arranged with the re-entry vehicle forward of the mission module. Some of the advantages are:

(1) Clean separation—only a single module is separated from the space vehicle in case of emergency.

(2) The arrangement allows for growth versions of Apollo without re-entry vehicle or escape system redesign.

(3) The heat shield on the forward-facing body furnishes good inherent protection against space radiation hazards.

(4) Guidance windows with a large field of view are possible.

(5) The re-entry vehicle attitude control nozzles may be used to reorient the spacecraft if the PEM system has failed and created an emergency.

(6) The command module need not be re-oriented after being separated from the remainder of the spacecraft.

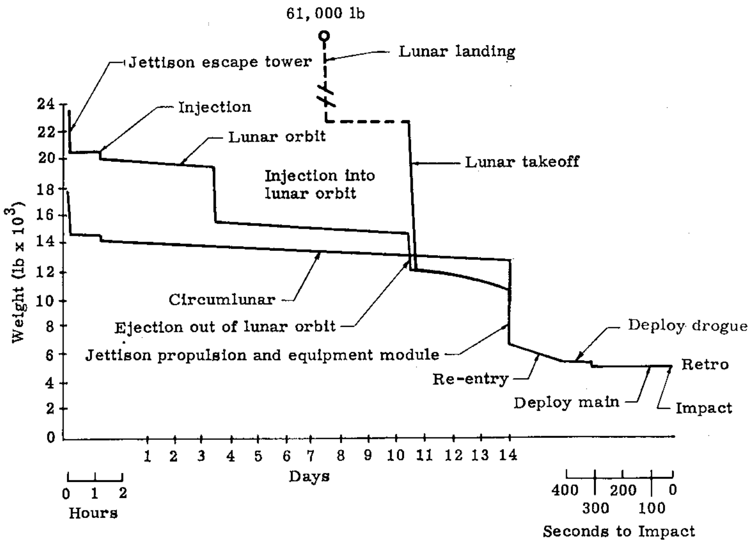
The main disadvantages of this arrangement is that the exposed heat shield is subject to damage by meteorites while in space. The effect of impingement on ablator performance must be studied by test. If the pits, etc., seriously degrade heat shield performance, some step such as the addition of a meteorite bumper may be necessary.

The orientation of the crew can be a significant factor in vehicle arrangement. However, life science studies have shown that the crew—properly restrained—can perform the necessary functions during re-entry, whether facing forward or aft if the accelerations are not excessive. The aerodynamic characteristics of the selected command module and the control methods used during re-entry will limit design re-entry load factors to 6g maximum.

The weight of Model 410 is presented in Table II–1.

TABLE II–1
WEIGHT SUMMARY

	EARTH POUNDS		
	CIRCUM LUNAR	LUNAR ORBIT	LUNAR TAKEOFF
COMMAND MODULE	(6954)	(6954)	(6954)
HEAT SHIELD	2078	2078	2078
STRUCTURE AND CONTROLS	1923	1923	1923
CREW AND EQUIPMENT	2953	2953	2953
PROPULSION AND EQUIPMENT MODULE	(5560)	(6080)	(5166)
MISSION MODULE STRUCTURE	399	399	399
EXTERNAL STRUCTURE	600	600	600
CONTROLS AND EQUIPMENT	3040	3560	2646
PROPULSION AND TANKS	1521	1521	1521
ADAPTER SKIRT	489	489	0
LAUNCH ESCAPE PROPULSION SYSTEM TOTAL	3091	3091	
LAUNCH ESCAPE PROPULSION SYSTEM EFFECTIVE	185	185	0
MISSION PROPELLANT	1812	7112	10452
EFFECTIVE LAUNCH WEIGHT	15000	20820	22572
SPACECRAFT INJECTION WEIGHT	14326	20146	—
START RE-ENTRY WEIGHT	6954	6954	6954
CHUTE DEPLOYMENT WEIGHT	5667	5667	5667
TOUCHDOWN WEIGHT	5352	5352	5352



III. COMMAND MODULE

This chapter presents the salient factors affecting the Command Module configuration. These factors include the operational concepts, aerodynamic heating and the thermal protection system required to protect against it, the aerodynamic characteristics of the vehicle, the guidance and control laws used during the reentry and the range and manuverability requirements. Other factors considered in the selection process were the effect of space environments and the crew hazards they may create, the type of landing system to be used and the number of crew men to be carried, their seating and display arrangements, and the amount, type and arrangement of equipment required to complete the mission.

The selected re-entry vehicle, the Model 410, has a hypersonic L/D ratio of 0.77 with a corresponding W/C_DA of 142. The vehicle geometry and characteristics are shown in Fig. III–1.

The general arrangement of the vehicle and its contents is shown in Fig. III–2.

The alternate re-entry body arrangements referred to in this report are shown in Chapter VII.

A. AERODYNAMIC CONSIDERATIONS

The aerodynamic considerations leading to the selection of the re-entry vehicle configuration are discussed here. In particular, it describes:

(1) The manner in which the design requirements of atmospheric maneuverability, maximum allowable load factor and corridor width define the L/D requirement for the reentry vehicle.

(2) The cost, in terms of total heat load to the vehicle associated with the selected criteria.

(3) The major aerodynamic considerations influencing the choice of configuration once the L/D requirement has been defined.

Consider first the relationships between the maximum allowable load factor, corridor width and L/D. Figures III–3 and III–4 compare the corridor performance of five representative reentry vehicles with maximum L/D of 0.35 to 0.8. All corridor widths shown are referenced to the positive $C_{L_{max}}$ overshoot boundary. The maximum load factor experienced during re-entry is shown as a function of the corridor width for several lift control techniques. For the lowest L/D configuration, Mercury, entry at constant (L/D) maximum of 0.35 will provide a 6 g limited corridor width of 12 naut mi. The maximum load factor for a 30-naut mi corridor is 11.5 g.

The L–2–C configuration, with a maximum L/D of 0.54, has somewhat g-limited corridors. The 6 g limited corridor width is 18 naut mi and the corridor maximum load factor is 9 g for entries at constant (L/D) maximum. The use of lift modulation during re-entry actually degrades the g-limited corridor performance of Mercury-type vehicles. The reason for this is described later in this chapter.

For the M–1, a forward facing cone with about the same maximum L/D as the L–2–C, the constant L/D g-limited corridor widths are essentially the same as those of the L–2–C. Maximum load factor experienced within the 30-naut mi corridor is about 9 g. However, if lift modulation can be used by this vehicle to increase g-limited corridor widths, 30-naut mi corridor entry would encounter a maximum load factor of 7 g with the use of full lift modulation.

The g-limited corridor performance of the remaining two vehicles, the W–1 and M–410, are nearly the same. The corridor performance with four types of lift control during re-entry are illustrated for both vehicles. These are:

(1) Constant C_L, C_D re-entry at $C_{L_{max}}$.

(2) Constant C_L, C_D re-entry at (L/D) max.

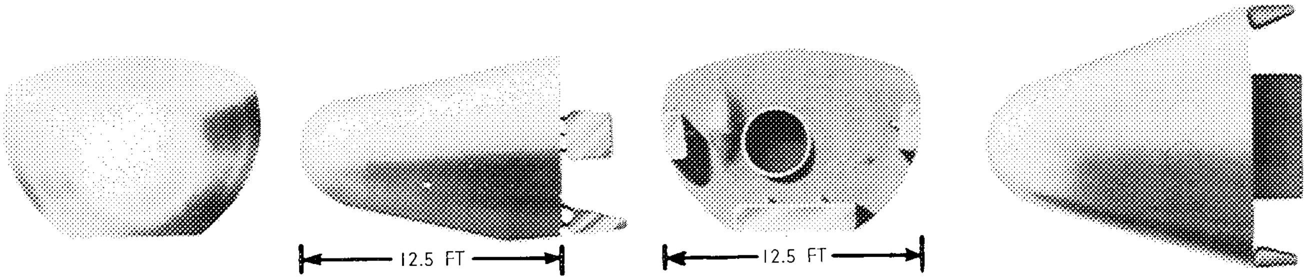
(3) Partial lift modulation from $C_{L_{max}}$ to (L/D) max.

(4) Full lift modulation from $C_{L_{max}}$ to $C_L = 0$.

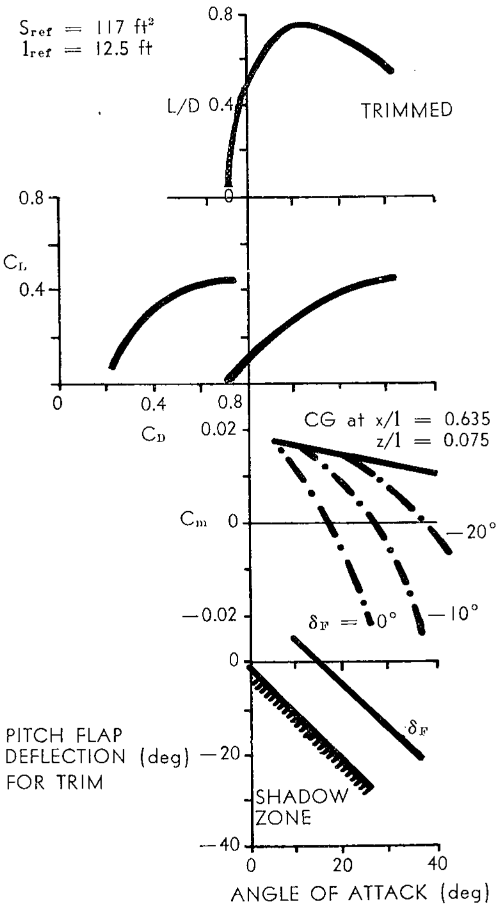
For the M–410 configuration, the 30-naut mi corridor maximum load factor is 9 g for constant $C_{L_{max}}$ re-entry. This can be reduced to 5 g for partial modulation from $C_{L_{max}}$ to $(L/D)_{max}$ or to 3 g for full modulation to $C_L = 0$. With full lift

PHYSICAL CHARACTERISTICS

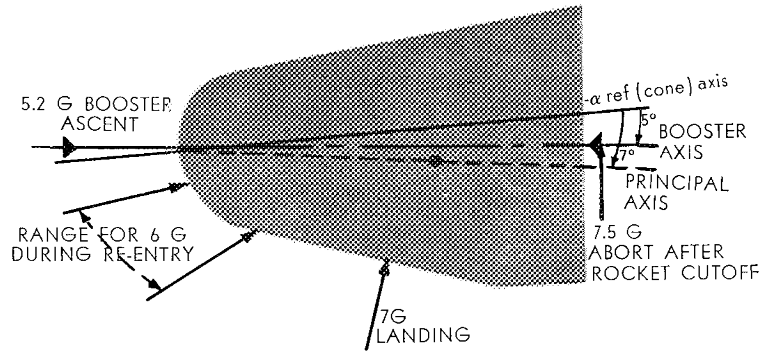
WEIGHT AT START OF RE-ENTRY	6954 LB	INERTIAS	I_{XX}	2054
WEIGHT AT END OF RE-ENTRY	5667 LB	AT START OF RE-ENTRY	I_{YY}	3253
LANDING WEIGHT	5352 LB	SLUG–FT²	I_{ZZ}	3415


12.5 FT
12.5 FT

HYPERSONIC CHARACTERISTICS



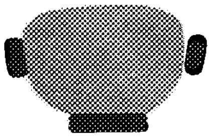
PEAK ACCELERATIONS FOR NORMAL FLIGHT CONDITIONS AND CENTER OF GRAVITY


5.2 G BOOSTER
ASCENT
-α ref (cone) axis
5°
BOOSTER
AXIS
7°
PRINCIPAL
AXIS
RANGE FOR 6 G
DURING RE-ENTRY
7.5 G
ABORT AFTER
ROCKET CUTOFF
7G
LANDING

		ALLOWABLE	ACTUAL
CG TRAVEL (%l)	AFT OF NOSE	61.5 TO 65.5	62.8 TO 65.2
	BELOW CONE AXIS	−6.5 TO 8.5	6.6

CONTROLS

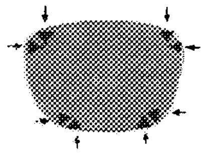
AERODYNAMIC FLAPS



YAW FLAPS $\frac{S_f}{S_{ref}} = 0.042$ (5 FT² EACH)

PITCH FLAPS $\frac{S_f}{S_{ref}} = 0.154$ (18 FT²)

REACTION JETS



THRUST = 50 LB EACH NOZZLE

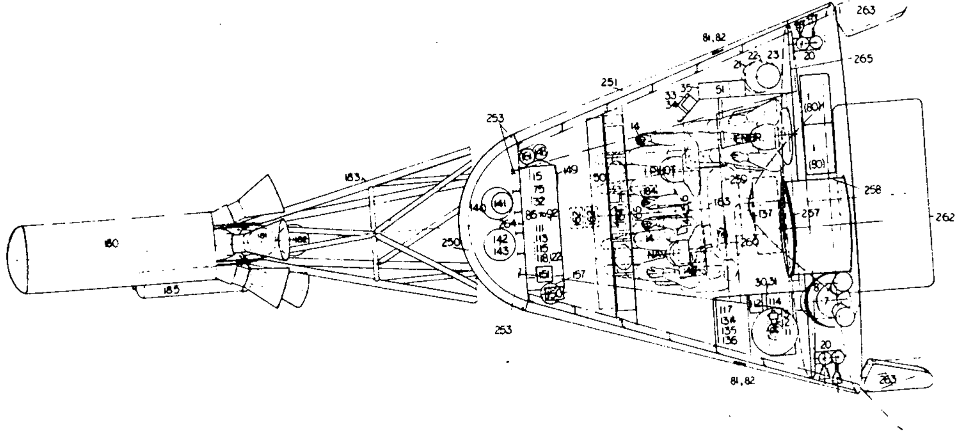
W_P = 120 LB FOR ROLL

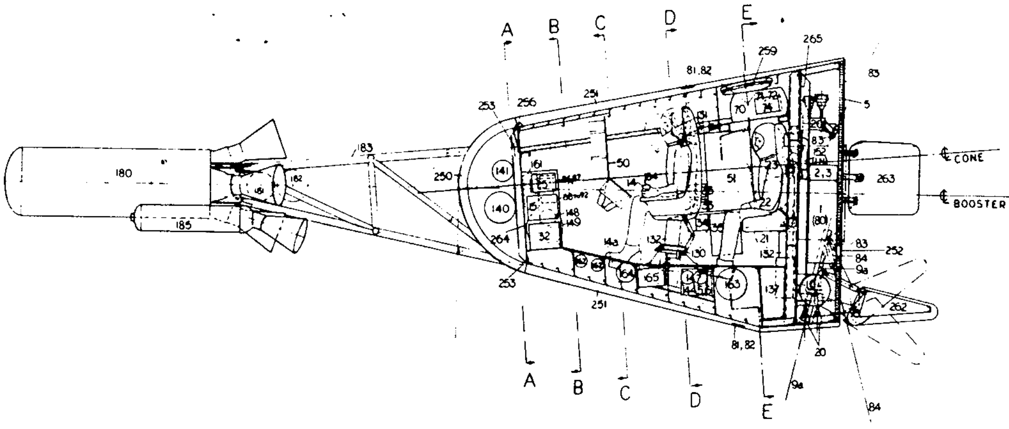
W_P = 40 LB FOR PITCH AND YAW

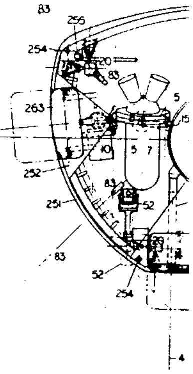
HYPERGOLIC PROPELLANTS: N_2O_4 AND UDMH/N_2H_4

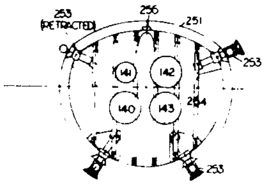
REDUNDANT (DUAL) SYSTEM

FIG. III–1. MODEL 410, RE-ENTRY VEHICLE

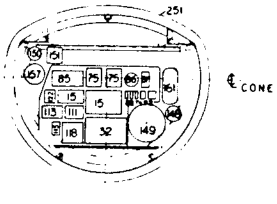

263
81,82
265
251
253
183
180
185
15
75
32
86 thru 92
111
113
115
118
122
141
142
143
149
50
PILOT
ENGR
NAV
137
257
258
262
(180)
30,31
117
134
135
136
20
81,82
253
263


A
B
C
D
E
259
265
83
81,82
251
253
256
5
℄ CONE
℄ BOOSTER
263
2,3
180
183
185
250
141
140
161
50
14
51
22
21
264
32
14a
132
130
163
165
164
(80)
252
84
9a
262
20
253
251
81,82
A
B
C
D
E
84
9a

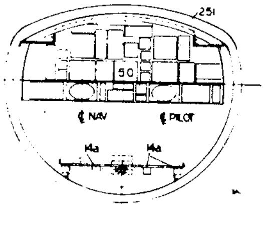

83
255
254
263
5
7
10
252
251
83
52
254
4


253 (RETRACTED)
256
251
253
141
142
140
143
264
253

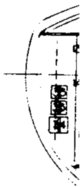
A-A


251
151
157
85
75
15
113
118
32
149
℄ CONE

B-B


251
50
℄ NAV
℄ PILOT
14a
14a

C-C



FOLDOUT FRAME 1

RECOVERY SYSTEM

1. MAIN CHUTES
2. DROGUE CHUTES
3. RISERS (DROGUE CHUTE)
4. PROBES
5. DISPOSABLE HEAT SHIELD PANELS
6. CONTROL AND DISPLAYS
7. RETROROCKET
8. ROCKET MOUNTING
9. STEERING EQUIPMENT

FLAP CONTROL SYSTEM

10. ACTUATOR PACKAGES
11. H_2O_2 TANK
12. GAS GENERATORS
13. ACCUMULATOR
14a. PRESSURE SENSORS, YAW CONTROL
14. PITCH AND ROLL CONTROL STICK
15. FLIGHT CONTROL UNITS

REACTION CONTROL

20. ATTITUDE NOZZLES
21. N_2O_4 TANK
22. UDMH/N_2H_2 TANK
23. He BOTTLE

ELECTRICAL POWER SYSTEM

30. APU
31. FUEL TANK, APU
32. BATTERIES
33. REVERSE CURRENT BREAKERS
34. VOLTAGE REGULATOR
35. DISTRIBUTION PANEL

INSTRUMENTS

50. MAIN INSTRUMENT PANEL
51. ENGINEER'S INSTRUMENT PANEL
52. TV CAMERA (EXTENSIBLE)

GUIDANCE

70. ASTRO-INERTIAL PLATFORM AND ELECTRONICS
71. MINIATURE PLATFORM
72. MINIATURE PLATFORM ELECTRONICS
73. TELESEXTANT (STOWED)
74. TELESEXTANT ELECTRONICS
75. DIGITAL COMPUTERS

COMMUNICATION AND TELEMETERING

80. RECOVERY ANTENNAS, HF AND UHF (WITH CHUTE)
81. C-BAND ANTENNAS
82. K_e-BAND ANTENNAS
83. VHF MONOPOLE ANTENNAS
84. HF ANTENNAS
85. K_e TRANSMITTER
86. C-BAND BEACON
87. HF TRANSMITTER
88. UHF TRANSMITTER
89. UHF RECEIVER
90. VHF RECEIVERS
91. VHF TRANSMITTERS
92. ANTENNA MULTIPLEXER
93. SOFAR BOMBS (WITH CHUTE)
94. DYE MARKER (WITH CHUTE)
95. RADAR CHAFF (WITH CHUTE)

INSTRUMENTATION AND SCIENTIFIC EQUIPMENT

111. TAPE RECORDER
112. RE-ENTRY INSTRUMENT PACKAGE
113. TEMPERATURE MEASUREMENT SYSTEM
114. PCM TELEMETRY SYSTEM
115. POWER SUPPLY
117. FILM (STOWAGE PROVISION)
118. RBE MEASUREMENT
122. DATA LINK SYSTEM

FURNISHING AND EQUIPMENT

130. SEAT ROTATION MECHANISM
131. SEAT TILT MECHANISM
132. SEAT SHOCK ABSORBER
134. SURVIVAL KIT
135. WATER TANK (PORTABLE)
136. FOOD STORAGE
137. WASTE STORAGE

ENVIRONMENTAL CONTROLS

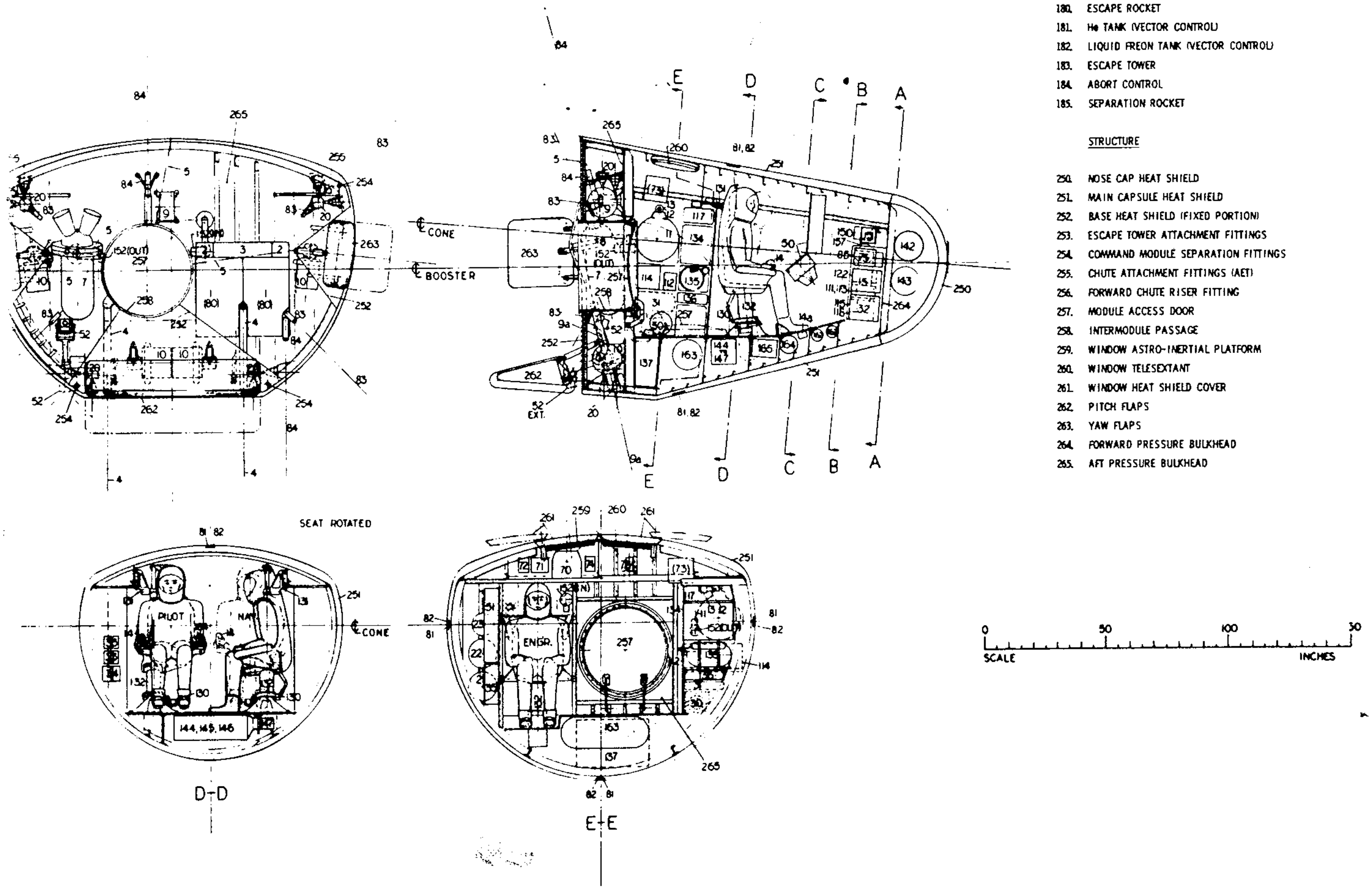
140. O_2 TANK
141. LOX TANK
142. N_2 TANK
143. LN TANK
144. FILTER ASSEMBLY
145. H_2O SEPARATOR
146. HEAT EXCHANGER
147. FAN
148. He TANK
149. H_2O TANK (CABIN AND EQUIPMENT COOLING)
150. PUMP AND MOTOR ASSEMBLY
151. CABIN PRESSURE REGULATOR
152. SNORKEL
157. ACCUMULATOR
161. WATER-GLYCOL TANK
162. PUMP AND MOTOR
163. WATER TANK (STRUCTURE COOLING)
164. ACCUMULATOR
165. HEAT EXCHANGER

LAUNCH ESCAPE SYSTEM

180. ESCAPE ROCKET
181. He TANK (VECTOR CONTROL)
182. LIQUID FREON TANK (VECTOR CONTROL)
183. ESCAPE TOWER
184. ABORT CONTROL
185. SEPARATION ROCKET

STRUCTURE

250. NOSE CAP HEAT SHIELD
251. MAIN CAPSULE HEAT SHIELD
252. BASE HEAT SHIELD (FIXED PORTION)
253. ESCAPE TOWER ATTACHMENT FITTINGS
254. COMMAND MODULE SEPARATION FITTINGS
255. CHUTE ATTACHMENT FITTINGS (AET)
256. FORWARD CHUTE RISER FITTING
257. MODULE ACCESS DOOR
258. INTERMODULE PASSAGE
259. WINDOW ASTRO-INERTIAL PLATFORM
260. WINDOW TELESEXTANT
261. WINDOW HEAT SHIELD COVER
262. PITCH FLAPS
263. YAW FLAPS
264. FORWARD PRESSURE BULKHEAD
265. AFT PRESSURE BULKHEAD



FOLDOUT FRAME 2

FIG. III-2. INBOARD PROFILE, COMMAND MODULE

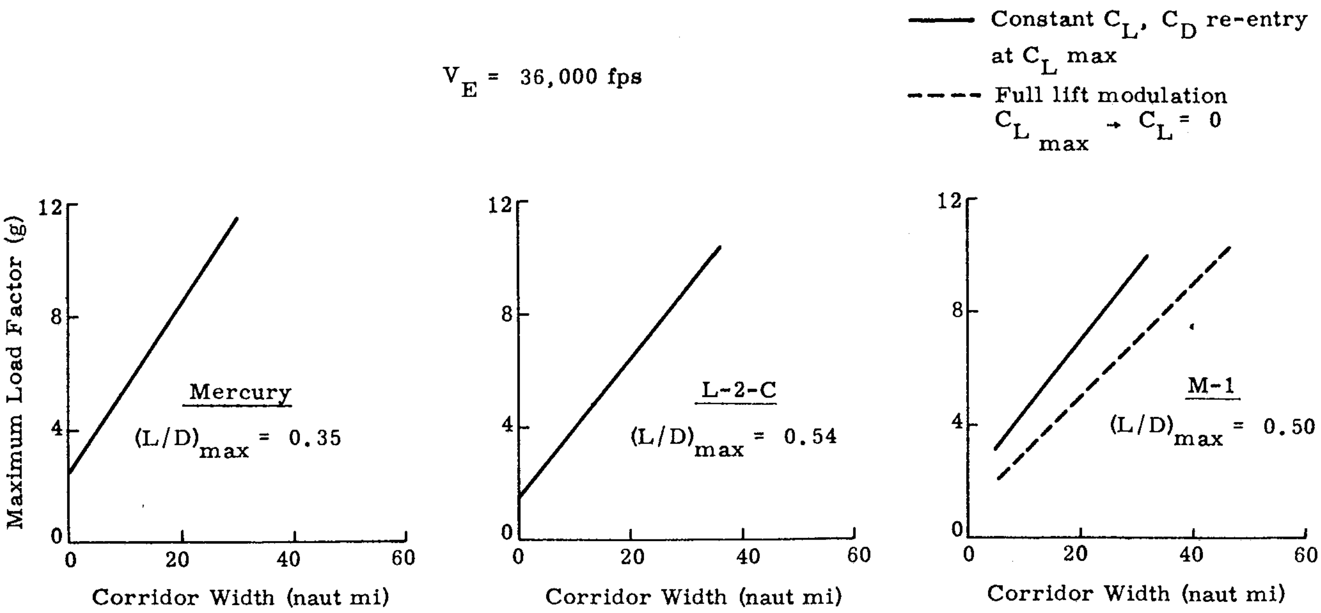


FIG. III-3. COMPARISON OF CORRIDOR PERFORMANCE

modulation, a 6 g limited corridor width of 55 naut mi is obtainable.

The use of lift modulation is usually associated with increased heating during re-entry. Some indication of the magnitude of this increase for the M-410 vehicle can be seen from Fig. III-5. This figure shows a plot of the maximum stagnation point equilibrium laminar convective and equilibrium radiative heat transfer rate versus corridor width. Two maximum heating rate curves are shown for each of the four lift control techniques. The constant $C_{L_{max}}$ re-entry gives the lowest maximum heating rates for a given corridor width. The maximum convective heating

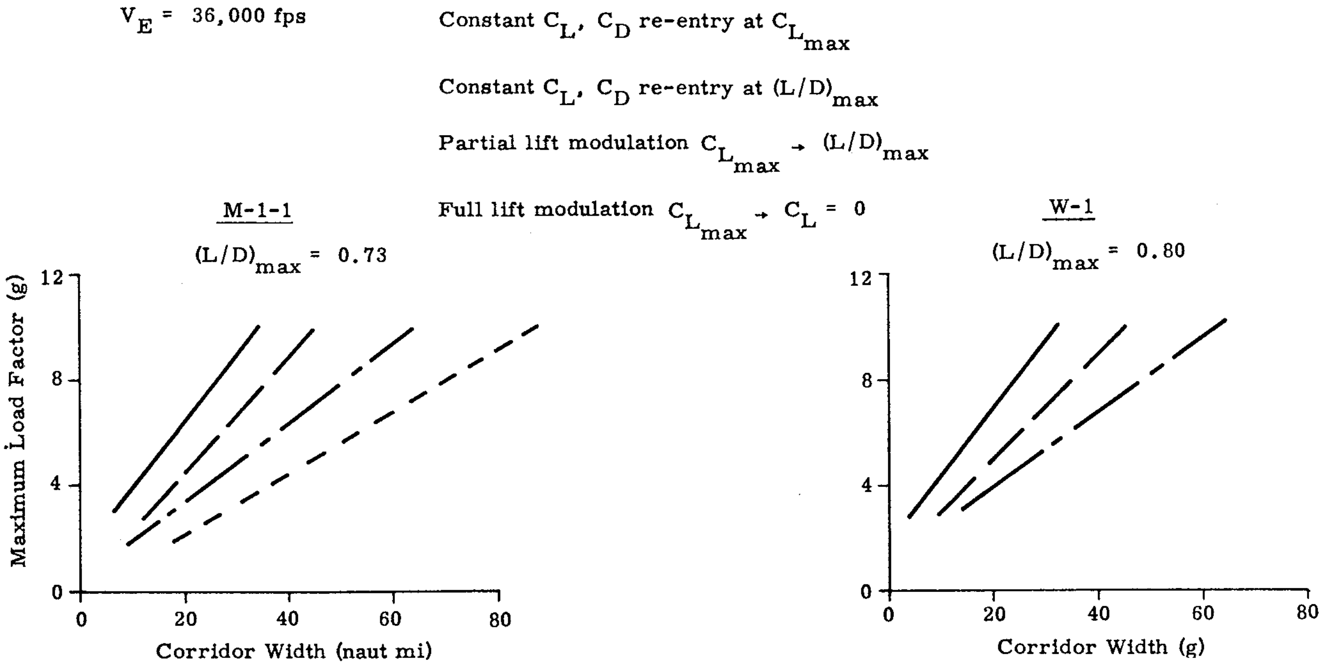


FIG. III-4. COMPARISON OF CORRIDOR PERFORMANCE

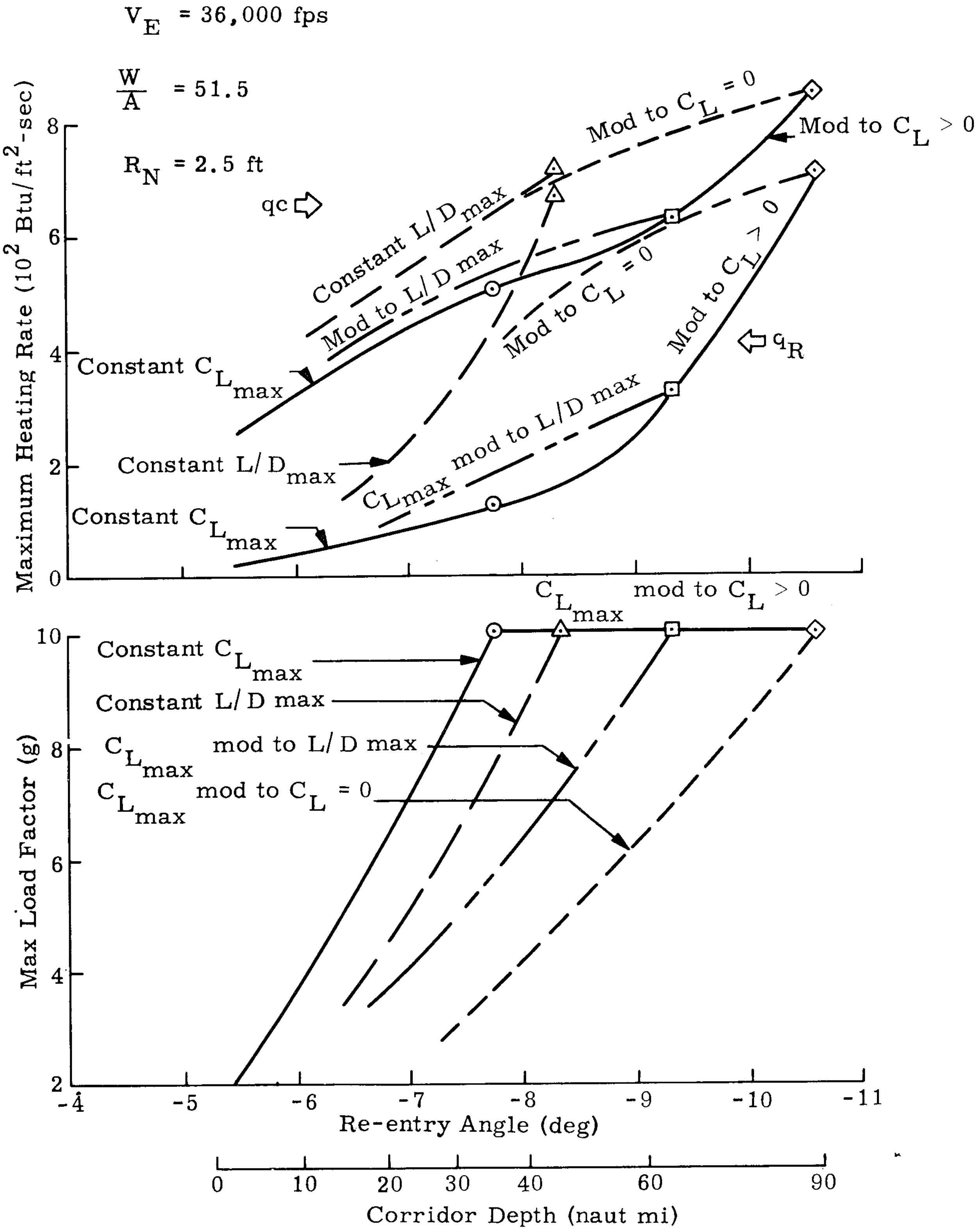


FIG. III-5. MODEL 410 MAXIMUM HEATING RATE AND MAXIMUM LOAD FACTOR VERSUS CORRIDOR DEPTH

rate is 480 Btu/ft^2/sec and radiation is 110Btu/ft^2/sec. To achieve the 55-naut mi corridor by full modulation, the maximum convective heating rate is increased to 740 Btu/ft^2/sec and the maximum radiative heating rate is increased to 560 Btu/ft^2/sec.

Partial lift modulation, however, can be used to obtain significant improvements in g-limited corridor widths with a relatively small increase in the maximum heating rates. For example, lift modulation from $C_{L_{max}}$ to $(L/D)_{max}$ can reduce the maximum load factor from 9 g at constant $C_{L_{max}}$ to 5 g for a 30-naut mi corridor. The maximum stagnation point convective heating rate is increased from 480 to 510 Btu/ft^2/sec and maximum radiative heating from 110 to 150 Btu/ft^2/sec. It is of interest to note that partial lift modulation can reduce *both* maximum load factor and maximum heating rates in comparison to constant $(L/D)_{max}$ re-entry for a given corridor width.

The corridor performance benefits attainable by lift modulation are not possible for vehicles restricted to operation on the "backside" of the drag curve as illustrated by Fig. III–6. The C_L versus C_D curve for the L–2–C is shown on the left side of the figure. It is assumed that the vehicle is restricted to angles of attack greater than that for $C_{L_{max}}$ to avoid exposure of the afterbody to excessive heating. To employ lift modulation during the re-entry, it is necessary to enter initially at a resultant force coefficient, C_R, that is greater than the minimum C_R available. In this way, C_R may be reduced as the dynamic pressure builds up, thus limiting the increase in load factor. It is seen from the drag polar that minimum C_R occurs at $C_{L_{max}}$ since the vehicle is restricted to the backside of the drag polar. Therefore, to re-enter initially at a C_R greater than the minimum, the initial lift coefficient must be less than the maximum available. This reduction in initial lift permits the vehicle to penetrate deeper into the atmosphere, resulting in higher dynamic pressures and thus offsets the g reduction obtained by modulating. It is found that the reduced lift coefficient during the initial re-entry more than offsets the g reduction by modulating so that the resulting g-limited corridor is narrower than that for constant $C_{L_{max}}$ re-entry. Figure III–6 shows the 10 g corridor widths obtained by modulation from C_{L_i} to $C_{L_{max}}$ as a function of C_{L_i}, the initial C_L at re-entry. It is possible to determine the corridor widths obtained for various values of C_{L_i} and the corresponding points on the drag polar. The maximum corridor is obtained when $C_{L_i} = C_{L_{max}}$ or in other words with no modulation. The g-limited corridor performance, therefore, of vehicles op-

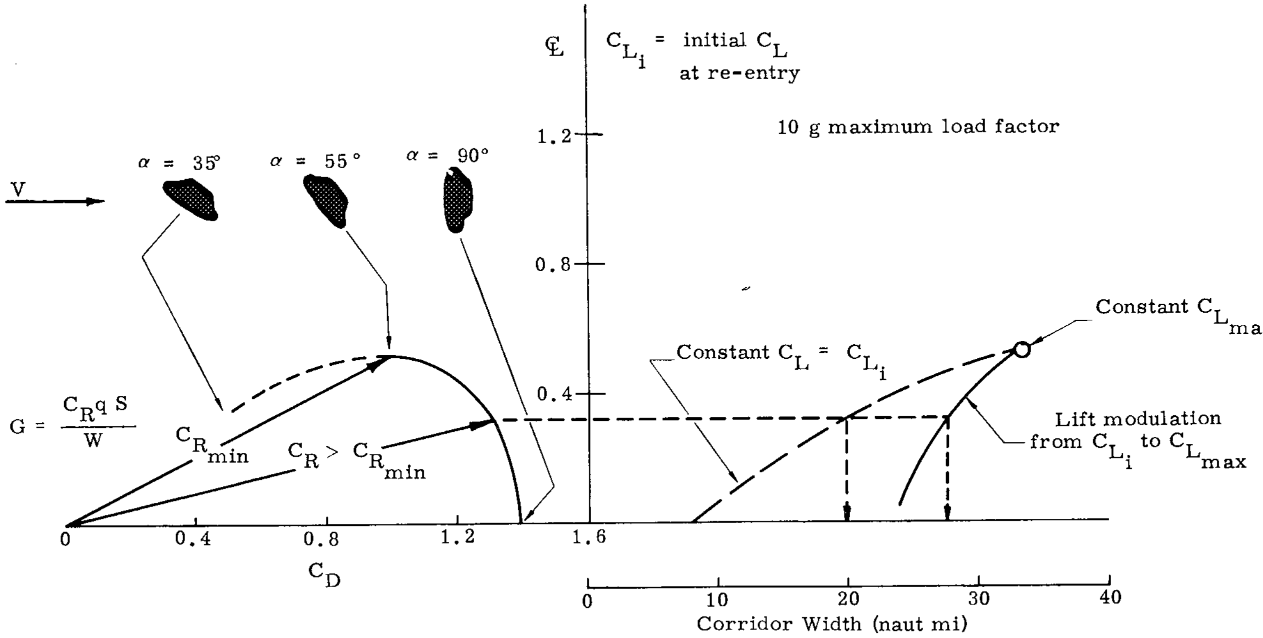


FIG. III–6. L–2–C LIFT MODULATION

erating on the "backside" of the drag curve is no better than that obtainable for entry at constant $C_{L_{max}}$.

Consider next the requirements on atmospheric maneuverability. The longitudinal re-entry range requirement is established by the selected trajectory inclination, the number of landing sites and the number of days during the lunar month on which flights are to be possible. A 35-degree inclination return trajectory has been selected in order to return along the Pacific Missile Range (PMR), making use of established tracking sites. Normal landings are to be made at a single landing site, Edwards AFB. Flights are to be possible any day of the lunar month. These requirements combine to yield a 10,000-naut mi re-entry range requirement as shown in Fig. III–7. This range must be available for anywhere in the design corridor.

To interpret this range requirement in terms of an L/D requirement, the relationship between range from an undershoot re-entry and L/D is shown in Fig. III–8 as a function of the corridor width. The corridor width used in this figure is measured from the positive $C_{L_{max}}$ overshoot boundary and is about five nautical miles below the more commonly used negative $C_{L_{max}}$ overshoot boundary. The use of this overshoot definition is based on emergency considerations. There is some indication from our analogue studies that the overshoot boundary may have to be lowered still further because of the sensitive range control conditions encountered near overshoot. Further lowering of the overshoot boundary would be reflected in reduced range capability for a given L/D.

Figure III–8 indicates that for constant L/D re-entry, a 30-naut mi corridor and a 10,000-naut mi re-entry range and an L/D of around 0.45 is required. Adding the requirements for lift modulation to reduce g and for lateral maneuverability at the 10,000-naut mi range increases the minimum L/D to greater than 0.5.

Figure III–7 can be used to interpret range requirements for other operational concepts in terms of L/D. For example, if the re-entry tracking requirements are relaxed, an increase in the return trajectory inclination to 90 degrees would reduce the required range to 5000 miles for operations on any day of the lunar month. A 5000-mile range requirement can be met by a vehicle with L/D = 0.4 within a 30-mi corridor. It can also be seen from Fig. III–8 that holding the tracking requirements but reducing the necessary launch time to one week per lunar month, will also require an L/D of only 0.4 to reach Edwards AFB.

The lateral maneuverability requirements for returns to Edwards AFB along the PMR are not severe because of the geometry of the trajectory. The approach to Edwards from due west requires longitudinal rather than lateral maneuvers to correct for errors in return time. The lateral maneuvering requirement is therefore determined by off-nominal condition.

The discussion thus far has indicated that vehicles with maximum L/D greater than 0.5 and less than 0.73 are required to meet the performance design criteria which have been selected, i.e., 10,000-naut mi re-entry range from any point within a 30-naut mi corridor with a maximum allowable load factor of 6 g. This capability is not obtained free of charge. Figure III–9 shows what meeting the selected design criteria means in terms of total heat load for the five vehicles considered earlier. Total heat load is used as the indicator of cost since it is the primary factor governing the weight of the heat shield. The comparison of total convective and radiative heat loads is made difficult by the additional variable of control technique which strongly influences the result. The present comparison has been restricted to the configuration effects by use of a single control technique consisting of an initial entry at $C_{L_{max}}$, using lift modulation, if necessary (and available), to limit the load factor to 6 g. At $\dot{h} = 0$, the constant lift coefficient necessary to transfer the vehicle to equilibrium glide conditions is applied. The desired range determines the point at which the equilbrium glide at $(L/D)_{max}$ is intercepted. This control technique is sufficiently representative of those being considered to compare total heat loads.

The major conclusion which can be drawn from Fig. III–9 is that the higher L/D configurations required to meet the performance criteria encounter about twice as much convective heat as the lower L/D vehicles at the 10,000-naut mi range and about four times that of the low L/D vehicle at the 2000-naut mi range. The radiation shielding analysis of ER 12018 indicates, however, that the additional heat shield weight serves the dual purpose of providing radiation protec-

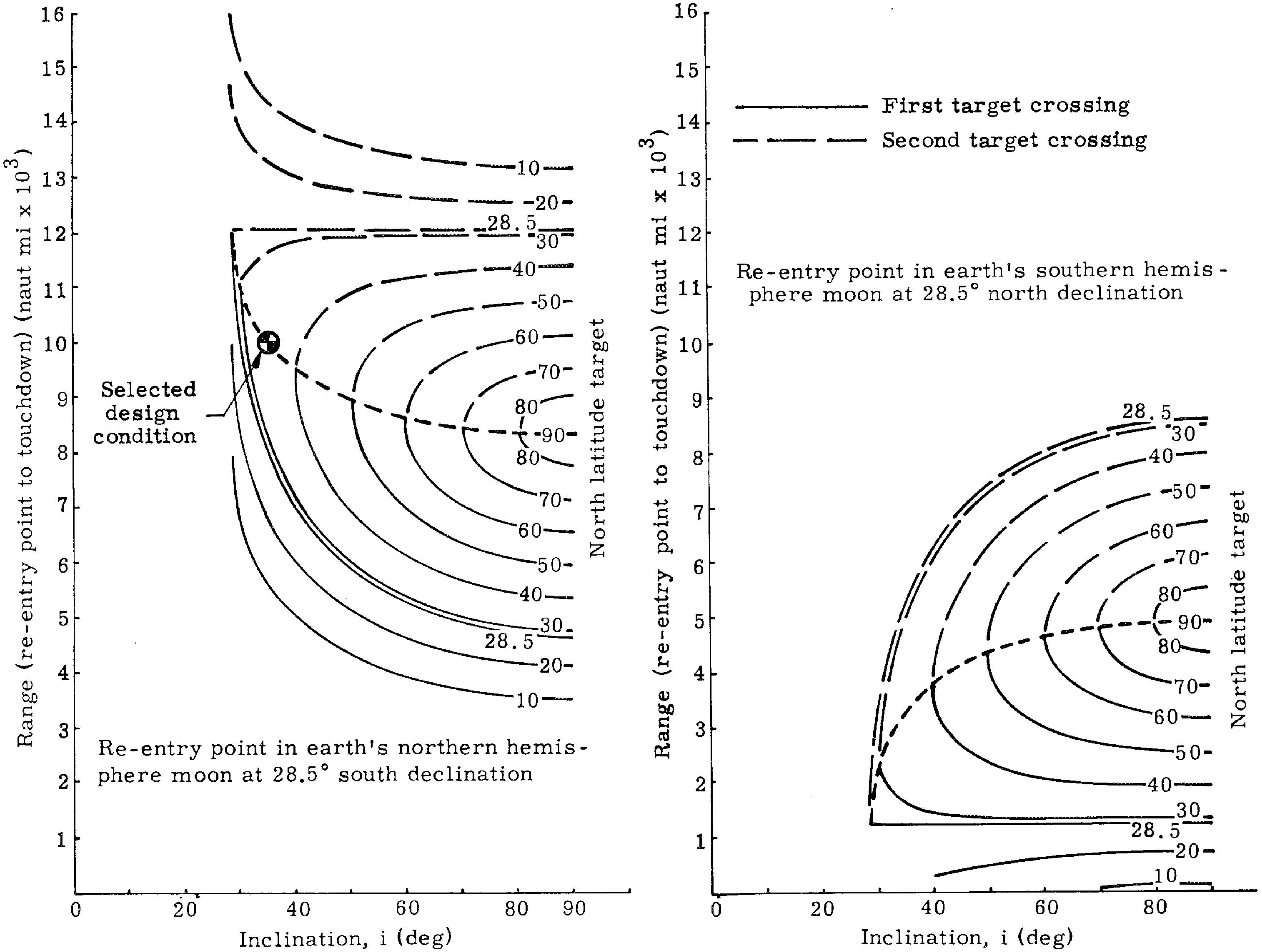


FIG. III–7. RE-ENTRY LONGITUDINAL RANGE REQUIREMENTS

tion. A detailed description of the methods used in preparing Fig. III–9 are contained in ER 12006.

The preceding discussion indicates the increase in total heating experienced by vehicles whose performance is sufficient to meet the selected design criteria as compared to a minimum vehicle capable of meeting the less stringent NASA guidelines.

The design performance requirements have been interpreted in terms of the L/D required to provide this performance and the cost of this capability has been indicated in terms of total heat load. The remaining step is to select the re-entry vehicle configuration to meet the L/D requirement. Both forward- and aft-facing cones were considered. The aft-facing cones had only marginal performance capabilities because of their lower L/D. In addition, a greater portion of the surface is exposed to significant radiative heating. The uncertainties of nonequilibrium radiative heating will therefore result in greater uncertainty in the vehicle total heat load. The forward-facing cones were therefore selected.

Forward-facing cones with L/D of 0.5 and 0.73 were evaluated in detail. These studies gave the unexpected result that no significant weight increase is incurred with the higher L/D. The reason for this is that the smaller cone angles associated with the higher L/D configuration result in greatly improved packaging arrangements. The improvement is sufficient to offset the increased total heat loads at the higher L/D. Therefore, the L/D of 0.7 to 0.8 was selected.

Two configurations have been considered which provide this L/D. These are the W–1 and M–410 configurations, one being essentially an inverted version of the other. The primary aerodyanamic factors affecting the choice between these two configurations concern heating and stability characteristics. Figure III–9 shows a distinct heating advantage for the W–1. This results almost entirely from the greater C_L and C_D at a given L/D for a flat bottom as opposed to the rounded bottom.

On the other hand, the flat bottom vehicle suffers from directional instability at high angles of attack and thus requires a greater stabilizing

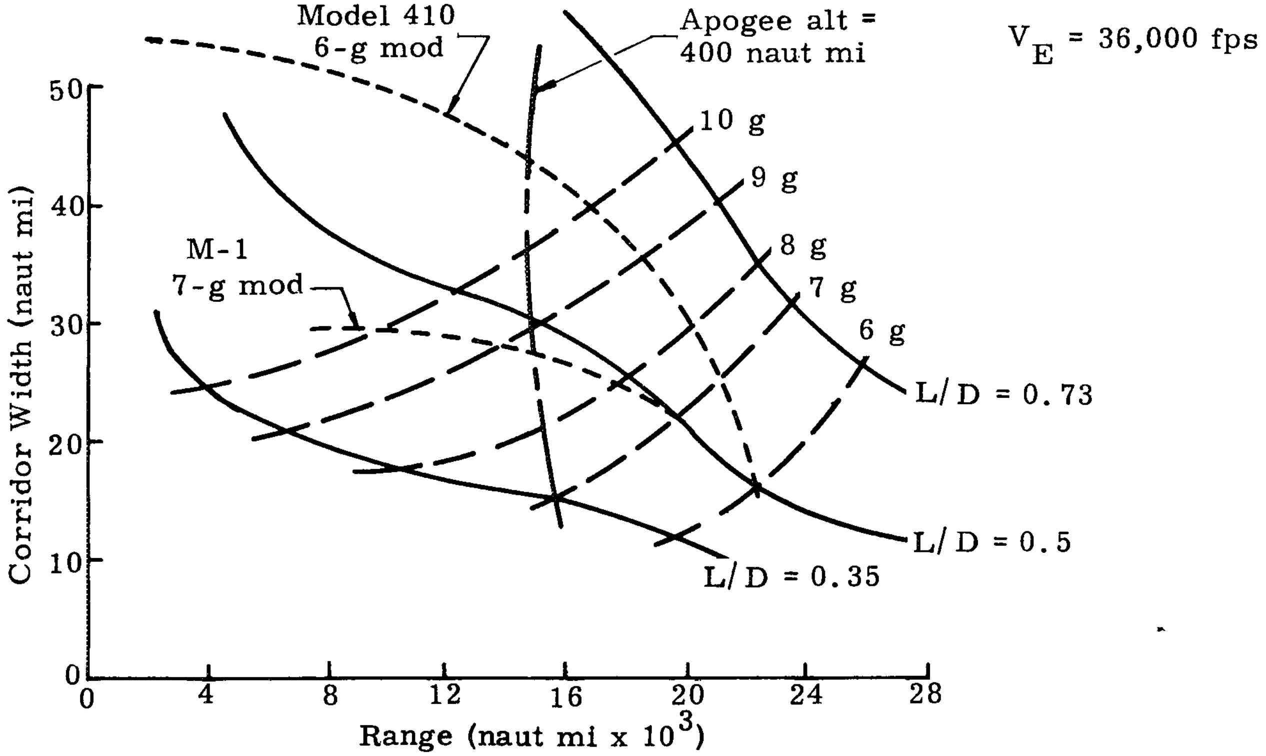


FIG. III–8. RE-ENTRY PERFORMANCE

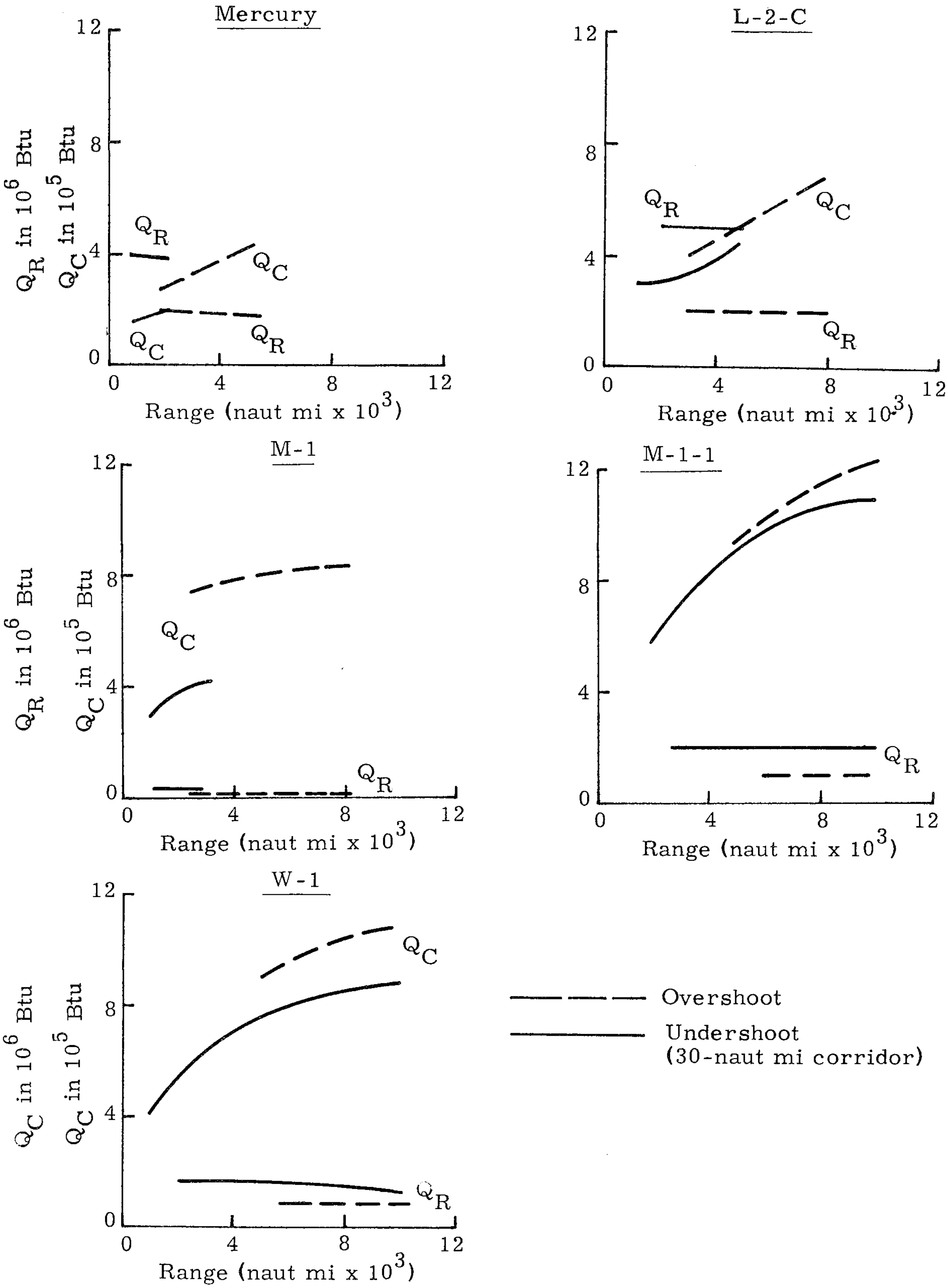


FIG. III-9. COMPARISON OF TOTAL CONVECTIVE AND RADIATIVE HEAT LOAD TO THE ENTIRE RE-ENTRY VEHICLE

fin area than the M–410. It is not possible to obtain sufficiently accurate estimates of both heating and stability with present theoretical methods on which to resolve the relative weight advantage. A selection on purely technical grounds would not be possible without appropriate experimental data. Furthermore, the weight advantage of one vehicle over the other appeared to be small regardless of the final result. Therefore, a round bottom vehicle has been selected because more applicable wind tunnel data on which to base preliminary design estimates are available. The resulting configuration, Model 410, is a spherically, blunted, flat-topped half cone of 18-degrees semivertex angle and is shown in Fig. III–1.

The spherical nose results in large aerodynamic normal and axial loads forward of the cg which affect trim and stability according to both the axial and the vertical location of the cg. The geometry of the nose and cg location can be adjusted to provide sensitive adjustments to trim and stability. The rounded bottom semicone results in large loads aft which provide basic stability to the vehicle. Appropriate pitch flaps are provided to maintain stability and control over the required range of angles of attack.

B. THERMAL PROTECTION SYSTEM

Minimum thermal protection system weight per pound of re-entry vehicle weight is achieved by use of a ballistic-type vehicle designed for only one re-entry condition. The application of such a vehicle to the Apollo mission as defined by the ground rules is not feasible since several of the Apollo operational concepts cannot be achieved except by a lifting vehicle. Since the thermal protection system weight is proportional to heat input and since heat input will increase with an increase in range and decrease in allowable load factor, the amount of heat shield required can be determined only if the operational ground rules are firmly established. No attempt is made in this section to justify the amount of lift or L/D required (see Section A, Chapter III) but only to discuss the thermal protection requirements of two basic vehicle types. These are the forward-facing cones (M–1, M–1–1, W–1, M–410) and the aft-facing cones (Mercury, L–2–C). A number of basic characteristics and their effects on thermal protection are shown.

1. DEGREE OF BLUNTNESS

A high degree of bluntness, similar to the Mercury or L–2–C, is desirable to absorb convective heating if the afterbody can be put in an area of low heating rate such that radiative-type heat shields can be utilized efficiently. However, this same high degree of bluntness will result in increased radiative heating in the nose area. The nose is ablative, therefore, radiative heating is not desirable. Since the magnitude of the radiation heating is not firmly established, particularly in view of the nonequilibrium conditions, this radiative heating could be an important consideration in choice of vehicles. A compromise in the degree of bluntness will probably yield the minimum weight.

2. LIFT MODULATION

Either type of vehicle (Mercury or lifting body) could be flown using either roll or pitch control. Studies to date indicate that the total heat input to the body will be the approximate equal for the same range. However, the fact that the heat distribution around the body changes with lift modulation will result in a heavier composite heat shield for the pitch-controlled vehicle.

3. EFFECTS OF ORIENTATION OF BOOSTER ON HEAT SHIELD

The two types of vehicles are oriented 180 degrees apart on the booster for the arrangements considered applicable to the selected design. The two orientations are re-entry nose forward (on ascent) and re-entry nose aft (on ascent). The orientation effect on the heat protection system is mainly concerned with three areas: ascent heating, LEPS attachment and main hatch details.

(1) The ascent heating is not a problem for either vehicle since heat inputs are small on ascent as compared to re-entry.

(2) The re-entry launch escape tower attachment is a more serious problem for the forward-facing vehicle than for the aft-facing one although a practical method of attaching through the main heat shield has been found.

(3) The use of the modular approach creates a greater problem for the aft-facing vehicle than for the forward-facing vehicle since the hatch leading to the mission module will pierce the main heat shield of the aft-facing body. (The consequences of placing the mission module forward of the command module have been discussed in chapter II.)

C. ARRANGEMENT CONSIDERATIONS

The number of crew members, the crew functions and the onboard equipment influence the size and shape of the re-entry vehicle in various degrees.

Variation in the number of crew members obviously requires variation in re-entry vehicle size. The degree of vehicle size change involved is a function of the particular type of re-entry vehicle being considered, and may vary, often as a step function, through a wide range.

Figure III–10 shows the results of a size comparison study for a typical lenticular-shaped re-entry vehicle, using two to five crew members. Crew head room is the controlling size factor for this type vehicle.

Figure III–11 shows the results of a similar study based on an M–1 re-entry vehicle.

The lenticular shape is relatively insensitive to changes in the number of crew members, while the M–1 shape is very sensitive.

Model 410 is more size-sensitive than the lenticular shape, but for the Apollo mission, with 3 crew members, it actually has a smaller volume. Model 410 is less sensitive than the M–1 shape since its cone angle is smaller. Size variation for the M–1 (and for M–410) occur as steps with changes from 2 crew members to either 3 or 4 and again at the change from 4 to either 5 or 6.

The re-entry vehicle shape and size are affected by the crew orientation with respect to each other, which is, in turn, dictated by duties to be performed by them, and their body positions relative to accelerations. Mission requirements suggest, for optimum utilization of displays, a side-by-side arrangement for two stations. Manual star sighting for navigation requires space for a manned station at a viewing port which, in turn, requires proper orientation of the antennas and space radiators to the viewing port. The crew body positions relative to launch, abort, re-entry and landing accelerations determines the kind of body support (seat or couch) and shock attenuation devices required.

In addition, many of the displays and controls must be movable so that they can be used if radical body re-orientation is required by ac-

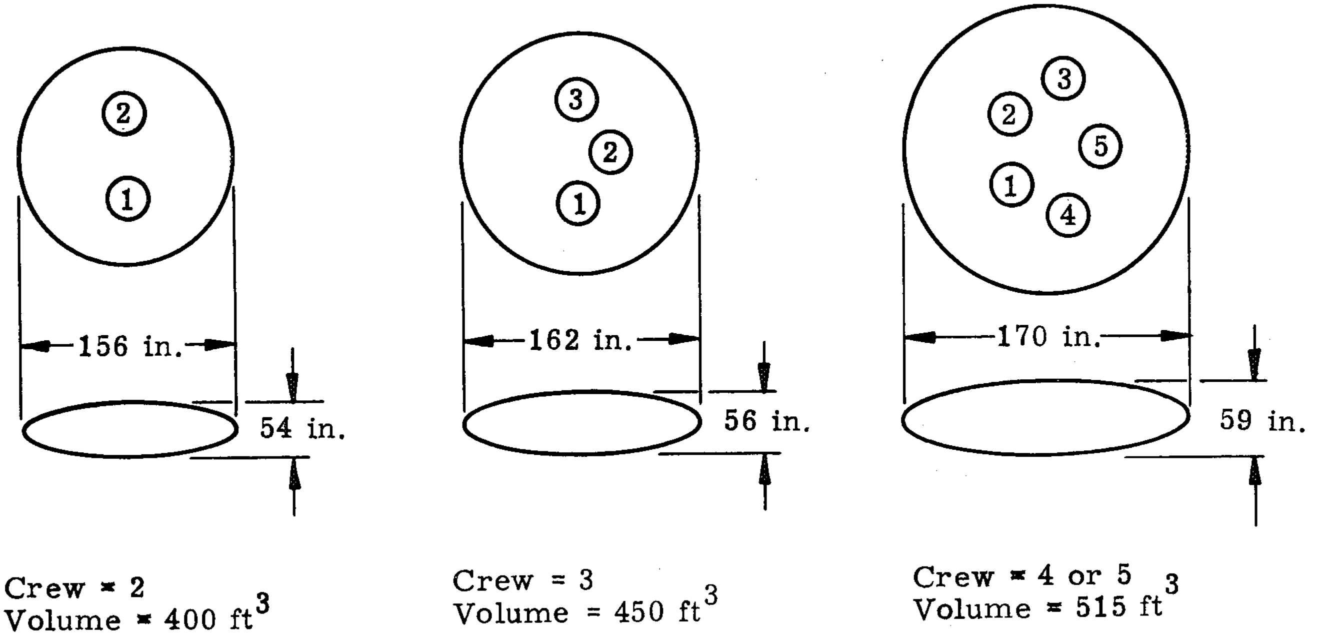


FIG. III–10. LENTICULAR CONFIGURATIONS

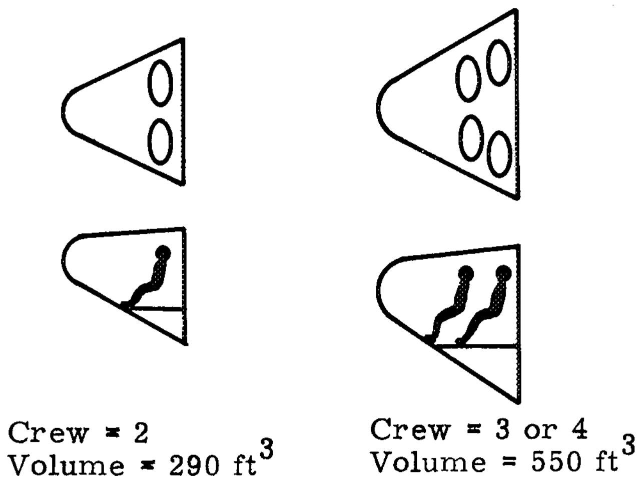


FIG. III-11. M-1 CONFIGURATIONS

celerations. The magnitude of the size change to accommodate a drastic change in body position is shown in Fig. III-12. The study is based on 3-man L-2-C re-entry vehicle. The basic size and arrangement is shown in Figure III-12a. A crew function performed by single crew member in the position sketched in Fig.III-12b necessitates an increase of 1½ ft in length. If the geometry of the vehicle is maintained, the diameter at the heat shield is increased by 3 ft, and the volume of the vehicle is approximately doubled.

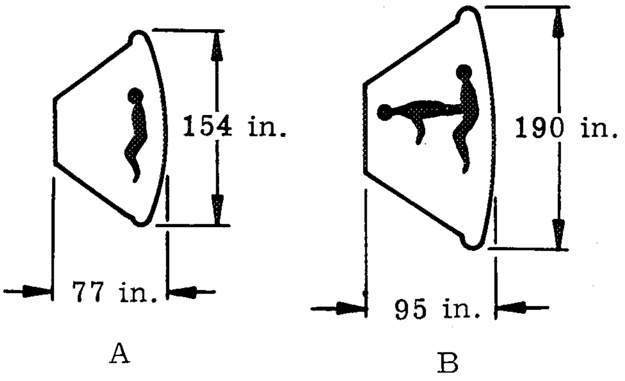


FIG. III-12. L-2-C CONFIGURATIONS

A considerably smaller volumetric change will result if the geometry of the vehicle is allowed to change sufficiently to hold the original diameters at each end and increase only the length, but this may expose the aft cone at high angles of attack during re-entry.

The arrangement of on-board equipments has been examined in several types of integrated and modular re-entry vehicles. Proper installation and arrangement is more difficult for the integrated types, but is not an insurmountable problem. Very large items, like the lunar camera, can be given primary consideration, and smaller equipment can be fitted into the remaining areas. Access for equipment maintenance is a design point that needs considerable attention, but no unusual or unsolvable problems are evident. The magnitude of the problem is somewhat greater for the integrated concept than for the modular type.

The volume of equipments in the inhabited modules does vary somewhat with mission duration, number of crew members and the scientific duties to be performed, but within the limits encountered with the Apollo, the overall change is small, and can be neglected in all but the most detailed studies. Studies have repeatedly shown that the crew is the most difficult "package" to be carried, and if the size of the re-entry vehicle is adequate for the crew, sufficient volume for the equipments is usually available.

1. COMPARISON OF MODULAR AND INTEGRATED CONCEPTS

A direct comparison of the modular concept and the integrated concept has been made with the L-2-C re-entry vehicle shape and a 3-man crew. The internal volume of the re-entry vehicle in the modular study was 380 cu ft, and the mission-module volume was 400 cu ft. For the integrated vehicle, a 600 cu ft re-entry vehicle was selected.

The study proved that all essential equipment could be properly installed in each vehicle. The integrated configuration demands considerably greater volumetric efficiency if it contains a privacy or exercise area. The integrated concept circumvents, however, at least one major problem which cannot be avoided with the modular configuration—provision for crew passage from one module to the other. This problem is especially difficult for the L-2-C (and other similar re-entry vehicle shapes) since the passage way must be located at, or near, the most critical part of the heat shield. The magnitude of the problem is obviously much smaller for the M–410 re-entry vehicle, since the heat shield is penetrated at the least critical area.

The versatility of Apollo is enhanced by the modular concept, and the growth potential is greatly extended. Operations on the lunar surface would be benefited. Maintenance is easier, since the equipment can be arranged in a more favorable manner. Privacy and exercise areas are readily available in the mission module, and the configuration virtually eliminates the possibility of interference (or annoyance) caused by simultaneous performance of the various crew functions. The increase in work area may also increase reliability by permitting some functions, such as lunar photography to be performed manually or semi-automatically rather than by completely automatic procedures.

The complications introduced by the modular concept are not great and are judged to be thoroughly worthwhile in view of the many benefits produced.

2. PERSONNEL INGRESS AND EGRESS

A single, circular, generously-sized hatch located near the center of the aft pressure bulkhead is the only personnel entrance to the M–410 re-entry vehicle. Components of the landing system, control system, and other equipment are arranged about this bulkhead so that the hatch is completely clear at all times. The 3-ft diameter of the hatch is adequate not only for personnel passage, but also will handle any equipment which must be placed inside the re-entry vehicle. Although M–410 has two stable flotation attitudes (upright and inverted), the hatch is above the water line regardless of orientation after a water landing.

Problems associated with other types of re-entry vehicles are numerous and in most cases are more serious.

The L–2–C re-entry shape, for example, was found to require two hatches. The shape has two stable flotation attitudes (small end submerged and heat shield end submerged) and is so shaped that a side hatch could be partially submerged regardless of the flotation attitude. The position of the stowed parachutes and of the LEPS tower make the small end impractical for a pre-launch entrance hatch, but the hatch is nevertheless necessary for egress after a water landing if the vehicle is floating with the heat shield end submerged. Consequently, this vehicle would need a hatch in the small end of the vehicle and another penetrating the heat shield at its most critical region.

D. LANDING SYSTEM

The ideal spacecraft configuration is one that can be completely controlled in acceleration, velocity and trajectory from launch to landing, but such a design is impossible within the established weight constraints.

Vehicle maneuverability in the post re-entry and landing phase somewhat reduces the need for accuracy in the guidance and control subsystems during re-entry (assuming that some form of position updating will be possible), permits minimizing wind drift, and allows the choice of the most desirable landing area within the available glide range. Several types of configurations were evaluated during the Apollo studies. They are described below.

1. CONFIGURATIONS

(1) *Airplane-type (fly-in landing vehicles using present pilot skills and control systems.* These included a Dyna-Soar-type winged vehicle and two lifting bodies, a flat topped cone (M–2B), and that interesting new family of shapes, the Lenticular, such as L–7. They all had the common problems of excessive spacecraft weight, booster compatibility and balance. They also require long, prepared-surface runways (over 5000 ft) because of their landing speeds, upwards of 130 knots. This high-speed landing requirement is not compatible with Apollo because of: (a) the "escape from the pad" situation, and (b) the possibility of all weather landing on poor terrain or rough water. These constraints can be met by auxiliary systems at the price of even more weight and a more complex development program. Accordingly, the airplane-type vehicles were discarded.

(2) *The Rogallo Kite flexible wing.* This device was particularly attractive in view of its recently proven feasibility from NASA model tests. In theory, this vehicle combines the long glide range, maneuverability and horizontal landing advantages with the light weight, compact stowage of the parachute. Considerable analytical effort was expended in evaluating its application to Apollo. (Reference ER 12005.) The

flexible wing was eliminated for the following reasons:

(a) Considerable development and testing would be required to "man-rate" the system.

(b) Erection under in-flight conditions is difficult since the required lengths of the keel and leading edge members are several times the length of the stowage area in the vehicle.

(c) Difficulties in packaging while attaining the required vehicle center of gravity.

(d) Absorption of landing loads, particularly under sea state 4 or unprepared terrain conditions with stall speeds above 40 knots. (Parachute vertical-descent velocity is essentially exchanged for a higher horizontal velocity.)

(e) Pilot control is always required for a safe landing.

(3) *Powered-rotor systems*—an example of this type of system features 30-ft diameter 3-bladed rotors with hydrogen peroxide-and-catalyst tip-rockets. The rotors are telescoped into a heat-shielded, trailing storage cylinder 3 feet in diameter and 9 feet long. The major features of the system are: (a) drag modulation for controllable rate of descent, (b) glide capability, (c) steering capability during the glide, and (d) about five minutes of helicopter flight to cruise to a desirable landing site.

However, the landing maneuver requires an accurate altitude sensor for height-above-ground measurement. Piloting technique requires excellent training and coordination with little chance of surviving an error. This is also true of an automatic flare-out device. The extremely high weight and volume places this system out of the application range for the Apollo vehicle. The stowage problems for the rotor blades would impose undesirable shape changes in the lifting body configuration. In addition, this system has no backup and depends on the rotor system to function without failure.

(4) *Parachute landing systems.* A number of assisted landing systems have been investigated for Apollo. These incorporated plain and steerable parachutes, various types of shock absorbing devices such as landing bags, and retrorockets. In view of the fact that some maneuverability is desirable for local obstacle avoidance, the steerable type parachute was chosen for Apollo. Selection of the retrorocket in conjunction with the parachute enables minimum touchdown velocity and crew load factors.

The main and backup parachutes systems are stowed on the aft bulkhead of the command module along with the retrorocket. The parachute is sized to give a nominal rate of descent of 40 fps. The retrorockets reduce the velocity at impact to a nominal 3 fps. In case of retrorocket malfunction, the landing shock will be absorbed by crushable structure incorporated in the crew seats. This system can overcome wind drift during its terminal descent, provides maneuverability to avoid local obstacles and requires minimum pilot skill.

2. LANDING LOADS

The best vehicle shape for minimizing vertical landing loads is a slender cone with the apex down. This configuration allows deceleration over a period of time as the body enters the more dense media of water or soil. Water stability of such a vehicle is dependent on a low center of gravity location with a maximum width base. Since these two criteria are incompatible, a compromise solution must be attained. Furthermore, the requirements of in-flight stability are primary; otherwise the ship would never have an opportunity to land. These considerations emphasize the need for minimum landing velocities to alleviate deceleration and dynamic stability problems. Figure III–13 shows a computed landing load factor comparison for four different vehicle shapes. A ten-degree variation in contact angle changes load factor considerably, the worst case being a perpendicular approach. It is apparent that the flatter-bottom shapes hit harder in emergency conditions. The short base of the M–1 makes it more susceptible to tumbling in the event of a horizontal velocity component. With the Rogallo Kite fly-in method, no appreciable change is apparent since the large, flexible wing damps any tumbling tendency and the resulting load factor is a function of flare-out accuracy.

3. EFFECT OF LANDING SYSTEM ON RE-ENTRY VEHICLE SHAPE

As mentioned earlier, the stowed rotor system and folding, rigid-keel flexikite cannot be used in conjunction with the required re-entry vehicle shape. The para-retro system causes little concern, except that it must be free to deploy behind the vehicle and means must be provided to reorient the vehicle to the most favorable landing attitude. The M–410 has a rounded nose, which, with the steerable parachute-yaw jet system pro-

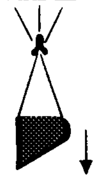
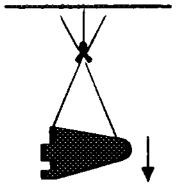
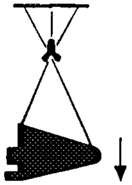
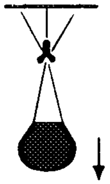
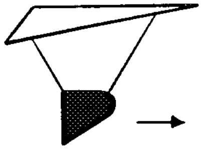
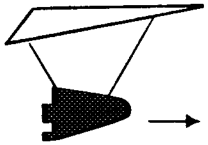
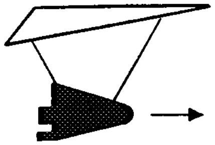
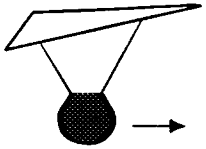
Type of Landing	M-1	Model 410	W-1	L-2C
(1) Vertical descent under parachute to:				
(a) Water landing under retrorocket (V = 8 fps)	≈ 1 g	≈ 1 g	≈ 1 g	≈ 1 g
(b) Water landing no retrorocket (V = 40 fps)	20 to 30 g (possible tumble)	25 to 35 g (possible tumble)	30 to 40 g	50 to 60 g
(c) Ground landing no retrorocket (V = 40 fps)	60 to 80 g (possible tumble)	60 to 80 g (possible tumble)	80 to 100 g	100 to 120 g
(2) Horizontal landing with Rogallo kite (V = 70 fps)				
(a) Water	5 to 10 g	5 to 10 g	5 to 10 g	5 to10 g
(b) Land (runway)	2 to 4 g	2 to 4 g	2 to 4 g	2 to 4 g

FIG. III-13. ESTIMATED NORMAL LOAD FACTOR

viding horizontal direction, has the ability to ride up over minor obstructions. The longitudinal geometry makes fore and aft overturning improbable, and the extended yaw flaps arrest lateral rotation.

E. CREW SAFETY AND DESIGN FLEXIBILITY

Other considerations in the final selection of the command module are:

(1) Radiation protection—The forward-facing vehicle, particularly in the modular approach, yields much more radiation protection for the crew than the aft-facing vehicle.

(2) Micrometeorites—The aft-facing vehicle has the advantage here in that the primary heat shield is protected from micrometeorites by the propulsion module.

(3) Flexibility—The forward-facing vehicle offers more flexibility for volume expansion since the aft end of the vehicle can be changed within center of gravity and aerodynamic limits. A volume expansion in the aft-facing vehicle would probably change all vehicle dimensions. Either of these approaches amounts to a complete redesign of the aft-facing vehicle.

F. WEIGHT CONSIDERATIONS

1. WEIGHT STUDIES

A number of command module configurations were investigated to determine whether or not a significant weight advantage applies to any particular configuration. This study was accomplished during the early part of the Apollo study program, and the weights, therefore, do not reflect the latest information regarding heat shields and systems, nor was any difference in operational concept included in the analysis. However, they are indicative of the results which might be expected on an up-to-date study. The types of re-entry bodies investigated are shown in Chapter VIII of this report. They included symmetrical bodies such as L–2–C and Mercury, unsymmetrical cones, such as W–1 and M–1; as well as fly-in lifting bodies, the lenticular L–7, and the M–2B. A summary of weights developed for the various configurations during this study is shown on Table III–1. The conclusions which were reached during this study were:

(1) The fly-in bodies, such as M–2B and lenticular L–7, are too heavy to be considered for the Apollo mission within the established weight limitations of 15,000 lb for the circumlunar mission.

(2) The integrated spacecraft is shown to be slightly heavier than the modular. In addition, the latter, disregarding weight considerations, lends itself to future missions.

(3) The total weight of any of the modular configurations is about equal.

(4) Weight, therefore, is not a significant criteria in establishment of the re-entry body aerodynamic configuration.

It appears from review of the aerodynamic heating regime and of the effects of L/D on the weights of these bodies, that an integrated spacecraft of the Mercury configuration, is somewhat lighter than the unsymmetrical forward-facing cones. However, the smaller re-entry corridors and larger load factors associated with the Mercury vehicle, as well as the lower radiation protection afforded by the heat shield at the back of the vehicle offsets these advantages. At the time of the study presented in the table, the integrated versions were sized for six crew men. A review of a smaller integrated vehicle showed that the weight saving possible through the modular approach was reduced somewhat. The weight difference between modular and integrated configurations is really not enough to force a decision one way or the other.

2. GROWTH FACTOR FOR COMMAND MODULE

Theoretical considerations of the growth factor associated with the command module show that there is some justification for choosing the modular approach from a weight standpoint.

The command module growth factor is defined as the total weight increase of the module resulting from a unit increase in module payload.

A pound of equipment added to the command module may require some or all of the following changes:

(1) Increase in internal volume
(2) Increase in structural strength
(3) Increase in the heat shield
(4) Increase in the landing system
(5) Increase in the control system and propellants
(6) Increase in environmental control
(7) Increase in electrical power
(8) Increase in LEPS capability.

Some of these changes in turn affect some of the others listed.

To get some idea of the possible magnitude of the growth factor, it is assumed that only the heat shield and the landing system have to change for an equipment change.

$$\Delta W_{RE} = \Delta W_E + \Delta W_{HS} + \Delta W_{IS} \qquad (1)$$

where W_{RE} = Change in initial re-entry weight
W_E = Given change in equipment weight
W_{HS} = Change in heat shield weight
W_{LS} = Change in landing system weight

Assume the heat varies as $\sqrt{\dfrac{W}{C_D A}}$

Therefore,

$$\Delta W_{HS} = W_{HS} \sqrt{\frac{W_{RE} + \Delta W_{RE}}{W_{RE}}} - W_{HS}$$

Where W_{HS} and W_{RE} are the initial "total" heat shield and re-entry weights.
Also, $W_{HS} = 0.3\ W_{RE}$ for the base design.

Therefore,

$$\Delta W_{HS} = 0.3\ W_{RE} \left(\sqrt{\frac{W_{RE} + \Delta W_{RE}}{W_{RE}}} - 1 \right)$$

Also from the base design,

$$\Delta W_{LS} = .07\ \Delta W_{RE}$$

Substituting in (1) results in:

$$\Delta W_{RE} = \Delta W_E + 0.3\ W_{RE} \left(\sqrt{\frac{W_{RE} + \Delta W_{RE}}{W_{RE}}} - 1 \right) + 0.07\ \Delta W_{RE}$$

Solving for $\dfrac{\Delta W_{RE}}{\Delta W_E}$, since W_{RE} is known, gives

$$\frac{\Delta W_{RE}}{\Delta W_E} = 1.28$$

Therefore, a pound of weight of equipment added to the command module that does not cause changes in volume, structure, electrical power, or other factors, but affects only the landing system and the heat shield results in a total increase of 1.28 lb.

This shows that moving 1 lb of equipment from the re-entry body to the mission module will save about 0.3 lb providing the volumes do not change. If volumes do change, the weight saving would be larger.

TABLE III-1

APOLLO

(B) RE-ENTRY BODY COMPARISON

	W-1	M-410	L-2-C	L-1	L-8	FLAPPED MERCURY	W-1 INTEGRATED	L-2-C INTEGRATED	LENTICULAR	M-2
HEAT SHIELD	1140	1148	1108	1190	1180	1158	1335	1477	1176	1841
HEAT SHIELD WATER AND SYSTEM	180	180	172	218	187	185	201	209	193	267
STRUCTURE	920	1075	870	1007	854	912	1156	1160	1247	1693
AERODYNAMICS SURFACE OR FLAP	251	221	173	188	295	155	320	251	371	872
SURFACE CONTROLS	258	236	290	155	315	234	330	403	425	600
REACTION CONTROLS	160	160	160	160	160	160	205	225	185	244
LANDING SYSTEM	450	450	450	450	450	450	575	630	622	845
AUXILIARY POWER SYSTEM	527	527	527	527	527	527	675	675	527	527
ENVIRONMENTAL CONTROL (P&E)	310	310	310	310	310	310	973	973	310	310
INSTRUMENTS	260	260	260	260	260	260	260	260	260	260
INSTRUMENTATION	98	98	98	98	98	98	122	122	98	98
COMMUNICATIONS	134	134	134	134	134	134	159	159	134	134
GUIDANCE	220	220	220	220	220	220	220	220	220	220
SCIENTIFIC EQUIPMENT	70	70	70	70	70	70	175	175	70	70
FURNISHINGS AND EQUIPMENT	239	239	239	239	239	239	413	413	239	239
CREW	630	630	630	630	630	630	630	630	630	630
TOTAL LAUNCH WEIGHT OF RE-ENTRY BODY	5847	5958	5711	5856	5929	5742	7749	7982	6707	8850

IV. MISSION MODULE

A. MISSION USES

The modular concept has been applied to the Apollo vehicle for the following reasons:

(1) The design of the spacecraft is oriented toward future missions including rendezvous and lunar exploration, both of which require airlocks. The mission module can serve as an airlock.

(2) The vehicle will be used for lunar reconnaissance in some of the missions leading to lunar landing and for general scientific investigations on all missions. The mission module arrangement provides space for the surveillance function. It also permits changes to be made in scientific instrumentation without the complication of vehicle weight, balance, and thermal protection system which would result if these changes occurred in the command module.

(3) The modular concept leads to a lighter overall vehicle. Designs with geometrically similar command modules were studied both with and without the mission module. In the modular designs, as much life support and equipment was stowed in the mission module as was possible. The weight analyses showed that the overall weight of the modular vehicles was from two to six percent lighter than the non-modular vehicle.

(4) The tanks for hypergolic storable propellants can be housed within the mission module insulation. These propellants must be held within close temperature limits, and absence of a mission module would require heaters for these tanks.

Studies of a lunar landing stage (see Chapter VIII) show that the mission module is an efficient airlock if it is used no more than 13 times. If it is used more often, auxiliary airlocks or pumping equipment would be necessary for minimum weight because operation as a simple airlock implies that all of the air is lost everytime the hatch is opened. (Of course, the volume of the selected mission module affects this answer greatly.)

B. SELECTION PROCESS

The criteria for the mission module design is as follows:

(1) The module is to be supported directly by the structural shell of the 154-in. diameter propulsion and equipment module.

(2) Entry to the module is by a hatch located at the outer propulsion and equipment module shell line.

(3) There should be 6½ ft head room in the module whether on the launch pad or on the moon.

(4) The minimum skin gage should be 0.040 to eliminate handling damage during the mission and for meteoritic protection.

(5) A cabin pressure of 12.2 psi is to be maintained. (shirt sleeve environment.)

(6) The module configuration should minimize spacecraft weight.

The module shapes investigated and the results obtained are shown in the table below.

The table shows that the transverse cylinder, which has been selected, is the lightest and permits the shortest spacecraft of any of the configurations which meet height requirements. The volume of the selected configuration is adequate to serve the intended purposes and yet is small enough to justify its use as a simple airlock.

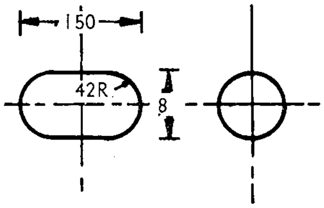
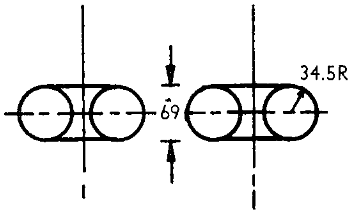
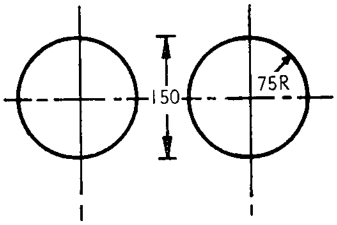
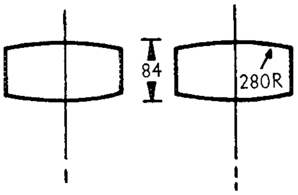
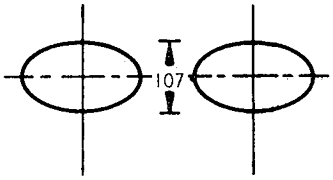
The importance of spacecraft length stems from the fact that the vehicle gross weight for the lunar orbit mission increases by about five pounds for every inch of added length.

The weight savings of the chosen configuration are greater than indicated in the table since tank internal structure is proportional to the wall weight, and some of the configurations require additional heavy framing.

The torus tank, which is the closest competitor weight wise, is lacking in height and, in addition, does not match well with the hatch location in the command module.

TABLE IV-1

MISSION MODULE CONFIGURATIONS

MODULE CONFIGURATION	VOLUME (CU FT)	MINIMUM HEIGHT	MAXIMUM HEIGHT	HEIGHT USED	TANK WEIGHT—WALL ONLY	VEHICLE LENGTH ADDED
	400	0	150	84	124	0
	545	0	69	69	220	—15*
	1020	150	150	150	284	66
	780	22.4	—	84	275	0
	720	106	212	106	200	22

* NOT SATISFACTORY HEIGHT

V. PROPULSION AND EQUIPMENT MODULE

A. REQUIREMENTS

1. SYSTEM

The propulsion and equipment module must provide space and optimum locations for such items as:

(1) Reaction control equipment.
(2) Electrical power system.
(3) Guidance equipment.
(4) Communication and telemetering equipment.
(5) Instrumentation and scientific equipment.
(6) Furnishings and equipment.
(7) Environmental controls.
(8) Module separation equipment.
(9) Propulsion system.
(10) Structure.

The arrangement of the above in the Model 410, propulsion and equipment module, is shown in Fig. V–1. The module must provide space for a mission module of 383 cu ft volume with an access hatch to the command module. Provision must be made for the propulsion system liquid hydrogen fuel tank with a volume of 400 cu ft and the liquid oxygen oxidizer tank of 125 cu ft volume. Space and structural mounting must be provided for the mission engine and its associated pumps, valves, fuel lines, etc. Mounting provisions must be provided for the vernier engines at four locations as well as for the module separation rockets and the attitude rockets. Space is required for two 27-in. diameter spherical vernier propellant tanks, for a 32-in. diameter spherical hydrogen and a 25-in diameter spherical oxygen tank for the fuel cells of the electrical power system, and a 16.5-in. diameter spherical reservoir containing high pressure helium. Adequate space is required for the three fuel cells, water tanks, and other containers. Sufficient space is also required to mount on the outer surface of the module, solar radiators of 135 sq ft for the environmental control system and 9 sq ft for the fuel cells. Mounting provisions must be provided for the guidance, communication, and telemetering equipment antennas which are in a retracted position during the boost phase and which are extended for use for the space mission. Windows, camera ports must also be provided in the module.

The total volume requirements of the larger items contained in the propulsion-equipment module are as follows:

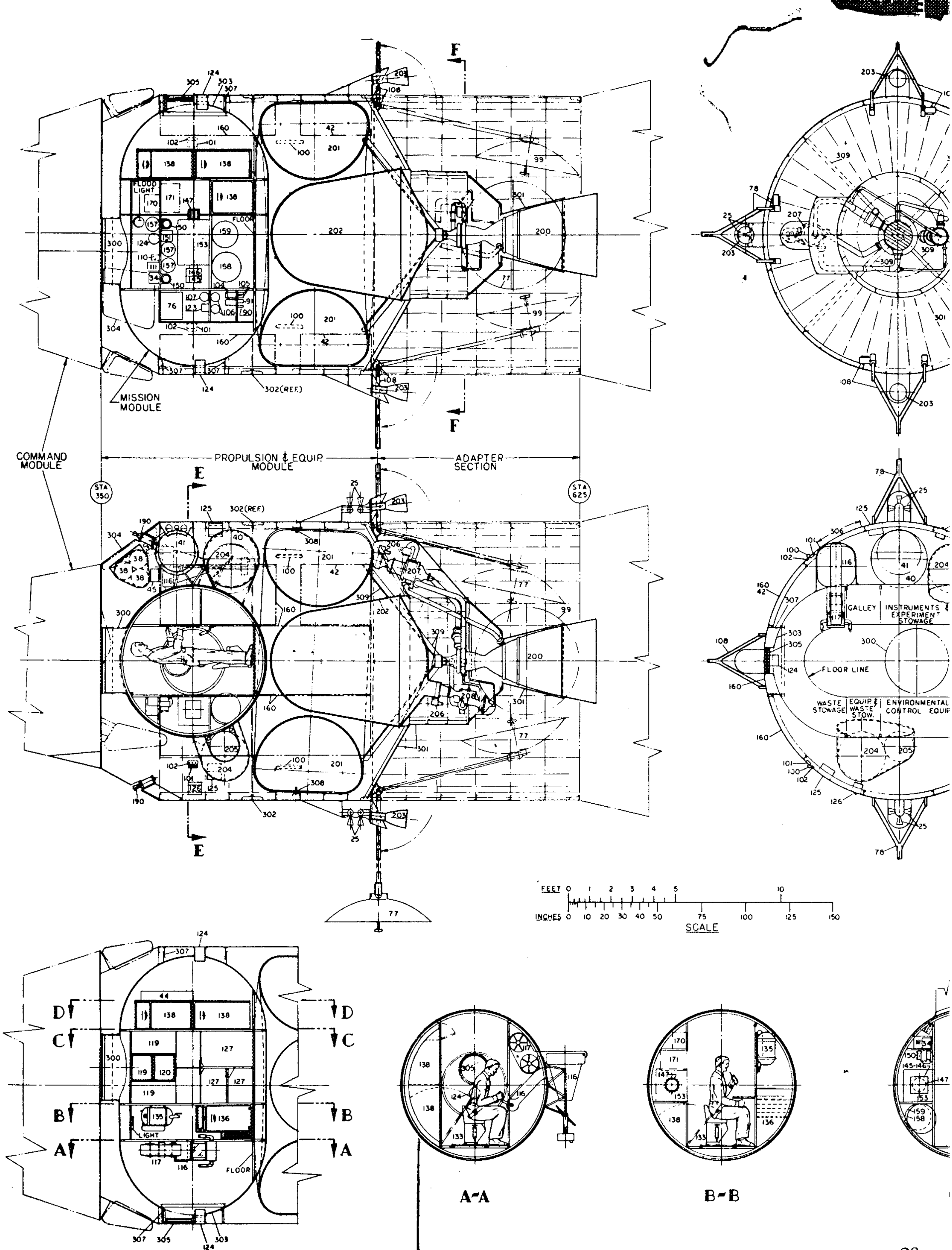
Item	*Volume (cu ft)*
Mission module	383
Hydrogen tank	400
Oxygen	122
Inter-module passage	7
Fuel cell hydrogen tank	13
Fuel cell oxygen tank	3.5
Camera	7
Vernier rocket fuel tanks	12
Helium tank	1.8
Fuel cells	5.5
Flaps (command module)	27.0
Total =	981.8

This volume does not include that required for miscellaneous equipment, plumbing, valves, structural members, etc. The available volume within the propulsion and equipment module (between the command module and the adapter section) is approximately 1,607 cu ft. This indicates that a volume usage of better than $\frac{981.8}{1{,}607}$ or 61% of the space available.

2. VEHICLE ORIENTATION

Proper orientation of the vehicle is required so that the radiators located on the outer surface of the propulsion-equipment module face into deep space. In addition, the guidance, communication and telemetering antennas must be oriented toward the earth. Another requirement is to orient the vehicle so that the fuel and oxidizer tanks, mission module, etc. are subjected to a minimum of solar heating.

The above requirements dictate that the vehicle be oriented during its space travel with the nose of the vehicle facing the sun. (See Fig. V–2.)


COMMAND MODULE
MISSION MODULE
PROPULSION & EQUIP. MODULE
ADAPTER SECTION
STA 350
STA 625
E
F
FLOOD LIGHT
FLOOR
LIGHT
GALLEY
INSTRUMENTS & EXPERIMENT STOWAGE
FLOOR LINE
WASTE STOWAGE
EQUIP & WASTE STOW.
ENVIRONMENTAL CONTROL EQUIP
302(REF.)
A
B
C
D
A-A
B-B
FEET 0 1 2 3 4 5 10
INCHES 0 10 20 30 40 50 75 100 125 150
SCALE
25
38
40
41
42
44
45
76
77
78
99
100
101
102
108
116
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120
124
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127
133
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147
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171
190
200
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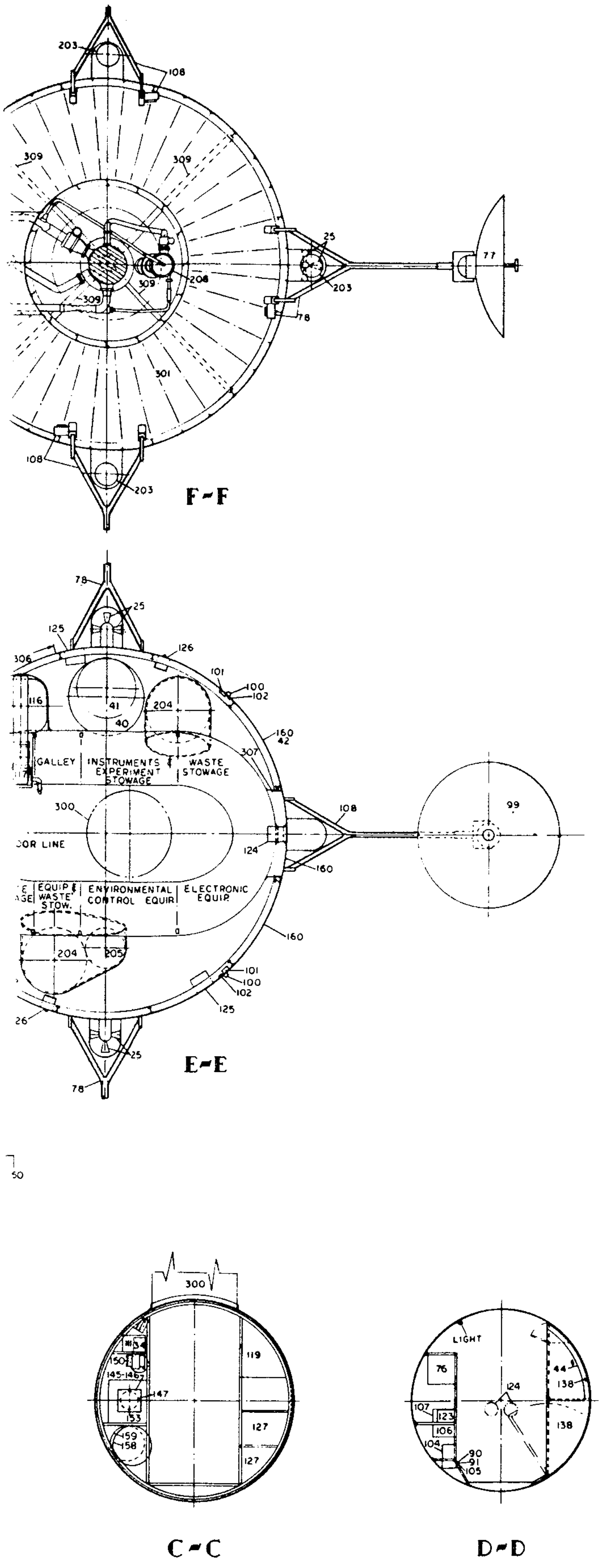
309

FOLDOUT FRAME 1


203
108
309
309
25
77
203
208
309
309
78
301
108
203
F-F
78
25
125
126
306
101
100
102
116
41
40
204
160
42
307
GALLEY
INSTRUMENTS & EXPERIMENT STOWAGE
WASTE STOWAGE
300
108
99
OR LINE
124
160
EQUIP & WASTE STOW.
ENVIRONMENTAL CONTROL EQUIP
ELECTRONIC EQUIP
160
204
205
101
100
102
125
26
25
78
E-E
300
119
150
145-146
147
153
127
127
159
158
C-C
LIGHT
76
44
138
124
107
123
106
104
138
90
91
105
D-D

REACTION CONTROL EQUIPMENT

25. ATTITUDE NOZZLES

ELECTRICAL POWER SYSTEM

34. VOLTAGE REGULATOR
38. FUEL CELL
39. F. C. & BUS CENTER
40. H_2 TANK
41. O_2 TANK
42. SPACE RADIATOR
44. ELECTRICAL DISTRIBUTION PANEL
45. WATER SEPARATOR

GUIDANCE EQUIPMENT

76. RADIO ALTIMETER
77. RADIO ALTIMETER ANTENNA
78. RADIO ANTENNA STRUT AND ERECTOR

COMMUNICATION AND TELEMETERING EQUIPMENT

90. V. H. F. RECEIVER
91. V. H. F. TRANSMITTER

99. S-BAND PARABOLIC ANTENNA
100. V. H. F. SHROUD ANTENNA
101. S-BAND ANTENNA
102. C-BAND ANTENNA

104. S-BAND TRANSMITTER
105. S-BAND RECEIVER
106. S-BAND TRACKING BEACON
107. C-BAND BEACON
108. ANTENNA STRUT AND ERECTOR

INSTRUMENTATION AND SCIENTIFIC EQUIPMENT

110. 35 MM CAMERA (SPACE PROVISION ONLY)
111. TAPE RECORDER

116. LUNAR CAMERA
117. FILM
119. EQUIPMENT CONTROL AND DISPLAY PANEL
120. EMERGENCY WARNING DISPLAY PANEL
123. SIGNAL CONDITIONING PACKAGE
124. RADIATION DETECTOR
125. MICROMETEORITE DETECTOR
126. SOLAR FLARE SENSOR
127. EXPERIMENT STOWAGE

FURNISHINGS AND EQUIPMENT

133. WASTE COLLECTION
135. WATER TANK
136. FOOD STOWAGE
138. WASTE STOWAGE

ENVIRONMENTAL CONTROLS

145. WATER SEPARATOR
146. HEAT EXCHANGER
147. FAN

150. PUMP AND MOTOR ASSEMBLY
151. CABIN PRESSURE REGULATOR
153. MOLECULAR SIEVE

157. ACCUMULATOR

158. OXYGEN TANK
159. NITROGEN TANK
160. SPACE RADIATOR

170. GAS SPECTROMETER (SPACE PROVISION ONLY)
171. GAS CHROMATOGRAPH (SPACE PROVISION ONLY)

MODULE SEPARATION EQUIPMENT

190. ROCKET

PROPULSION SYSTEM

200. MISSION ENGINE
201. TOROIDAL FUEL TANK (LH)
202. OXIDIZER TANK (LOX)
203. VERNIER ENGINE
204. VERNIER PROPELLANT TANK
205. H TANK

206. SHUT-OFF VALVE
207 HYDROGEN PUMP
208. OXYGEN PUMP

STRUCTURE

300. INTER-MODULE PASSAGE
301. METEORITE BUMPER
302. STRUCTURAL SPLICE
303. GAIN HATCH (INGRESS AND EGRESS)
304. EQUIPMENT ACCESS HATCH
305. VIEWING PORT (PERSONNEL)
306. VIEWING PORT (CAMERA)
307. MISSION MODULE SUPPORT STRUCTURE
308. H_2 TANK SUPPORT

309. O_2 TANK AND ENGINE SUPPORT

FIG. V-1. INBOARD PROFILE, PROPULSION AND EQUIPMENT MODULE

FOLDOUT FRAME 2

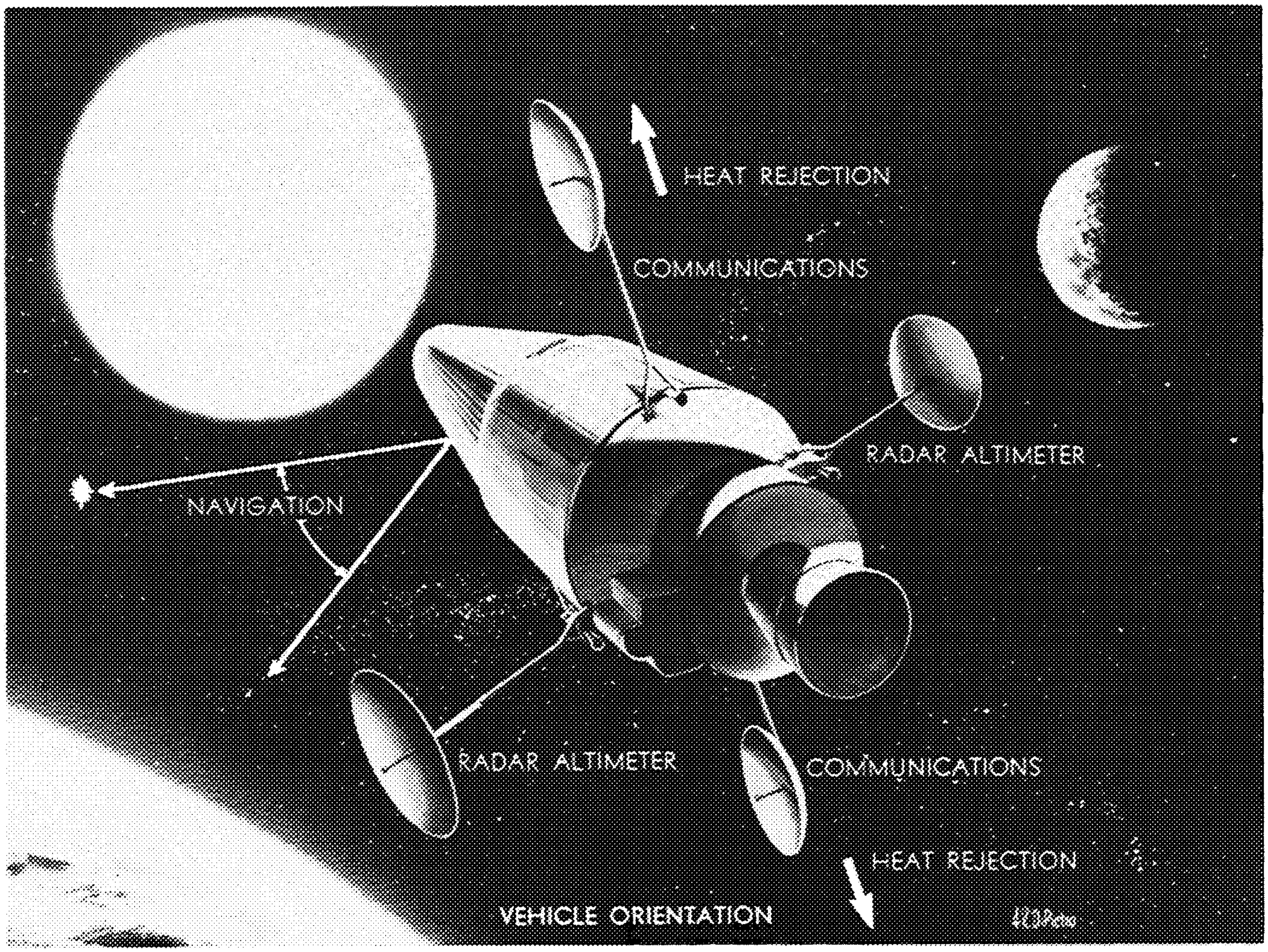


FIG. V–2. VEHICLE ORIENTATION

B. MISSION ENGINE CONSIDERATIONS

The desired number of mission engines to be used for Apollo has been established by analyzing the possible reasons for using two or more engines instead of one.

They are:

(1) If desired thrust level is not available in one engine.

(2) If an improved thrust to weight is available with two engines.

(3) If an improved control arrangement is available with two engines.

(4) If an improved installation is available with two engines.

(5) If improved reliability is available with two engines.

A study of the Apollo propulsion system showed that the first three reasons stated were not applicable. The engine considered for the Apollo mission control propulsion system is an advanced version of the Pratt and Whitney LR–115 engine delivering a nominal thrust of 15,600 lb. Two half-size versions of this engine might tend to show some vehicle structure weight advantage because of the shorter length. However, a 7500-lb thrust version of the LR–115 engine does not currently exist and would require a completely new development in order to realize the smaller size and weight required in order to gain a net system advantage. Furthermore, as the design thrust level of rocket engines is reduced, the thrust to weight ratio of the engine also tends to decrease. This effect would further nullify the weight advantages of two half-size engines as opposed to one full size engine.

For the Apollo vehicle, the remaining question is whether an increase in reliability can be gained by use of multiple engines. This consideration is the most significant factor in selecting either a one- or two-engine configuration.

At first glance it would seem that two engines would offer a higher reliability than one engine provided that any one engine was adequate to do the job. The possible failures which may occur in a rocket engine are as follows:

(1) Thrust chamber failure.
(2) Turbopump failure.
(3) Control system failure.
(4) Valve failure.

Thrust chamber and turbopump failures are very rare in qualified rocket engines. This is particularly true in the case of the turbopump assemblies of a liquid hydrogen engine, since the turbines run at low temperatures. The thrust chambers also run at low temperatures due to the great cooling capacity of liquid hydrogen. However, the thrust chamber could be vulnerable to meteorite strikes which would puncture the coolant flow tubes. In the Model 410 installation, the mission control engine is protected by a combined flame shield and meteorite bumper.

An analysis of the effect of meteorite strikes on the engine by Pratt and Whitney and by the Martin Apollo studies has shown that the hazard is not great. The probability of *no* penetrations has been estimated at 0.969. Calculations by Pratt and Whitney indicate that up to 21 penetrations can be sustained in the cooling tubes without deterioration of the engine performance. It is concluded, therefore, that thrust chamber and turbopump failures are highly improbable for the burning time of the mission control engine (285 sec for lunar takeoff—the maximum case).

The two remaining sources of failure are control system failure and valve failure. Three types of control failure may occur. These are turbine overspeed, faulty start sequence, and unstable operation. Valve failures will be either failure to open or failure to close.

Control failures may be approached in two ways. If two engines are used, the engine with the faulty control system can simply be shut down. The other approach is to design redundancies and fail-safe features into the basic system. The redundant system approach is lighter but must be carefully considered so as to avoid "sneak circuits" or other subtle interactions which would jeopardize system operation. If the system can be highly developed with extensive operating time prior to operational use, the redundant system approach is preferred because of its weight advantage.

The possibility of valve failure points up one of the strongest arguments against a two-engine installation. The reason for this is that with two engines, the number of propellant shutoff valves is doubled (4 instead of 2). If the failure is a fail-closed type of malfunction during engine starting, the engine with the faulty valve will suffer a false start and can simply be shut down, closing the operable propellant valve. The engine with two properly operating valves will have started normally and the required maneuver (ejection from lunar orbit) can be accomplished using the remaining operative engine. All that is required of this system is sufficient extra propellant to account for that loss during the false start and an indication of valve failure. However, if the failure should be a fail-open type of malfunction during engine shutdown, then all remaining propellant in the tank connected to that valve will be lost. If this occurred after injection into lunar orbit, the result would be disastrous. Since the number of valves has been doubled in the two-engine case, the probability of this type of failure is doubled. It can be argued that this type of failure can be overcome with a redundant valve system. It is an equally valid argument, however, that a single engine with a redundant system can show equally good reliability at substantially lower weight.

It is concluded, therefore, that a single engine with carefully designed systems and redundancies in selected areas is the preferred approach to the Apollo mission control propulsion system. It should be noted, however, that this discussion has been limited to a choice between one or two engines with a single propellant tankage system. If there were no weight limitation, two complete and independent propulsion systems (engine plus tankage), would, of course, offer a significant improvement in system reliability. Such a system would impose a weight penalty on the order of 10,000 lb on the Apollo vehicle.

C. TANKAGE STUDIES

Because of the space requirements for the propulsion system liquid hydrogen and oxygen tanks and the need to keep the weight to a minimum, a study was made of various propellant tank designs. Five of the designs that were studied are shown on Figs. V–3–A, B, C, D and E. A weight comparison was made to establish the tank weights for lunar orbit and lunar takeoff requirements.

The study of the weight of the various propellant tank configurations for a Lunar takeoff is shown in Table V–1. It indicates that the configuration in Fig. V–3–A, which is the configuration incorporated in the Model 410, is the lightest. The totals shown on the chart indicate that the configuration arrangement used in Model 410 is at least 58 lb lighter than any other configuration. When the weight of the adapter is included, Fig. V–3–A, is at least 37 lb lighter.

A review of the weight summary indicates the configuration would also be the lightest if lunar orbit (ΔV of 6100 fps) were to be the design criteria.

It should be noted that in all cases, except Fig. V–3D (Torus), the adapter weight could be reduced 144 lb if the 36-in. clearance between the engine nozzle on Model 410 and the propellant dome on Saturn were eliminated.

Configuration, Fig. V–3D, (Torus) was the shortest in overall length, however, a larger part of the skirt remains on Model 410 at separation and the corresponding cost in propellant to carry this skirt results in a heavier total weight than the configuration in Fig. V–3A.

Selected Arrangement. The arrangement of the propulsion and equipment module that was selected to meet the requirements of section A of this chapter is shown in Fig. V–1. The overall diameter of the module has been held to a diameter of 154 in. This diameter was selected to permit the Apollo spacecraft to be attached to the Saturn booster at a splice in the S–IV stage, and also because this diameter makes the L/D of the module about one $\left(\frac{155}{154} = 1.008 \right)$. It was found that a $L/D = 1$ will give a minimum weight of the external shell as follows:

(1) The external shell is used for structure load carrying and for a meteorite bumper.

(2) With the 0.040 in. skin required as a minimum because of the "bumper" requirements, the weight per square foot of the shell is constant regardless of the diameter for practical diameters.

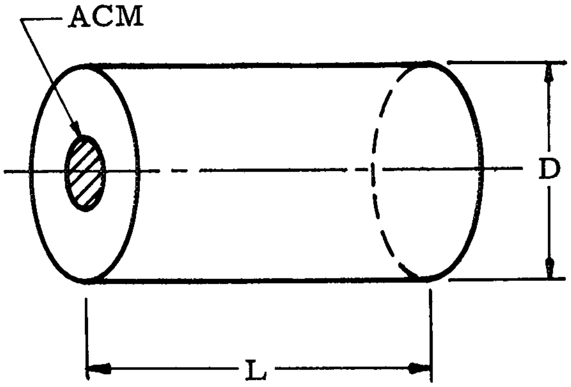
(3) The minimum weight will occur when the total wall area is a minimum (for a constant volume requirement).

TABLE V–1

PROPELLANT TANK CONFIGURATION STUDIES

(Refer to Fig. V–3)

	A	E	B	C	D
APOLLO SKIRT LENGTH (IN.)	(102)	(134)	(171)	(122)	(154)
WEIGHT AT 4 LB/IN.	408	536	684	488	616
ENGINE MOUNT	158	60	50	158	50
VERNIER ENGINE SUPPORTS	10	—	—	10	—
METEORITE SHIELD—ENGINE AND PIPING	131	131	50	25	10
AFT BULKHEAD AND FLAME SHIELD	—	127	127	—	127
OXYGEN TANK INCLUDING INSULATION	146	146	196	192	192
HYDROGEN TANK INCLUDING INSULATION	416	416	309	291	345
TRAPPED OXYGEN	91	91	182	273	273
TRAPPED HYDROGEN	53	53	53	35	53
HYDROGEN BOILOFF	77	77	54	49	66
PROV FOR EXPANDABLE ANTENNAS	—	—	+40	—	+40
TOTAL FOR MODEL 410—(DRY WT, LB)	1490	1637	1745	1521	1772
PROP. REQUIRES FOR ΔV 8600 FT/SEC (LUNAR TAKEOFF)	—	+131	+227	+27	+251
TOTAL MODEL 410 (LB)	1490	1768	1972	1548	2023
ADAPTER	480	288	184	460	84
TOTAL WEIGHT ON SATURN (LB)	1970	2056	2156	2008	2107
ON BASIS OF LUNAR ORBIT					
TOTAL MODEL 410 (LB)	1490	1719	1887	1538	1930
TOTAL WEIGHT ON SATURN (LB)	1970	2007	2071	1998	2014



(4) Wall area of the shell is:

$$A = \frac{\pi D^2}{4} + \left(\frac{\pi D^2}{4} - A_{cm}\right) + \pi DL$$

approximately.
Area A_{cm} is the area of the back face of the command module. Minimization of the area requires that L=D.

(5) The actual L/D is $\frac{155}{154} = 1.008$ for the propulsion equipment module. Therefore, the configuration of the shell is nearly the optimum.

D. ARRANGEMENT FEATURES

The PEM was arranged to provide:

(1) Incorporation of the engine-thrust support, the rear meteoritic bumper and the propulsion tankage support in a single unit for weight minimization.

(2) Thermal and physical isolation of the propellant tanks to avoid boiloff losses and to increase safety.

(3) Arrangement to enable installation and checkout of the mission engine and tankage prior to assembly to the rest of the module. A splice is provided for this purpose.

(4) Avoidance of common domes in propulsion tankage design.

(5) Simplicity of maintaining the N_2O_4 N_2H_4—UDMH vernier and attitude-control pro-

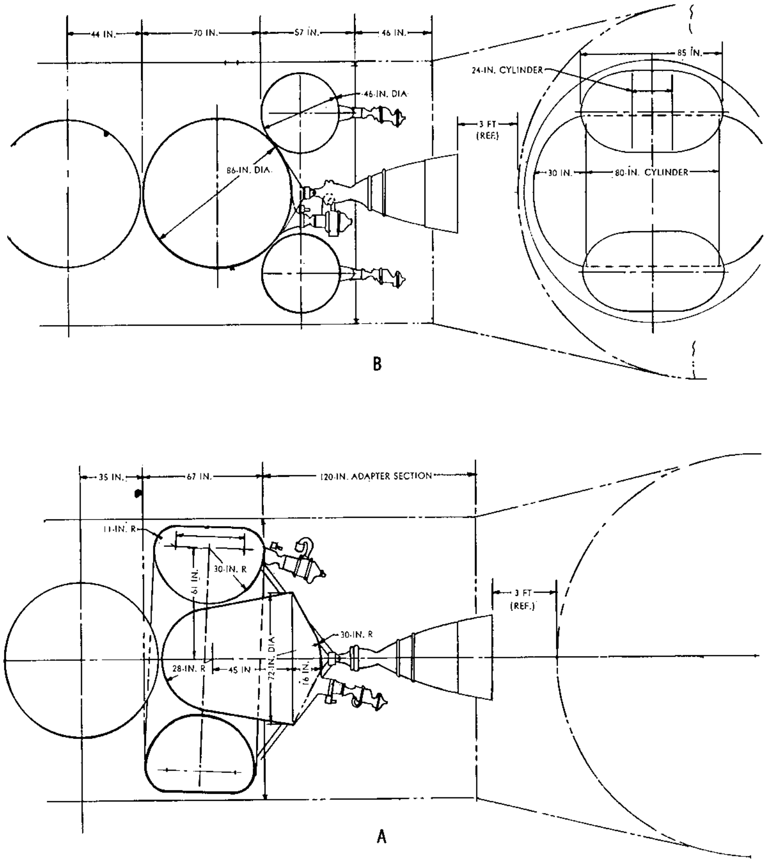

44 IN.
70 IN.
57 IN.
46 IN.
85 IN.
24-IN. CYLINDER
46-IN. DIA
3 FT (REF.)
86-IN. DIA.
30 IN.
80-IN. CYLINDER
B
35 IN.
67 IN.
120-IN. ADAPTER SECTION
11-IN. R
30-IN. R
61 IN.
3 FT (REF.)
30-IN. R
28-IN. R
45 IN
72-IN. DIA
16 IN.
A

FIG. V–3. PROPULSION TANK CONFIGURATIONS

pellants within the required temperatures limits. These tanks are attached to a container inside the external insulation of the mission module, which is environmentally controlled.

(6) Minimization of unusable fuel weight and of tankage surface area. Spheres are used for all except for the mission tankage. Considerations of overall spacecraft length and effective weight led to the configuration of the centrally located oxygen tank with consequent minimum ullage weight (oxygen being the heavier fuel) and the toroidal hydrogen tank, which is tipped three degrees for ullage minimization. The flat sides of the torus give a maximum volume-to-length ratio without requiring gages above the established minimum.

The torus-shaped liquid hydrogen tank allows the cone-shaped oxygen tank to be placed in the center of it. The mission module is a cylinder tank of 84 in. diameter with hemispheric ends and is arranged close to the command module. Sufficient space is available between the outside diameter of the propulsion and equipment module and the mission module to allow location of the vernier propellant tanks, the electrical power system hydrogen and oxygen tanks, and the fuel

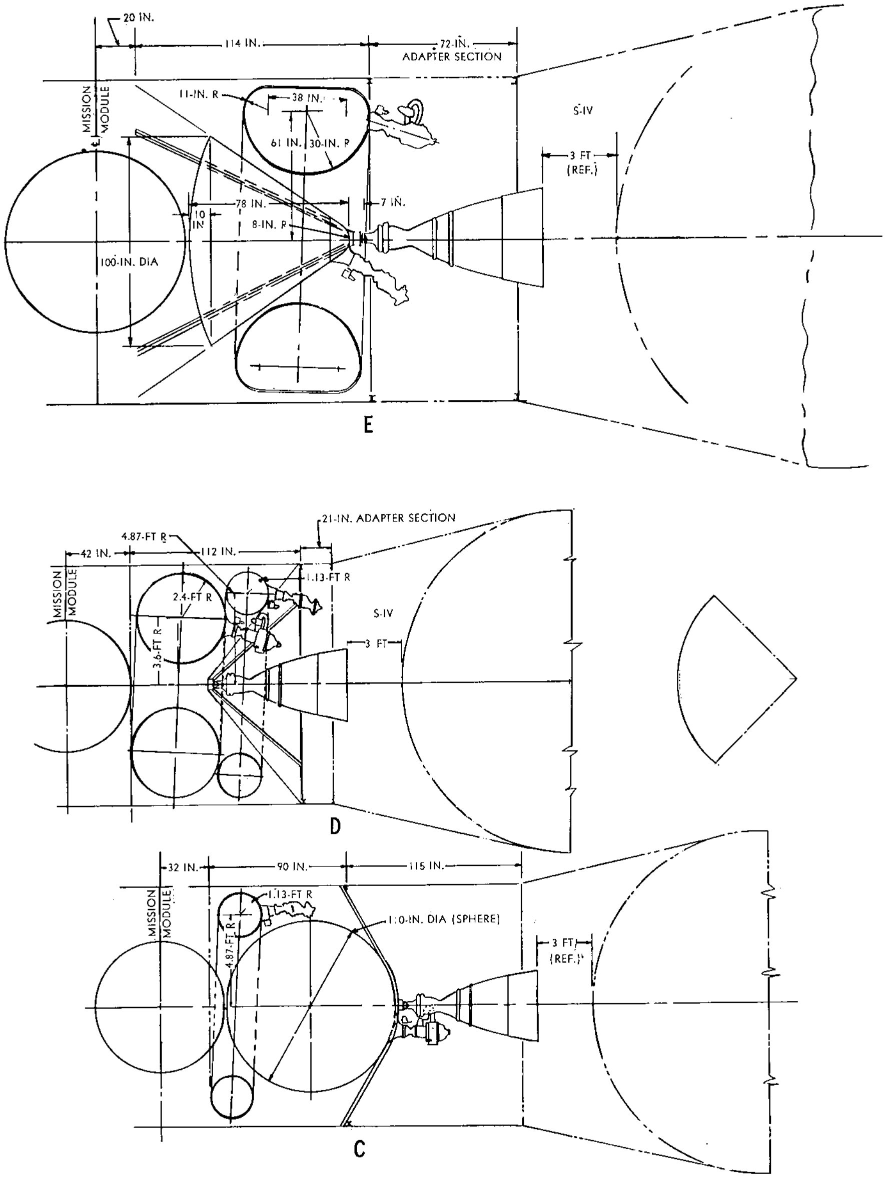

20 IN.
114 IN.
72-IN.
ADAPTER SECTION
MISSION MODULE
℄
11-IN. R
38 IN.
61 IN.
30-IN. R
S-IV
3 FT (REF.)
78 IN.
7 IN.
10 IN
8-IN. R
100-IN. DIA
E
21-IN. ADAPTER SECTION
4.87-FT R
42 IN.
112 IN.
1.13-FT R
2.4-FT R
3.6-FT R
S-IV
3 FT
D
32 IN.
90 IN.
115 IN.
1.13-FT R
4.87-FT R
110-IN. DIA (SPHERE)
3 FT (REF.)
C

FIG. V–3. PROPULSION TANK CONFIGURATION—CONTINUED

cells. The mission engine is arranged in the center of the module and is mounted on four struts which pickup stringers on the module's outside diameter. The retractable antennas are mounted at the rear end of the propulsion and equipment module and are folded down around the mission engine components. The 0.040-in. thick aluminum alloy skin of the module serves a dual purpose of a structural load carrying cover and a meteorite "bumper" shield.

The arrangement of the propulsion and equipment module as shown in Fig. V–1 affords a compact and serviceable design with a minimum weight and which meets all its requirements.

VI. ARTIFICIAL "G" STUDIES

During the course of the design studies, an investigation was made of ways to obtain artificial g in the Apollo spacecraft as limited by the crew's tolerance.

Limited investigations in the Navy Slow Rotation Room at Pensacola have indicated certain limitations to man's tolerance to pure rotation:

(1) With a random population, rotation at 1 to 2 rpm, little if any motion sickness occurs.

(2) At 5 rpm, most subjects show motion sickness symptoms but can adapt to some degree.

(3) At 10 rpm, serious problems of motion sickness arise and few can adapt.

Several approaches to gravity simulation within the above limits have been instigated briefly (see Fig. VI–1) encompassing a range from low (2) rpm and 1.0 g simulation to high (5) rpm and minimum (0.1) g simulation.

It would appear so far from this preliminary investigation that, if the motivation for simulating gravity is sufficient, a method similar to those shown may feasibly be integrated into a spacecraft configuration. However, overall weight will increase and reliability and safety will be degraded.

A weight study of the total weights for providing artificial g for circumlunar vehicles similar to those shown in Fig. VI–1 is given in Table VI–1. The weight of the model L–2–C version of the command module has been used as the base line for these comparisons.

The method using the last stage of the booster as a counter balance mass seems the most promising at present. It requires the least structural modification from the presently conceived spacecraft configuration and does not have the serious safety problem associated with a solution utilizing separation of modules from the spacecraft, or separation of the men from each other.

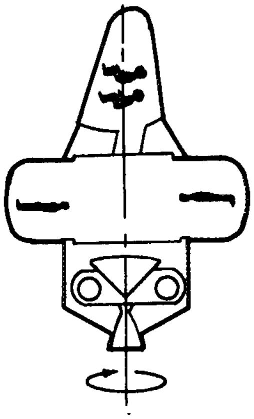
To achieve gravity simulation with this method, the two bodies are separated to the required distance, being connected by four cables. The manned vehicle would be oriented and reaction motors on the manned vehicle and the booster would spin up the combination to the required

TABLE VI–1

ARTIFICIAL G VERSIONS—WEIGHT OF VEHICLES FOR CIRCUMLUNAR MISSION

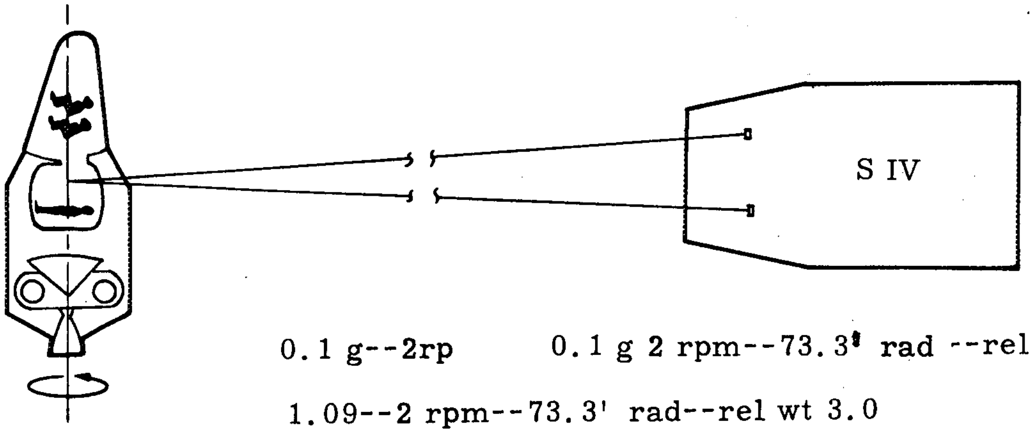
	BASIC	"BOLO"	EXPANDED MODULE		BOOSTER COUNTER BALANCE
	L2C (LB)	0.1 G (LB)	VERSION NO. 2A 0.1 G (LB)	VERSION NO. 2B 0.1 G (LB)	0.1 G (LB)
COMMAND MODULE	(5,866)	(5,866)	(5,866)	(5,866)	(5,866)
MISSION MODULE					
STRUCTURE	1,106	2,101	1,676	1,606	1,281
REACTION CONTROL	805	2,570	940	936	5,275
AUXILIARY POWER SYSTEM	514	580	514	514	1,000
COMMUNICATION, TRACKING, AND INSTRUMENTATION	100	115	100	100	100
INSTRUMENTS		15			
SCIENTIFIC PAYLOAD	205	205	205	205	205
FURNISHING AND EQUIPMENT	218	332	261	261	218
ENVIRONMENTAL CONTROL	798	1,063	948	948	798
SEPARATION SYSTEM	10	10	10	10	10
TOTAL MISSION MODULE	(3,756)	(6,991)	(4,654)	(4,580)	(8,887)
LAUNCH ESCAPE SYSTEM	433	433	433	433	433
PROPULSION SYSTEM	1,224	1,314	1,254	1,251	1,554
PROPELLANT					
LAUNCH ESCAPE	664	664	664	664	664
TURNAROUND	2.530	3,200	2,740	2,720	5,270
TOTAL CIRCUMLUNAR	(14,473)	(18,468)	(15,611)	(15,514)	(22,674)

Expanded module



0.05 - 0.1 g - 5 rpm - 12 ft rad rel wt 1.1

Booster--
counterbalance

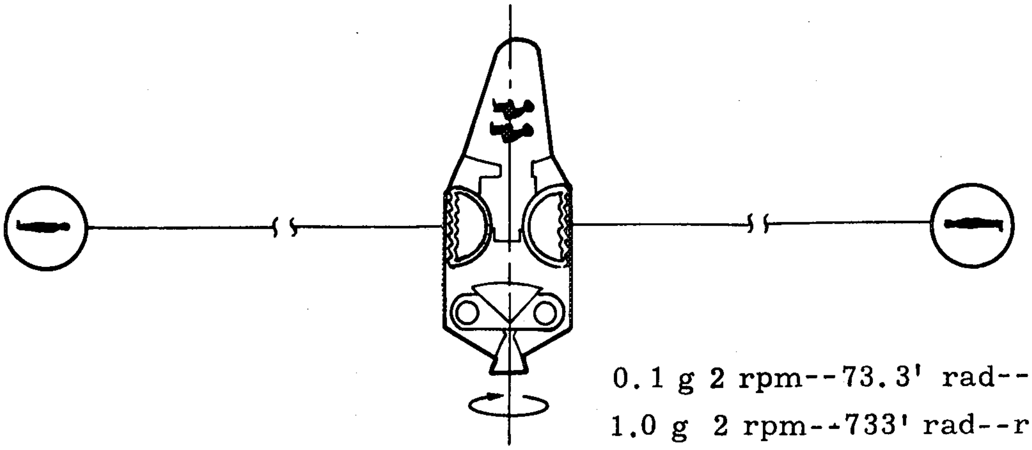


0.1 g--2rp

0.1 g 2 rpm--73.3' rad --rel wt 1.5

1.09--2 rpm--73.3' rad--rel wt 3.0

BOLO



0.1 g 2 rpm--73.3' rad--rel wt 1.15

1.0 g 2 rpm--733' rad--rel wt 1.25

FIG. VI-1. GRAVITY SIMULATION

velocity. For course corrections and lunar injection and ejection, the combination would have to be stopped from rotation and the manned vehicle reoriented so that the booster body is in line with the main engine-thrust.

Simulating 1.0 g would be prohibitively heavy; at least 300 to 400% increase in spacecraft weight results from the large spin up and main engine fuel requirements. Simulating 0.1 g incurs a lesser but still substantial weight penalty—at least 50% increase in vehicle weight—but seems technically feasible.

With payload efficiency as a key to the feasibility of the whole Apollo program, gavity simulation should not be incorporated unless conclusive evidence shows that man requires it.

VII. ALTERNATE CONFIGURATIONS

Figures III–1 through VII–16 show most of the configuration concepts which have been considered during the Apollo studies. These studies have shown the major advantages and disadvantages of the overall spacecraft concepts. The most desirable features of each have been considered and, wherever possible, incorporated in the M–410 configuration.

Integrated and modular type vehicles are shown in the figures. Re-entry vehicle studies are evident in the drawings and include controlled ballistic types and lifting body types. Both symmetrical and unsymmetrical type lifting bodies have been used, and, for the unsymmetrical lifting bodies those featuring flat tops and those with flat bottoms have been examined.

Various geometric shapes have been used for mission modules, as well as for fuel and oxidizer tanks. The affects of size changes have been considered and are evident in the drawings. Liquid and solid propellants have been studied for the launch escape propulsion system, and several schemes and arrangements for this system are indicated. Several different concepts are evident in the air-lock arrangements shown. Landing systems considered include landing on air bags, on crushable structure, on a retrorocket system, and consideration has been given to various means of shock attenuation by the seat/couch. The relative merits of various crew arrangements and of various body positions are made obvious by these drawings.

RECOVERY SYSTEM

1. MAIN CHUTES
2. DROGUE CHUTES
3. RISERS (DROGUE CHUTE)
5. SEQUENCING EQUIPMENT
6. CONTROL AND DISPLAYS

FLAP CONTROL SYSTEM

10. ACTUATOR PACKAGES
15. FLIGHT CONTROL UNITS

REACTION CONTROL

20. ATTITUDE NOZZLES
21. N_2O_4 TANK
22. UDMH/N_2H_2 TANK
23. H_e BOTTLE
24. CONTROLS AND DISPLAYS
25. ATTITUDE NOZZLES

ELECTRICAL POWER SYSTEM

30. APU
31. APU FUEL TANK
32. BATTERIES
34. VOLTAGE BOOSTER
35. DISTRIBUTION PANEL
39. BUS CENTER
40. H_2 TANK
41. O_2 TANK
42. SPACE RADIATOR
43. CONTROL PANEL

GUIDANCE

70. ASTRO-INERTIAL PLATFORM W/TRACKER AND ELECTRONICS
71. MINIATURE PLATFORM
72. MINIATURE PLATFORM ELECT.
73. AUTO MANUAL TRACKER
74. AUTO MANUAL TRACKER ELECT.
75. DIGITAL COMPUTERS
76. ALTIMETER
77. ALTIMETER ANTENNAS

COMMUNICATION AND TELEMETERING

80. RECOVERY ANTENNAS HF AND UHF
81. C-BAND ANTENNAS
82. K_e-BAND ANTENNAS
83. VHF MONOPOLE ANTENNAS
84. HF ANTENNAS
85. K_e TRANSMITTER
86. C-BAND BEACON
87. HF TRANSMITTER
88. UHF TRANSMITTER
89. UHF RECEIVER
90. VHF RECEIVERS
91. VHF TRANSMITTERS
92. ANTENNA MULTIPLEXER
93. SOFAR BOMBS
94. DYE MARKER
95. RADAR CHAFF
96. CONTROL PANEL, COMM.
97. FLASHING LIGHT
98. INTERPHONE CONTROL
99. S-BAND PARABOLIC ANTENNAS
100. VHF SHROUD ANTENNAS
101. S-BAND ANTENNAS
102. C-BAND ANTENNAS
104. S TRANSMITTERS
105. S RECEIVERS
106. S TRACKING BEACON
107. C BEACON

INSTRUMENTATION AND SCIENTIFIC EQUIPMENT

110. 35 MM CAMERA
111. TAPE RECORDER
112. RE-ENTRY INSTRUMENT PACKAGE
113. TEMPERATURE MEASUREMENT SYSTEM
114. PCM TELEMETRY SYSTEM
115. POWER SUPPLY
118. RBE MEASUREMENT
119. SCIENTIFIC EQUIPMENT CONTROL PANEL
120. SOLAR FLARE DISPLAY
121. RECORDING EQUIPMENT CONTROL PANEL
122. DATA LINK SYSTEM
123. SIGNAL CONDITIONING PACKAGE
124. RADIATION DETECTORS
125. MICROMETEORITE DETECTORS
126. SOLAR FLARE SENSORS

FURNISHING AND EQUIPMENT

130. SEATS
131. SURVIVAL KIT
132. FOOD STORAGE
133. WASTE COLLECTION
134. PANELS AND CONSOLES
135. WATER TANK (PORTABLE)
136. FOOD STORAGE
137. WASTE STORAGE
138. WASTE STORAGE

ENVIRONMENTAL CONTROLS

140. O_2 TANK
141. LOX TANK
142. N_2 TANK
143. LN TANK
144. FILTER ASSEMBLY
145. H_2O SEPARATOR
146. HEAT EXCHANGER
147. FAN
148. H_e TANK
149. H_2O TANK COOLING
150. PUMP AND MOTOR ASSEMBLY
151. CABIN PRESSURE REGULATOR
152. SNORKEL
153. MOLECULAR SIEVE
154. CONTROLS AND DISPLAYS
155. SUIT PRESSURE DISPLAYS
156. ALARM DISPLAYS
157. ACCUMULATOR
158. LOX TANK
159. LN TANK
160. SPACE RADIATOR

LAUNCH ESCAPE SYSTEM

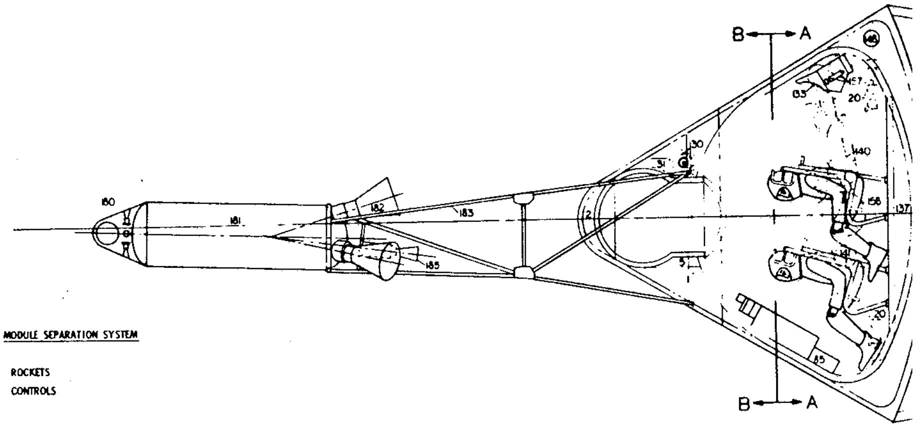
180. ABORT CONTROL
181. THRUST CHAMBER
182. NOZZLES
183. MOUNTING PROVISIONS
184. CONTROL AND DISPLAYS
185. SEPARATION ROCKET

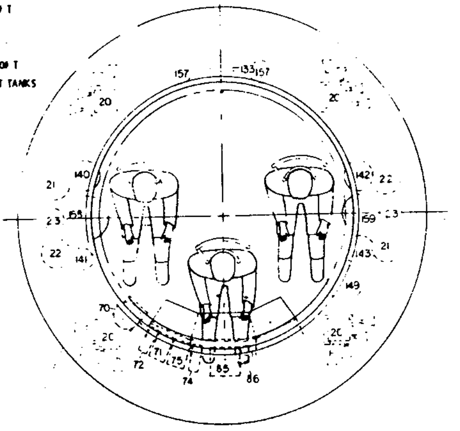
MODULE SEPARATION SYSTEM

190. ROCKETS
191. CONTROLS

PROPULSION

200. ENGINE MAIN 15,000# T
201. FUEL TANK
202. OXIDIZER TANK
203. VERNIER ENGINES 600# T
204. VERNIER PROPELLANT TANKS
205. H_e TANK





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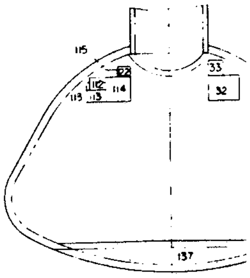

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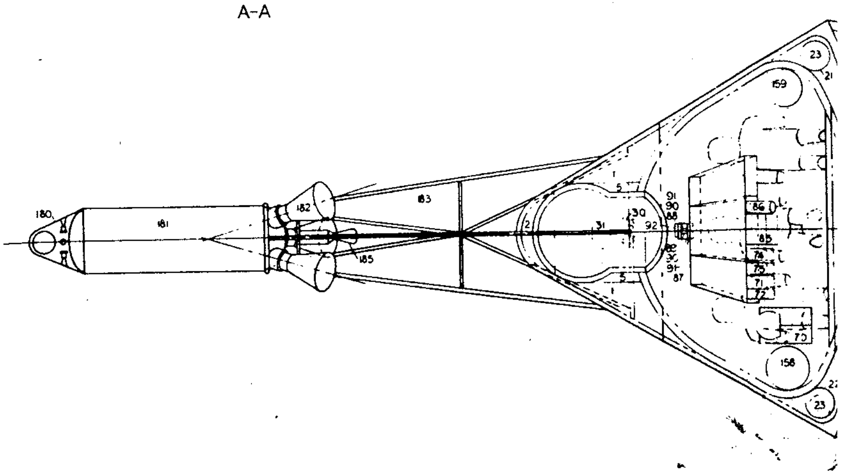

FOLDOUT FRAME /

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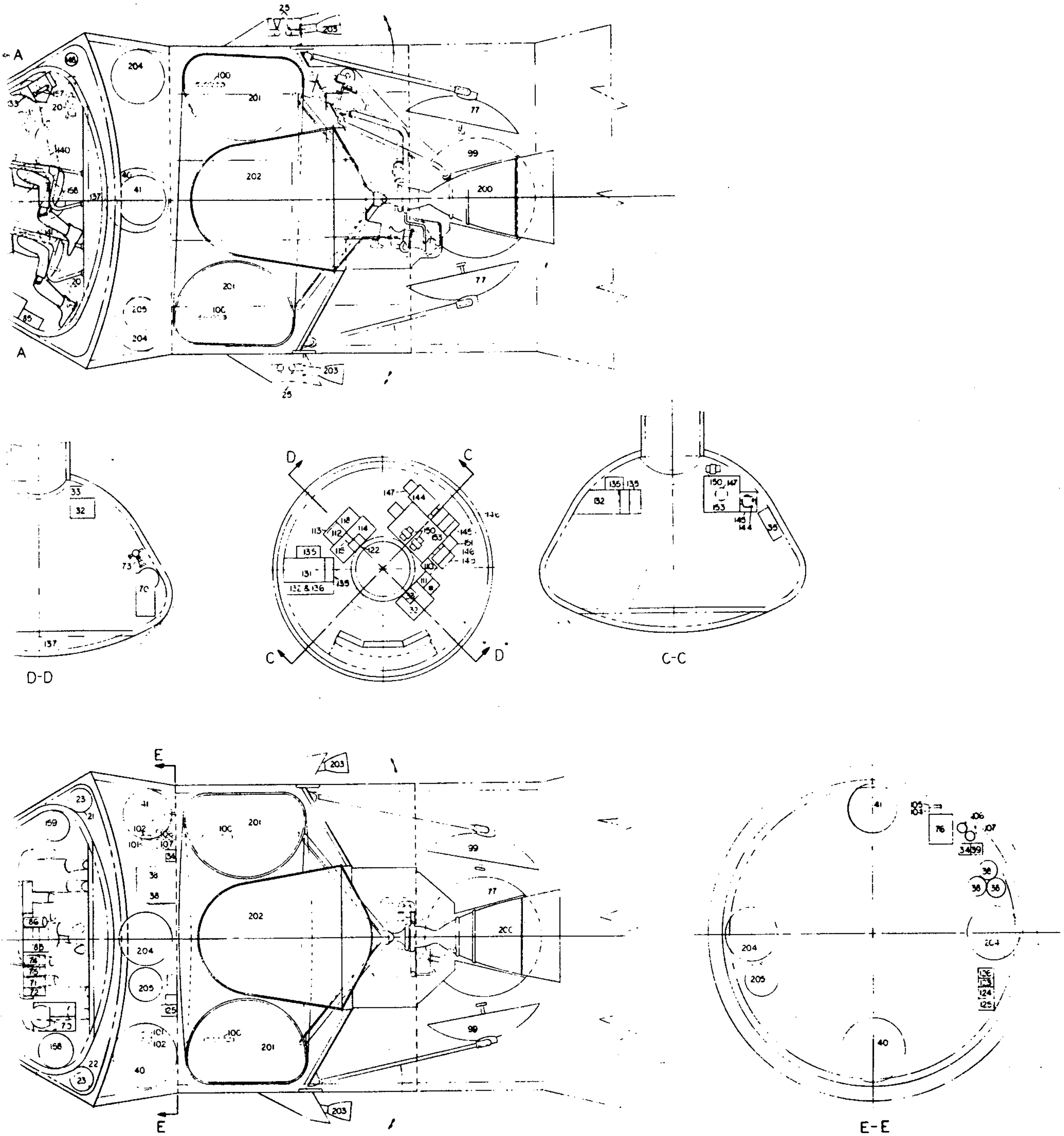
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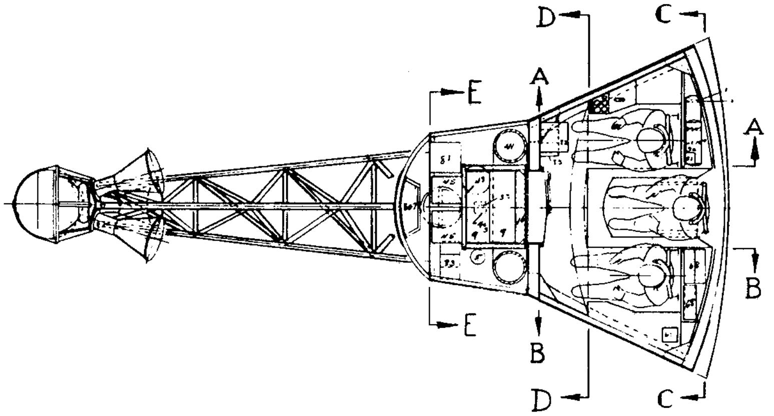
FOLDOUT FRAME 2

FIG. VII-1. MODIFIED MERCURY



RECOVERY SYSTEM

1. H.F TRANSCEIVER
H.F & U.H.F RESCUE LOCATOR BEACONS
2. U.H.F TRANSCEIVER
3 FLASHING LIGHT
4 BATTERY PACKAGE
5. SOFAR BOMBS
6 DYE MARKER
7 CHAFF
8 DROGUE PARACHUTE(S)
9 MAIN PARACHUTE(S)
10 SEA ANCHOR

COMMAND RECEIVER SYSTEM

11 U.H.F COMMAND RECEIVER
12. L.F COMMAND RECEIVER
X-BAND TRANSP- DECODER
OTHER DECODERS

VOICE COMMUNICATION SYSTEM

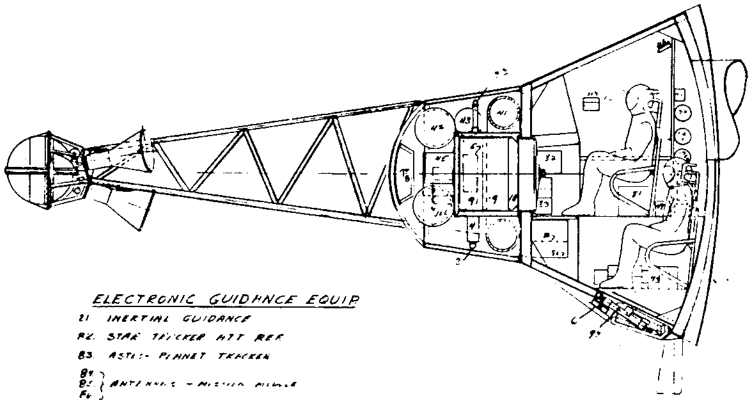
21 H.F RADIO } MISSION MOD.
22. U.H.F RADIO

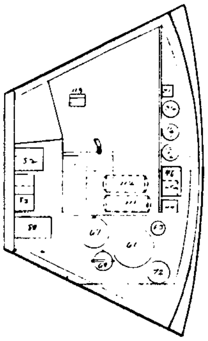
TRACKING S

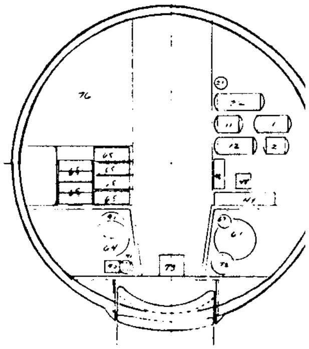
31. MINITRACK BEACON
32. C-BAND & X-BAND TRA
33. FLASHING LIGHT

ELECTRICAL

41. FUEL FOR A.P.U (H_2)
42 O_2 FOR A.P.U.
43. LUBE OIL FOR APU.
44 SOLAR ARRAY - NOT
45. A.P.U.
46. BUS CENTER
47. INVERTER
48. VOLTAGE BOOSTER
49 BATTERY
50 BATTERY - MISSION M




A-A


C-C

ELECTRONIC GUIDANCE EQUIP

81 INERTIAL GUIDANCE
82. STAR TRACKER ATT REF
83. ASTRO- PLANET TRACKER
84 } ANTENNAS - MISSION MODULE
85
86

HYDRAULIC SYSTEM

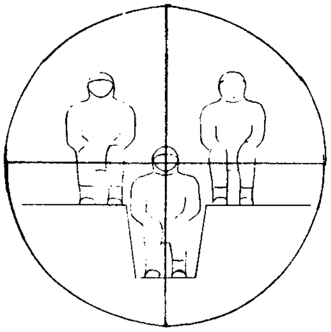
90 PUMP
91 RESERVOIR
92 ACCUMULATOR
93 ACTUATOR

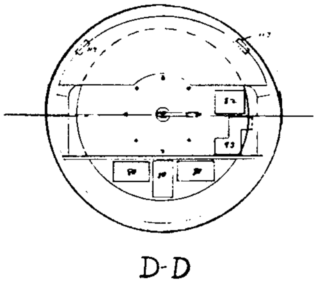
REACTION CONTROL SYSTEM

100 PROPELLANT BTL
101 PITCH - ROLL - YAW NOZZLES - NOT SHOWN
102. ULLAGE VERNIER NOZZLES UNDEFINED

INST SYSTEM

110. CAMERA(S)
111. TELEMETRY ~ VEH. & SCIENTIFIC DATA.
112. X-BAND TRANSMITTER
113 TAPE RECORDER(S)
114 MISC. (17 SMALL PKGS - NOT SHOWN)

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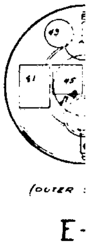
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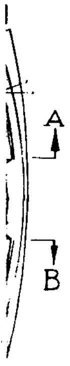


RECOVERY SYSTEM

1. H.F TRANSCEIVER
   H.F & U.H.F RESCUE LOCATOR BEACONS
2. U.H.F TRANSCEIVER
3. FLASHING LIGHT
4. BATTERY PACKAGE
5. SOFAR BOMBS
6. DYE MARKER
7. CHAFF
8. DROGUE PARACHUTE(S)
9. MAIN PARACHUTE(S)
10. SEA ANCHOR

COMMAND RECEIVER SYSTEM

11. U.H.F COMMAND RECEIVER
12. L.F COMMAND RECEIVER
    X-BAND TRANSP-DECODER
    OTHER DECODERS

VOICE COMMUNICATION SYSTEM

21. H.F RADIO } MISSION MOD
22. U.H.F RADIO }

TRACKING SYSTEM

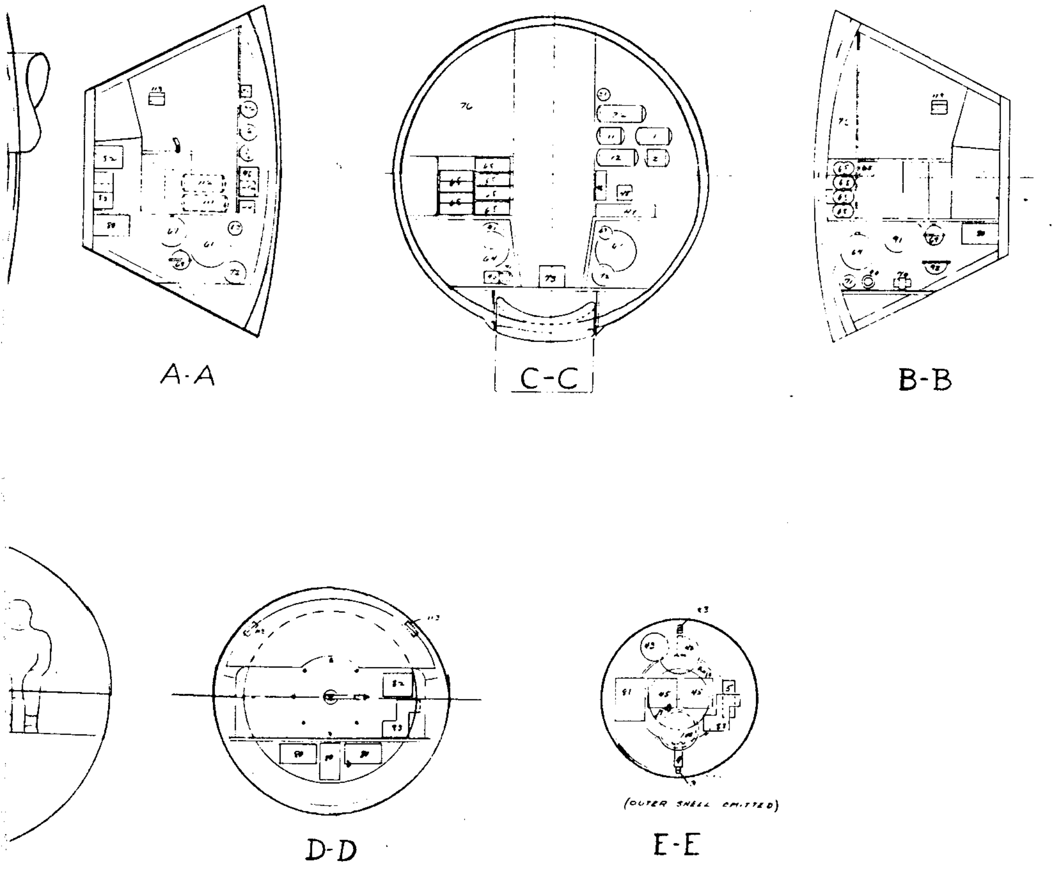
31. MINITRACK BEACON
32. C-BAND & X-BAND TRANSPONDERS.
33. FLASHING LIGHT.

ELECTRICAL POWER

41. FUEL FOR A.P.U (H_2)
42. O_2 FOR A.P.U.
43. LUBE OIL FOR A.P.U.
44. SOLAR ARRAY - NOT SHOWN
45. A.P.U.
46. BUS CENTER
47. INVERTER
48. VOLTAGE BOOSTER
49. BATTERY
50. BATTERY - MISSION MODULE

ENVIRONMENTAL CONTROL SYSTEM

61. O_2 CONTAINER
62. O_2 CONTAINER
63. O_2 PURGE GAS
64. NITROGEN BOTTLE
65. CABIN REG. VALVE
66. WATER RECOVERY UNIT - MISSION MOD.
67. H_2O STOWAGE.
68. LITH. HYDROX. FILTER(S)
69. ACCUMULATOR(S)
70. DUAL PUMP PKG.
71. N_2 BTL.
72. H_2O BTL.
73. FAN
    FILTER
    EXCHANGER
    EVAP. EXCH'R
    H_2O SEPERATOR



A-A

C-C

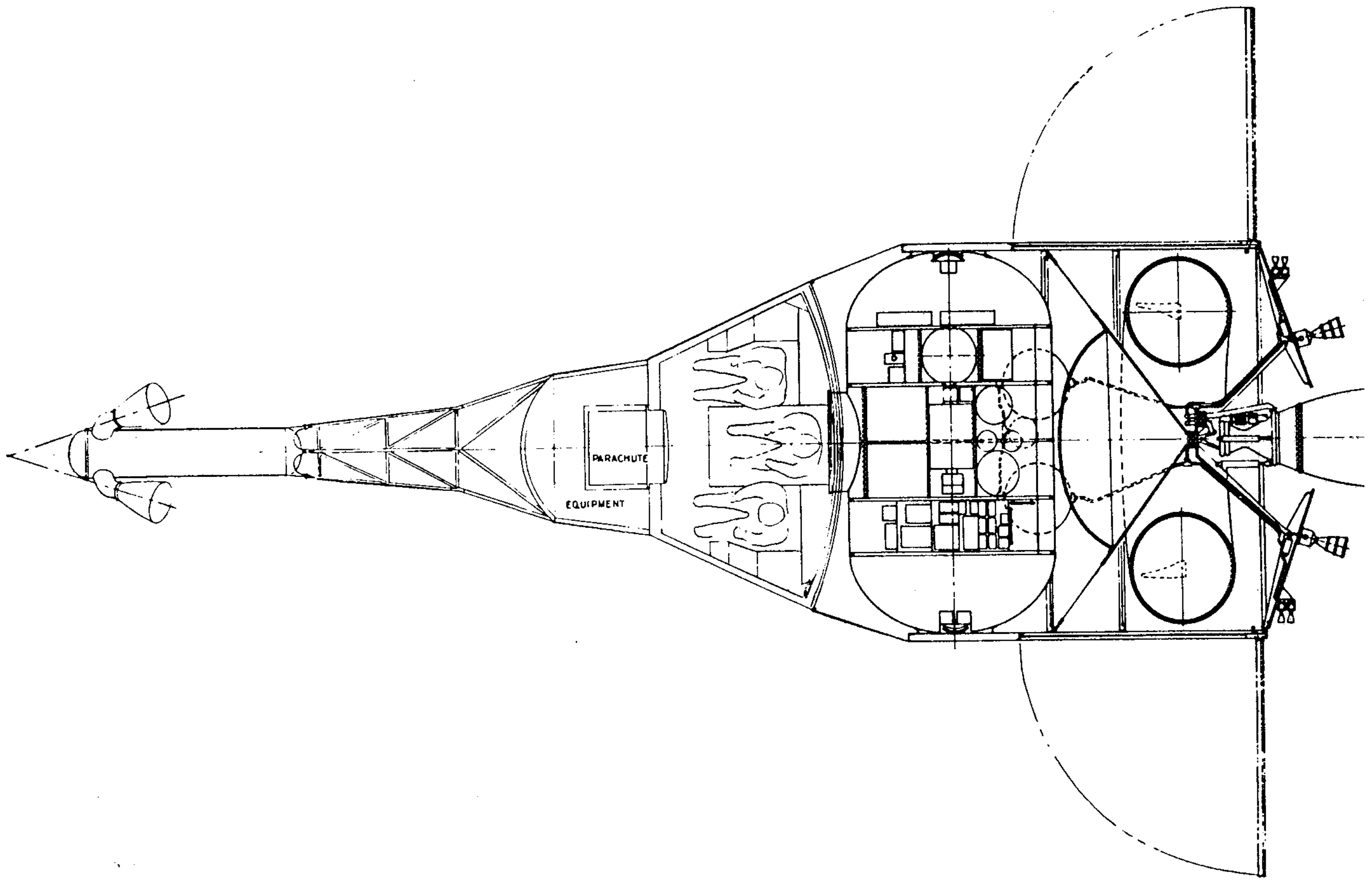
B-B

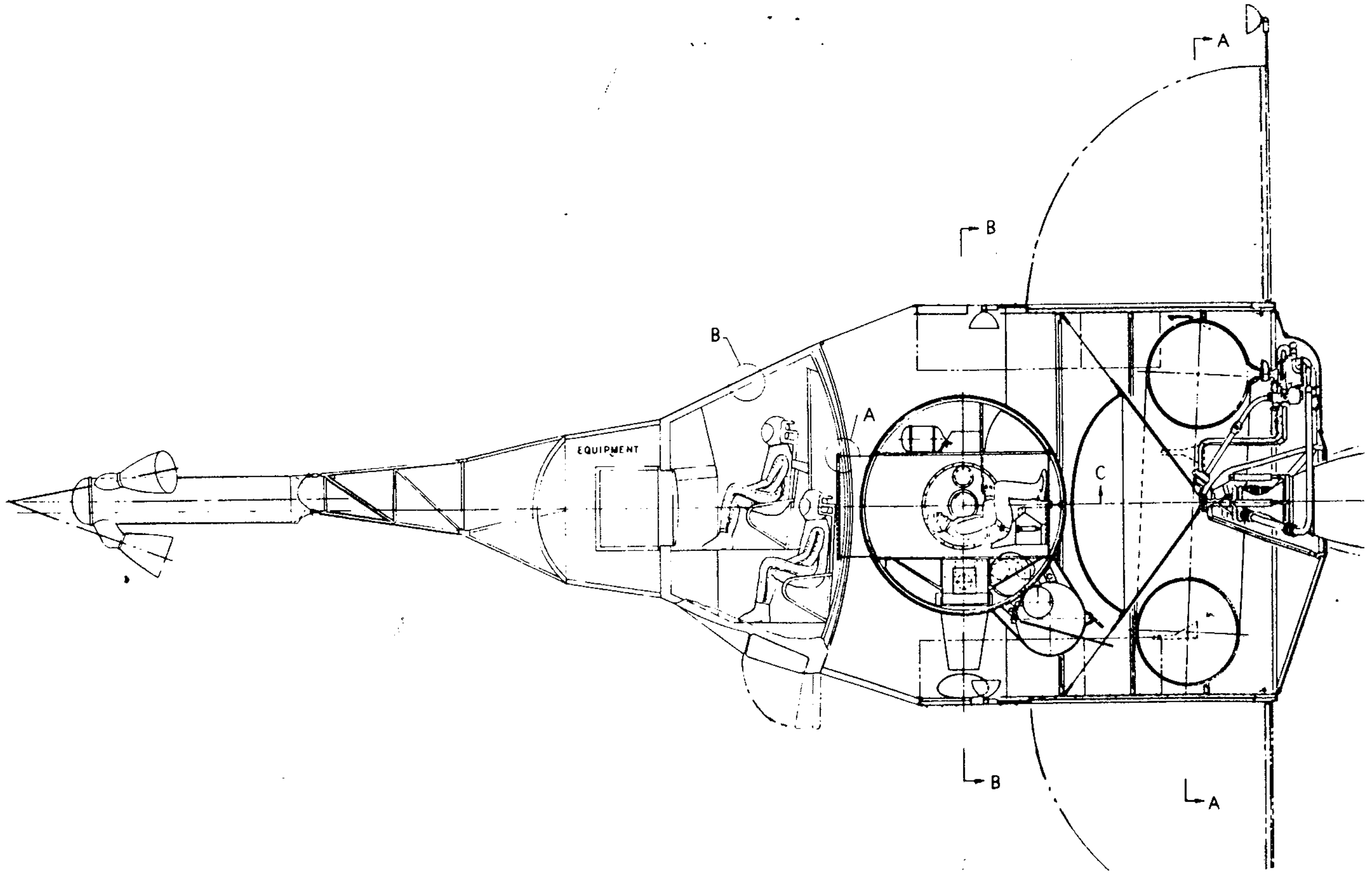
D-D

E-E
(OUTER SHELL OMITTED)

FOLDOUT FRAME 2

FIG. VII-2. FLAPPED MERCURY, SHEET 1


PARACHUTE
EQUIPMENT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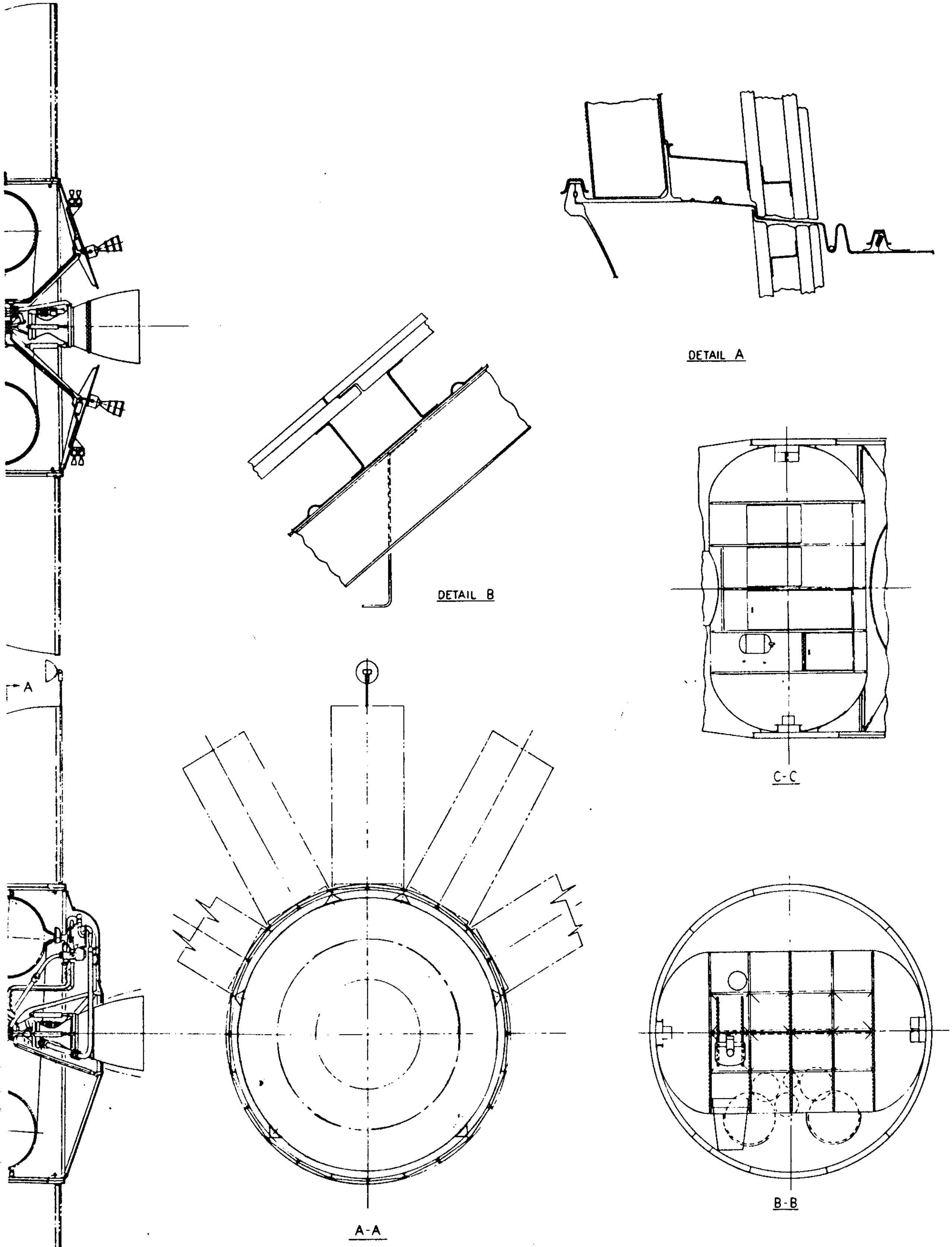
DETAIL A
DETAIL B
C-C
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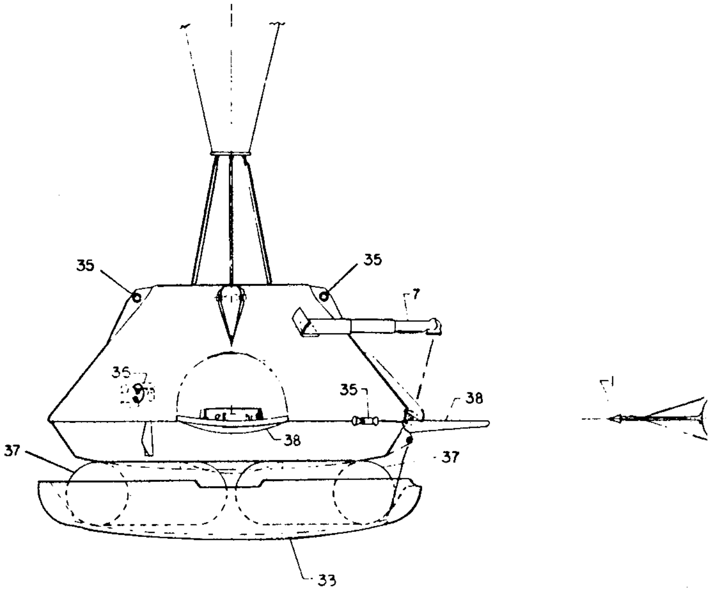
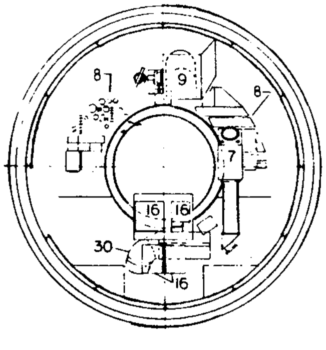
FOLDOUT FRAME 2

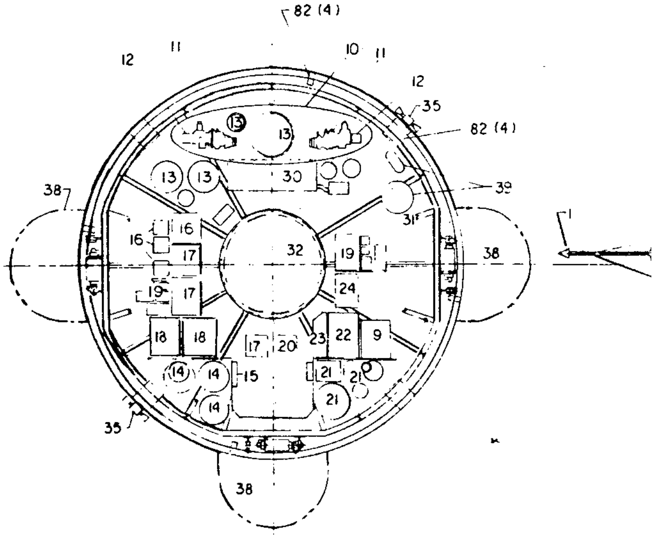
FIG. VII–3. FLAPPED MERCURY, SHEET 2

1. PITOT HEAD
2. ESCAPE ROCKET
3. ESCAPE SYSTEM SEPARATION ROCKET
4. ESCAPE ROCKET SUPPORT TRUSS
5. PARACHUTE PACKAGE
6. PARACHUTE SEQUENCING EQUIPMENT
7. PERISCOPE AND DRIFT SIGHT
8. INSTRUMENT AND CONTROL PANEL
9. GUIDANCE EQUIPMENT
10. SAFETY COMPARTMENT
11. AUXILIARY POWER UNIT
12. ELECTRICAL GENERATOR
13. FUEL AND OXIDIZER TANKS FOR APU AND CONTROL SYSTEMS
14. O_2 AND N_2 SUPPLY TANKS
15. FILM STOWAGE
16. ELECTRICAL SYSTEM COMPONENTS
17. COMMUNICATION SYSTEM COMPONENTS
18. GUIDANCE AND NAVIGATION EQUIPMENT
19. TELEMETRY EQUIPMENT
20. RBE MEASURING EQUIPMENT
21. ENVIRONMENTAL CONTROL SYSTEM COMPONENTS (CABIN AND EQUIPMENT)
22. SURVIVAL KIT -3 MAN
23. WASTE DISPOSAL PACKAGE
24. DRINKING WATER SUPPLY
30. ENVIRONMENTAL CONTROL SYSTEM COMPONENTS (STRUCTURAL)
31. PRESSURE BULKHEAD
32. HATCH
33. HEAT SHIELD
34. WINDOW
35. REACTION CONTROL NOZZLES AND SERVOVALVES
36. SNORKEL
37. LANDING BAG
38. AERODYNAMIC CONTROL SURFACE AND ACTUATOR
39. GAS GENERATING EQUIPMENT
40. SEAT
41. HEAT SHIELD RELEASE MECHANISM
42. MODULE SEPARATION DEVICE
43. MODULE SEPARATION ROCKETS
50. MISSION MODULE
51. EQUIPMENT SUPPORTING STRUCTURE
52. CAMERA
53. SCIENTIFIC DATA GATHERING EQUIPMENT
54. FOOD AND WATER STOWAGE
55. WASTE COLLECTION
56. WATER RECOVERY UNIT
57. MAIN ENTRANCE HATCH
58. VIEWING PORT
60. H PRESSURE BOTTLE
61. FUEL AND OXIDIZER TANKS--REACTION CONTROL AND VERNIER ROCKET SYSTEM
62. MAIN LOX TANK
63. MAIN LH TANK
64. TANK SUPPORT AND INSULATING STRUCTURE
65. MAIN ENGINE (XLR-119 MODIFIED)
66. GIMBAL ACTUATORS
67. PUMP
68. VERNIER ROCKET NOZZLE
69. FLAME SHIELD
80. SOLAR CELL ARRAY
81. SPACE RADIATOR
82. ANTENNA
83. HINGED RADIATION SHIELD





E-E



D-D

FOLDOUT FRAME 1

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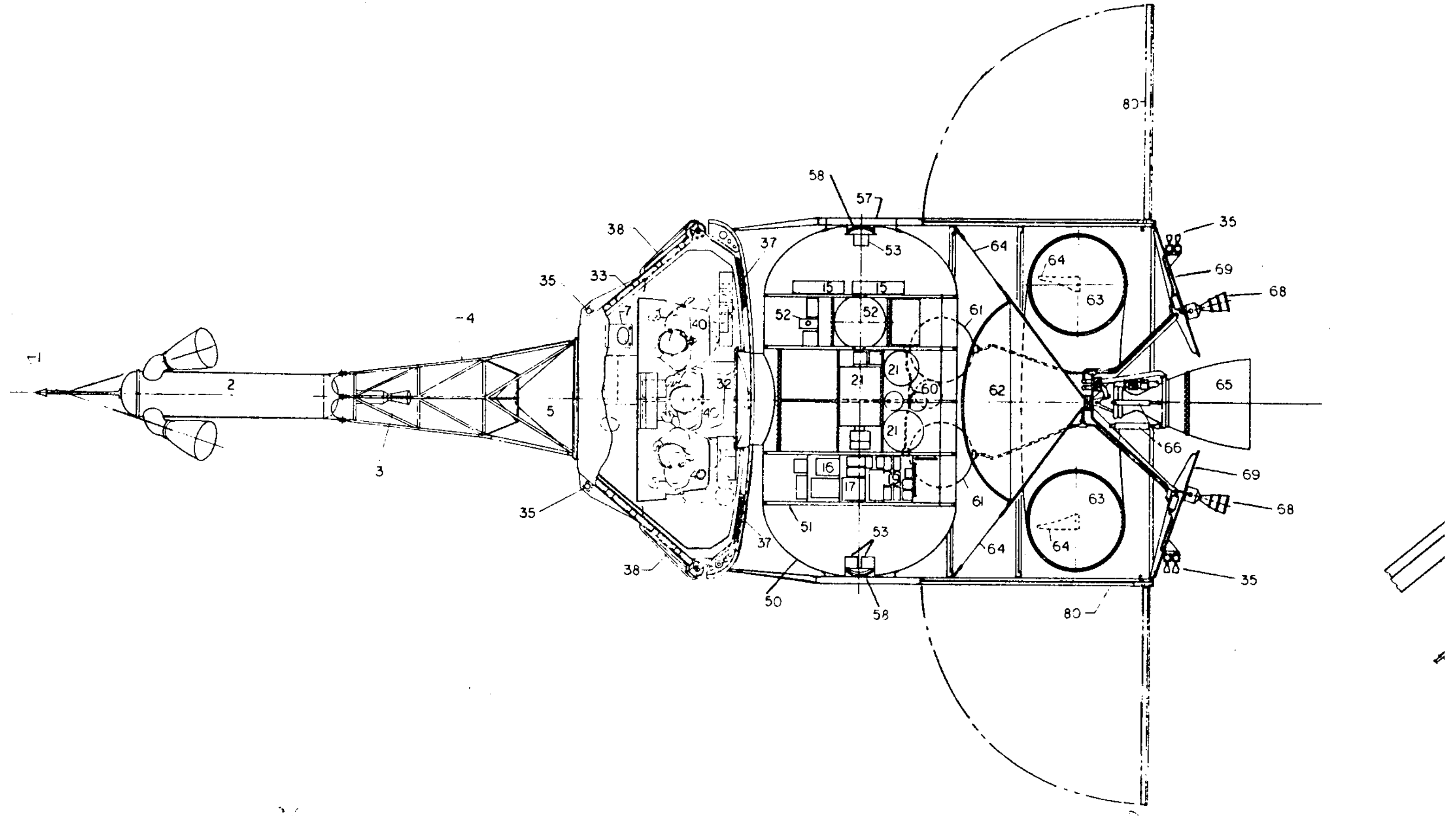

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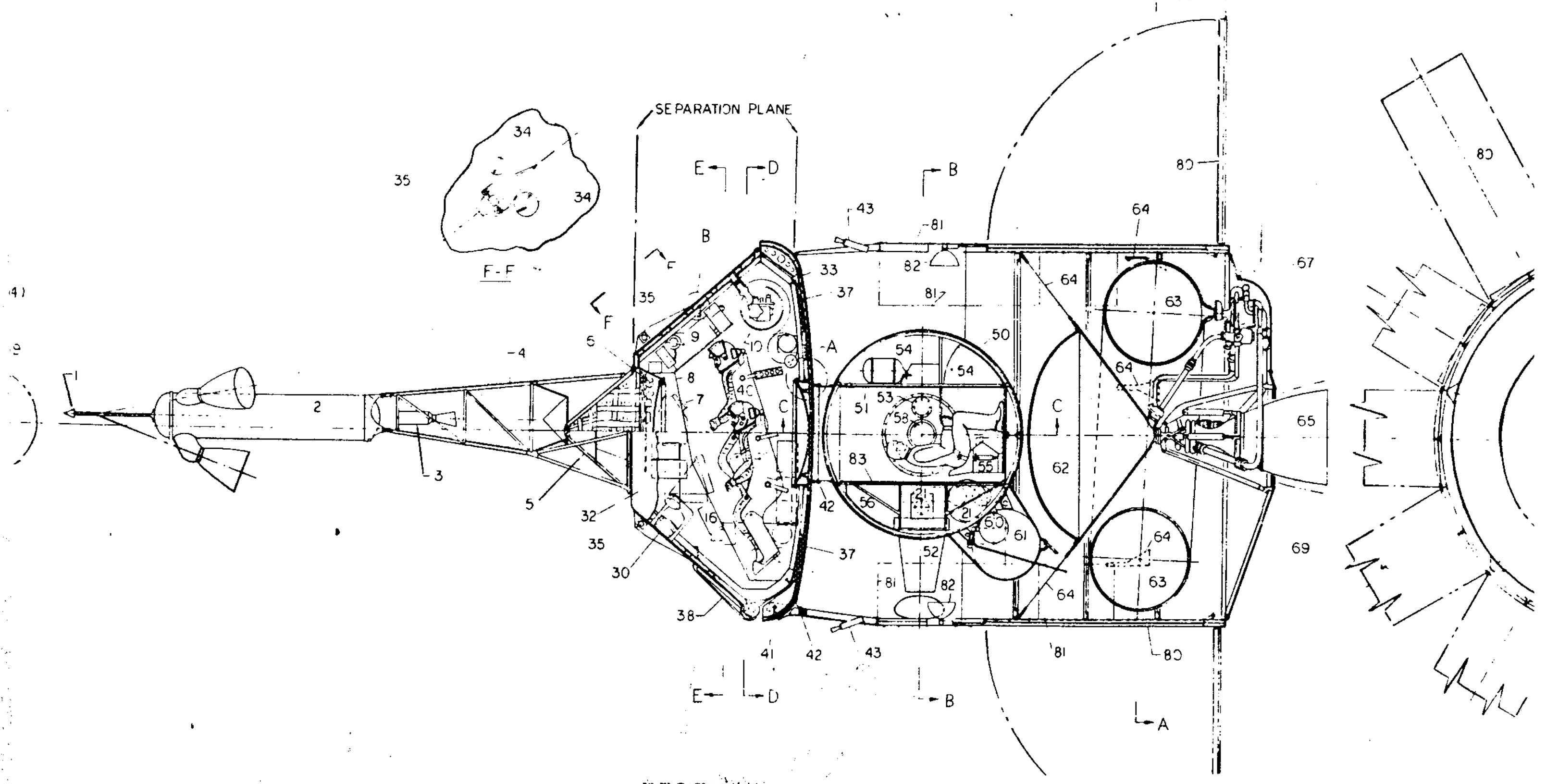

SEPARATION PLANE
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FOLDOUT [illegible] 2

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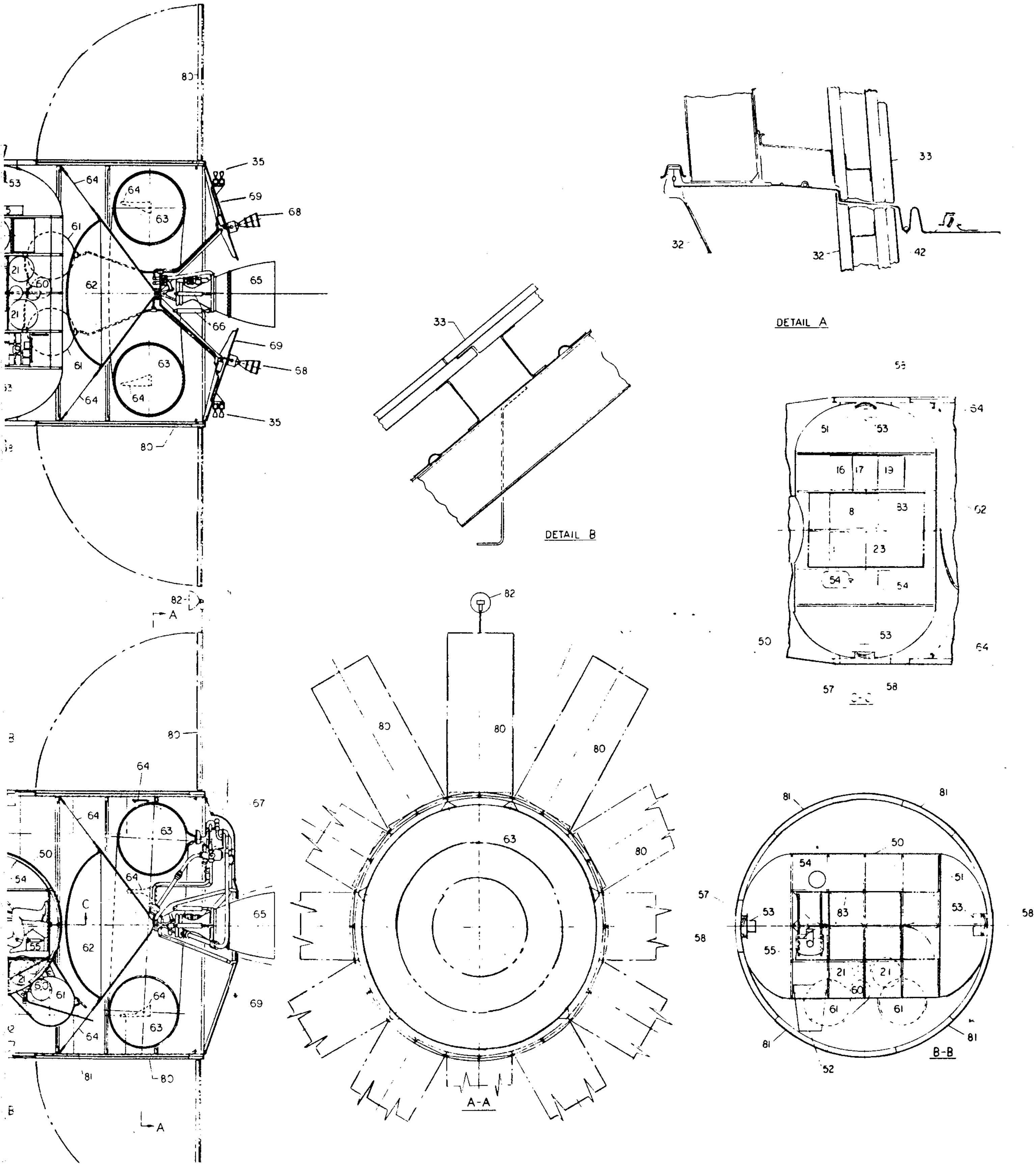

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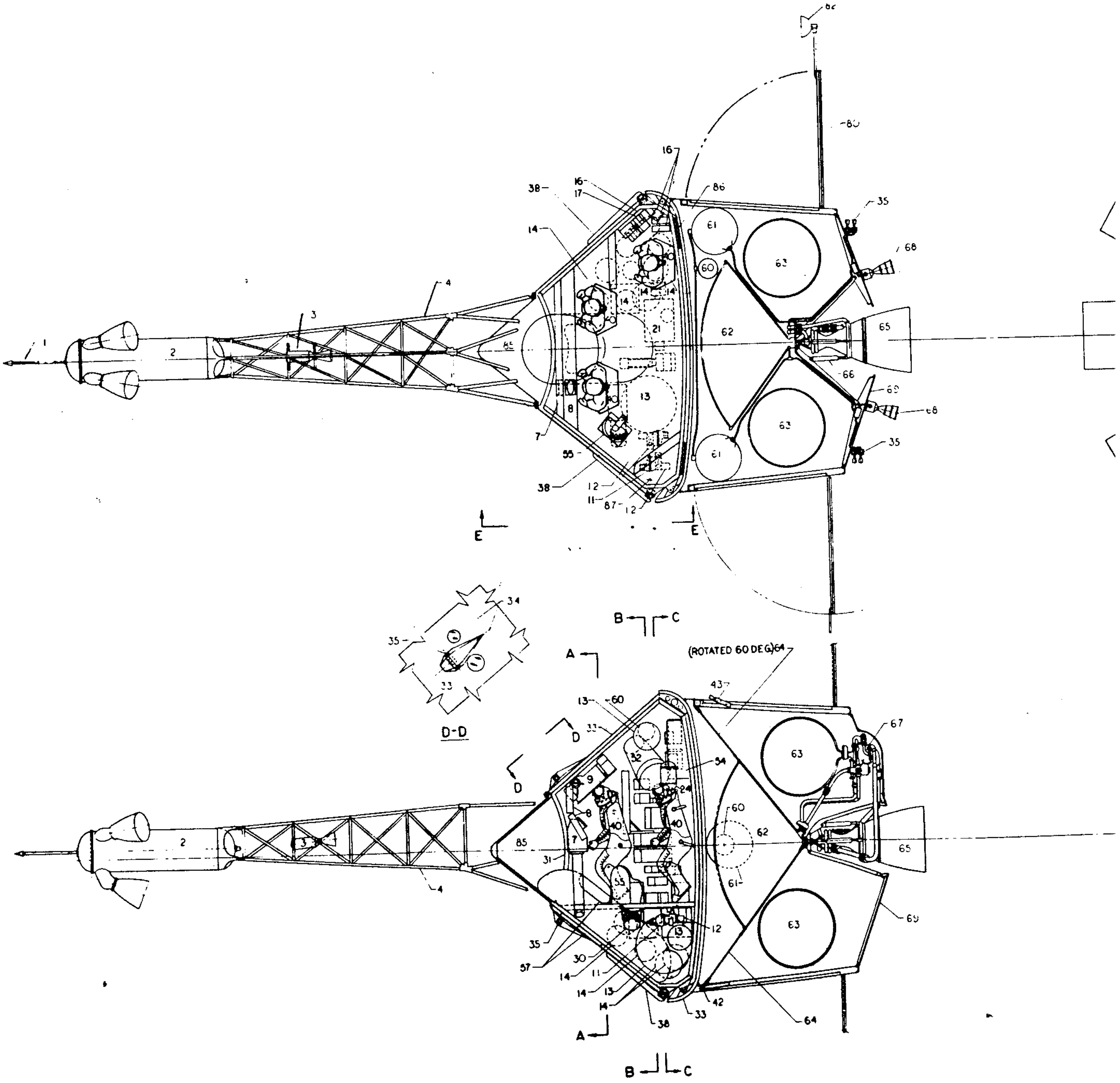
FOLDOUT FRAME 3

FIG. VII–4. L–2–C BODY, MODULAR



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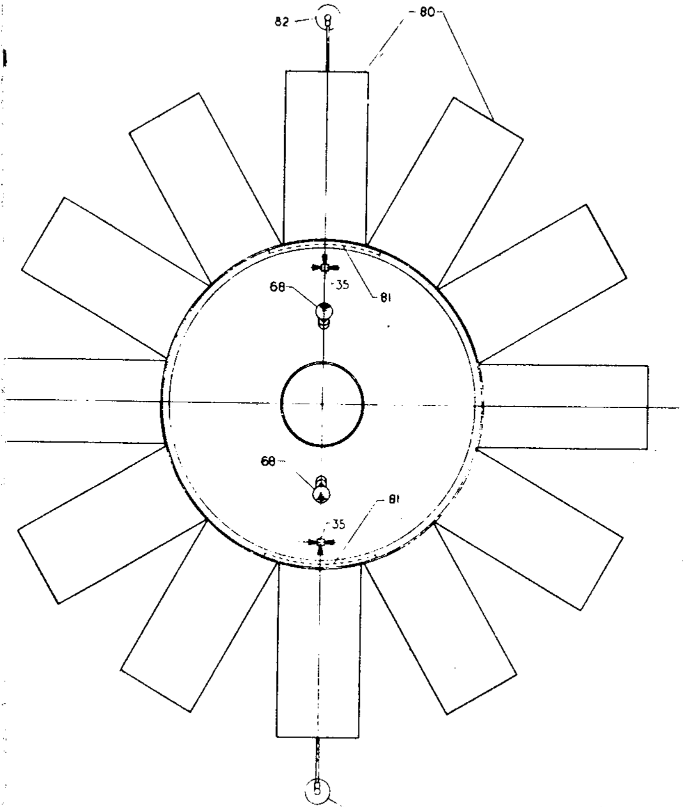
FOLDOUT FRAME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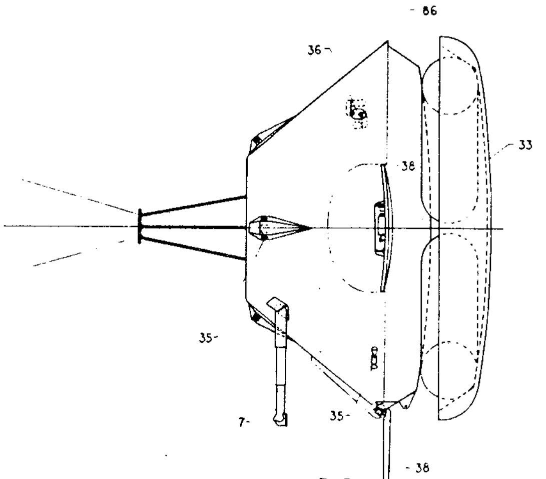
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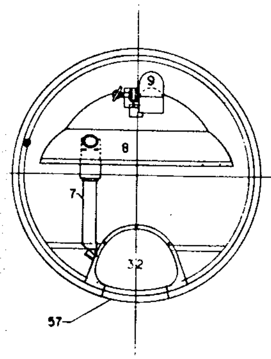
E-E

ALTERNATE ARRANGEMENT
SHOWING 6 PLACE VERSION



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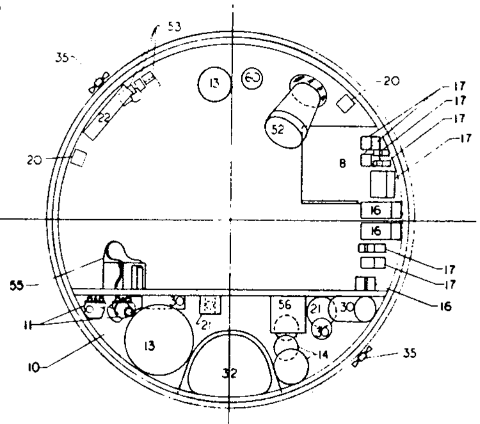
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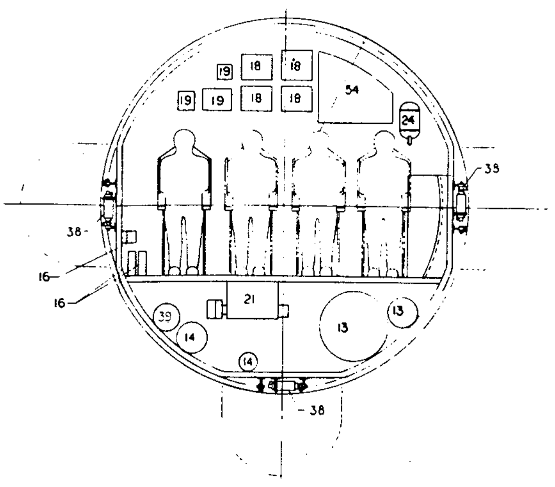
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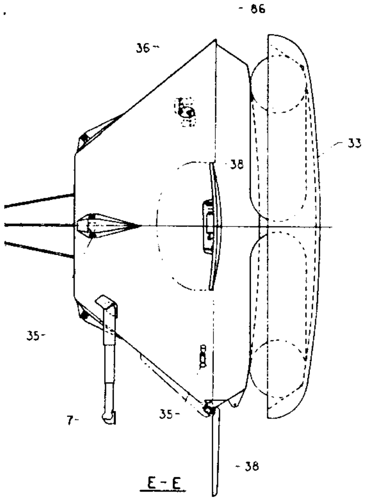
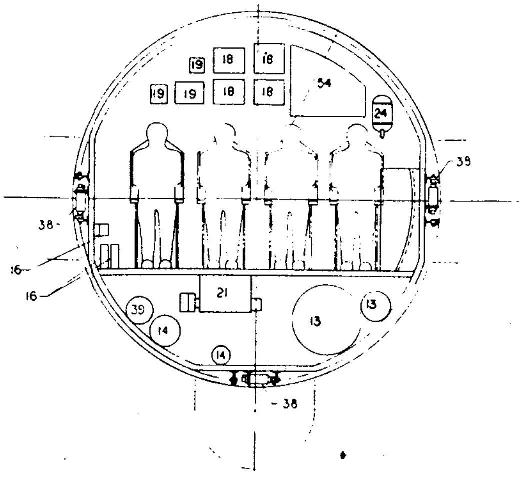
C-C

FOLDOUT FRAME 2



E-E

ALTERNATE ARRANGEMENT – SHOWING 6 PLACE VERSION

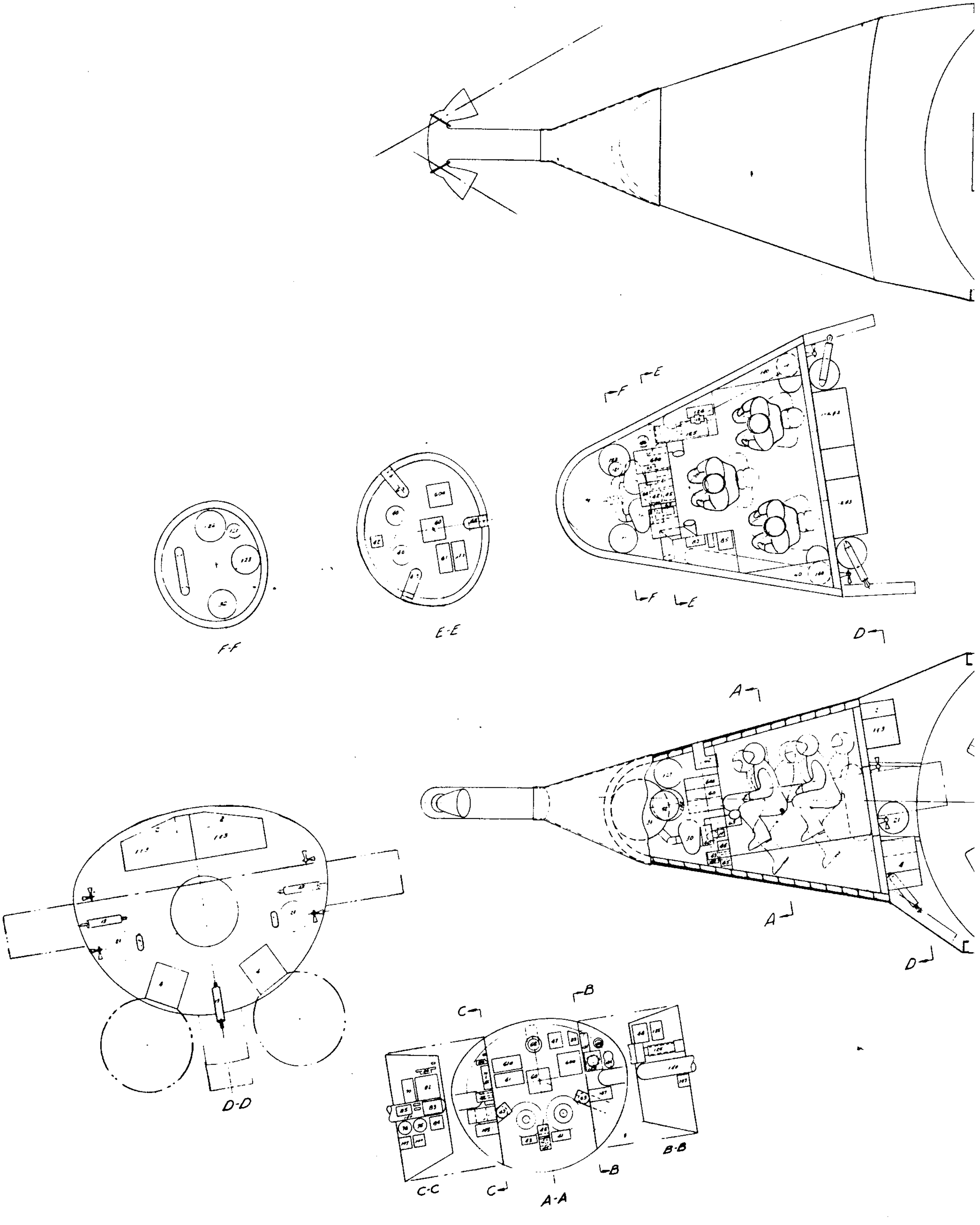


C-C

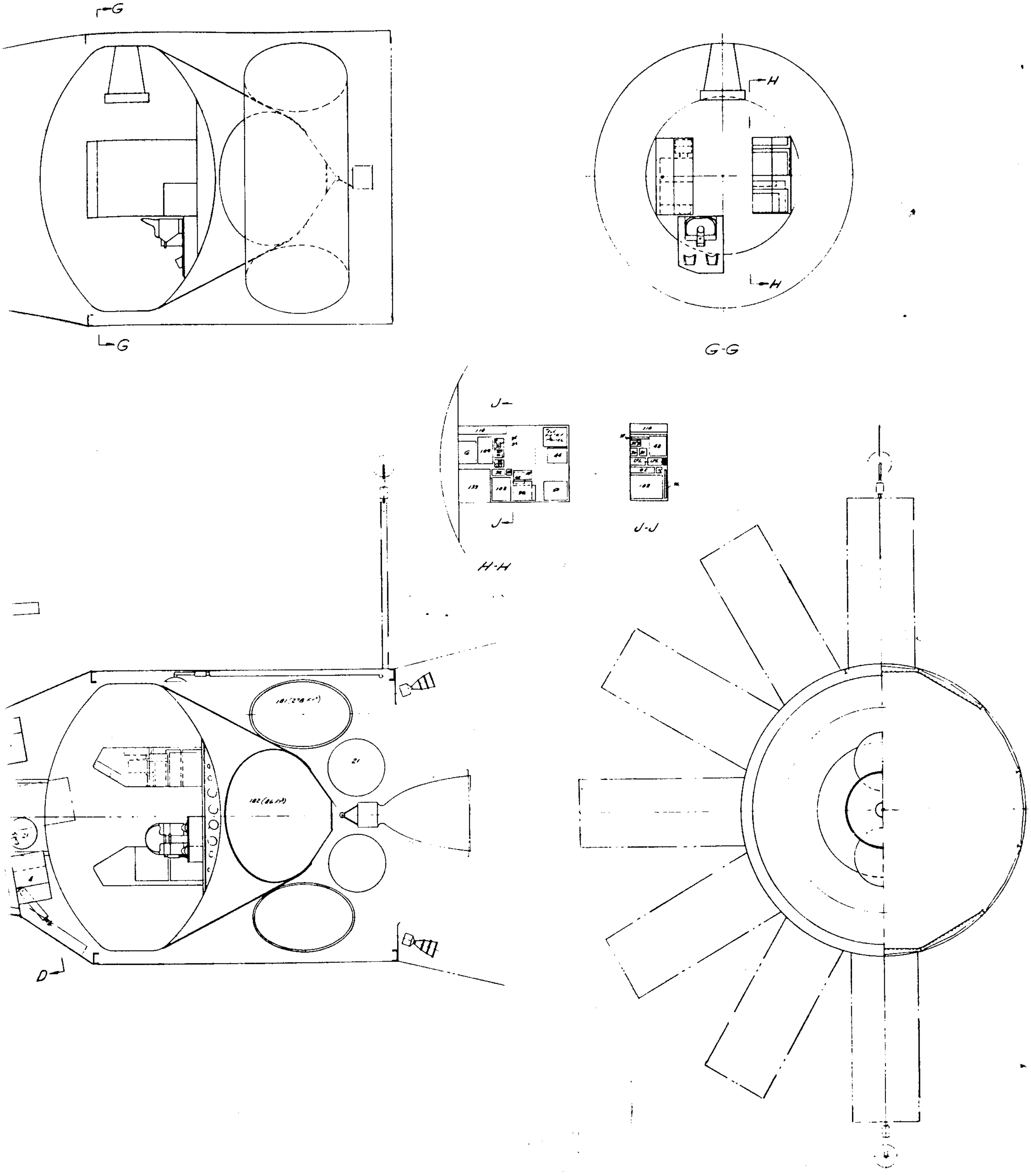
1. PITOT HEAD
2. ESCAPE ROCKET
3. ESCAPE SYSTEM SEPARATION ROCKET
4. ESCAPE ROCKET SUPPORT TRUSS
7. PERISCOPE AND DRIFT SIGHT
8. INSTRUMENT AND CONTROL PANEL
9. GUIDANCE EQUIPMENT
10. SAFETY COMPARTMENT
11. AUXILIARY POWER UNIT
12. ELECTRICAL GENERATOR
13. FUEL AND OXIDIZER TANKS FOR APU AND CONTROL SYSTEMS
14. O_2 AND N_2 SUPPLY TANKS
16. ELECTRICAL SYSTEM COMPONENTS
17. COMMUNICATION SYSTEM COMPONENTS
18. GUIDANCE AND NAVIGATION EQUIPMENT
19. TELEMETRY EQUIPMENT
20. RBE MEASURING EQUIPMENT
21. ENVIRONMENTAL CONTROL SYSTEM COMPONENTS (CABIN AND EQUIPMENT)
22. SURVIVAL KIT
24. DRINKING WATER SUPPLY
30. ENVIRONMENTAL CONTROL SYSTEM COMPONENTS (STRUCTURAL)
31. PRESSURE BULKHEAD
32. AIR LOCK
33. HEAT SHIELD
34. WINDOW
35. REACTION CONTROL NOZZLES AND SERVOVALVES
36. SNORKEL
38. AERODYNAMIC CONTROL SURFACE AND ACTUATOR
39. GAS GENERATING EQUIPMENT
40. SEAT
42. MODULE SEPARATION DEVICE
43. MODULE SEPARATION ROCKETS
52. CAMERA
53. SCIENTIFIC DATA GATHERING EQUIPMENT
54. FOOD AND WATER STOWAGE
55. WASTE COLLECTION
56. WATER RECOVERY UNIT
57. MAIN ENTRANCE HATCH
60. H_e PRESSURE BOTTLE
61. FUEL AND OXIDIZER TANKS--REACTION CONTROL AND VERNIER ROCKET SYSTEM
62. MAIN LOX TANK
63. MAIN LH TANK
64. TANK SUPPORT AND INSULATING STRUCTURE
65. MAIN ENGINE (XLR-119 MODIFIED)
66. GIMBAL ACTUATORS
67. PUMP
68. VERNIER ROCKET NOZZLE
69. FLAME SHIELD
80. SOLAR CELL ARRAY
81. SPACE RADIATOR
82. ANTENNA
85. PARACHUTE PACK AND SEQUENCING EQUIPMENT
86. LANDING BAG
87. WASTE DISPOSAL

FOLDOUT FRAME 3

FIG. VII-5. L-2-C BODY, INTEGRATED

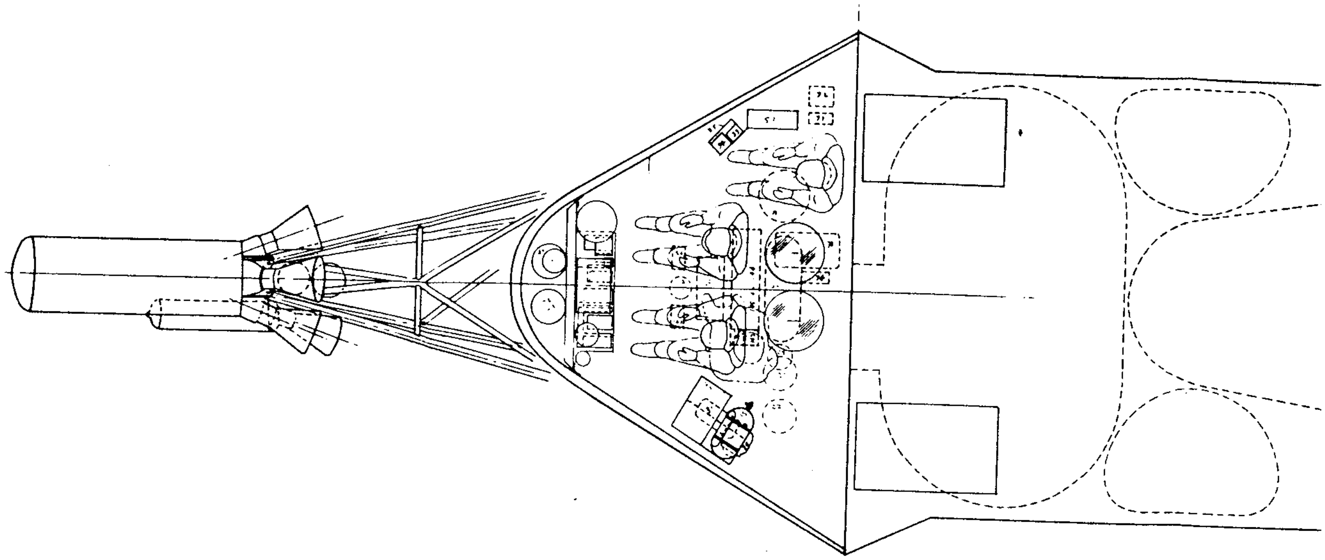

F
E
F
E
D
F-F
E-E
A
A
D
D-D
C
B
C
B
C-C
A-A
B-B

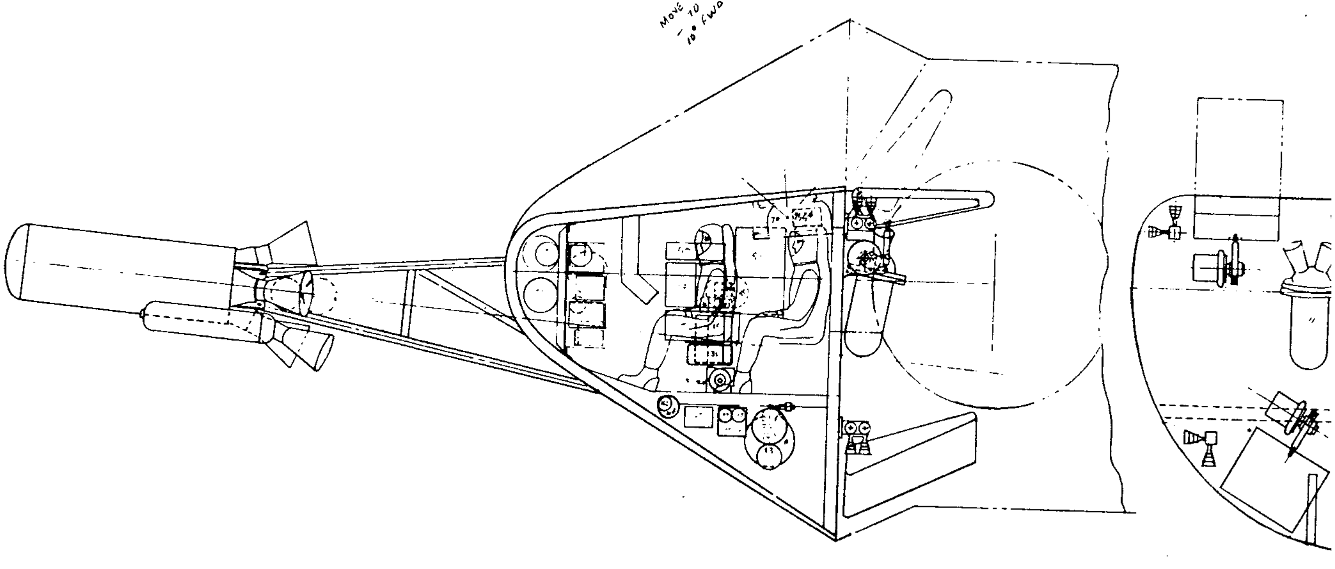
FOLDOUT FRAME 1


G
G
H
H
G-G
J
J
118
G
108
44
139
102
118
48
102
J-J
H-H
181(278 ft³)
21
182(86 ft³)
D

FOLDOUT FRAME 2

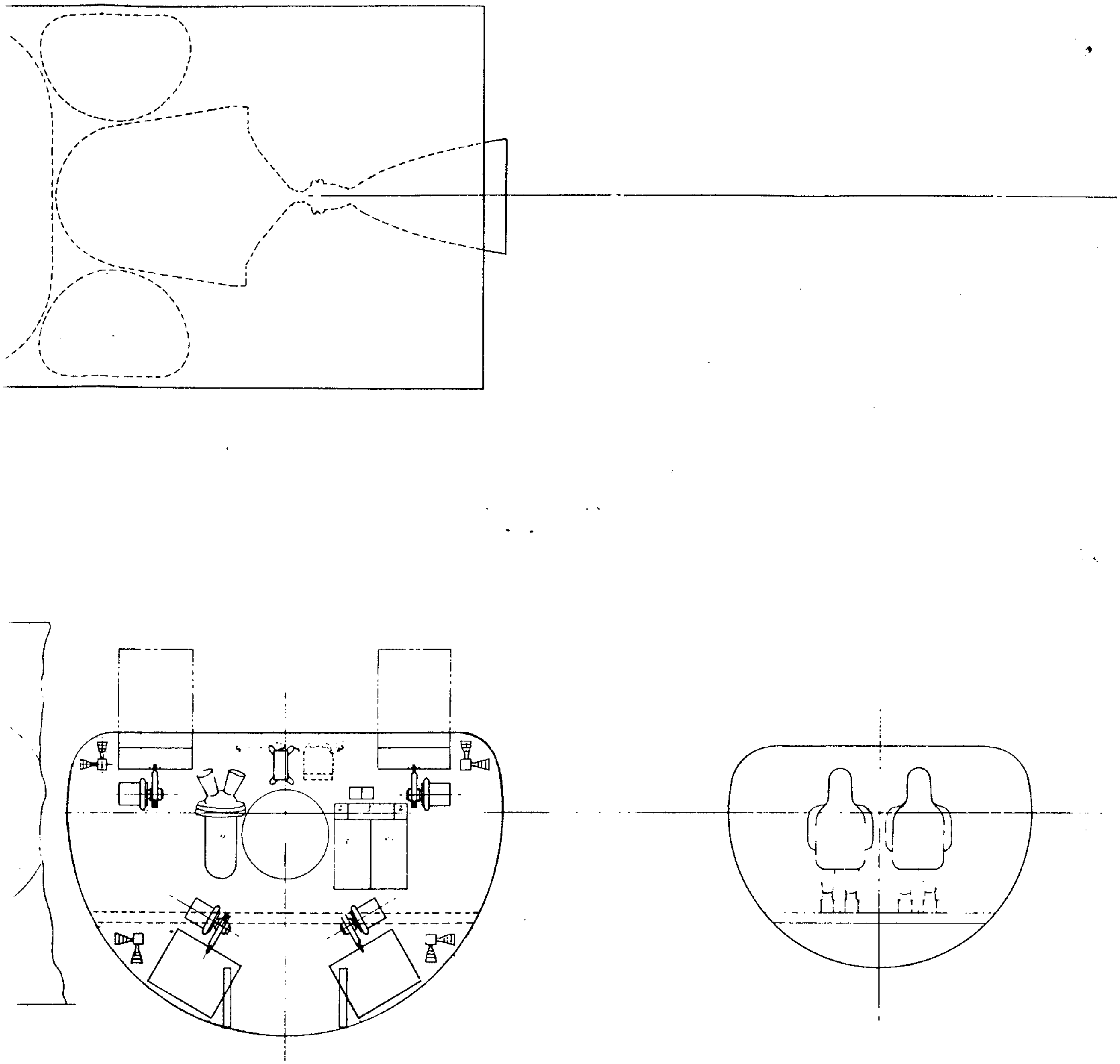
FIG. VII-6. M-1-1 LIFTING BODY, MODULAR





MOVE MEN ⊥ TO THRUST LINE 10° FWD FOR LAUNCH

FOLDOUT FRAME 1



FOLDOUT FRAME 2

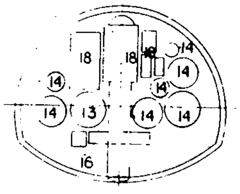
FIG. VII–7. M–1 LIFTING BODY, INTEGRATED

1. PITOT HEAD
2. ESCAPE ROCKET
3. ESCAPE SYSTEM SEPARATION ROCKET
4. ESCAPE ROCKET SUPPORT TRUSS
5. PARACHUTE AND ROCKET PACKAGE
6. PARACHUTE SEQUENCING EQUIPMENT
7. PERISCOPE AND DRIFT SIGHT
8. INSTRUMENT AND CONTROL PANEL
9. GUIDANCE EQUIPMENT
10. SAFETY COMPARTMENT
11. AUXILIARY POWER UNIT
12. ELECTRICAL GENERATOR
13. FUEL AND OXIDIZER TANKS FOR APU AND CONTROL SYSTEMS
14. O_2 AND N_2 SUPPLY TANKS
15. FILM STOWAGE
16. ELECTRICAL SYSTEM COMPONENTS
17. COMMUNICATION SYSTEM COMPONENTS
18. GUIDANCE AND NAVIGATION EQUIPMENT
19. TELEMETRY EQUIPMENT
20. RBE MEASURING EQUIPMENT
21. ENVIRONMENTAL CONTROL SYSTEM COMPONENTS (CABIN AND EQUIPMENT)
22. SURVIVAL KIT -3 MAN

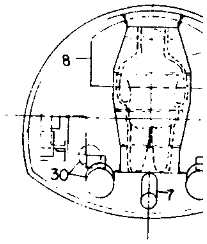
24. DRINKING WATER SUPPLY
30. ENVIRONMENTAL CONTROL SYSTEM COMPONENTS (STRUCTURAL)
31. PRESSURE BULKHEAD
32. HATCH
33. HEAT SHIELD
34. WINDOW
35. REACTION CONTROL NOZZLES AND SERVOVALVES
36. SNORKEL

38. AERODYNAMIC CONTROL SURFACE AND ACTUATOR
39. GAS GENERATING EQUIPMENT
40. SEAT

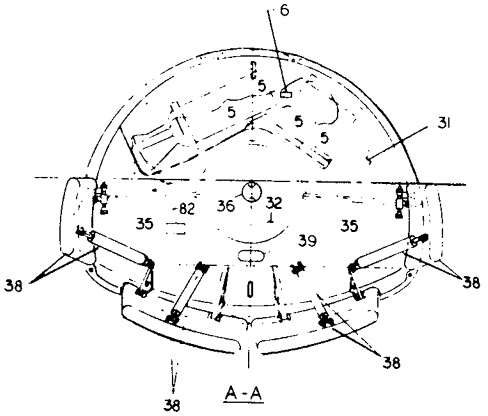
42. MODULE SEPARATION DEVICE
43. MODULE SEPARATION ROCKETS
50. MISSION MODULE
51. EQUIPMENT SUPPORTING STRUCTURE
52. CAMERA
53. SCIENTIFIC DATA GATHERING EQUIPMENT
54. FOOD AND WATER STOWAGE
55. WASTE COLLECTION
56. WATER RECOVERY UNIT
57. MAIN ENTRANCE HATCH
58. VIEWING PORT
60. H PRESSURE BOTTLE
61. FUEL AND OXIDIZER TANKS--REACTION CONTROL AND VERNIER ROCKET SYSTEM
62. MAIN LOX TANK
63. MAIN LH TANK
64. TANK SUPPORT AND INSULATING STRUCTURE
65. MAIN ENGINE (XLR-119 MODIFIED)
66. GIMBAL ACTUATORS
67. PUMP
68. VERNIER ROCKET NOZZLE
69. FLAME SHIELD
80. SOLAR CELL ARRAY
81. SPACE RADIATOR
82. ANTENNA
83. HINGED RADIATION SHIELD



D-D

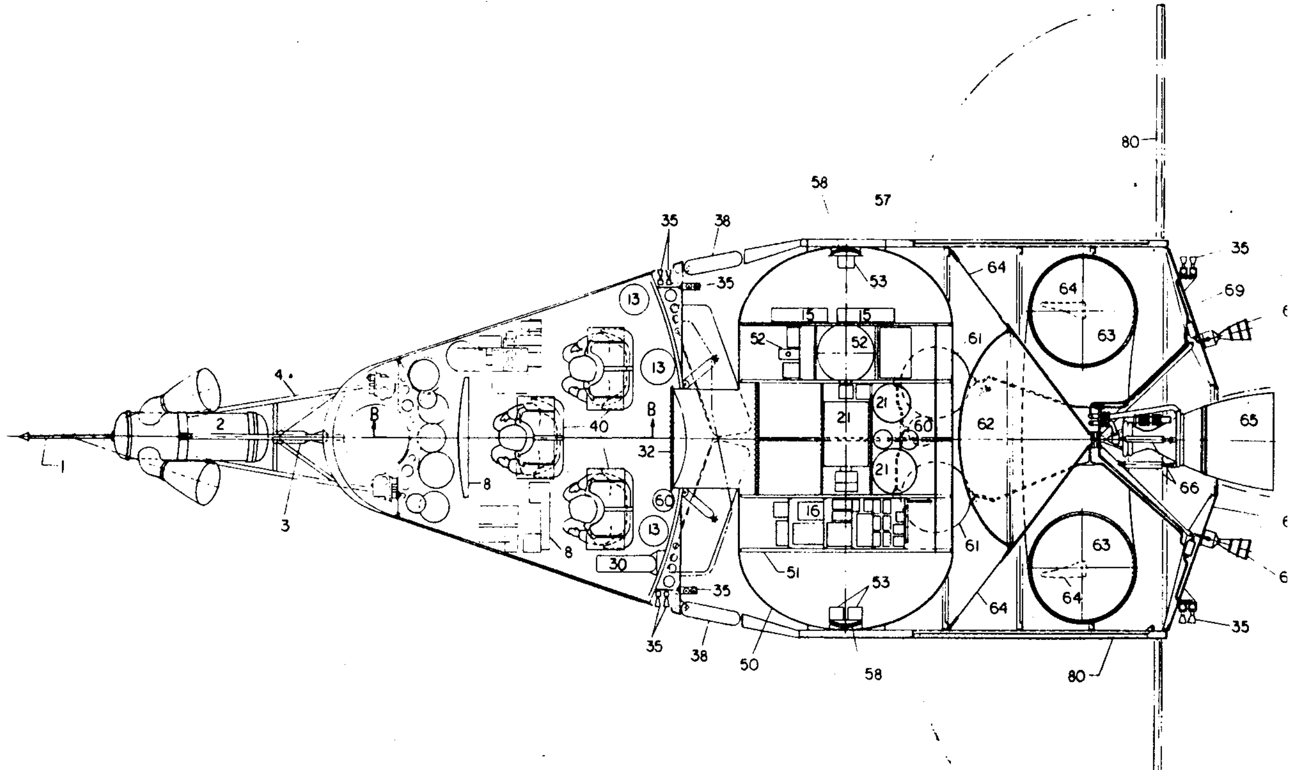


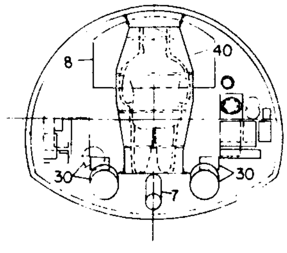
C-C



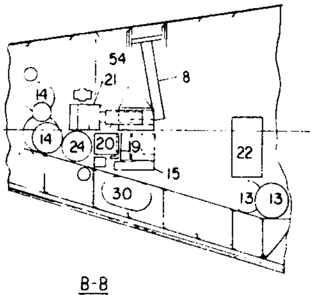
A-A

FOLDOUT FRAME 1

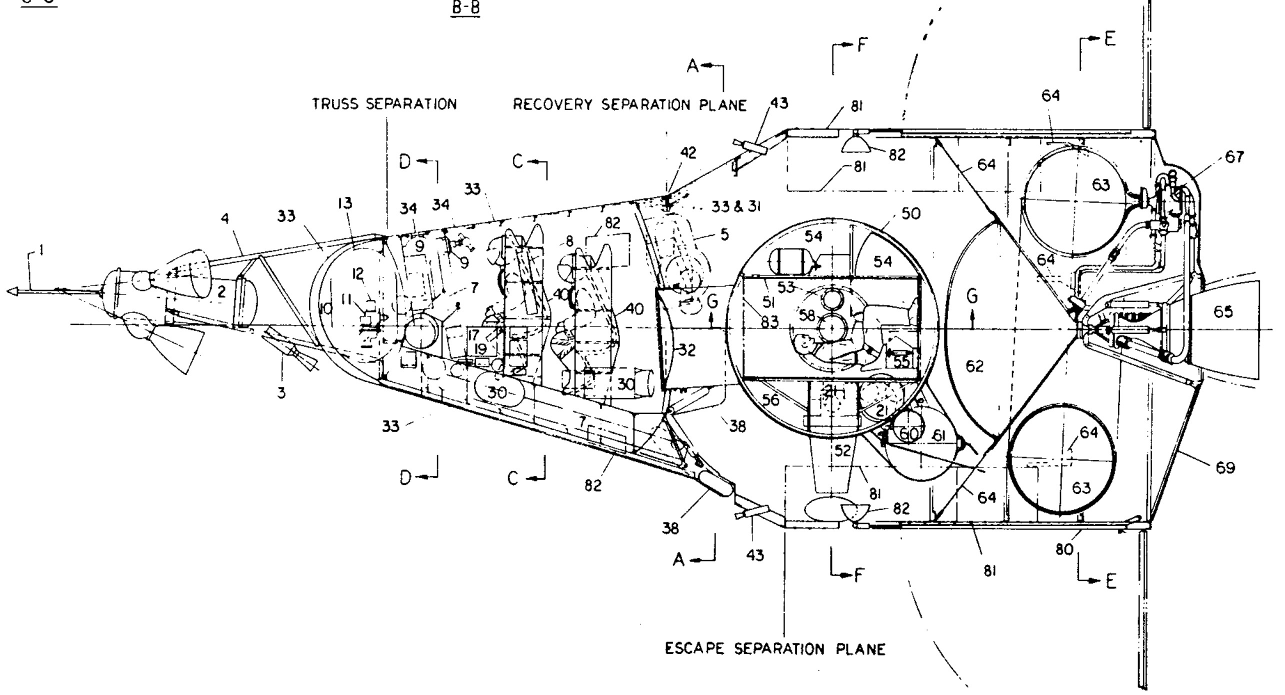

80
58
57
35
38
35
53
64
64
63
15
15
52
52
61
13
13
21
21
60
62
65
4
2
B
40
B
32
21
21
L1
8
60
16
66
3
8
13
61
30
51
63
64
64
35
53
35
35
38
50
58
80



C-C


54
21
8
14
14
24
20
19
22
15
30
13
13

B-B


82
80
F
E
A
TRUSS SEPARATION
RECOVERY SEPARATION PLANE
43
81
64
67
D
C
42
82
81
63
1
4
33
13
34
34
33
33 & 31
50
12
9
9
8
82
5
54
54
64
2
11
7
10
40
40
G
51
83
53
58
G
65
17
19
30
30
32
55
62
3
33
7
56
38
21
21
60
61
52
64
D
C
82
81
82
64
63
69
38
A
43
F
81
80
E
ESCAPE SEPARATION PLANE

38

FOLDOUT FRAME 2

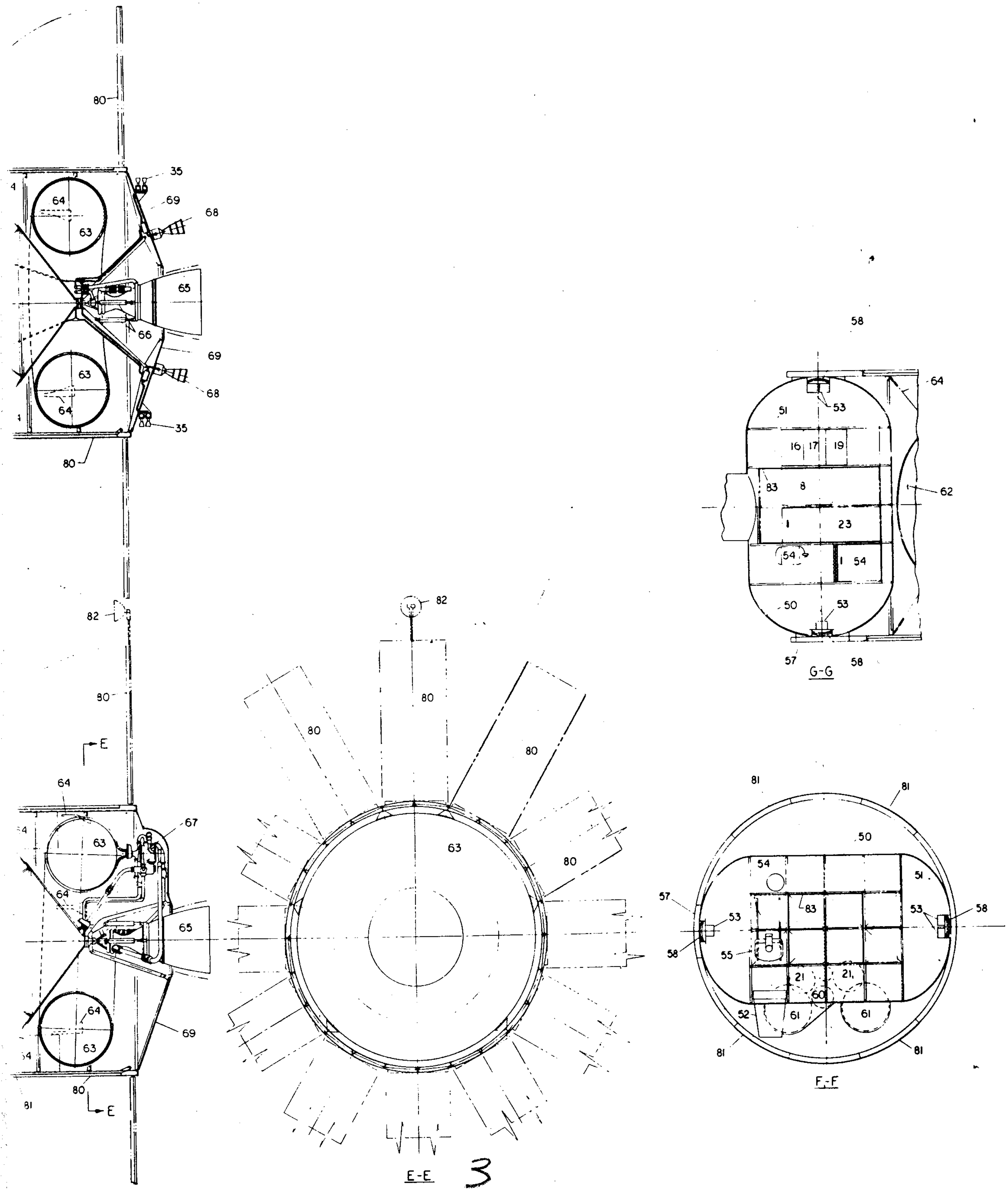

80
35
64
69
68
63
65
66
69
63
68
64
35
80
58
64
51
53
16
17
19
83
8
62
1
23
54
1
54
50
53
57
58
G-G
82
82
80
80
80
80
E
64
67
63
64
65
64
63
69
80
E
81
63
80
81
81
50
54
51
57
53
83
53
58
58
55
21
21
60
52
61
61
81
81
F-F
E-E
3

FIG. VII-8. W-1 LIFTING BODY, MODULAR

FOLDOUT FRAME

2. ESCAPE ROCKET
3. ESCAPE SYSTEM SEPARATION ROCKET
4. ESCAPE ROCKET SUPPORT TRUSS
5. PARACHUTE AND ROCKET PACKAGE
6. PARACHUTE SEQUENCING EQUIPMENT
7. PERISCOPE AND DRIFT SIGHT
8. INSTRUMENT AND CONTROL PANEL
9. GUIDANCE EQUIPMENT
10. SAFETY COMPARTMENT
11. AUXILIARY POWER UNIT
12. ELECTRICAL GENERATOR
13. FUEL AND OXIDIZER TANKS FOR APU AND CONTROL SYSTEMS
14. O_2 AND N_2 SUPPLY TANKS
15. FILM STOWAGE
16. ELECTRICAL SYSTEM COMPONENTS
17. COMMUNICATION SYSTEM COMPONENTS
18. GUIDANCE AND NAVIGATION EQUIPMENT
19. TELEMETRY EQUIPMENT
20. RBE MEASURING EQUIPMENT
21. ENVIRONMENTAL CONTROL SYSTEM COMPONENTS (CABIN AND EQUIPMENT)
22. SURVIVAL KIT

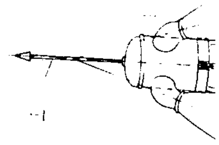
24. DRINKING WATER SUPPLY
30. ENVIRONMENTAL CONTROL SYSTEM COMPONENTS (STRUCTURAL)
31. PRESSURE BULKHEAD
32. HATCH
33. HEAT SHIELD
34. WINDOW
35. REACTION CONTROL NOZZLES AND SERVOVALVES
36. SNORKEL

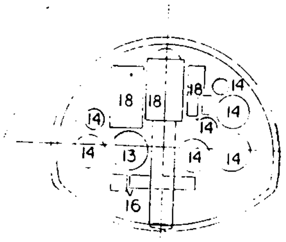
38. AERODYNAMIC CONTROL SURFACE AND ACTUATOR
39. GAS GENERATING EQUIPMENT
40. SEAT

42. MODULE SEPARATION DEVICE
43. MODULE SEPARATION ROCKETS

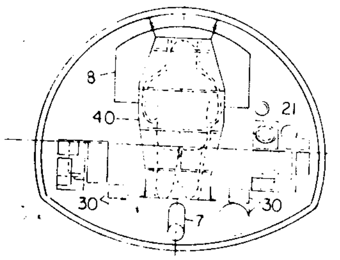
52. CAMERA
53. SCIENTIFIC DATA GATHERING EQUIPMENT
54. FOOD AND WATER STOWAGE
55. WASTE COLLECTION
56. WATER RECOVERY UNIT
57. MAIN ENTRANCE HATCH

60. H_e PRESSURE BOTTLE
61. FUEL AND OXIDIZER TANKS—REACTION CONTROL AND VERNIER ROCKET SYSTEM
62. MAIN LOX TANK
63. MAIN LH TANK
64. TANK SUPPORT AND INSULATING STRUCTURE
65. MAIN ENGINE (XLR-119 MODIFIED)
66. GIMBAL ACTUATORS
67. PUMP
68. VERNIER ROCKET NOZZLE
69. FLAME SHIELD
80. SOLAR CELL ARRAY
81. SPACE RADIATOR
82. ANTENNA

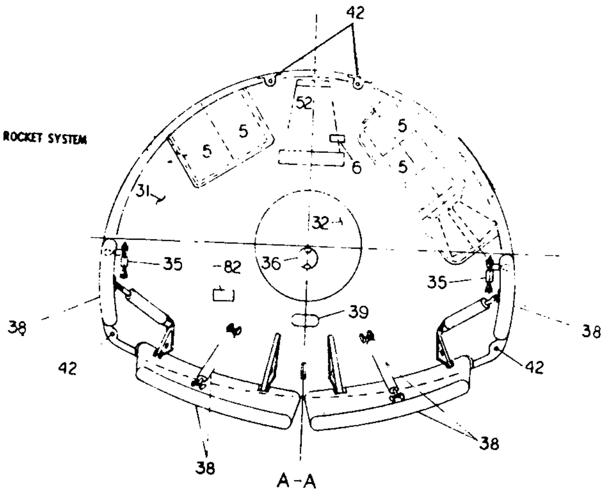




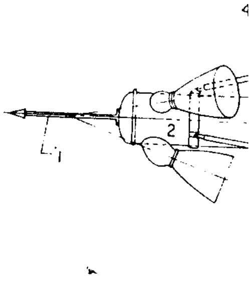
C-C



B-B

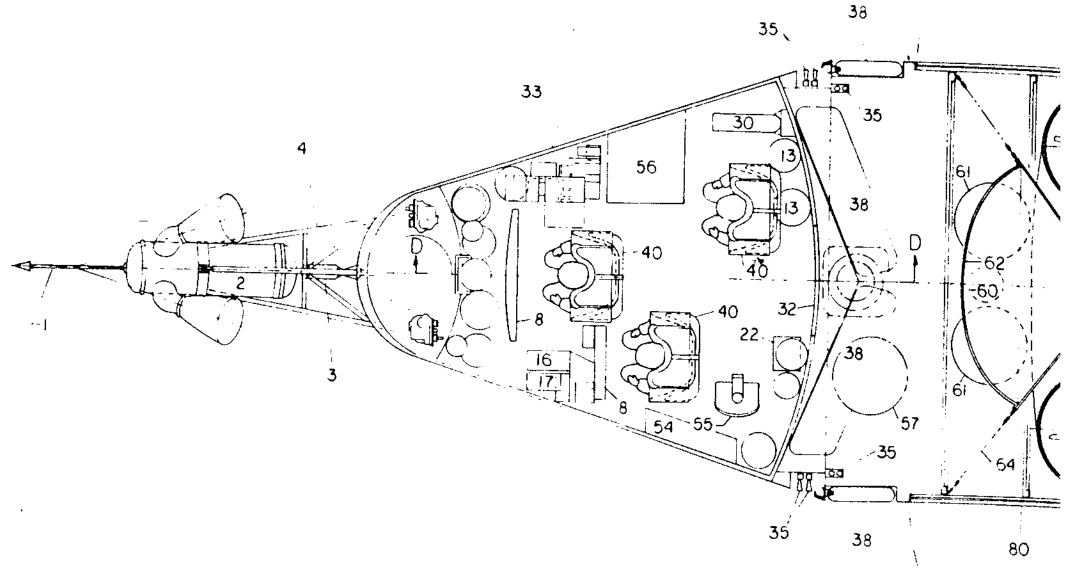


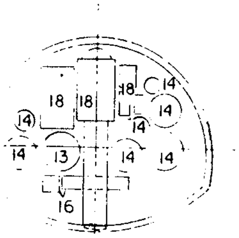
A-A



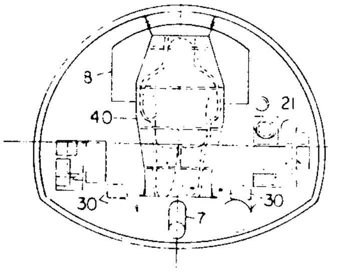
FOLDOUT FRAME 1

[redacted]

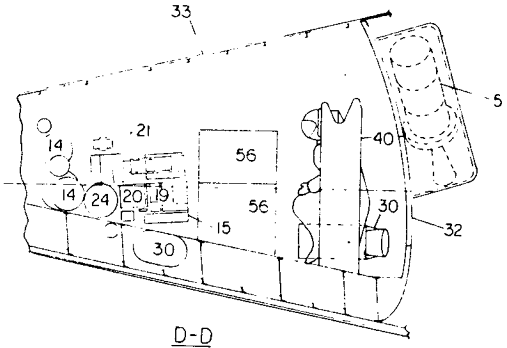

38
35
33
4
30
56
13
13
38
35
D
D
40
40
2
1
8
40
32
22
16
17
8
3
54
55
38
57
35
61
62
60
61
64
35
38
80


14
18
18
18
14
14
14
14
13
14
14
16

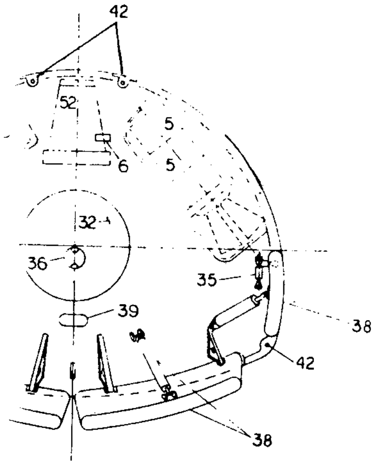
C-C


8
40
21
30
30
7

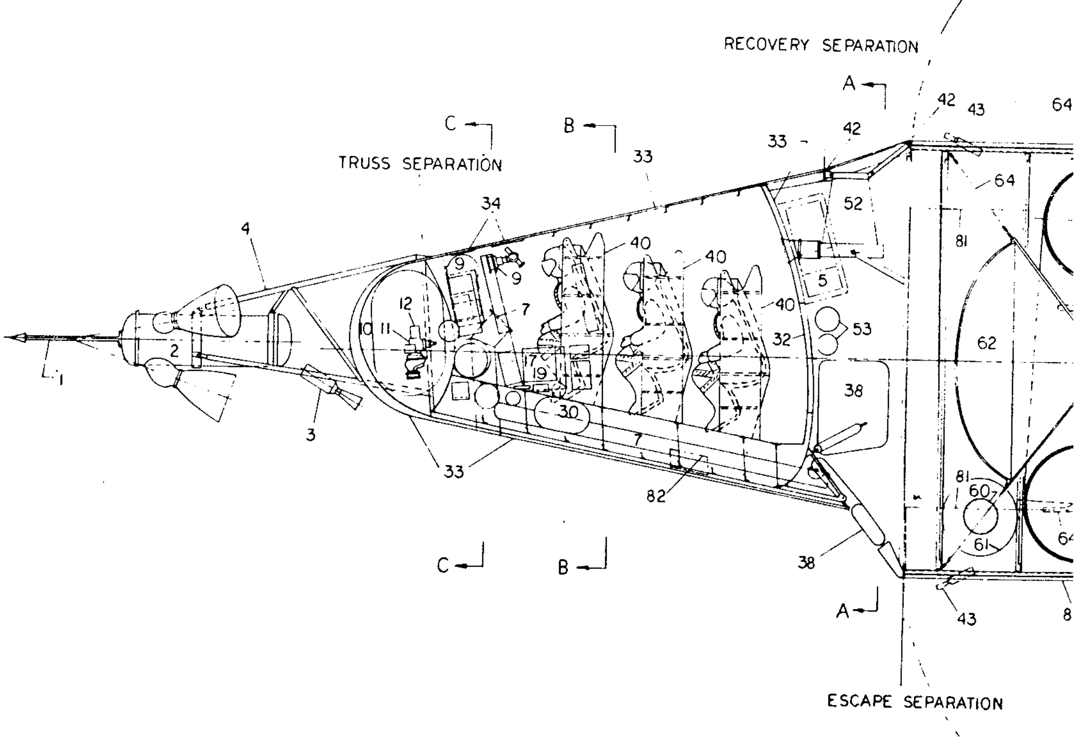
B-B


33
5
40
21
14
56
56
14
24
20
19
30
15
30
32

D-D

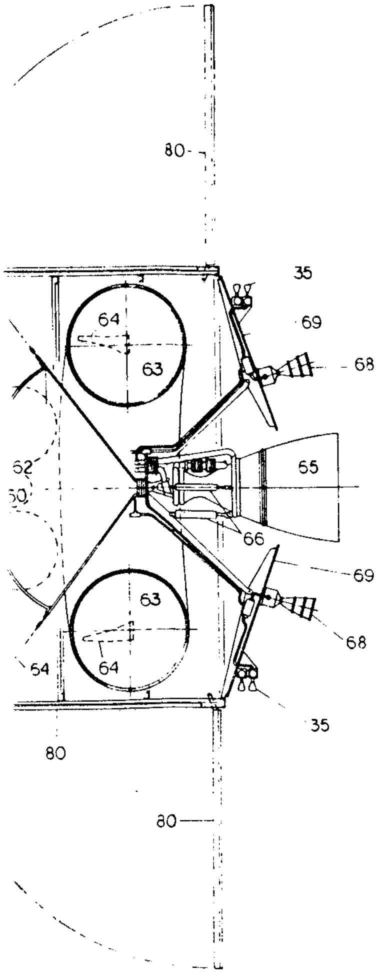

42
52
5
6
5
32
36
35
39
38
42
38

A-A

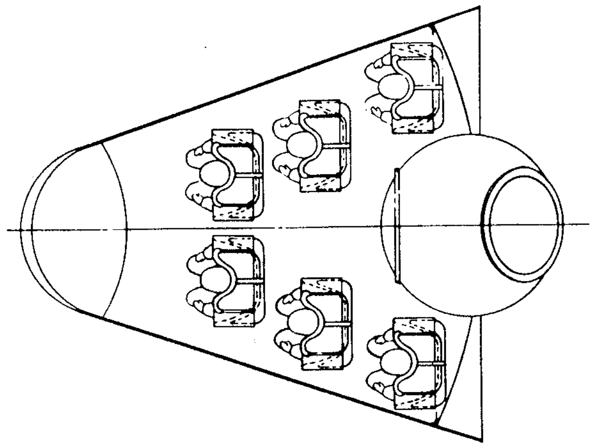

RECOVERY SEPARATION
A
42
43
64
C
B
33
42
TRUSS SEPARATION
33
34
52
4
9
9
40
40
81
12
10
11
7
40
5
53
32
62
2
1
17
19
30
38
3
7
33
82
81
60
61
64
38
C
B
A
43
8
ESCAPE SEPARATION

FOLDOUT FRAME 2

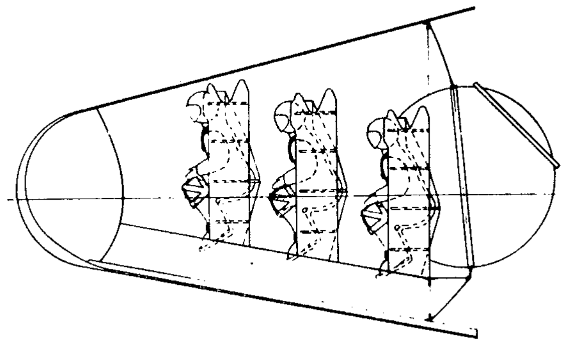
[redacted]

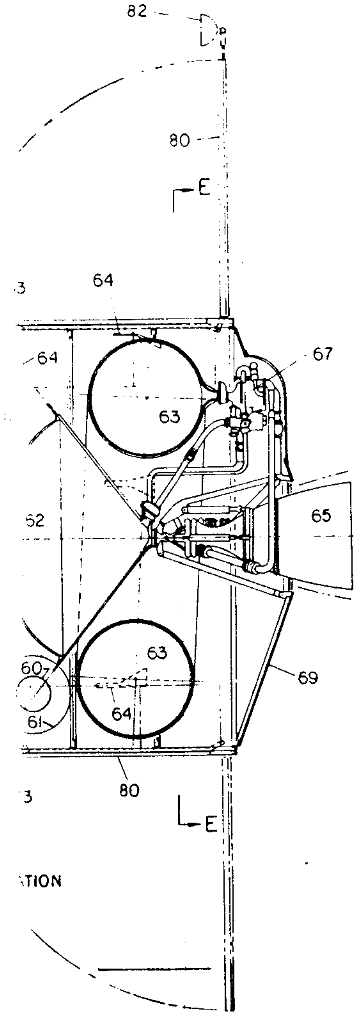


80
35
69
64
63
68
62
60
65
66
69
63
68
64
64
35
80
80

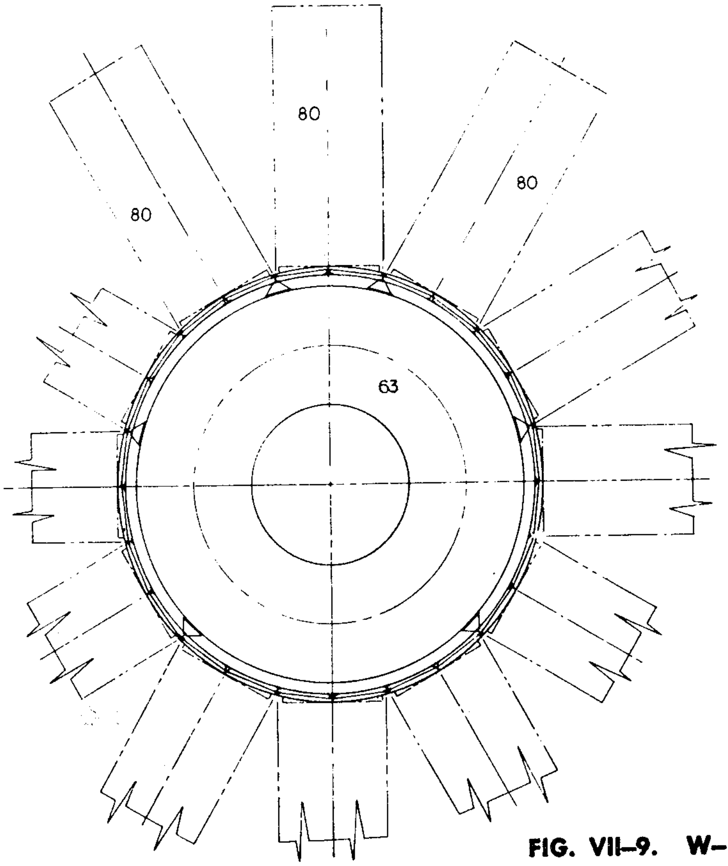


ALTERNATE 6 MAN ARRANGEMENT





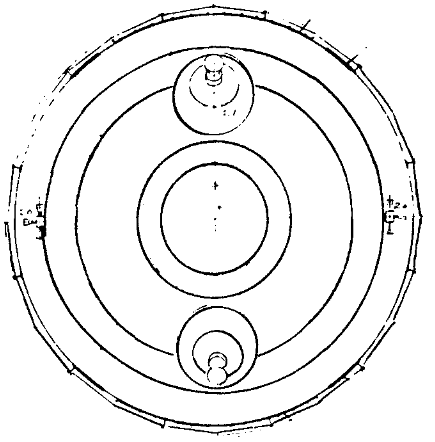
82
80
E
3
64
64
67
63
62
65
60
63
61
64
69
3
80
E
TION

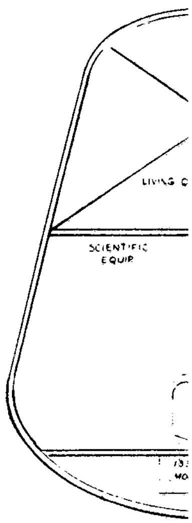


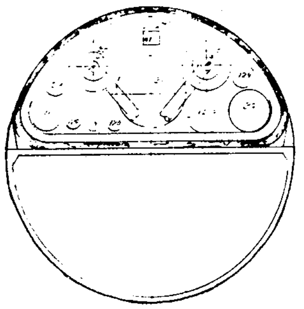
E-E

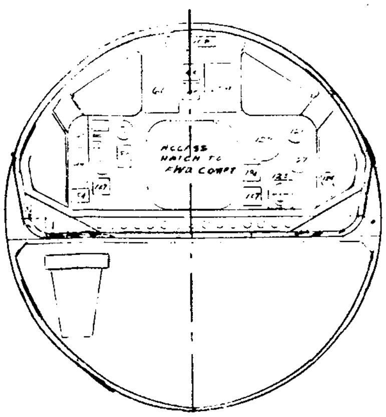
FIG. VII-9. W-1 LIFTING BODY, INTEGRATED

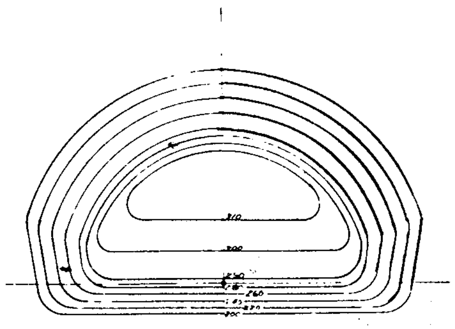
FOLDOUT FRAME 3




LIVING C
SCIENTIFIC
EQUIP

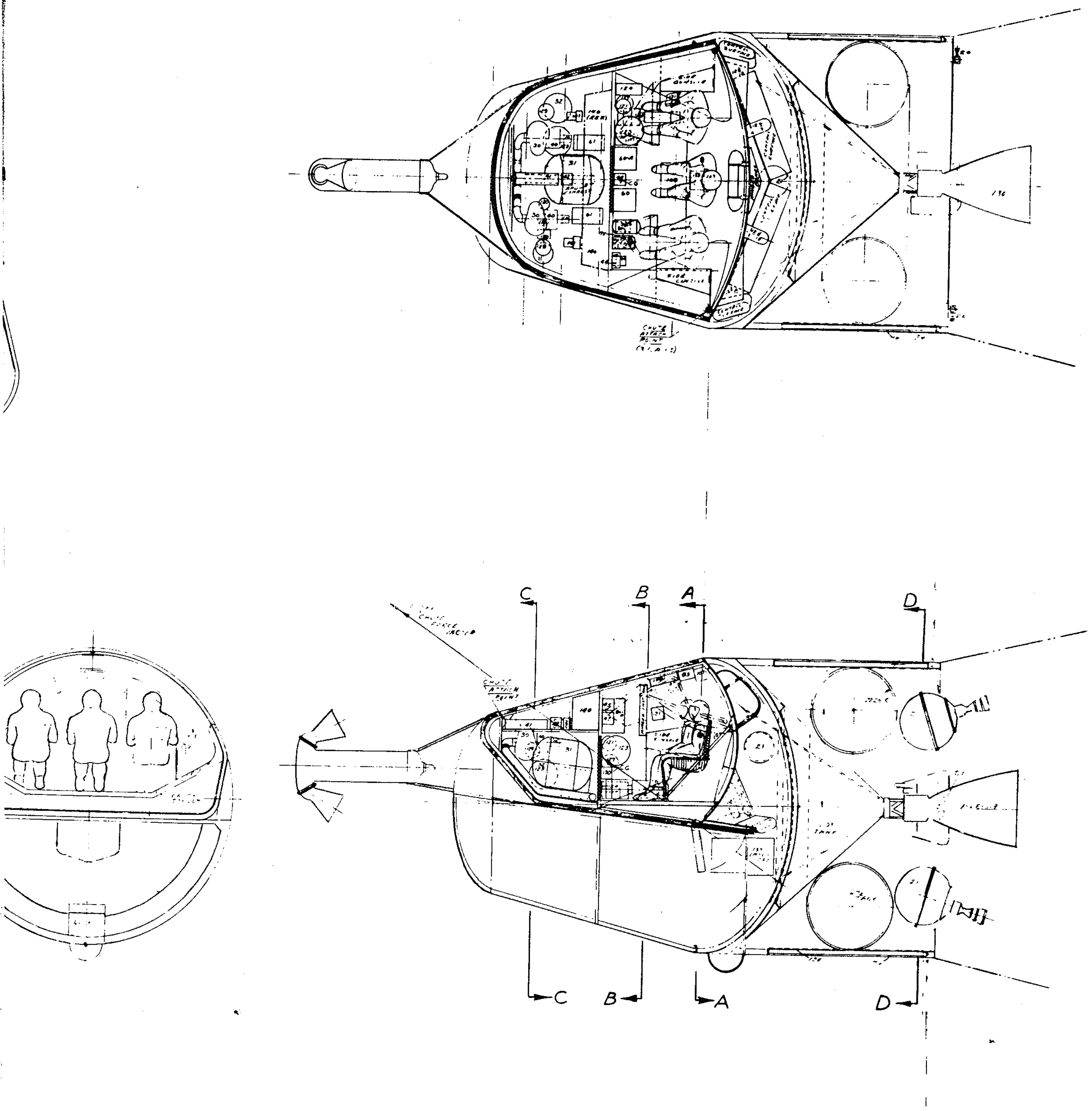



ACCESS
HATCH TO
FWD COMPT
54
127
117


310
300
250
260

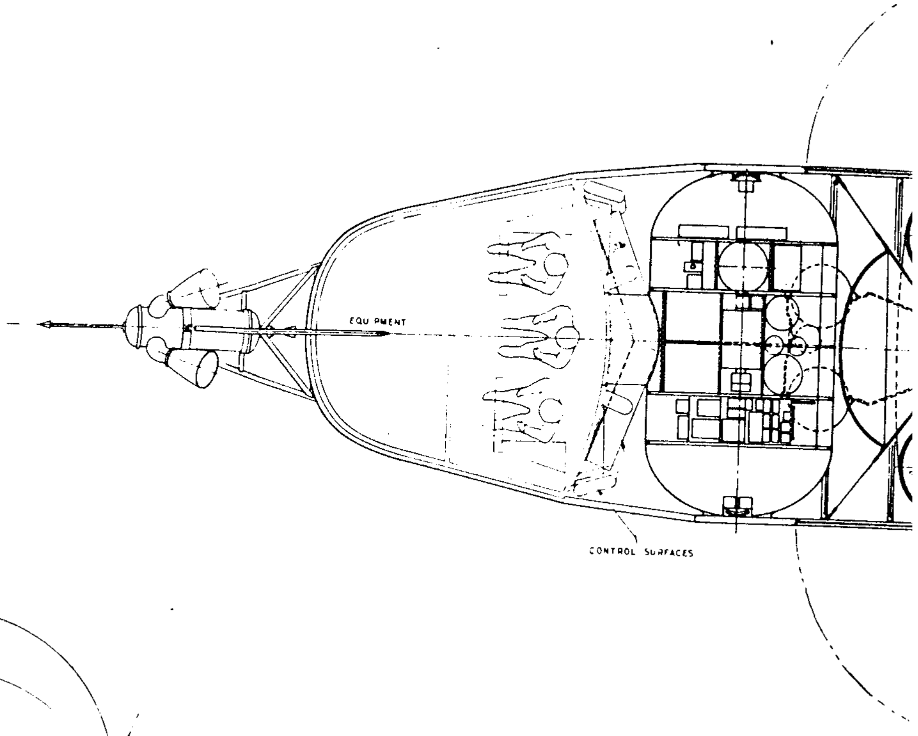
EXTERNAL LINES

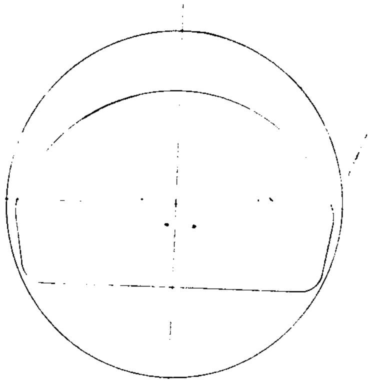
FOLDOUT FRAME


SIDE CONSOLE
CHUTE ATTACH POINT
C
B
A
D
C
B
A
D
SIDE CONSOLE
CHUTE ATTACH POINT
130

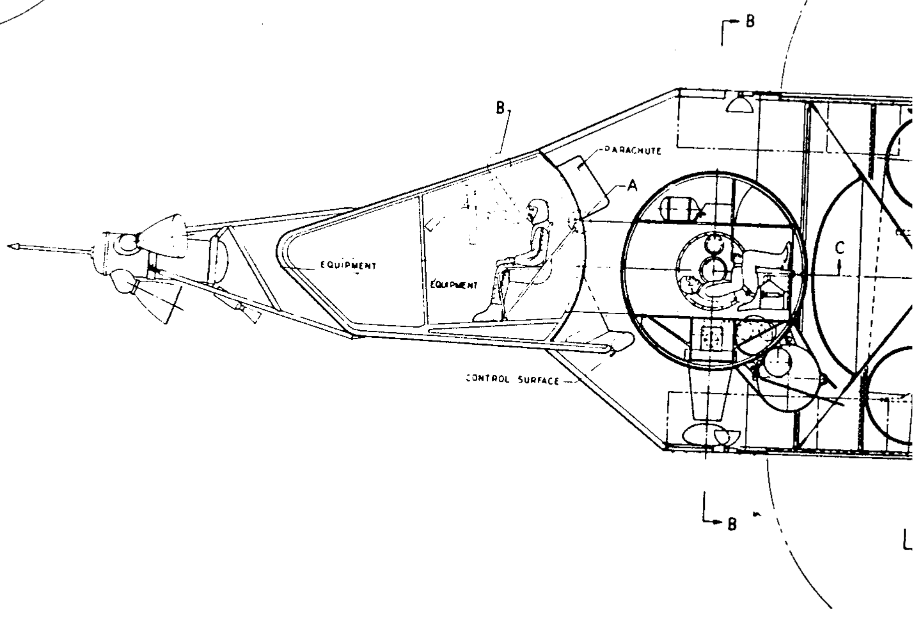
FOLDOUT FRAME 2

FIG. VII-10. L-1 LIFTING BODY, S


EQUIPMENT
CONTROL SURFACES



FRONT VIEW


B
B
PARACHUTE
A
EQUIPMENT
EQUIPMENT
C
CONTROL SURFACE
B
L

FOLDOUT FRAME 1

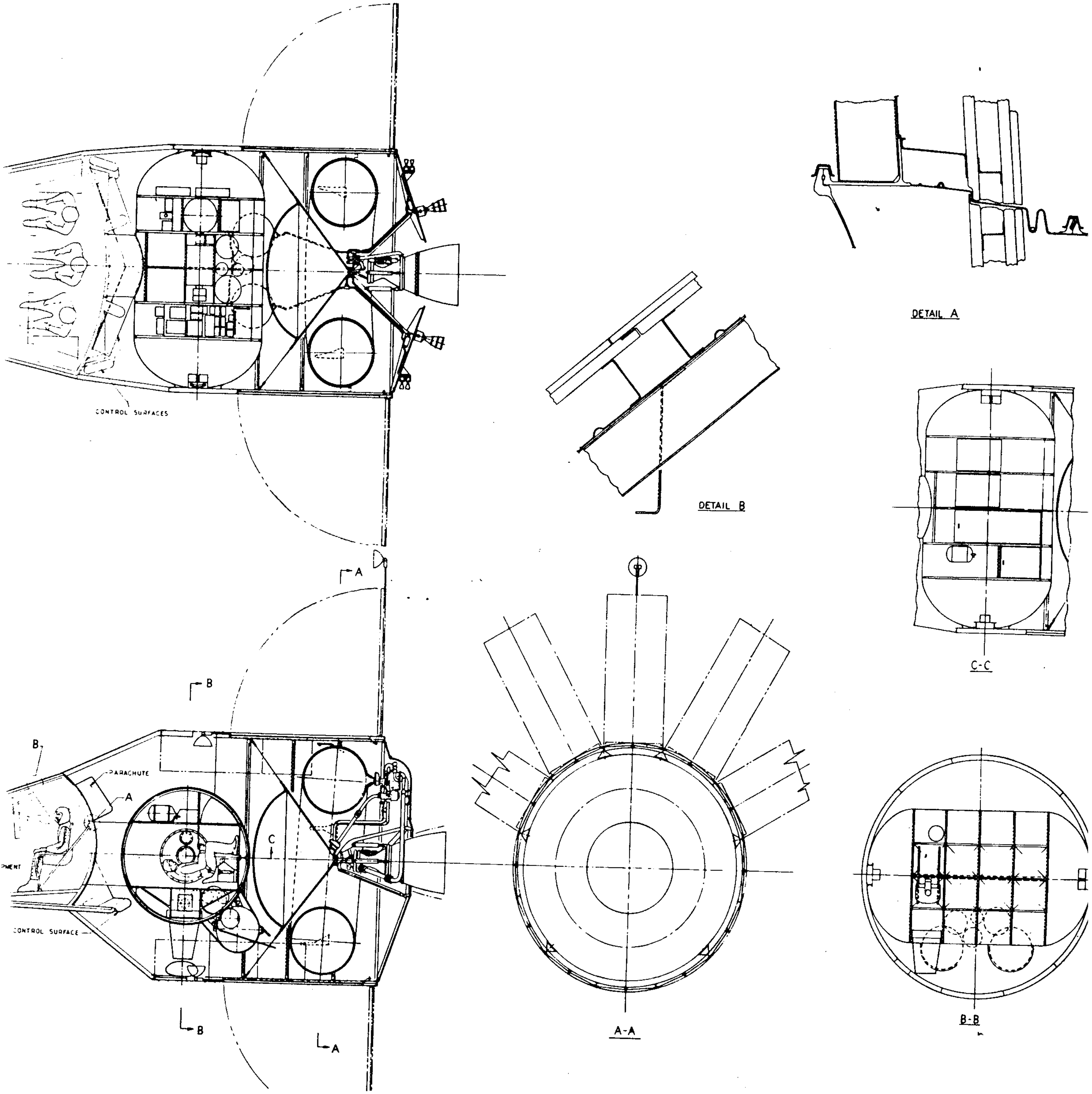
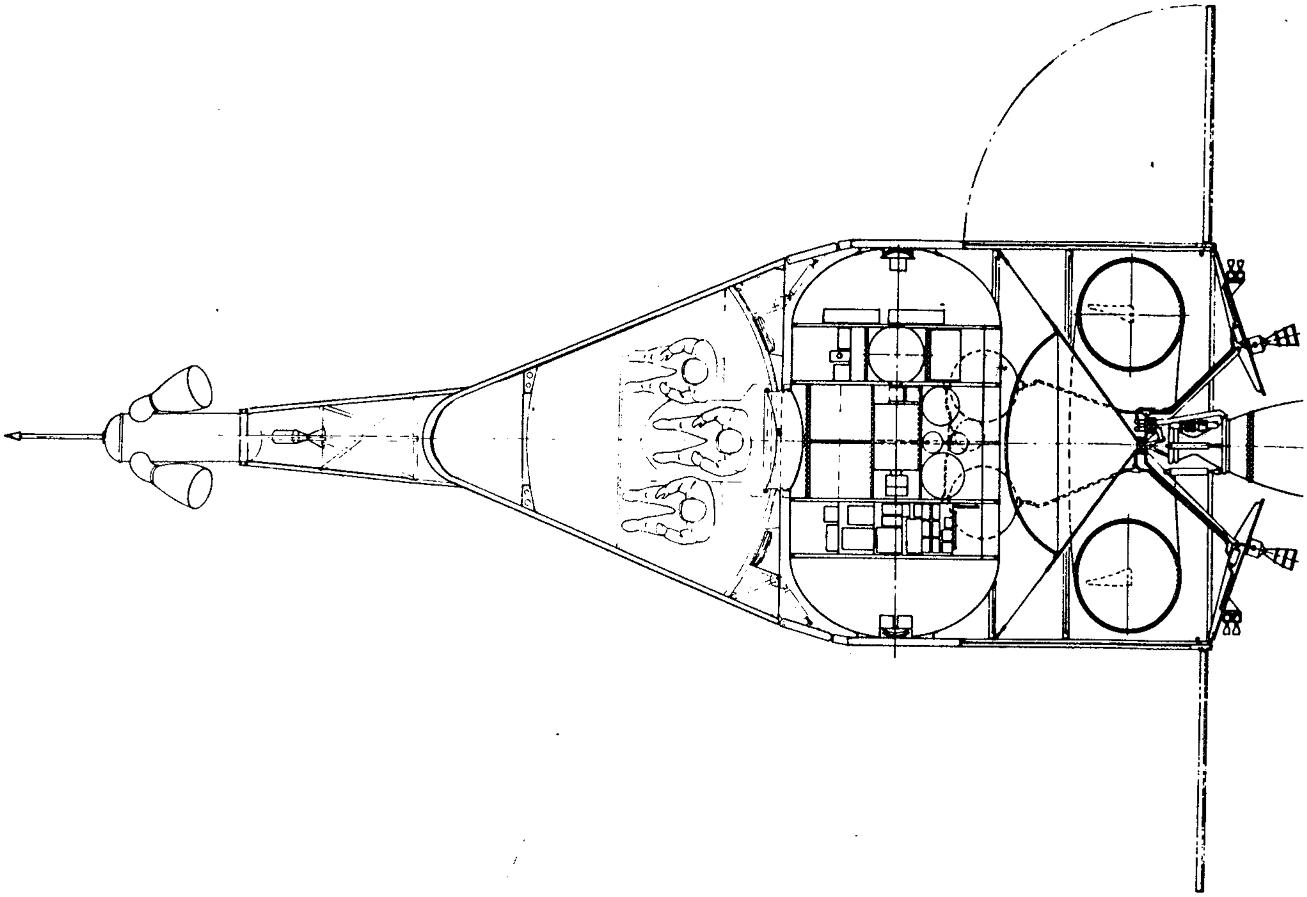
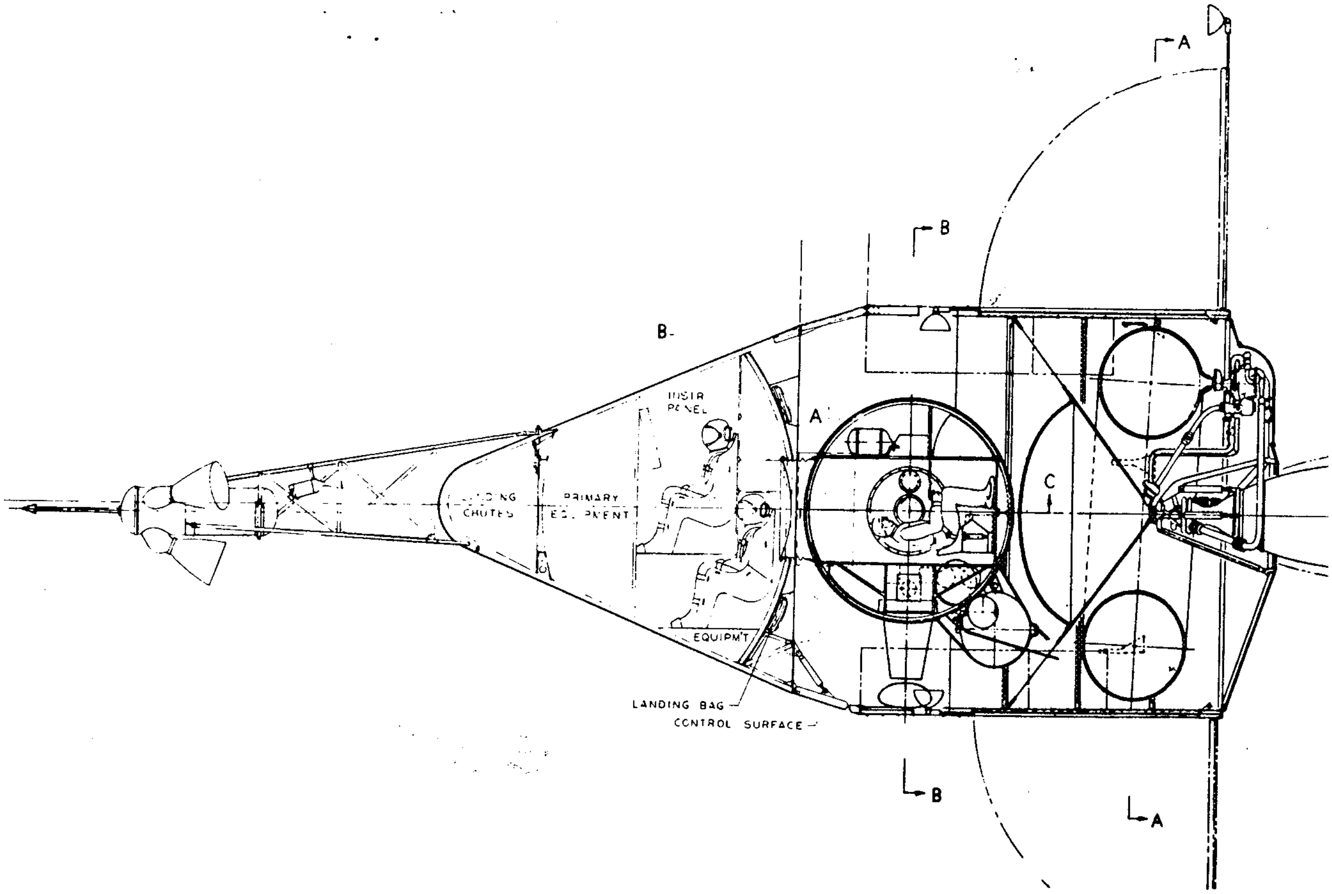

CONTROL SURFACES
DETAIL B
DETAIL A
C-C
A
B
B
PARACHUTE
A
C
PMENT
CONTROL SURFACE
B
A
A-A
B-B

FIG. VII-11. L-1 LIFTING BODY, T

FOLDOUT FRAME 2




A
A
B
B-
B
A
C
INSTR PANEL
LANDING CHUTES
PRIMARY EQUIPMENT
EQUIPM'T
LANDING BAG
CONTROL SURFACE
B
A

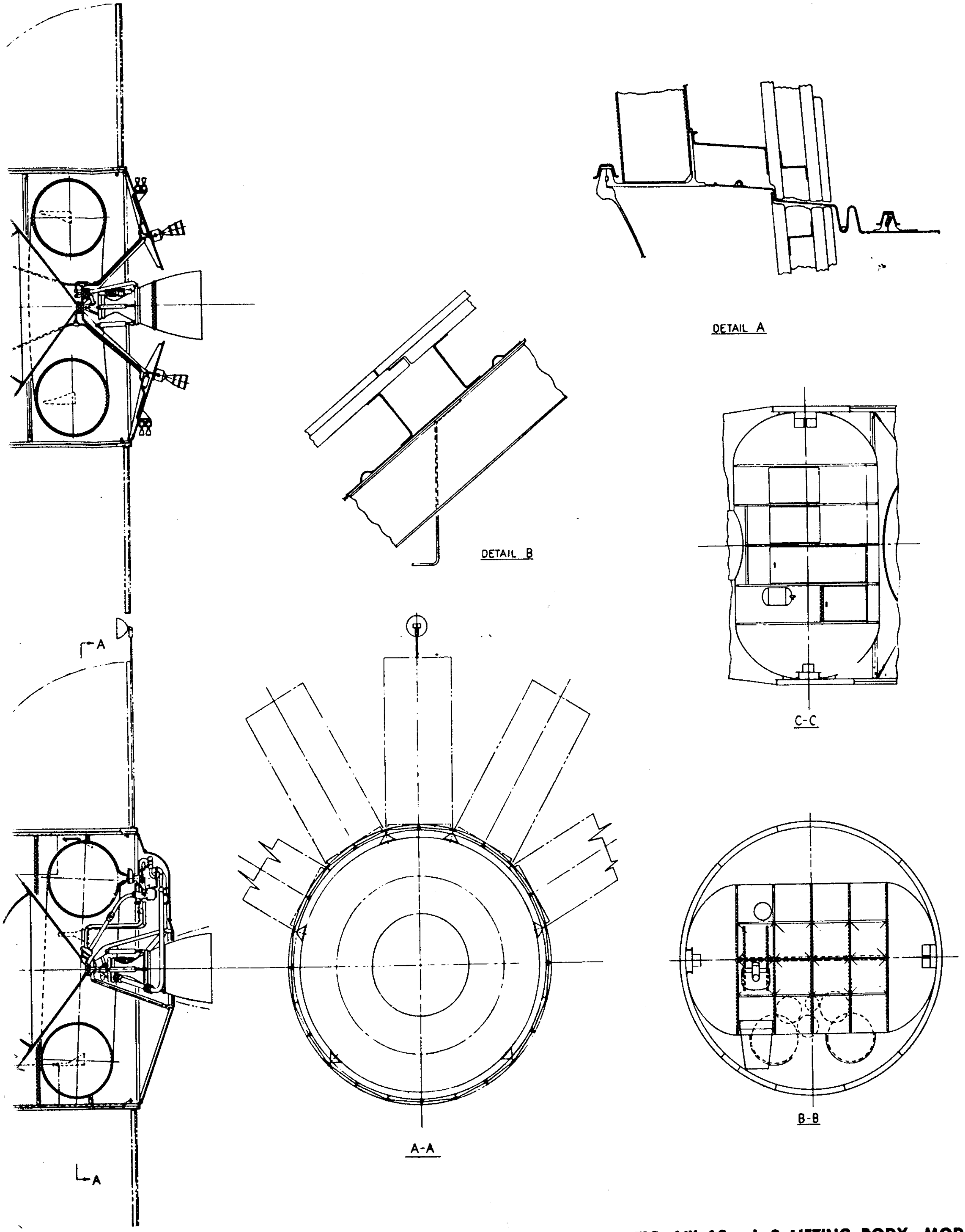
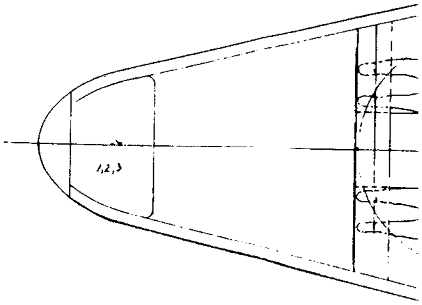
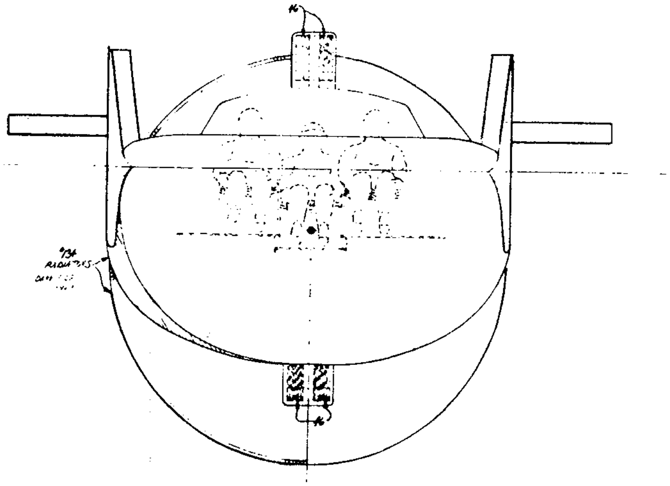
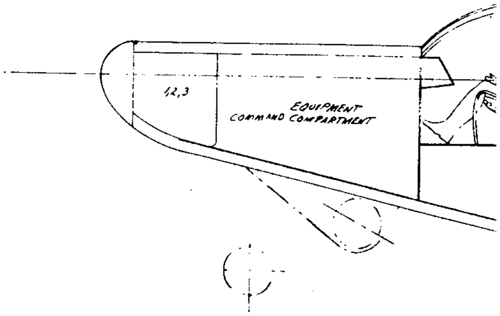

DETAIL A
DETAIL B
C-C
A
A-A
B-B
A

FIG. VII-12. L-8 LIFTING BODY, MODULAR




14
#34
RADIATORS
14


1,2,3
EQUIPMENT
COMMAND COMPARTMENT

FOLDOUT FRAME 1

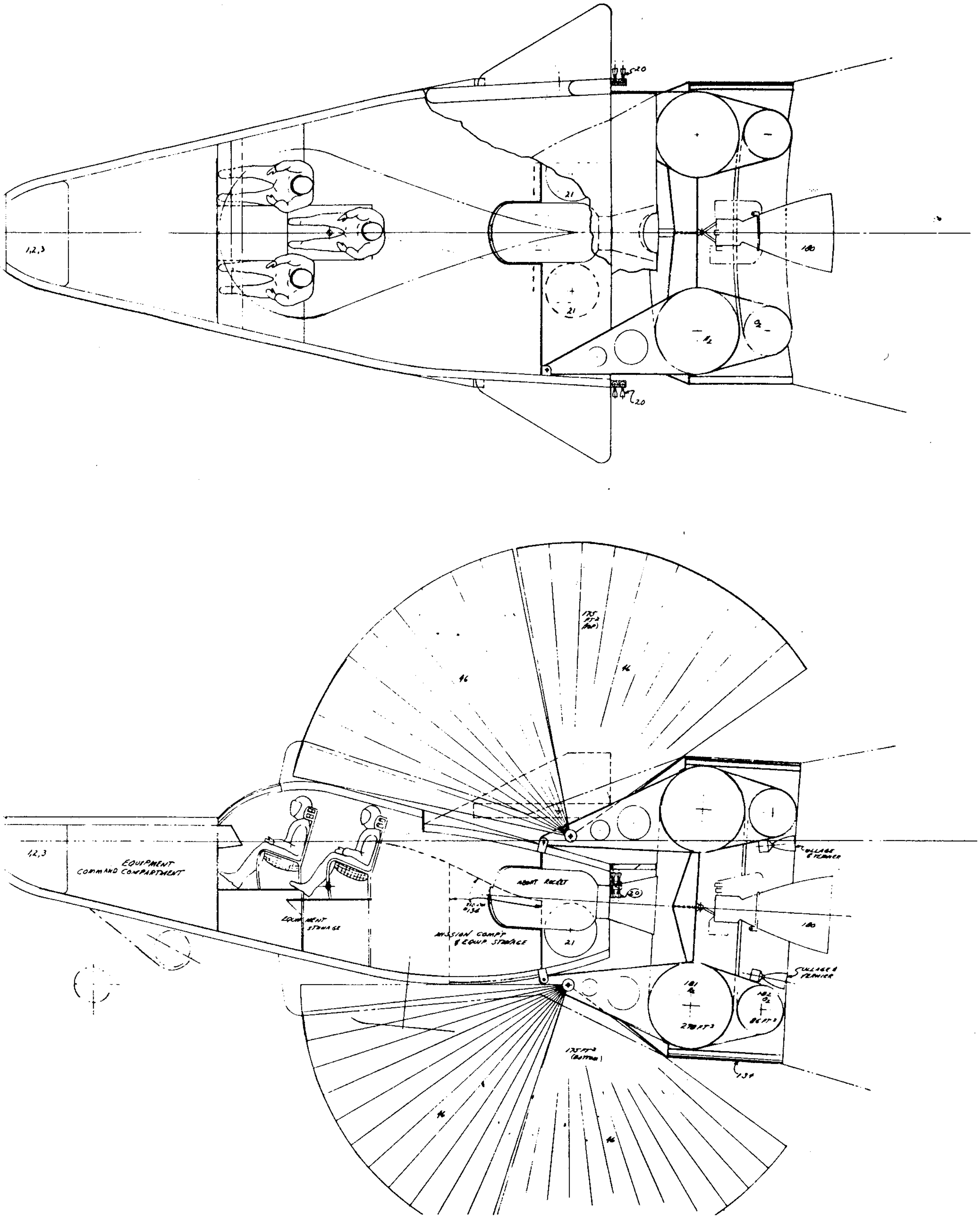
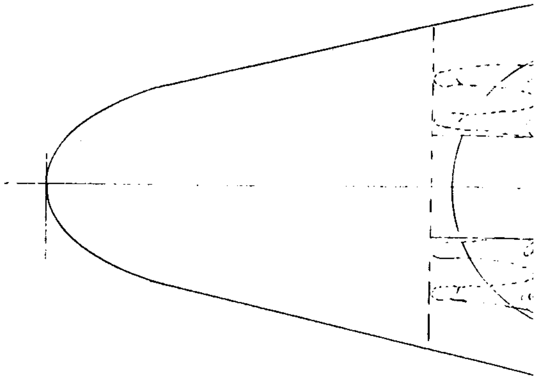
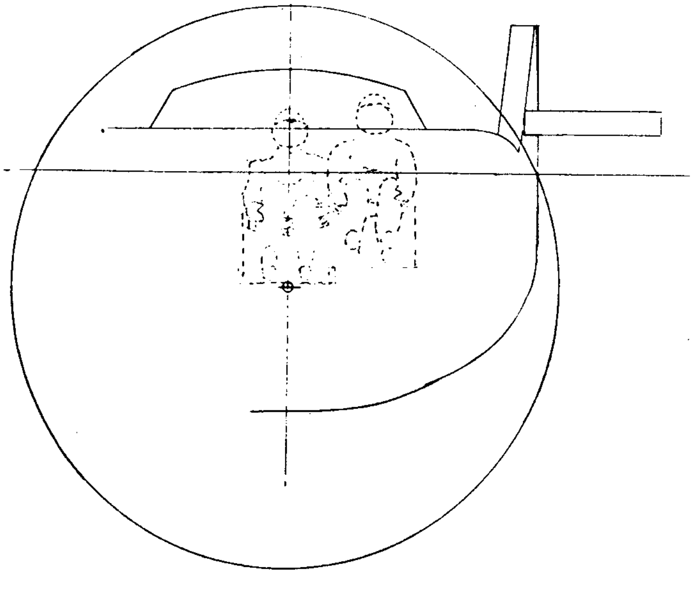
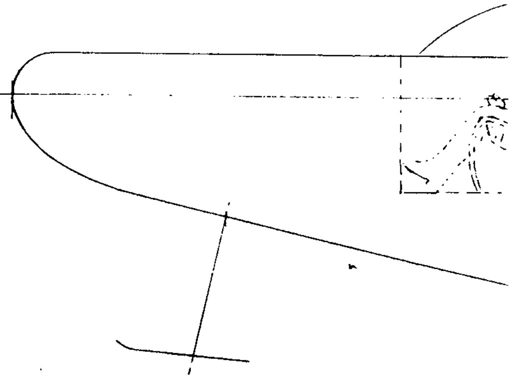

20
1,2,3
21
H2
O2
180
20
1,2,3
EQUIPMENT
COMMAND COMPARTMENT
EQUIPMENT STORAGE
MISSION COMPT & EQUIP STORAGE
ABORT ROCKET
21
175 FT² (TOP)
46
46
ULLAGE & VERNIER
180
ULLAGE & VERNIER
181 H2
278 FT³
182 O2
86 FT³
134
175 FT² (BOTTOM)
46
46

FIG. VII-13. M-2B FLY-IN TYPE BODY, NO. 1







FOLDOUT FRAME

[redacted]

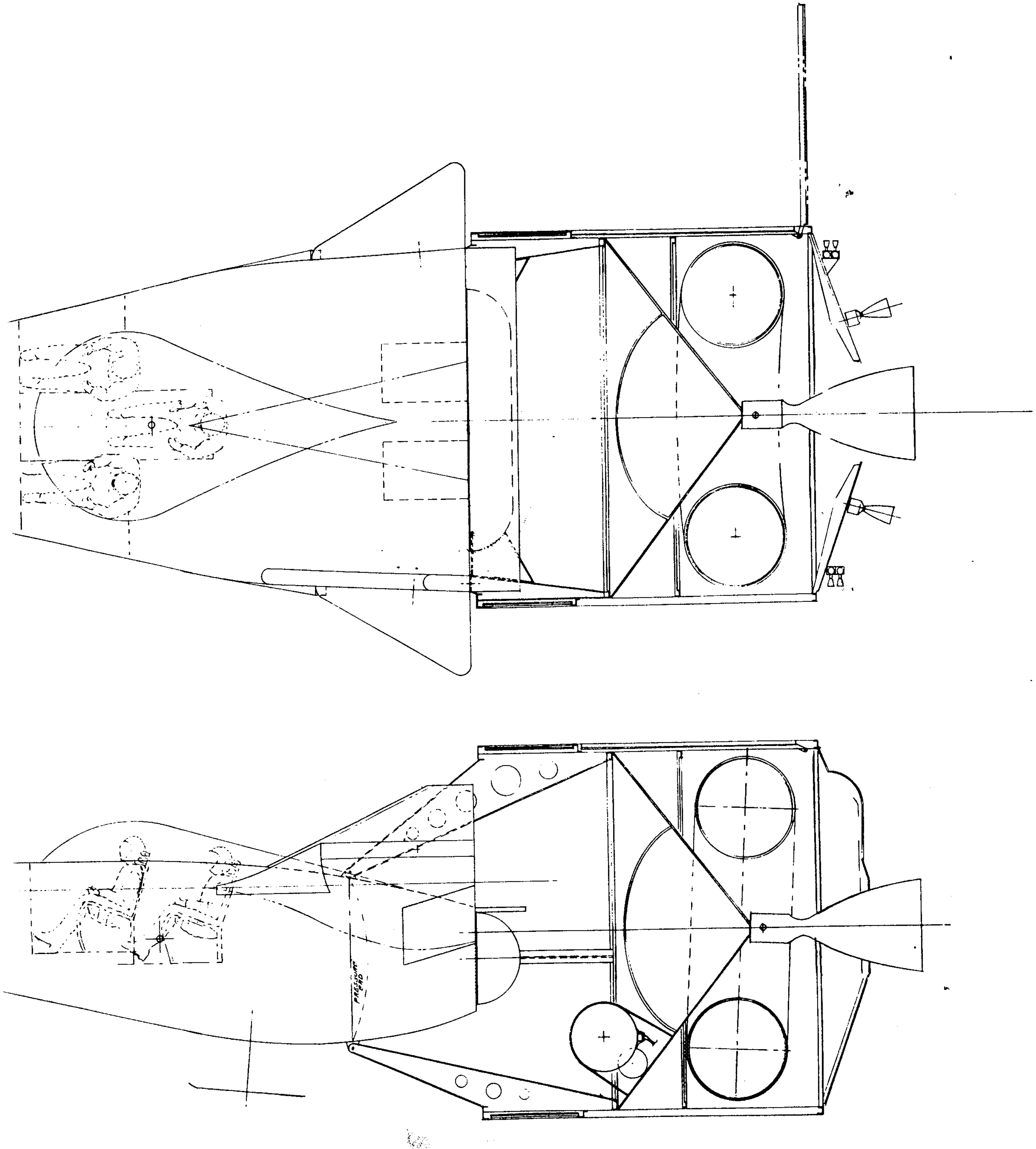
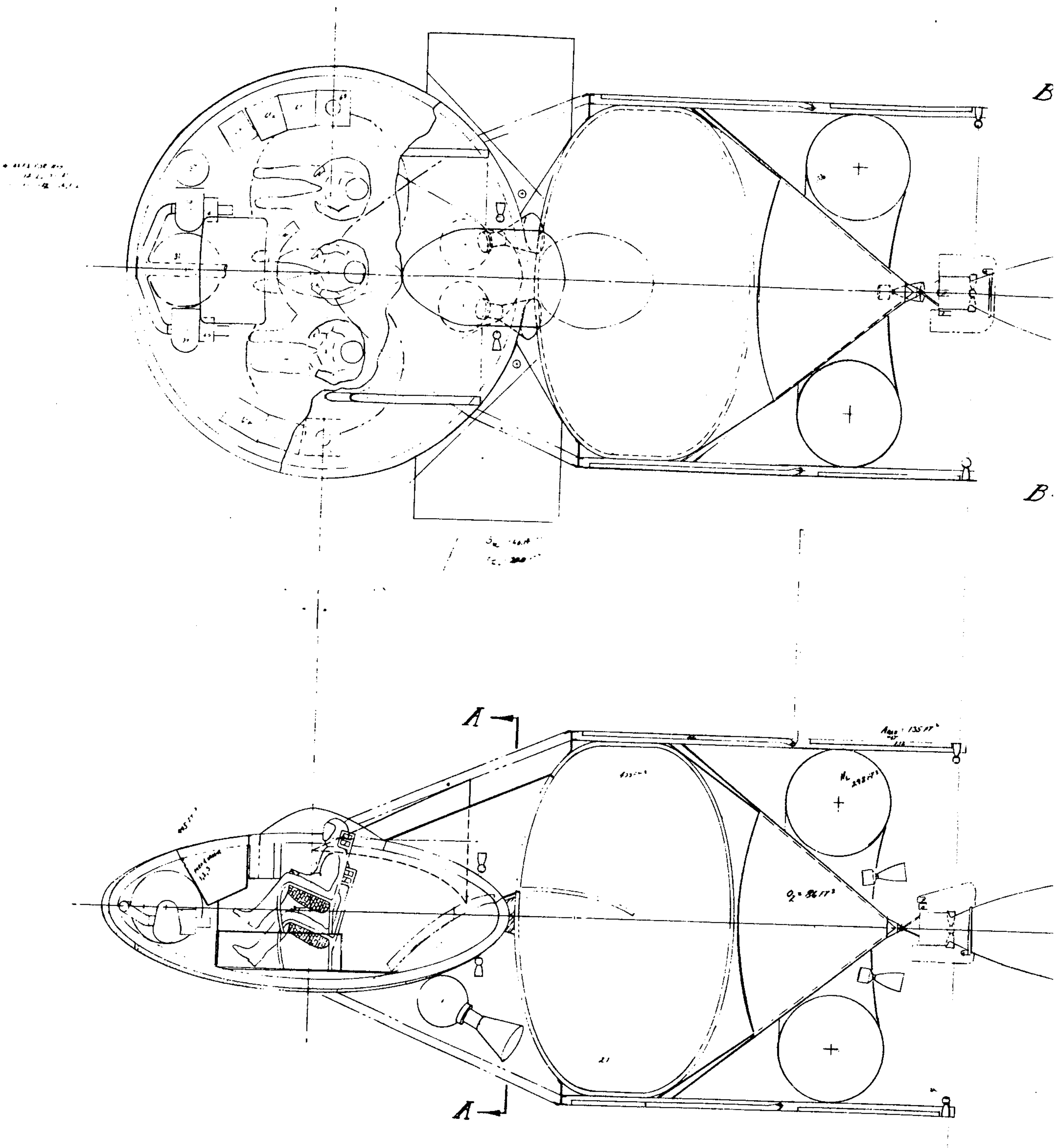

PRESSURE END

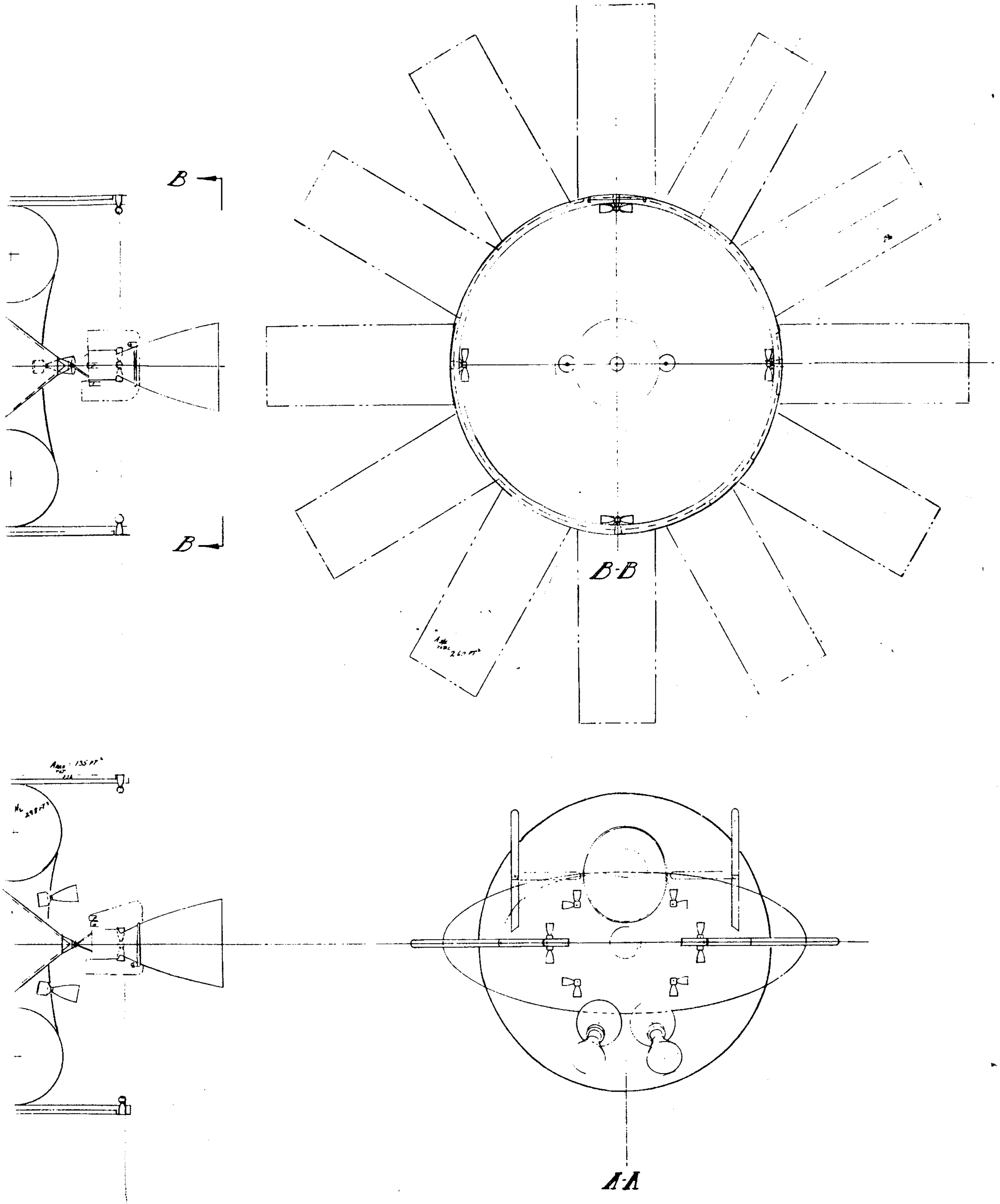
FIG. VII–14. M–2B FLY-IN TYPE BODY, NO. 2

FOLDOUT FRAME 2

[redacted]

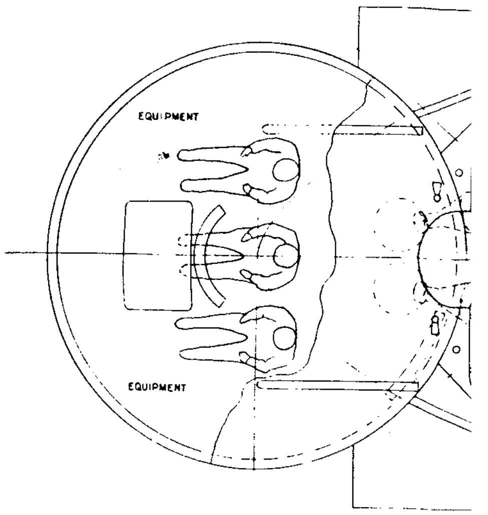

B
B
A
A
O₂ = 86 FT³
A = 135 FT²

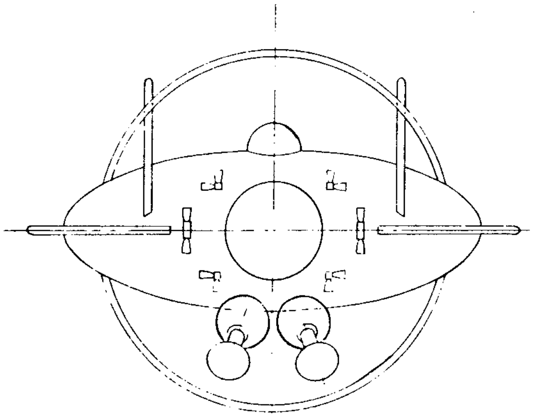
FOLDOUT FRAME 1


B
B
B-B
260 FT²
135 FT²
29 FT²
A-A

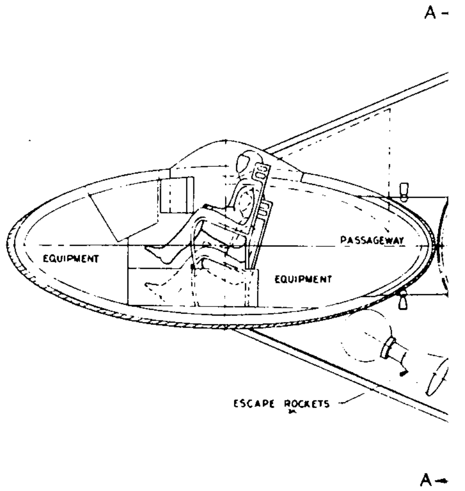
FOLDOUT FRAME 2

FIG. VII-15. LENTICULAR BODY, MODULAR NO. 1


EQUIPMENT
EQUIPMENT



A-A


A -
EQUIPMENT
PASSAGEWAY
EQUIPMENT
ESCAPE ROCKETS
A -

FOLDOUT FRAME 1

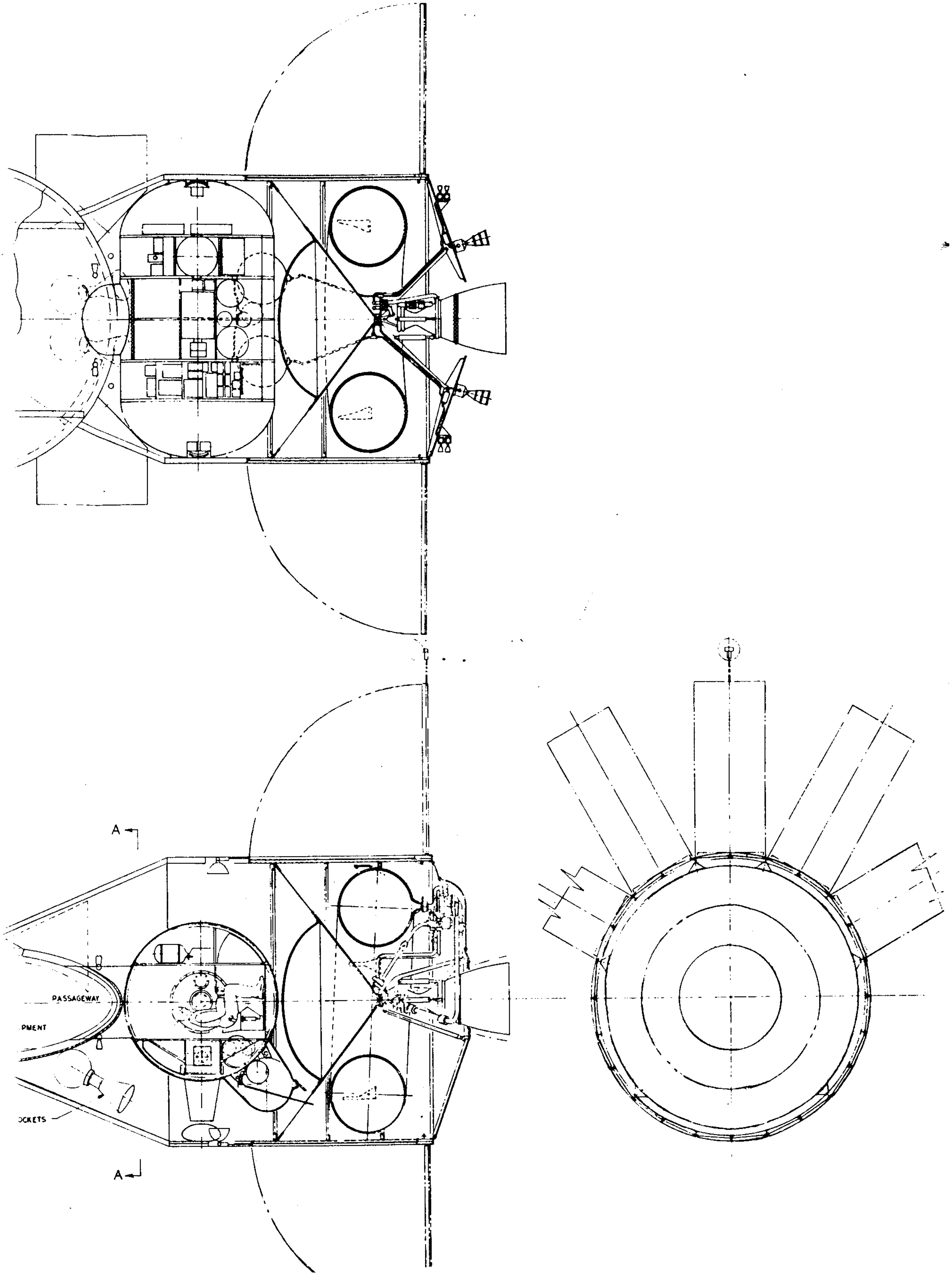

A
PASSAGEWAY
PMENT
CKETS
A

FIG. VII–16. LENTICULAR BODY, MODULAR NO. 2

VIII. LUNAR LANDING STUDIES

A. INTRODUCTION

While the design mission of the Apollo spacecraft is that of manned lunar orbital flight, studies early indicated that with minor modifications to the basic Apollo, the goal of manned lunar landings might also be achieved. Three alternate methods were considered:

(1) Provide Apollo with both landing and takeoff capability ($\Delta V = 17{,}200$ fps).

(2) Provide a separate smaller craft to perform the landing and rendezvous with Apollo in a lunar orbit.

(3) Provide a separate module for landing which can be left behind on the moon and provide the basic Apollo with only takeoff capability ($\Delta V = 8{,}600$ fps).

The weight growth using method (1) was prohibitive since design for this capability in the original vehicle prevented meeting the 15,000-lb weight limit for the circumlunar flights.

The difficulties in launch timing and rendezvous capabilities together with the complexity of the overall system using method (2) led to discarding this approach for Apollo. Such an approach may be applicable to vehicles with larger gross weights.

Consideration of method (3), however, indicated that by providing the basic Apollo with tankage for takeoff and introducing a separate module with all the requirements for the landing and the stay on the moon, the original weight goals for the circumlunar flight could be met. Some studies of the necessary modifications to the basic Apollo spacecraft were carried out to determine their compatibility with some preliminary configurations of the landing module and these are presented herein.

B. LUNAR LANDING MODULE CONFIGURATIONS

Initial studies were directed toward using the Centaur stage as the landing module, but the required structural and propulsion modifications made the landing stage designed specifically for Apollo so much more attractive that this approach was abandoned. Three of the configurations studied are shown in Figs. VIII–1, 2, and 3 which are subsequently described as configurations I, II and III. Weights for the three configurations are shown in Section E. Common characteristics of the three configurations which were adopted to limit the study are as follows:

(1) The ΔV requirement for the landing module vernier system will be 200 fps. This corresponds to the established for the translunar trajectory for Model 410.

(2) The ΔV for the lunar landing is 8,600 fps.

(3) The mission engines should be the LR–115 with 15,600 lb thrust and $I_{sp} = 427$ throttable to one-half thrust. These are the same as for the M–410 except for the throttling feature which Pratt and Whitney states is a modification rather than a developmental change. The provision of a six to one thrust modulation is adequate for a controlled soft lunar landing.

(4) The overall mission time is to be 14 days of which 3 days are to be spent on the moon.

(5) Tankage of the Model 410 Apollo shall be adequate to provide sufficient energy for takeoff from the lunar surface to the requisite transearth return trajectory ($\Delta V = 8{,}600$ fps). In this manner, only one propulsion system will be developed for Apollo which will have cislunar and lunar orbit mission capabilities at lighter gross weights by means of propellant off-landing.

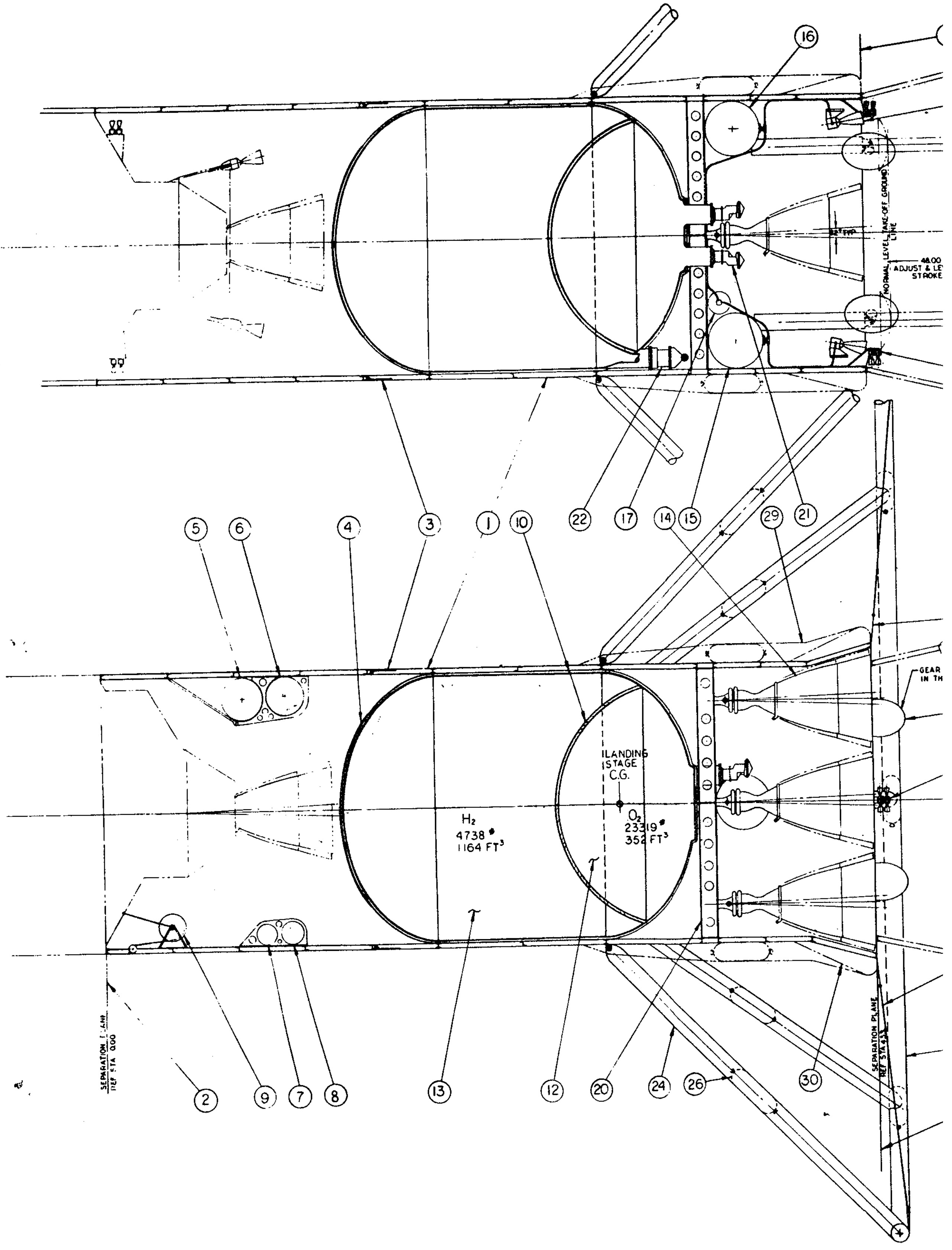
(6) Provision of a landing system to absorb the lunar landing impact and to support the vehicle on the surface of the moon.

(7) Use of the basic Apollo spacecraft as the earth return vehicle with as few modifications as possible.

A description of each of the configurations together with their advantages and disadvantages follows:

1. CONFIGURATION I

This configuration (shown in Fig. VIII–1) has the same overall diameter of 154-in. as the Apollo spacecraft. The propulsion is furnished by the



SEPARATION PLANE
REF STA 0.00
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④
⑤
⑥
⑦
⑧
⑨
⑩
⑫
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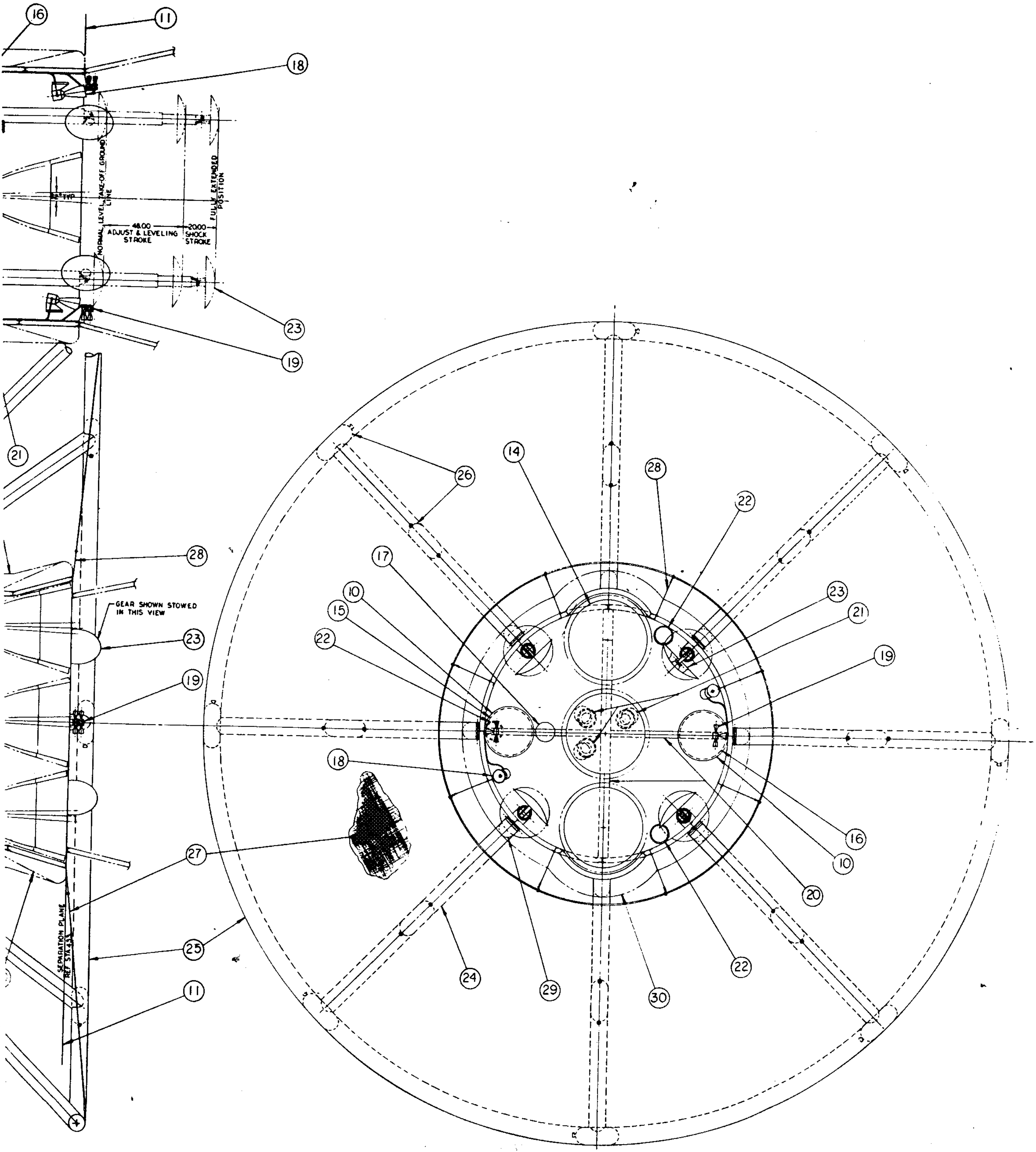
㉚
H₂
4738#
1164 FT³
LANDING
STAGE
C.G.
O₂
23319#
352 FT³
22° TYP
NORMAL LEVEL TAKE-OFF GROUND LINE
48.00
ADJUST & LE
STROKE
GEAR
IN TH
SEPARATION PLANE
REF STA 433

FOLDOUT FRAME 1


16
11
18
NORMAL LEVEL TAKE-OFF GROUND LINE
FULLY EXTENDED POSITION
12° TYP
48.00 ADJUST & LEVELING STROKE
20.00 SHOCK STROKE
23
19
21
28
GEAR SHOWN STOWED IN THIS VIEW
23
19
27
SEPARATION PLANE REF STA 433
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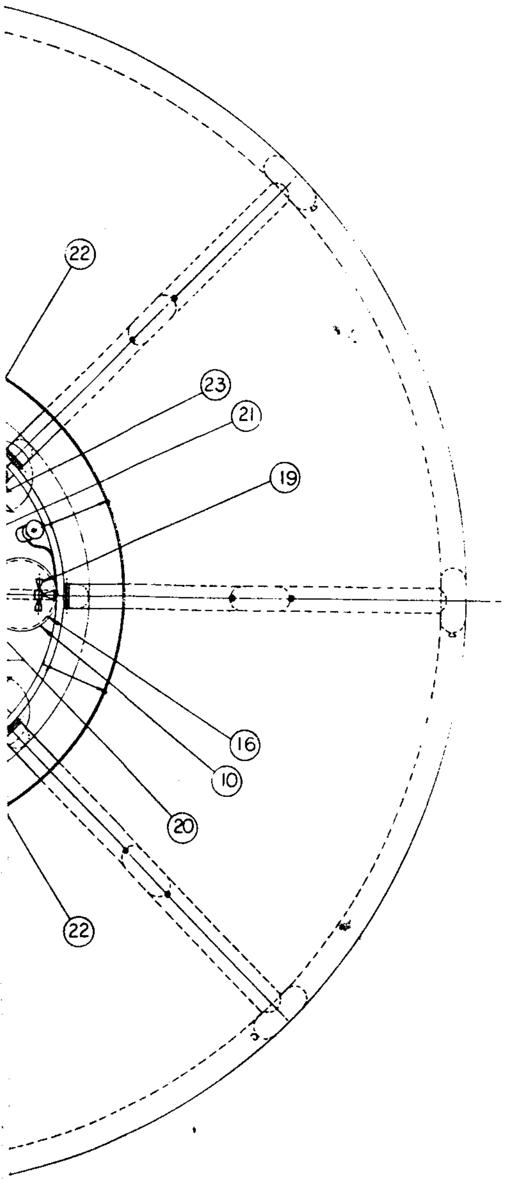
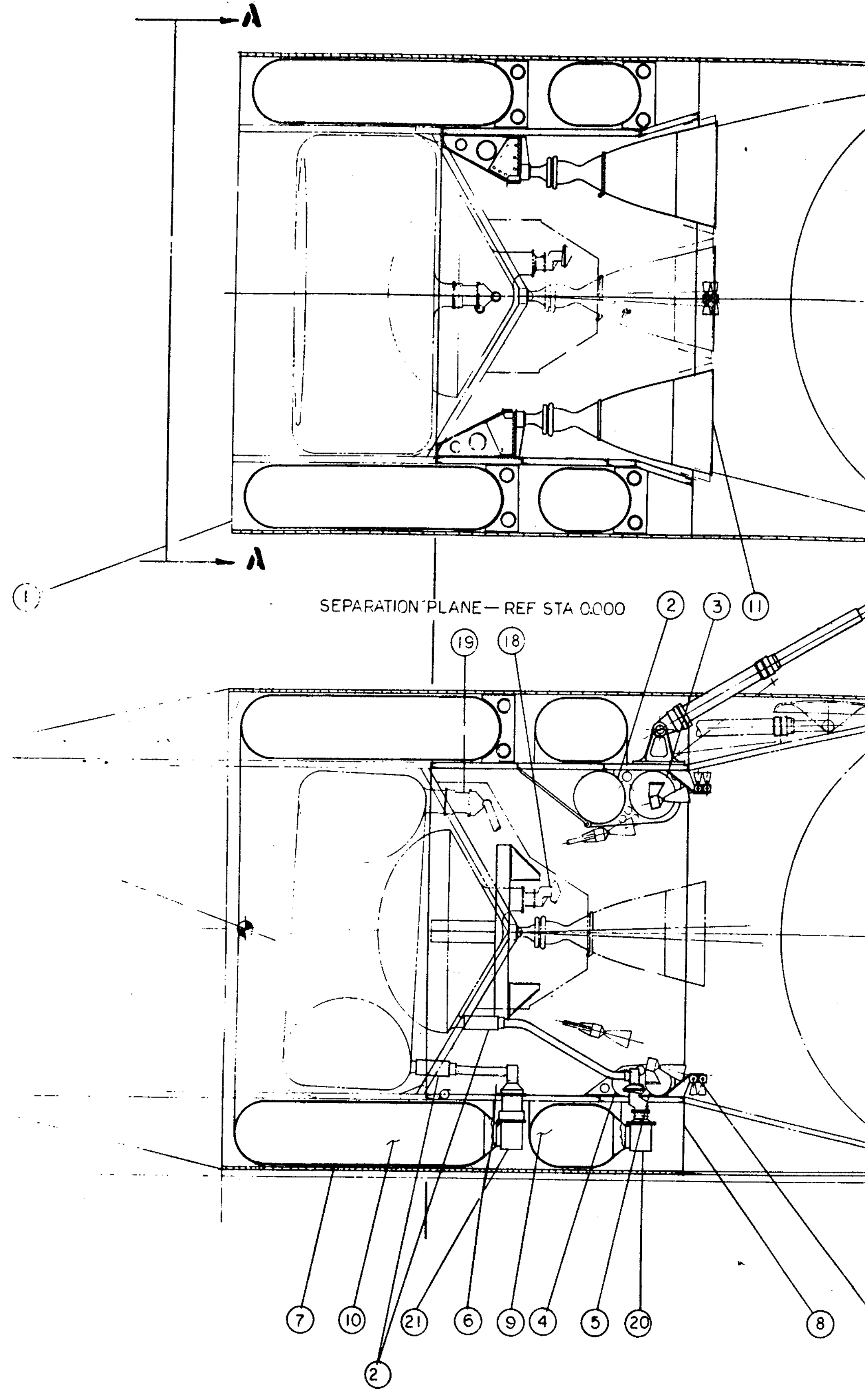
30

ITEM	NOMENCLATURE	ZONE
1	LUNAR LANDING STAGE	6C
2	LUNAR TAKE-OFF STAGE SEPARATION PLANE	7A
3	BLOWOUT PORTS	6C
4	TANK-ABLATIVE COATING	6C
5	MISSION MODULE REPLENISHMENT OXYGEN	6C
6	" " " NITROGEN	6C
7	ADDITIONAL FUEL-CELL OXYGEN TANKAGE	6A
8	" " " HYDROGEN TANKAGE	6A
9	SPACESUIT ELECTRICAL CABLE PAYOUT REEL	6A
10	TANKAGE INSULATION	6C
11	BOOSTER STAGE SEPARATION PLANE	5A
12	MIDCOURSE & LUNAR LANDING OXYGEN TANK	6A
13	" " " " HYDROGEN TANK	6A
14	" " " " ENGINES (3)	5C
15	VERNIER & ATTITUDE CONTROL FUEL TANK	5C
16	" " " " OXIDIZER TANK	5D
17	" " " " NITROGEN PRESSURE TANK	5C
18	VERNIER ENGINES (2)	4D
19	ATTITUDE CONTROL NOZZLES (12)	4C
20	MAIN ENGINE MOUNTING STRUCTURE	6A
21	" " OXYGEN PUMPS (3)	5C
22	" " HYDROGEN PUMPS (3)	6C
23	LUNAR LANDING SHOCK STRUTS (TELESCOPING)	4C
24	LANDING-STABILIZATION FOAM-INFLATED STRUTS	5A
25	" " " " BASE RING	4A
26	INFLATION-FOAM LIQUID STOWAGE & INJECTION BOTTLES	5A
27	ANNULAR BASE-WEB	4A
28	VEHICLE BASE-STABILIZATION "GUYS"	4B
29	STABILIZATION-STRUTS STOWAGE FAIRINGS	5C
30	BASE-RING & WEB STOWAGE FAIRING	5A

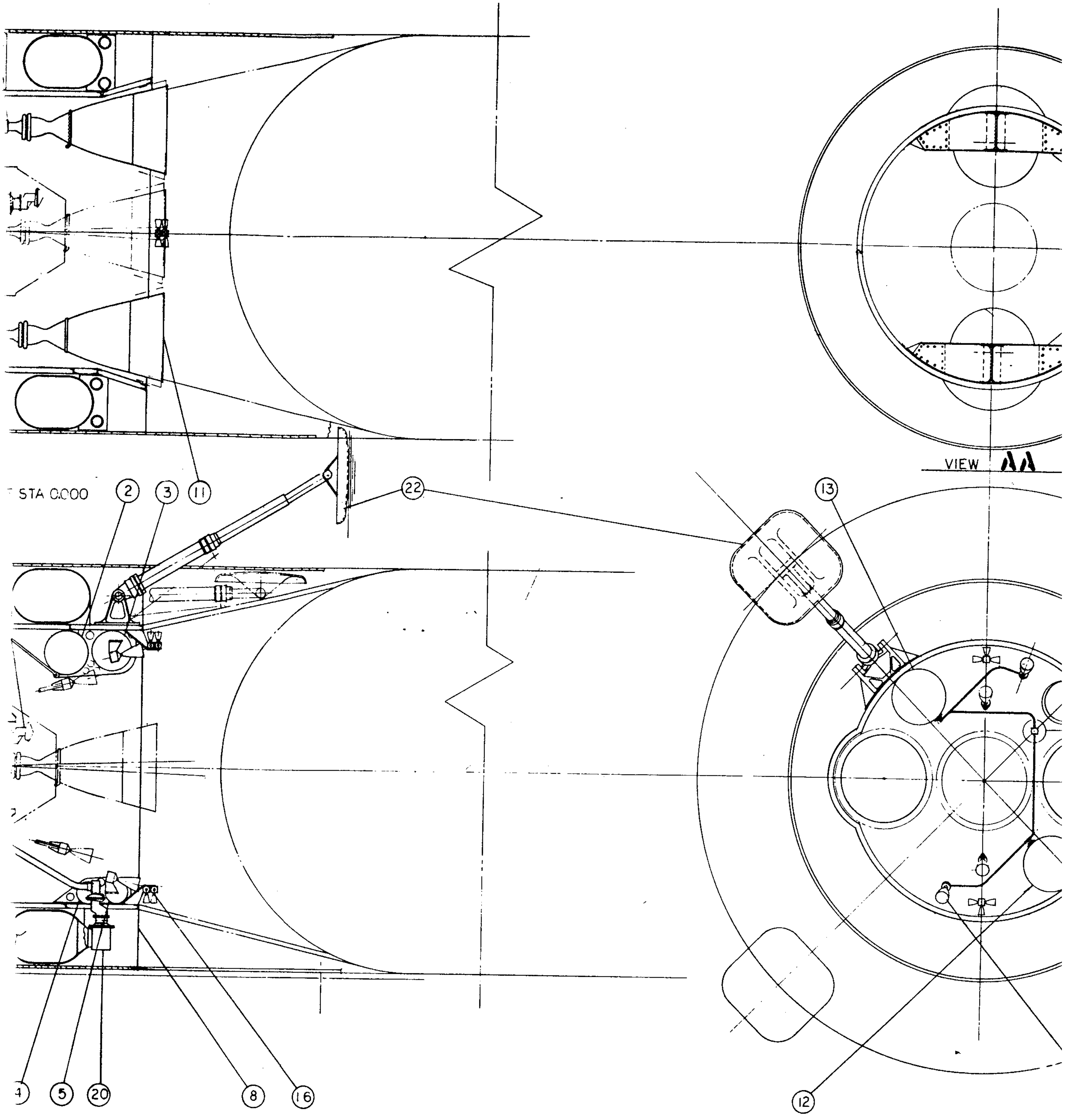


FOLDOUT FRAME 3

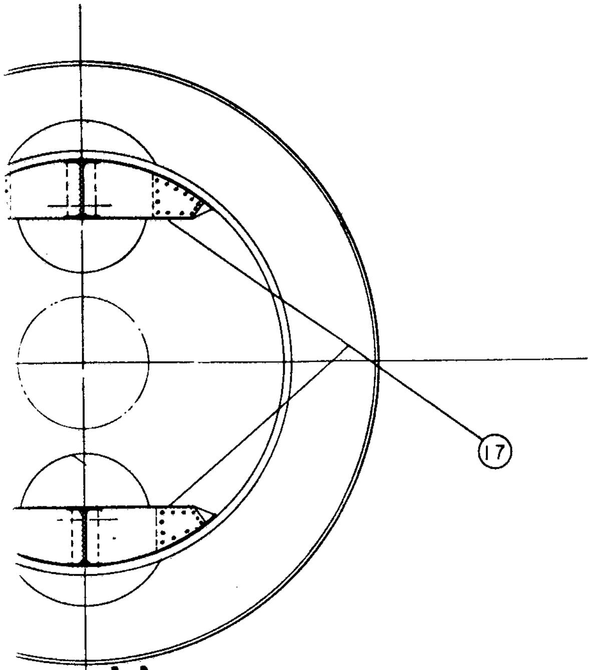
FIG. VIII-1. LUNAR LANDING MODULE I


A
A
1
SEPARATION PLANE — REF STA 0.000
2
3
11
19
18
7
10
21
6
9
4
5
20
8
2

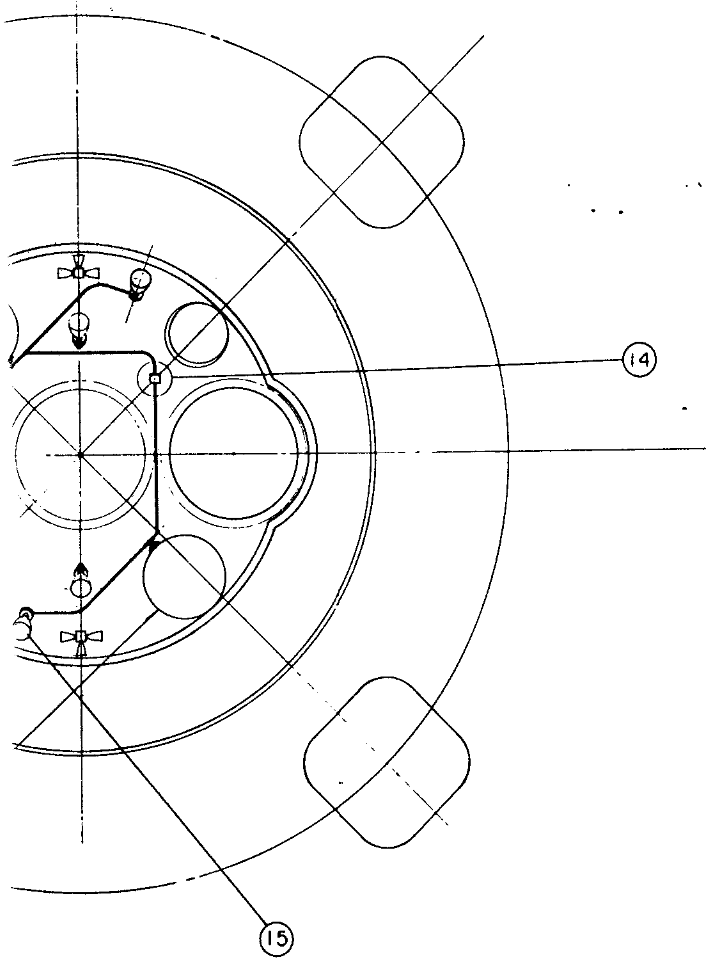
FOLDOUT FRAME 1


STA 0.000
2
3
11
22
VIEW AA
13
4
5
20
8
16
12

FOLDOUT FRAME 2



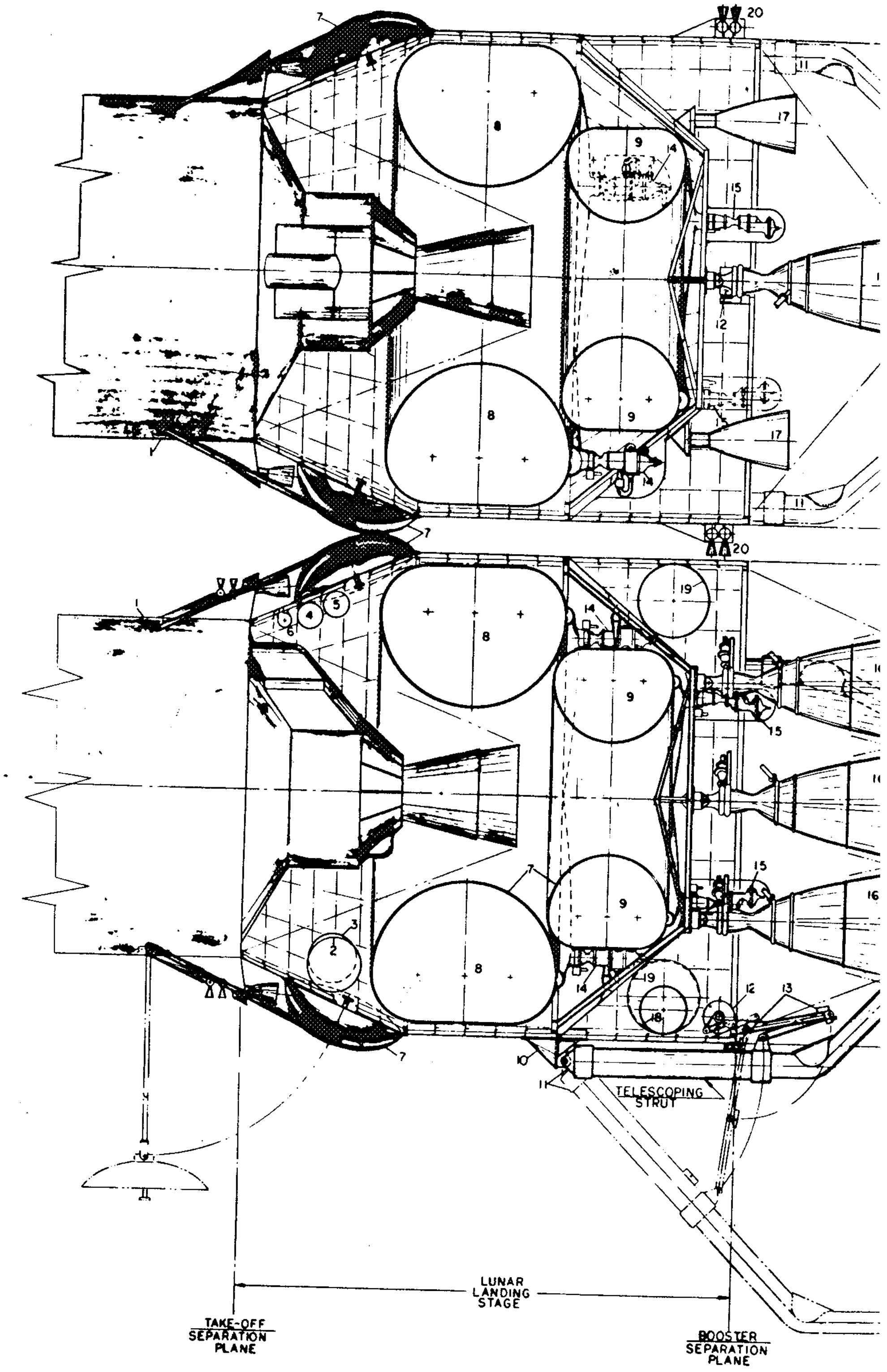
VIEW AA



ITEM	NOMENCLATURE
1	LUNAR LANDING STAGE
2	MISSION MODULE REPLENISHMENT OXYGEN
3	" " " NITROGEN
4	ADDITIONAL FUEL-CELL OXYGEN TANKAGE
5	" " " HYDROGEN TANKAGE
6	SPACESUIT ELECTRICAL CABLE PAYOUT REEL
7	TANKAGE INSULATION
8	BOOSTER STAGE SEPARATION PLANE
9	MIDCOURSE & LUNAR LANDING OXYGEN TANK
10	" " " " " HYDROGEN TANK
11	" " " " " ENGINES (3)
12	VERNIER & ATTITUDE CONTROL FUEL TANK
13	" " " " OXIDIZER TANK
14	" " " " NITROGEN PRESSURE
15	VERNIER ENGINES (2)
16	ATTITUDE CONTROL NOZZLES (12)
17	MAIN ENGINE MOUNTING STRUCTURE
18	" " OXYGEN PUMPS (3)
19	" " HYDROGEN PUMPS (3)
20	OXYGEN TRANSFER PUMP (1)
21	HYDROGEN TRANSFER PUMP (1)
22	LUNAR LANDING SHOCK STRUTS
23	FUEL & OXIDIZER AUTOMATIC DISCONNECT

FIG. VIII-2. LUNAR LANDING MODULE II

FOLDOUT FRAME 3


20
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4
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3
2
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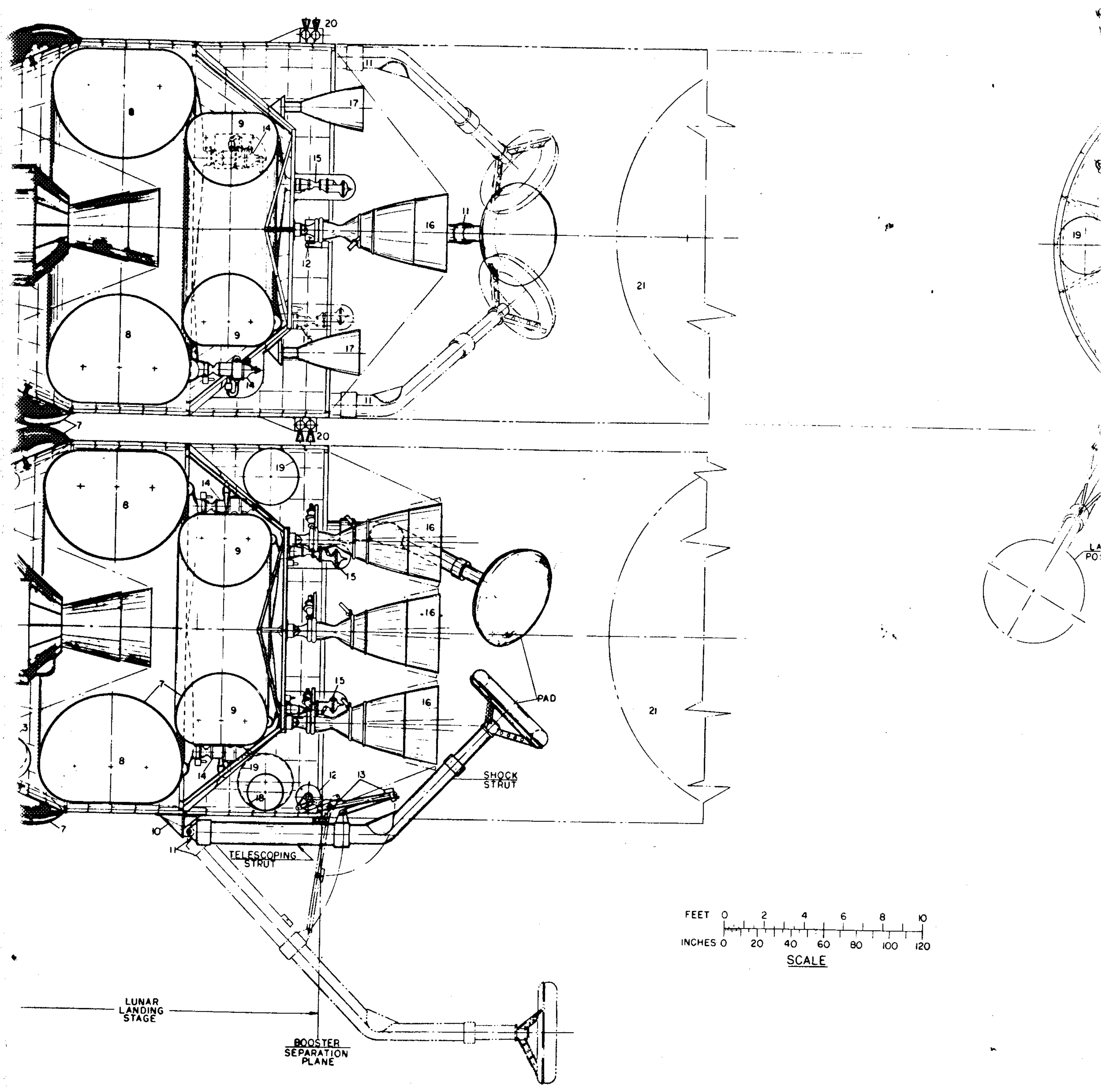
9
15
16
7
9
8
14
19
18
12
13
10
11
TELESCOPING STRUT
LUNAR LANDING STAGE
TAKE-OFF SEPARATION PLANE
BOOSTER SEPARATION PLANE

FOLDOUT FRAME 1

[redacted]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PAD
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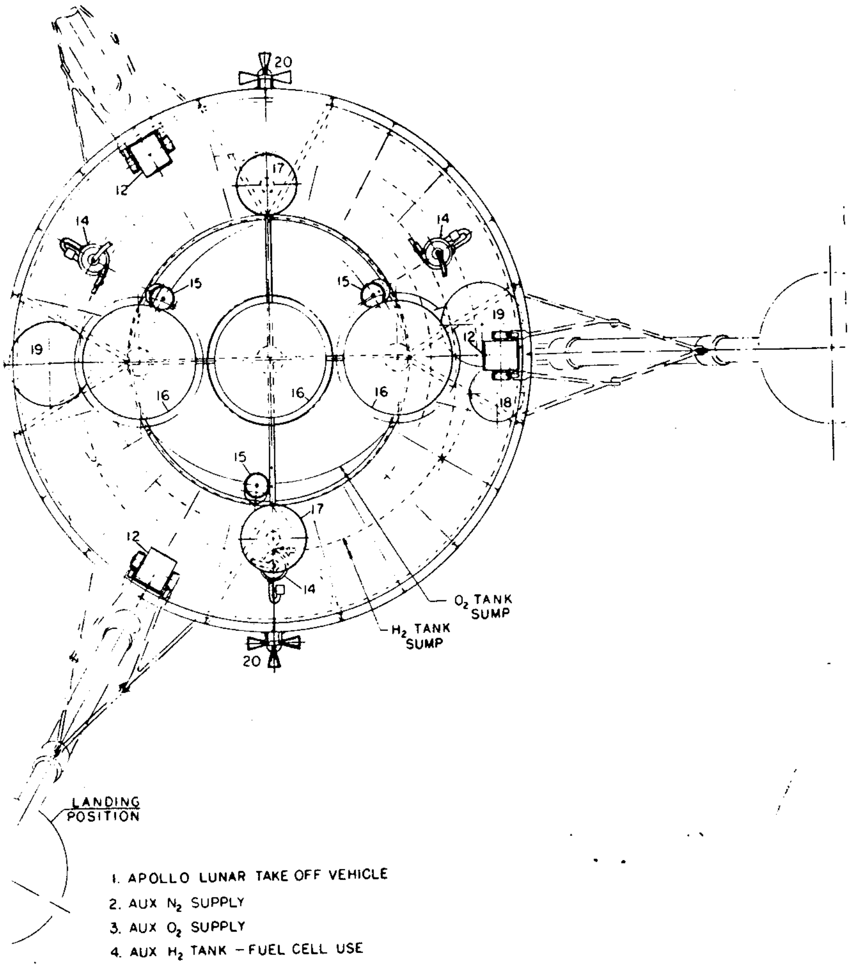
19
18
12
13
SHOCK STRUT
7
10
11
TELESCOPING STRUT
LA
PO
FEET 0 2 4 6 8 10
INCHES 0 20 40 60 80 100 120
SCALE
LUNAR LANDING STAGE
BOOSTER SEPARATION PLANE

FOLDOUT FRAME 2

[redacted]


20
17
12
14
14
15
15
19
12
19
16
16
16
18
15
17
12
14
O_2 TANK SUMP
H_2 TANK SUMP
20
LANDING POSITION

1. APOLLO LUNAR TAKE OFF VEHICLE
2. AUX N_2 SUPPLY
3. AUX O_2 SUPPLY
4. AUX H_2 TANK – FUEL CELL USE
5. AUX O_2 TANK – FUEL CELL USE
6. REEL - ELECTRICAL CONNECT.
7. ANTENNA COVER - JETTISONABLE
8. TOROIDAL H_2 TANK
9. TOROIDAL O_2 TANK
10. LANDING GEAR ATTACH. F'T'G.
11. LANDING GEAR
12. LD. GEAR ACTUATOR
13. LD. GEAR LOCKS
14. H_2 PUMP
15. O_2 PUMP
16. MIDCOURSE & LUNAR LANDING ENGINES
17. VERNIER ENGINES
18. H_e BOTTLE
19. VERNIER & CONTROL ROCKET PROPELLANT
20. ATTITUDE CONTROL ROCKETS
21. LAUNCH VEHICLE

FIG. VIII–3. LUNAR LANDING MODULE III

FOLDOUT FRAME 3

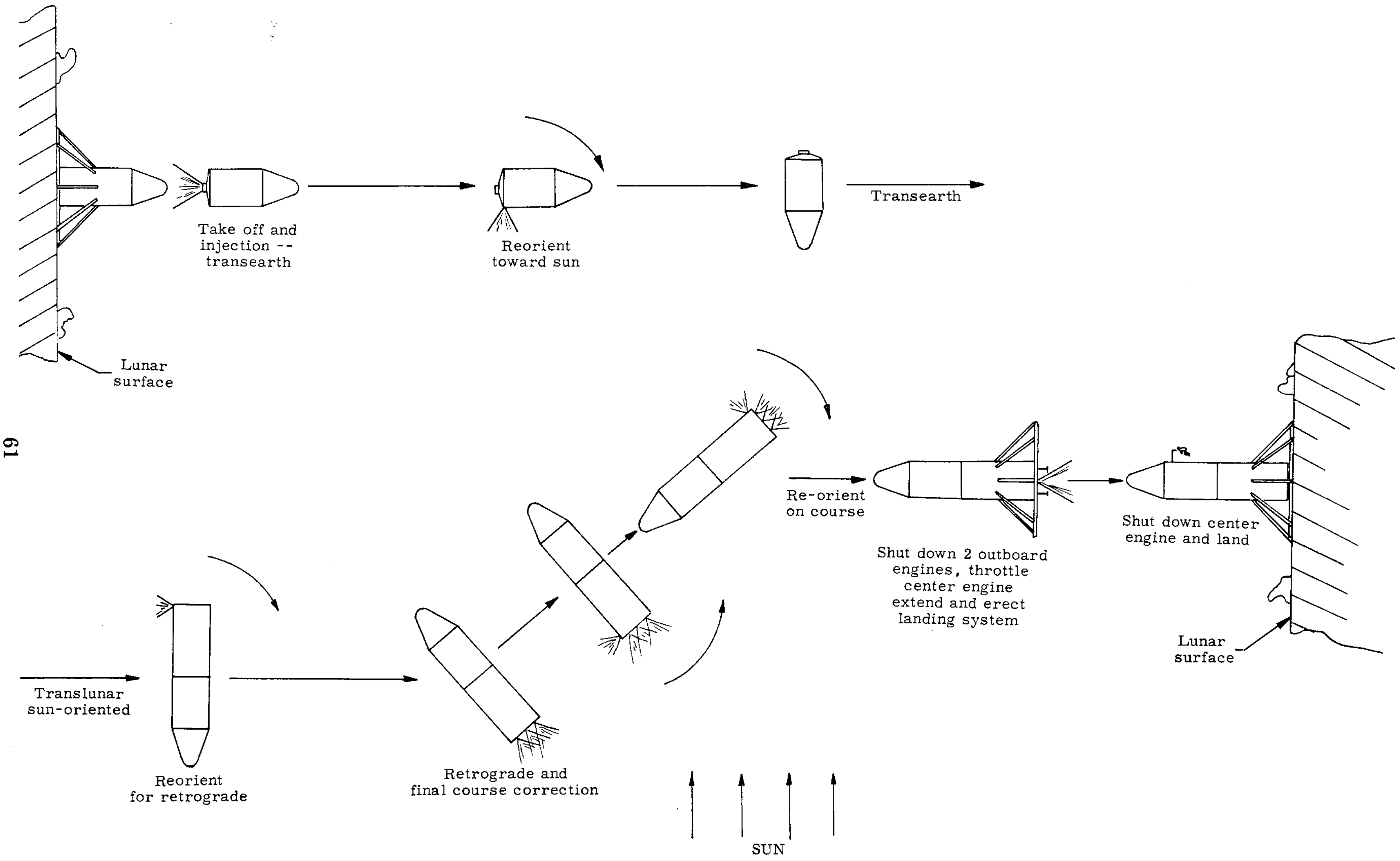

Lunar surface
Take off and injection -- transearth
Reorient toward sun
Transearth
Translunar sun-oriented
Reorient for retrograde
Retrograde and final course correction
Re-orient on course
Shut down 2 outboard engines, throttle center engine extend and erect landing system
Shut down center engine and land
Lunar surface
SUN

FIG. VIII–4. LUNAR LANDING OPERATIONAL SEQUENCE

three LR-115 engines, arranged side by side with one engine on the centerline of the vehicle. The propulsion tankage consists of a cylindrical common-dome hydrogen tank of 1164 cu ft capacity and an oxygen tank of 352 cu ft capacity. Landing touchdown provisions consist of a set of four shock absorbers which are supplemented by an inflatable structure which is stowed in fairings prior to deployment before touchdown and is rigidized by a foam-in-place material carried in cylinders. The base of the structure is closed by annular web which provides a large area of contact which is not significantly affected by reasonable surface irregularities.

The interstage volume at the forward end of the module is utilized for stowage of items such as replacement oxygen and nitrogen which are to be expended during the lunar surface mission. Blowout ports located in the external shell are designed to allow firing of the Apollo engine "in the hole." The forward dome of the hydrogen tank is provided with an ablative coating to insulate against the engine blast during the lunar takeoff sequence. The aft end of the landing module contains mounting provisions for the main rocket engines, the four telescoping shock struts, vernier and ullage engines, and attitude base rings, including the foam-producing liquid stowage and injection bottles. The overall length of this configuration is about 450 in.

Disadvantages are the use of a common dome tank, complex landing gear, larger booster bending moments due to module length, the need for blowout ports and tank insulation for engine blast during lunar takeoff. Relatively few changes are required to the Apollo spacecraft.

2. CONFIGURATION II

The arrangement of this configuration is shown in Fig. VIII-2. The outside diameter of the landing module is 220 in. overall length is approxi-

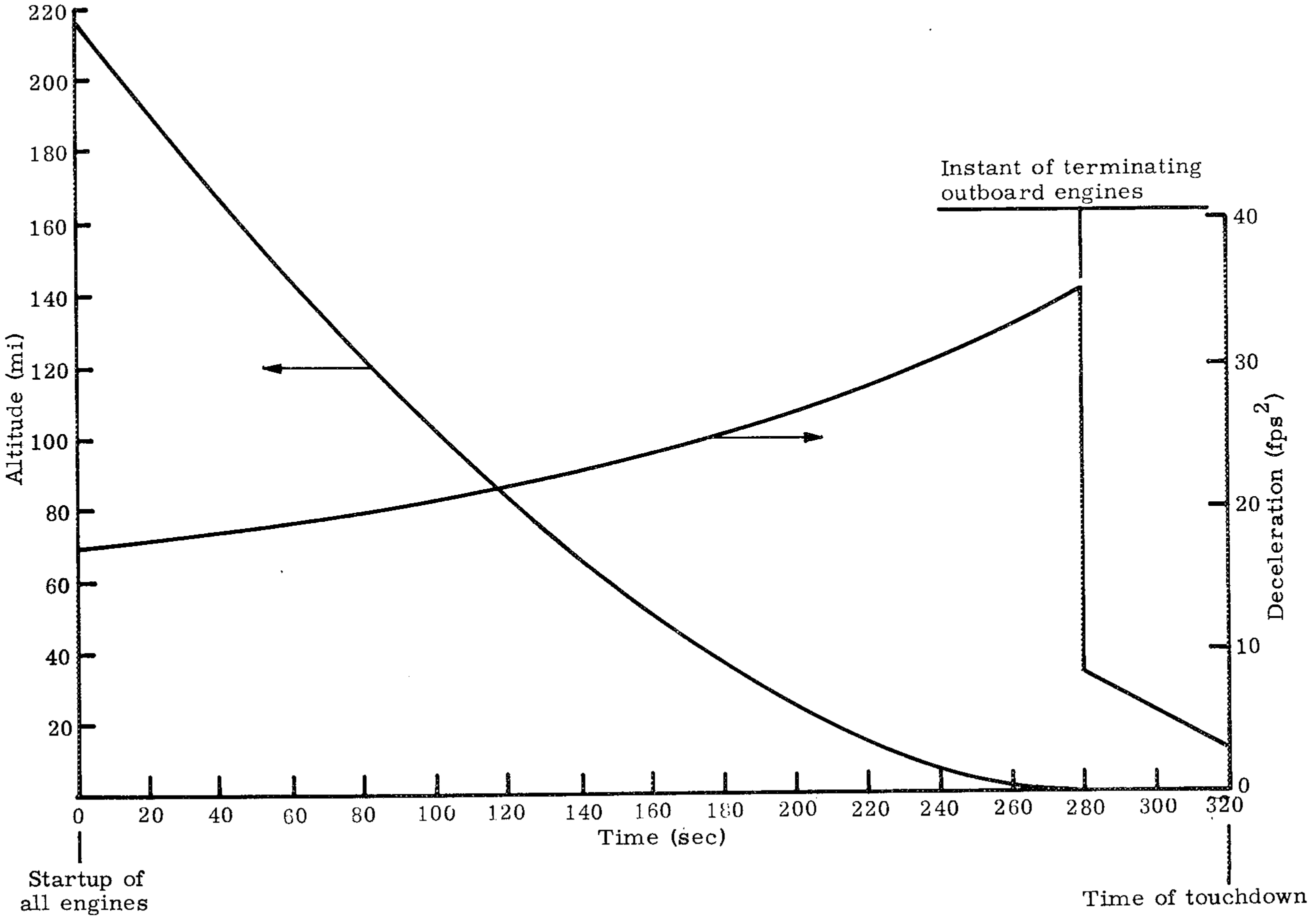


FIG. VIII-5. LUNAR LANDING, ALTITUDE AND DECELERATION VERSUS TIME

mately 330 in. The reduced overall length of this configuration is a result of utilizing toroidal oxygen and hydrogen tanks which are wrapped around the basic 154-in. diameter of the Apollo spacecraft and are contained in the space between this diameter and the 220-in. diameter of the booster vehicle.

Propulsion is furnished by three LR–115 engines, one of which is the Apollo spacecraft mission engine. The use of the Apollo LR–115 engine is accomplished by the transfer of fuel between the landing module tanks and the hydrogen and oxygen tanks of the Apollo spacecraft. Landing provisions are provided by four shock struts with a pad at each end for support at the lunar surface. The clearance between the Apollo spacecraft and the landing module as it leaves for the return to earth will present a problem for this configuration. Another problem is the need to disconnect the fuel transfer lines between the Apollo spacecraft and landing module at takeoff from the moon.

Disadvantages are: Clearance problem during lunar takeoff; fuel transfer between the Apollo spacecraft and the landing module; problem of quick release in case of aborted landing; the poor tank structure arrangement; it is heavier than the other versions; and it requires more modifications to the Apollo spacecraft.

Advantages are its shorter length, which give smaller bending moments on the booster, its better landing gear and the utilization of separate tanks.

3. CONFIGURATION III

Configuration III in Fig. VIII-3 also utilizes torus shaped tanks for the main propulsion system (contained within an outside diameter of 220 in.), allowing the mission engine of the Apollo spacecraft to fit inside of the landing module. In this case, the mission engine is not used for landing. Three LR–115 engines are used to provide the needed landing thrust and are mounted aft of the oxygen tank. The three landing shock struts are provided with an inflatable bag at the ends for bearing support at the lunar surface. Structural transition section is provided to match the 154-in. diameter of the Apollo spacecraft and the 220-inch diameter of the lunar landing module. The overall length of the landing module for this configuration is about 370 in.

Disadvantages are that the in-hole engine firing impinges on many parts and that the antennas must be relocated.

Advantages are that relatively minor changes are required to the Apollo spacecraft, it possesses good clearances for lunar takeoff, it has less complex landing gear, lighter weight, and it has separate tanks for the hydrogen and oxygen.

Figure VIII–4 shows the operational sequence during the moon takeoff and landing.

C. SYSTEM DESCRIPTION AND OPERATION

The lunar landing module contains a major propulsion system, a vernier and attitude control system, a landing or touchdown system and an electrical power system in addition to the basic structure of the module. Analysis of the sensors and guidance system available for a soft landing indicates that means are available to adequately guide the vehicle into a landing and to select a landing surface with a specified maximum roughness.

The study here concerned itself with the preliminary performance requirements and weight estimates for these systems.

1. PROPULSION SYSTEM

The main propulsion system consists of three in-line advanced LR–115 rocket engines with $I_{sp} = 427$, expansion ratio 60 to 1 and maximum thrust of 15,600 lb each. Flexible propellant lines and clearance are provided to allow 2° of angular movement of the thrust chambers for vector control. Capabilities for thrust termination and engine restart are provided in these units and the central engine is capable of being throttle to 50% of its rated thrust value. The propellants are delivered to the thrust chambers by pumps mounted on the tanks within the vehicle structure The total amount of usable propellants necessary for the lunar landing mission for configuration I is about 28,000 lb with a mixture ratio of 5 to 1. A weight breakdown is presented in table VIII–1.

Operation. The lunar landing powered phase of operation will start with all three engine operating together. After the lunar landing vehicle has reached a specified velocity and altitude the outboard engines are terminated and the central engine is operated alone. Thrust of the central rocket engine is modulated during thi period as required to execute a soft landing. The propulsion system will retard the lunar landing vehicle to essentially zero velocity at the lunar surface.

The resulting motion of the Apollo during this landing sequence is shown in Fig. VIII–5 wherein the altitude above the surface and the deceleration experienced are shown. With all three engines operating for 280 sec, the spacecraft is brought from an altitude of over 200 mi to within a mile of the surface with a deceleration varying from approximately 1/2 to 1 earth g's. Final descent is then made with the one engine operating for an additional 40 sec.

2. VERNIER AND ATTITUDE CONTROL SYSTEM

The attitude control system consists of six 25-lb thrust nozzles used in such combination as to provide yaw, pitch and roll control. These nozzles are backed up with an identical set of nozzles for purposes of redundancy. The vernier system consists of a pair of 1000-lb nozzles which are canted to act through the center of gravity of the vehicle so as to minimize moment errors due to a malfunction in one of the units. The systems utilize a hypergolic propellant of N_2O_4 and a mixture of 50% UDMH and 50% N_2H_4 contained in two 28-in. spheres mounted on the beam structure supporting the main propulsion system. The tanks are maintained at a constant pressure with a 4500-psia helium tank through a pressure regulator. An insulation blanket around the tanks and auxiliary heating means are provided to maintain the propellants at proper operating temperature.

Solenoid valves at the engines control the release of the propellants.

The total weight of the attitude and vernier control system is estimated to be 1182 lb.

3. LANDING GEAR

The landing gear must be capable of accommodating an uneven surface on the moon and at the same time to absorb the final landing impact of the spacecraft. Several different types of gear have been investigated. These include oleo-type shock absorbing struts with inflatable or rigid foot pads and inflatable struts with an inflatable ring distending a fabric surface. The estimated weights are shown in the weight summary of this section.

Further investigation of the landing system will be required as more information regarding the characteristics of the lunar surface becomes available.

4. ELECTRICAL POWER SYSTEM

An investigation was made into the added electrical requirements if a landing on the moon were accompolished with Apollo. For the investigation, the following assumptions were made:

(1) The moon stay will be limited to 72 hr.

(2) Two crewmen in space suits will be outside the vehicle during the entire stay on the moon.

(3) Space suits can operate from other than 28 vdc.

(4) Load power analysis is based on an average continuous load without regard to short duration peak loads. Repetitive loads of one to ten minute duration are averaged over the 72-hr stay.

(5) Air loss or storage are based on opening the mission module hatch once every 6 hr for entrance and exit.

The result of a study showed that it is more feasible to replenish the air lost through airlock use rather than retain it for the number of airlock uses in three days. Space suit power is to be supplied by a 500-ft line from the landing stage. Each man will carry two batteries which are adequate in case of line failure and for extended exploration.

If it is desired to increase the stay beyond 72 hr, a weight penalty will exist amounting to 4.9 lb per hour stay-time. This factor is valid if the total mission for the earth-moon-earth trip does not exceed a 14-day total. Otherwise, the basic Apollo tanks for environmental and fuel cell supply must be increased on a proportional basis.

The averaged load analysis considered is as follows:

	(Watts)
Telemetry	80
Power system and losses	240
Lighting	40
Communications	60
Displays and panels	100
Environmental Control	870
2 space men, 425 watts each from converter @ 90% efficiency	945
Charging of small silver zinc spacesuit batteries through dc to dc converter	60
Consumption for 72 hr	2345

As a result of the study, the weight of additional equipment and materials for supplying electrical power amounted to:

	(LB)
DC TO DC CONVERTER, 850 WATTS	12
500 FT CABLE NO. 16	20
REEL ASSEMBLY FOR CABLE	10
2 BATTERIES PER MAN BY 3 SETS AT 10 LB EACH	60
MOTOR WINCH ASSEMBLY	4
AIR-REPLACEMENT: NITROGEN	101
NITROGEN 18.7 IN. SPHERE	8.5
OXYGEN	215
OXYGEN 21.5 IN. SPHERE	11.5
FUEL CELL ADDITIONAL FUEL	
OXYGEN	19.6
OXYGEN 9.5 IN. SPHERE	2
HYDROGEN	2.4
HYDROGEN 12.1 IN. SPHERE	3.3
ADDED TOTAL WEIGHT	469.3

D. MODIFICATIONS TO THE APOLLO SPACECRAFT

As discussed previously, the basic model 410 Apollo spacecraft has been designed with propulsion tankage of sufficient capacity to allow a lunar takeoff and return to earth. To accomplish a lunar landing and a subsequent 3-day stand on the moon, other modifications to the Apollo spacecraft will be needed as follows:

(1) Additional electrical circuits to control the systems within the landing module.

(2) Additional display panels within the command module and possibly within the mission module.

(3) A means to lower and raise personnel and equipment to and from the lunar surface (the mission module will serve as an airlock for crew egress-ingress).

(4) Provisions for replenishing air supply expended during lunar surface exploration.

(5) Scientific equipment required for the lunar landing mission (located in the mission module).

(6) The means of attaching the Apollo spacecraft to and releasing it from the landing module will be identical to the attachment to the lunar orbit mission launch vehicle.

(7) Possible remounting of the guidance, telemetering and communications antenna within a fairing outside of the 154-in. diameter of the Apollo spacecraft.

E. WEIGHT

As shown in Figures VIII–1, VIII–2, and VIII–3 the lunar landing and takeoff spacecraft consists of the basic Apollo spacecraft and a lunar landing module. A summary of the weights of the various configurations is given in Table VIII–1.

TABLE VIII–1

WEIGHT OF LUNAR LANDING SPACECRAFT

	CONFIGURATION I (TANDEM INST.) (LB)	CONFIGURATION II (WRAP AROUND INST.) (LB)	CONFIGURATION III (TORUS TANKS) (LB)
PROPULSION SYSTEM	2,640	3,106	2,810
STRUCTURE	2,183	2,814	1,632
REACTION CONTROL	1,182	1,182	1,182
ELECTRICAL SYSTEM	519	519	519
LANDING GEAR	2,280	1,560	1,560
PROPELLANT	(28,983)	(29,339)	(28,063)
H_2 SYSTEM	(5,063)	(5,104)	(4,907)
UNUSABLE	404	382	404
USABLE	4,659	4,722	4,503
O_2 SYSTEM	(23,920)	(24,235)	(23,156)
UNUSABLE	631	626	631
USABLE	23,289	23,609	22,525
LANDING MODULE	37,787	38,520	35,766
LUNAR T.O. STAGE	22,572	22,572	22,572
LUNAR LANDING SPACECRAFT	60,359	61,092	58,338
REACTION PROPELLANT	—545	—545	—545
LUNAR VEHICLE (EFFECTIVE)	59,814	60,547	57,793
MODIFY SATURN ADAPTER			+390

IX. GROWTH VERSION

The basic Apollo mission used during the study has been the lunar orbit mission. Alternate missions which should be considered in establishing the final Apollo design include:

(1) Earth orbit both for test and for alternate missions such as scientific observations.

(2) Rendezvous and orbit around the earth.

(3) Lunar landing and takeoff.

(4) Extended periods on the moon's surface.

The selected Apollo design has been chosen and arranged so as to provide many of the features necessary to accomplish these alternate missions. For example, incorporation of the mission module, establishment of the tankage size for the lunar takeoff, utilization of fuel cells rather than solar arrays. However, there are certain system changes which would be required in order to accomplish the missions as defined above. Table IX–1 shows the various alternate missions as well as a brief description of the systems revisions required to accomplish the required mission. The evaluation of the lunar landing vehicle has been shown in Chapter VIII of this report. The lunar landing vehicle has been established on the basis of a total 14-day mission with three days spent on the moon. Staying for longer periods on the moon may require other changes to the vehicle, such as incorporation of a separate air lock. Our studies have shown that the mission module chosen is an adequate airlock for the short periods of time involved in the three days on the moon. Assuming that the mission module is used as an airlock once every six hours, the total weight of air expended by utilization of the module in this fashion is no greater than would be required for incorporation of alternate methods of conserving air, such as pumps, bladders, etc. Probably the largest problem involved in the lunar takeoff and landing consists of control of the takeoff and monitoring of the various systems prior to takeoff such that the launch from the moon can be accomplished by the Apollo crew consisting of three men.

Growth of the Apollo re-entry vehicle to a four-man version can be accomplished by minor revisions.

TABLE IX–1

SUMMARY SYSTEM CHANGES RELATIVE TO LUNAR ORBIT MISSION

ALTERNATE MISSION →	EARTH ORBIT (NO RENDEZVOUS)
SYSTEM ↓	SYSTEM CHANGE CONSIDERATIONS
ENVIRONMENTAL CONTROL	FOR 200-MI ORBITAL, EXTRA COOLING LOAD DUE TO EARTH SHINE EQUALS 2.27 LB WATER PER ORBIT. HUMAN CONSUMPTION EQUALS 0.73 LB WATER PER ORBIT. FUEL CELLS GENERATE 2.05 LB WATER PER ORBIT. WATER DEFICIENCY TO BE MADE UP FROM STORAGE EQUALS 0.95 LB PER ORBIT. NORMAL WATER STORAGE CAPACITY EQUALS 50 LB OF WHICH 25 LB IS RESERVED FOR RE-ENTRY. 25 LB ALLOTTED TO ORBIT COOLING WILL LAST FOR 26 ORBITS.
LIFE SUPPORT	SAME SYSTEMS AS FOR LUNAR ORBIT MISSION.
GUIDANCE AND NAVIGATION	SAME SYSTEMS AS FOR LUNAR ORBIT MISSION.
COMMUNICATIONS	ADD VHF TRANSMITTER-RECEIVER FOR GREATER VOICE AND DATA LINK.
POWER SUPPLY	SAME SYSTEM AS FOR LUNAR ORBIT MISSION.
PROPULSION	NORMAL RETRO-THRUST REQUIRED TO DE-ORBIT USES 500 TO 700 LB OF PROPELLANT. SPACE PROPULSION SYSTEM USED FOR RETRO. TANKS OFF-LOADED. COMMAND MODULE DETACHED FOR RE-ENTRY AFTER RETRO MANEUVER AND TURNAROUND.
BASIC VEHICLE	NO BASIC CHANGE TO STRUCTURE IF TOTAL SPACECRAFT IS USED. PAYLOAD CAN VERY WITH MISSION OBJECTIVES. ADDITIONAL WATER STORAGE REQUIRED FOR MISSION GREATER THAN 26 ORBITS (1.7 DAYS) AS INDICATED.

TABLE IX-1 (CONT)

SUMMARY SYSTEM CHANGES RELATIVE TO LUNAR ORBIT MISSION

ALTERNATE MISSION →	EARTH ORBIT (WITH RENDEZVOUS)
SYSTEM ↓	SYSTEM CHANGE CONSIDERATIONS
ENVIRONMENTAL CONTROL	ADDITIONAL WATER REQUIRED FOR MORE THAN 26 ORBITS (SAME AS EARTH ORBITS NO RENDEZVOUS)
LIFE SUPPORT	PORTABLE LIFE SUPPORT UNITS FOR SPACE SUITS ARE ADDED. RESERVE AIR IS REQUIRED FOR AIR LOCK LOSSES AND PORTABLE EQUIPMENT CHARGING. SUPPLY CAPACITY ADEQUATE FOR A NUMBER OF TRANSFERS.
GUIDANCE AND NAVIGATION	ALTIMETER ADAPTED AS RANGE RADAR. TV CAMERA ADDED FOR TRACKING AND DOCKING. FLASHING BEACON ON COMPANION SATELLITE FOR TV TRACKING IN EARTH'S SHADOW. STATION KEEPING RADAR FOR CLOSE RANGE MEASUREMENT. ADDITIONAL PROGRAM FOR ONBOARD COMPUTER FOR ORBIT TRANSFER.
COMMUNICATIONS	VHF TRANSMITTER-RECEIVER ADDED (SAME AS EARTH ORBIT, NO RENDEZVOUS).
POWER SUPPLY	SAME SYSTEM AS FOR LUNAR ORBIT MISSION.
PROPULSION	RENDEZVOUS PROPELLANT EQUALS ABOUT 1/8 THE WEIGHT OF VEHICLE. POSITIVE EXPULSION TANKS AND 6 TRANSLATION JETS ADDED AS EXPENDABLE UNITS. GROUND LOADED HYPERGOLIC PROPELLANTS USED FOR RETRO IMPULSE. LH_2 AND LOX TRANSFERRED IN ORBIT.
BASIC VEHICLE	DOCKING MECHANISMS, AIRLOCK COUPLING, PROPELLANT TRANSFER MECHANISMS, AND SYSTEMS CHANGES AS NOTED, ARE REQUIRED AS MODIFICATIONS ACCORDING TO MISSION OBJECTIVES. POSITIVE EXPULSION OF LH_2 AND LOX FROM TANKER SATELLITE REQUIRED.

TABLE IX-1 (CONT)

SUMMARY SYSTEM CHANGES RELATIVE TO LUNAR ORBIT MISSION

ALTERNATE MISSION →	LUNAR LANDING (LIMITED STAY 3 DAYS) AND TAKEOFF
SYSTEM ↓	SYSTEM CHANGE CONSIDERATIONS
ENVIRONMENTAL CONTROL	GREATER COOLING LOAD FROM POSSIBLE RENDEZVOUS IN EARTH ORBIT AND RADIANT HEAT FROM LUNAR SURFACE. COOLING SUPPLIES IN LANDING MODULE.
LIFE SUPPORT	PORTABLE LIFE SUPPORT UNITS FOR SPACE SUITS. LUNAR SURFACE RESCUE EQUIPMENT.
GUIDANCE AND NAVIGATION	AFT LOOKING TV CAMERA AND STATION KEEPING RADAR REQUIRED FOR RENDEZVOUS, ALSO USED FOR LUNAR LANDING PROGRAM FOR ONBOARD COMPUTER REQUIRED.
COMMUNICATIONS	SAME SYSTEM AS FOR LUNAR ORBIT MISSION.
POWER SUPPLY	SUPPLEMENTARY FUEL FOR POWER GENERATION CARRIED IN LANDING MODEULE (BASIC FUEL CELLS USED)
PROPULSION	LH_2, LOX ROCKET ENGINE IN A LANDING MODULE MUST BE SUFFICIENT FOR LUNAR RETRO LANDING OF 22,600 LB. VEHICLE. (EARTH ORBIT REFUELING OF SPACECRAFT AND LANDING MODULE ASSUMED, HOWEVER, AT PRESENT THE TECHNIQUES HAVE NOT BEEN STUDIED IN DETAIL.) LANDING MODULE LEFT ON MOON AT LUNAR LAUNCH.
BASIC VEHICLE	LUNAR TAKEOFF VEHICLE BASICALLY THE SAME AS FOR LUNAR ORBIT WITH FULL PROPELLANT TANKS. RENDEZVOUS IN EARTH ORBIT EQUIPMENT CAN BE ADDED TO LANDING MODULE. IF REFUELING OF BOOSTER IS REQUIRED, LANDING MODULE CARRIES ADDED SUPPLIES AND MUST BE SELF ERECTING ON THE MOON.

TABLE IX-1 (CONT)

SUMMARY SYSTEM CHANGES RELATIVE TO LUNAR ORBIT MISSION

ALTERNATE MISSION →	LUNAR LANDING EXTENDED MISSION (16 ADDITIONAL DAYS ON THE MOON'S SURFACE)
SYSTEM ↓	SYSTEM CHANGE CONSIDERATIONS
ENVIRONMENTAL CONTROL	200 LBS OF EXTRA SKIN THICKNESS FOR CRITICAL AREAS OF SPACECRAFT ADDED FOR METEOR PROTECTION (HOLDING 95 PROBABILITY CONSTANT FOR EXTENDED TIME IN SPACE). RADIATION PROTECTION FROM SOLAR FLARE ASSUMED PROVIDED BY LOCAL LUNAR FEATURES. ADDITIONAL COOLING CAPACITY PROVIDED BY LANDING MODULE 200 = LB WATER.

ALTERNATE MISSION →	LUNAR LANDING EXTENDED MISSION (16 ADDITIONAL DAYS ON THE MOON'S SURFACE)
SYSTEM ↓	SYSTEM CHANGE CONSIDERATIONS
LIFE SUPPORT	DUPLICATE PORTABLE LIFE SUPPORT UNITS FOR SPACE SUITS PROVIDED = 150 LB. DRINKING WATER TAKEN FROM FUEL CELLS AND CABIN AIR CONDENSATE. 16 DAYS O_2 SUPPLY = 100 LB (FOR BREATHING). MAKE UP OF AIR LOCK LOSS = 150 LB AND NORMAL AIR LEAKAGE = 20 LBS FOR ADDITIONAL 16 DAYS. 73 LBS OF DRY WATER MIX FOOD REQUIRED. ALL ADDITIONAL SUPPLIES STOWED IN LANDING MODULE. METABOLIC REQUIREMENTS BASED ON 3200 KCAL PER MAN DAY FOR LUNAR SURFACE ACTIVITIES.
GUIDANCE AND NAVIGATION	CONTROLS FOR TAKEOFF DESIGNED FOR TWO MAN NORMAL OPERATION. CAN BE OPERATED BY ONE MAN WITH EARTH ASSISTANCE IN EXTREME EMERGENCY.
COMMUNICATIONS	SAME AS FOR LUNAR ORBIT MISSION.
POWER SUPPLY	FUEL FOR POWER GENERATION TAKEN FROM LANDING MODULE. FUEL CELLS USED AT CONSTANT 1.5 KW LEVEL BURN 47 LB H_2 AND 400 LBS OF O_2 IN 16 DAYS.
PROPULSION	OPTIMIZED H_2 BOILOFF FOR 16 DAYS = 312 LB, 47 LBS ARE RECOVERABLE IN FUEL CELLS. O_2 BOILOFF OPTIMIZED AT FUEL CELL REQUIREMENTS. 100 LBS TANK INSULATION ADDED TO LANDING MODULE.
BASIC VEHICLE	EFFICIENT AIR LOCK SYSTEM ADDED TO MISSION MODULE. STRUCTURAL WEIGHT = 190 LB. METEOR PROTECTION FOR INCREASED TIME IN SPACE = 200 LB, PROPELLANT TANK WEIGHT ADDED 65 LB FOR HOUSING 665 LB PROPELLANT NEEDED AS ADDITIONAL FUEL. LANDING MODULE REQUIRES 100 LB OF MISC. STRUCTURE PLUS ADDITIONAL TANKAGE REFLECTED IN PROPELLANT AND TANKAGE REQUIRED TO RETRO LAND OVERALL VEHICLE ON LUNAR SURFACE. TOTAL INCREASE IN LANDING MODULE PAYLOAD = 2985 LB. LANDING PROPELLANT AND TANKAGE = 2985 LB. TOTAL VEHICLE INCREASE IN WEIGHT FOR 16 DAYS ON THE MOON = 5970 LB.