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APOLLO

FINAL STUDY REPORT

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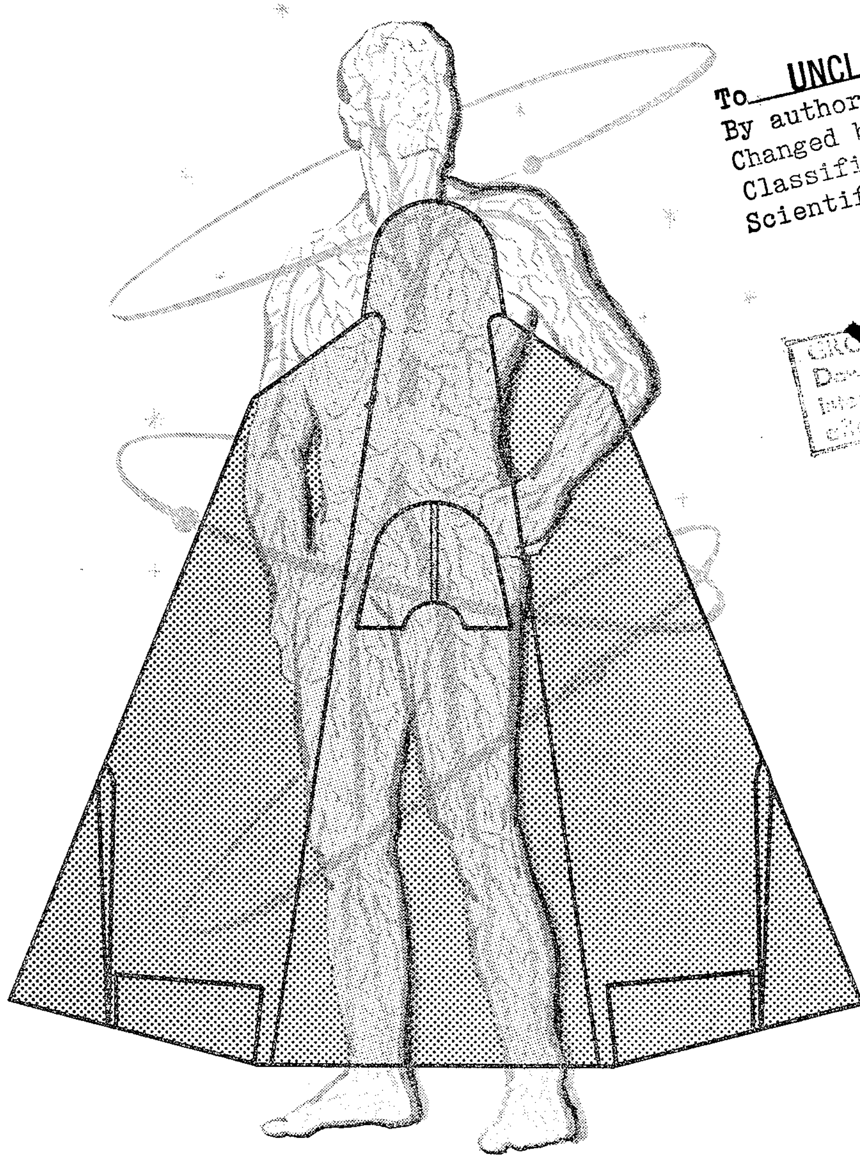

APOLLO

FINAL STUDY REPORT

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Report No. AE61-0363

15 May 1961

Contract NAS 5-304

Volume IV

GROWTH AND ADVANCED CONCEPTS

[U]

Prepared by:

GD GENERAL DYNAMICS

CONVAIR (ASTRONAUTICS) DIVISION
GENERAL DYNAMICS CORPORATION
and
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VOLUME III | SUPPORTING DESIGN ANALYSES

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BOOK 2 | HEAT CONTROL SYSTEMS

BOOK 3 | STRUCTURAL ANALYSES AND MATERIALS

BOOK 4 | PROPULSION AND AUXILIARY POWER

BOOK 5 | TRACKING, COMMUNICATIONS AND DATA SYSTEMS

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1 INTRODUCTION

It is important to evaluate a selected design as to growth applications both immediate and long-term. The M-1 re-entry vehicle coupled with the modular concept configuration provides limitless versatility for manned space systems.

First, the M-1 re-entry vehicle will be refined using flight experience and additional research testing. The heat shielding, control, and landing systems offer the greatest areas of design evolution. The propulsion system selected is not likely to be superseded throughout the Apollo program. Alternate ablative materials and ceramics are under development and will be progressively applied to the Apollo vehicle as heat shielding technology advances. The parachute landing system may be superseded by the paraglider to permit glide control and reduced impact. The paraglider warrants continued study for the M-1 vehicle recovery and landing system. Stability and control testing of the M-1 shape should yield applicable data for reducing flap size, weight, and cross coupling effects. The elimination of control flaps is the goal of feasibility studies currently underway on the re-entry roll maneuver.

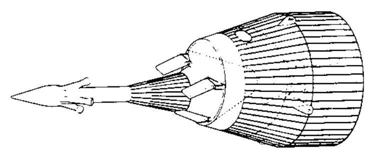
Secondly the Apollo vehicle will offer a logical development system for cooperative manned space vehicle rendezvous techniques. The interchangeable modular design provides distinct assembly advantages for special subsystems related to the rendezvous technique. One such subsystem, the retractable airlock, is included.

Subsequently, the manned lunar landing vehicle could emerge from the Apollo program. The propulsion module with jettisonable tanks then becomes the dominant module as shown in Section 2.4.

A highly tapered M-1 type vehicle possessing the same re-entry stability and heating characteristics and which also achieves an $L/_D$ adequate for glide landings is under test. The experience to be gained with the M-1 shape will build a technical foundation for successively larger and cleaner re-entry shapes such as the M-2.

Eventually, the requirement to re-enter, fly in, and land at an existing airport, or to ferry vehicles from one base to another will be justifiable.

As re-entry vehicles grow in size and complexity, the ferry version will inevitably appear. The M-3 vehicle with foldable wings is one logical evolution from the M-2. Ferry flights would be powered and fueled by detachable engines and fuel pods. Thus, the M series vehicles, as created by NASA research, provide a versatile spectrum of space re-entry vehicles. Each advancement will not require a complete new concept, and experience obtained by development of its predecessors can be used.



2 GROWTH POTENTIAL OF SELECTED CONFIGURATION

An outstanding feature of the recommended Apollo vehicle is the provision for versatility and growth. It is impossible to predict much less enumerate all the missions within the capability of all or selected modules of the selected design.

Although the re-entry vehicle is sized for three men, it is roomy with adequate consideration for confinement psychology. Alternate interior arrangements are possible to include two additional men if needed for a rescue mission.

The volume requirements of space station living, crew exchange, highly instrumented, or specially equipped missions are all adequately provided for with the mission module concept presented. The modular design permits a volume to weight ratio that is significantly higher than other integrated vehicle designs.

Incorporation of an ellipsoidal pressure-stabilized bulkhead in the mission module is one reason for high structural efficiency. The Atlas program stands as evidence to the proven reliability of pressure-stabilized structures, when carefully designed and quality produced. It also exemplifies the method to best eliminate the air leakage loss typical of this airframe structure.

2.1 RE-ENTRY MODULE

2.1.1 Heat Shield Improvements

2.1.1.1 Composite Heat Shield Materials. While the simple concept of a single ablator heat shield, as used in the selected design (Figure 1), represents what is considered the most reliable use of presently available materials, it seems entirely reasonable that lighter weight and higher performance heat shields can be obtained by combining the desirable properties of a number of different materials. The Avcoat X-5000 series ablation materials, as has been noted earlier, represent an attempt to obtain the char stability of the best high temperature ablators while at the same time obtaining insulation effectiveness in a single material. These goals are not entirely compatible, and the resulting material represents a compromise solution to the problem.

Ultimately the best solution would be to use a composite material in which all constituents are employed under optimum conditions. It seems certain that a composite consisting of the best possible ablator and the best possible insulator used in series would out-perform any compromise material which might be developed. Some evidence that this is the case was given by the RAD-Q material, which represented a first approximation to the ideal system just described. Under some conditions, this composite was definitely superior to the reference heat shield materials.

The major difficulties in reliable application of this principle lie in the areas of material compatibility and fabrication and attachment difficulties. In some cases, there are no materials presently available with the necessary properties at a given temperature to permit optimum use of a composite. Thus, it is not difficult to outline extensive developmental programs aimed at improving known materials and developing new ones for use in composite heat shields. The first steps in this area should be in the order indicated:

a. Devise suitable test techniques for evaluation of heat shield performance. (This is currently well under way with the Hyperthermal Wind Tunnel.)

b. Prepare test specimens of composites such as high and medium temperature ablators-over-insulator, and low temperature ablators-over-reradiating metal skins without great regard to ultimate fabrication and attachment problems - but with the purpose of determining how good or bad the composite may be for the intended heat shielding.

c. Choose best composite types from these tests and concentrate on property improvement for optimum performance.

d. When materials choices are considerably narrowed, concentrate on attachment and fabrication problems.

As noted, there is some progress in the earliest step of this planned development of the composite materials concept, and it is anticipated that marked advances will be made by this means in the optimum utilization of both plastic and ceramic ablation materials, insulations, and metal and ceramic radiating surfaces.

2.1.1.2 Improved High Temperature Ablators. Improvement of char stability appears to be the optimum approach to the development of improved organic or organic-ceramic high-temperature ablators. In the

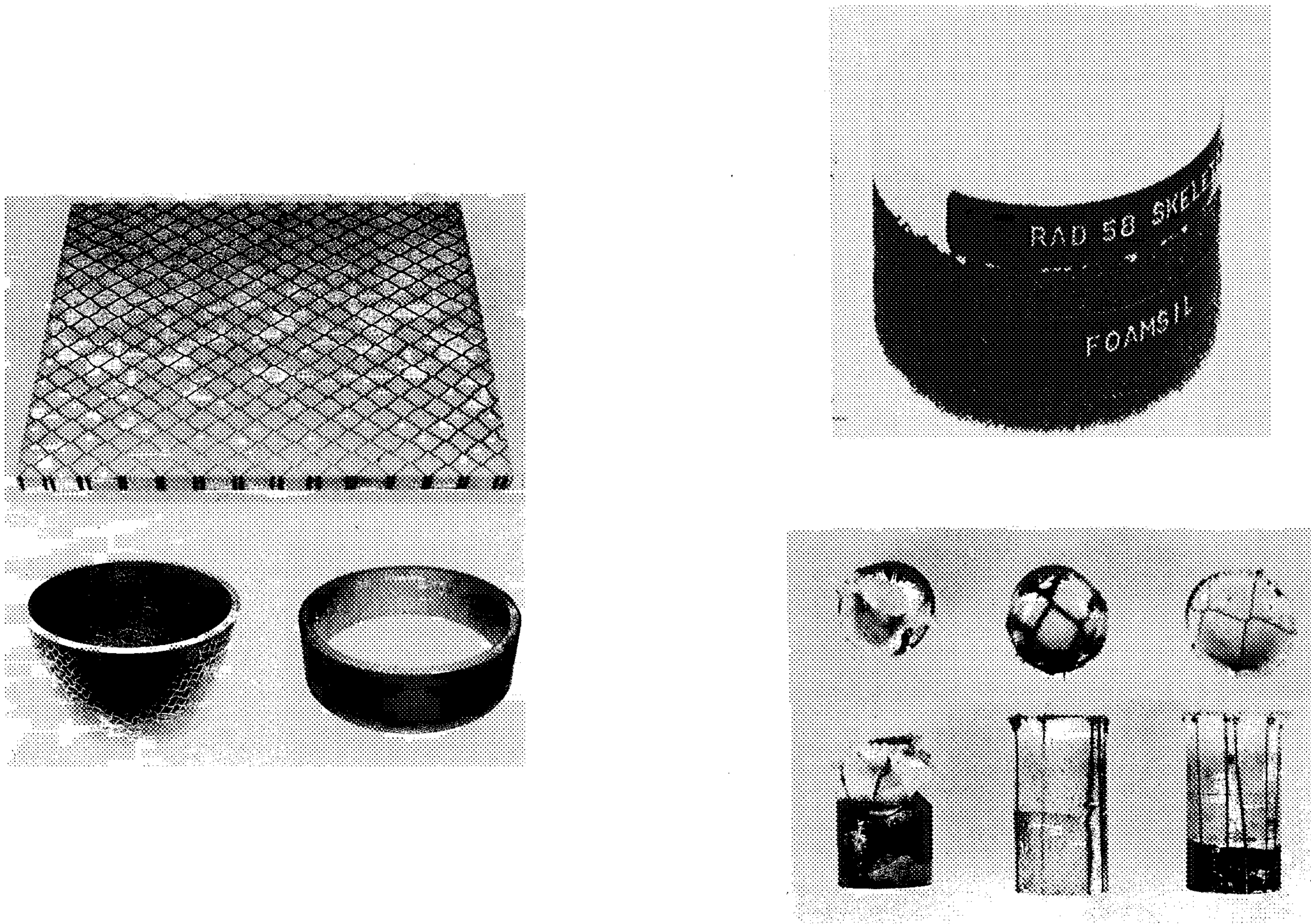


Figure 1. Composite Heat Shields

field of all-organic materials, one effort which shows some promise is the reinforcement with or incorporation of a pre-charred structure in the ablator. By oven heating or "coking" a fibrous organic laminate, for example, a fairly strong carbonized skeleton is obtained. When this material is impregnated with resin and cured, a material results which in ablation tests gives a considerably stronger char layer than would the original laminate had it been impregnated without the pre-charring step.

This phenomenon is not well understood and much work is required to reveal the mechanism of the strengthening and thereby permit the design of optimum ablation materials based on this principle.

The development of the Avcoat X-5000 series of organic-ceramic ablators also had the goal of char stabilization by the addition of silica fibers and phenolic resin to an epoxy resin. The extent of these additions is fixed at present by the physical inability to blend or mold the mixture if certain compositional limits are exceeded. Thus, in this case the next step is one of development of processes by which greater amounts of stability additions can be incorporated.

2.1.1.3 Ceramic Heat Shield Materials. Ceramics, particularly oxides, appear to offer great potential advantages as heat shield components particularly for very high temperature re-radiation of heat. Ideally, a porous ceramic should permit re-radiation from a very hot surface combined with excellent insulation properties for protection of the structure behind. That presently available materials fail to perform in this way is primarily the result of the inability of the materials to withstand thermal stresses. As discussed in an earlier section, the foamed silica materials by virtue of their low expansion coefficients offer some hope of operation under fairly severe stress conditions.

One disadvantage of silica as a high temperature heat shield is the fact that it is fairly transparent with regard to radiant energy. Thus, a major contribution toward ceramic development can be made in the area of opacifying silica based foams. Some simple preliminary studies have been made in spraying various ceramics to give opacified surface layers, but it seems probable that more sophisticated approaches will prove to be the answer. The spray technique employed in forming the RAD-58-DS skeleton seems amenable to modification so that opacifiers could be dispersed throughout the silica layer. Additional work might include studies of the effects of impurity elements on the optical properties of silica in hope that significant changes could be effected without affecting its excellent ablative properties.

2.1.2 Alternate Flap Configuration. An alternative arrangement of the flaps gives direct yaw and pitch control without cross-coupling effects. This is achieved by mounting the flaps centered on and operating normal to the horizontal and vertical axes of the command module, as shown in Figure 2. Prior to re-entry the flaps are latched in retracted position. They are actuated similarly to the selected method previously described.

2.1.2.1 Alternate Flap Configuration Analysis. Flap control design is based on moving flaps at max q through an angle from 0 to 30 degrees in 0.5 second.

Design Assumptions:

Max q = 1000 lb/ft^2. Duty cycle: move 3 flaps to 30 degrees in 0.5 second.

Flap area = 10.7 ft^2 pause 30 seconds.

Moment arm = 1.58 feet.

Flap angle = 30 degrees to free stream velocity vector.

Max hinge moment, $M_h = 10.7 \times 1000 \times 1.583 \times \sin^2 30$ degrees = 4275 lb-ft.

Actuator area based on 2000 psi load pressure and 96 percent of 6-inch useful actuator moment arm.

$$A_p = \frac{4275}{2000 \times 0.5 \times 0.96} = 4.3 \text{ in.}^2$$

Using an unbalanced area actuator design, Cylinder ID = 2.375 inches. Rod diameter = 1.25 inches.

Volume required to extend three flaps to extreme deflection from initial position (parallel to Z axis).

$$V_o = 2\,(4.43 \times 4.6) + 4.43 \times 2.6 = 52.3 \text{ in.}^3$$

Accumulator gas volume required at 2000 psi precharge.

$$V_g = \frac{52.3}{1 - \left(\frac{2000}{3000}\right)^{1/1.4}} = 208 \text{ in.}^3$$

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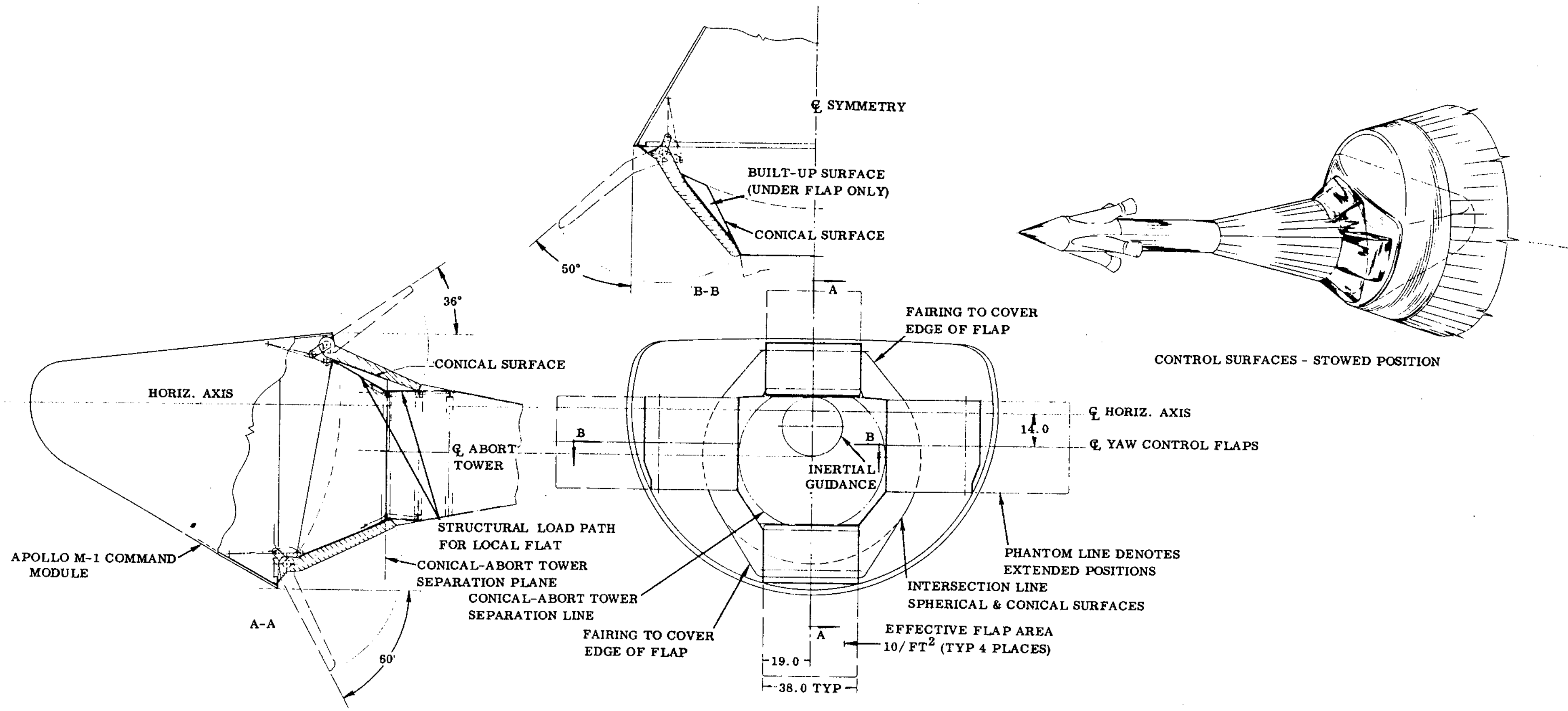


Figure 2. Alternate Flap Configuration.

Max actuator flow rate, $Q = \frac{3.11 \text{ (stroke) } 4.43 \text{ (area)}}{0.5 \text{ seconds}} = 27.8 \text{ in.}^3$ or $Q = 7.22$ gpm. Hence, the servo valves would be rated at 7.5 gpm for a differential cylinder port pressure of 1000 psi, working pressure of 3000 psig, proof 4500 psi and burst of 6000 psi; and a leakage at 3000 psi of 0.15 gpm total.

Assuming a flow factor, K, of the valves to be constant, and with the formulae below,

$$Q = K\sqrt{P_s - P_l}$$

$$p_l = 10.7 \times 1000 \times 1.583 \sin^2\theta$$

$$p_s = 3000\left(\frac{V_g}{V_g + \Delta V}\right)^{1.4}$$

where:

Q = valve rated flow,

p_s = supply pressure,

p_l = load pressure,

θ = flap deflection,

V_g = gas volume at 3000 psi, and

V = volume to displace flaps some angle θ

For 3 flaps moving through a 30 degree angle, a flow vs flap deflection is shown in Figure 3.

A flap schematic is included as Figure 4.

2.1.3 Re-entry Roll Maneuver. Consideration has been given to the use of a vehicle without aerodynamic controls which is trimmed, by means of center of gravity location, to fly at an L/D ratio of 0.5. This vehicle would be stabilized in roll, pitch, and yaw by means of a gas reaction control system. The bank angle and yaw angle would be controlled by roll jets in conventional fashion.

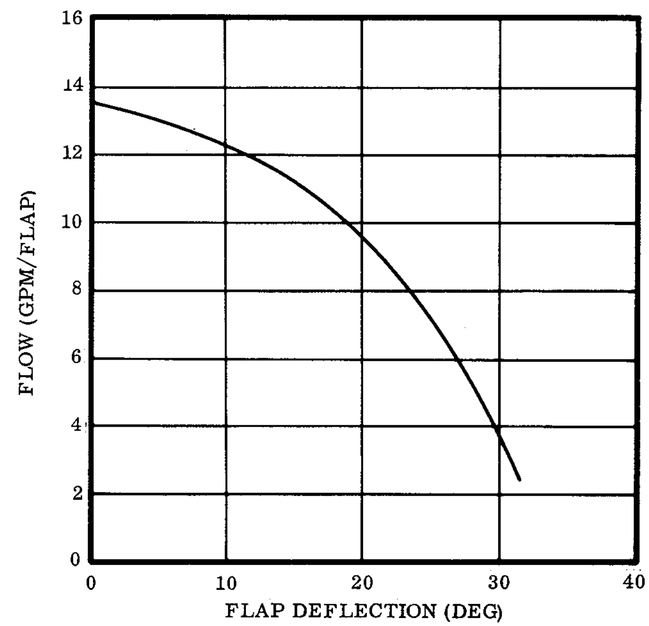


Figure 3. Flap Actuator Requirements

The angle-of-attack would remain at the trim value for L/D = 0.5, except for disturbances, and the control over the effective L/D would be accomplished by roll modulation using the reaction roll control system. Thus, if an L/D of zero is desired, for example, the vehicle could be rolled back and forth about a bank angle of zero with an amplitude which results in an average vertical component of lift equal to zero. For an effective positive L/D between zero and 0.5, the amplitude of the roll oscillation would be reduced appropriately, and for effective bank angles other than zero, the mean value for the oscillation would be the desired effective bank angle. Such a system has the significant advantage of eliminating the vehicle control surfaces and associated control systems, but this advantage is countered by the weight of the reaction system additional propellants and tankage. Since this system is already onboard for attitude control, no added complexity is involved. The weight, speed of response, and other aspects of this technique are discussed more fully in the following paragraphs.

2.1.3.1 Response Rate. The most critical point on the trajectory occurs at the bottom of pull-up from the lifting undershoot boundary. Here the lift

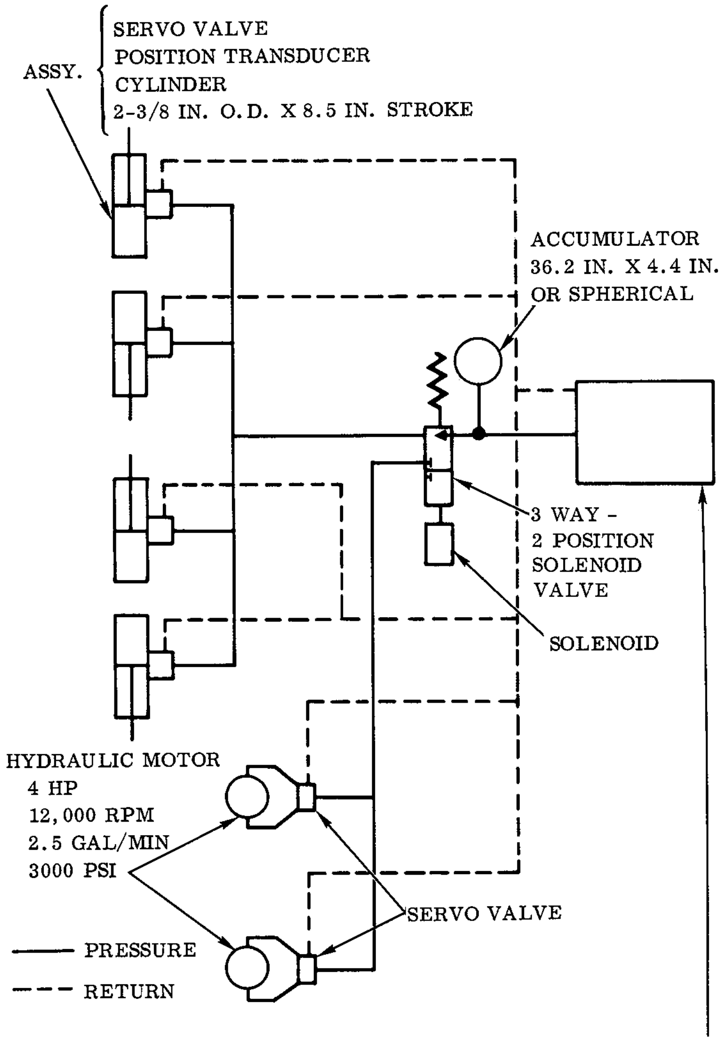


Figure 4. Flap Schematic

must be reduced from a maximum to zero in about 5 seconds. This requires rolling the vehicle from a bank angle of zero to 180 degrees in 5 seconds to give an average lift of zero in the vertical plane, and then rolling the vehicle to an equal amplitude in the opposite direction, etc. The problem is somewhat more complicated than this because during the 5 second roll to about 180 degrees there will be a horizontal lift component which is not desired. If this is not quickly nulled out by rolling in the opposite direction, the vehicle may proceed to an altitude at which the lift has become too small to counteract the large effect at the bottom of pull-up.

It appears that the ability to roll 180 degrees in 5 seconds or 360 degrees in 10 seconds, is a minimum requirement, and that this speed may have to be doubled in the vicinity of the bottom of pull-up. Computer simulation has not been adequate to fully verify this conclusion, but the following analysis will be based on these assumptions.

2.1.3.2 System Operation. Roll control is accomplished by two roll jets on opposite sides of the vehicle which operate simultaneously to provide a rolling moment. If it is desired to oscillate the vehicle through a roll angle of ± 90 degrees, for example, the roll jets provide a positive moment until the proper angular velocity has been reached. The vehicle coasts until the roll angle approaches 90 degrees, at which point the thrust is reversed and maintained until an appropriate angular velocity is reached in the opposite direction, and so on. If it is desired to increase or decrease the amplitude or to change the mean or average roll angle, the switching time is adjusted.

The thrust level and duration of the thrust impulse are determined by the performance requirements of the system, to be discussed below. Once these values are established, the time of turning the thrust on and off becomes a simple switching criterion which is a function of the desired bank angle and L/D ratio. The outputs of the guidance computer are the desired bank angle and L/D ratio; the switching network receives these as inputs along with actual bank angle and bank angle rate, and these quantities determine the proper time and direction for operating the thrust.

The switching sequence operates so that if a given impulse duration is required to achieve a limit cycle with ± 90 degrees amplitude and complete a cycle in a given period of time, the impulse duration will be adjusted for other amplitudes so that they too will complete one cycle in the same length of time. Consequently, the maximum fuel requirement will exist when the vehicle must be rolled through ± 180 degrees. It is evident that instead of rolling through ± 180 degrees, the vehicle could be rolled continuously in the same direction, if zero lift is desired, with essentially no expenditure of fuel after the initial impulse.

Unfortunately, it will in general be necessary to use magnitudes of lift which vary from zero to the maximum, therefore it is necessary to design for the worst case. It should also be mentioned that rolling through ± 180 degrees results in zero lift only in the case of an infinitely short impulse. For longer impulses the roll angle is decreased, since the vehicle spends proportionately more time at the extreme amplitudes where its angular velocity is lower.

2.1.3.3 Reaction System Weight. Figure 5 shows the weight of propellant required to roll through 360 degrees, starting and ending with zero angular velocity. Several curves are drawn corresponding to different thrust durations, and the corresponding thrust levels are indicated. It can be seen that the minimum fuel corresponds to a zero impulse time with infinite thrust and the maximum fuel corresponds to continuous thrust (no coast time between thrust reversals).

The flight time for a maximum range trajectory is just under 1000 seconds when the altitude for deploying the parachute is reached. Using a roll rate of 360 degrees in 10 seconds as a minimum requirement, it is seen that a propellant weight of 140 to 280 pounds is required. A practical impulse time of 2 seconds gives a fuel weight of 160 pounds and requires 200 pounds of thrust. If the roll rate is doubled, an impulse time of 2 seconds requires a propellant weight of 710 pounds and a thrust of 445 pounds. It is obvious that the latter weight is so large that the system is most unattractive.

It is certainly not necessary that the vehicle be required to roll 360 degrees in 5 seconds for the entire flight. In fact, it is not necessary to roll one cycle in 10 seconds during portions of a long range flight, because that part of the trajectory above 250,000 feet is in a region of such low dynamic pressure that lift forces are very small and can be modulated very slowly.

Consider now the following requirements:

a. For 10 seconds either side of pull-up the vehicle must be able to roll 360 degrees in 6 seconds.

b. For the remainder of the flight below 250,000 feet a roll rate of 360 degrees in 10 seconds is required.

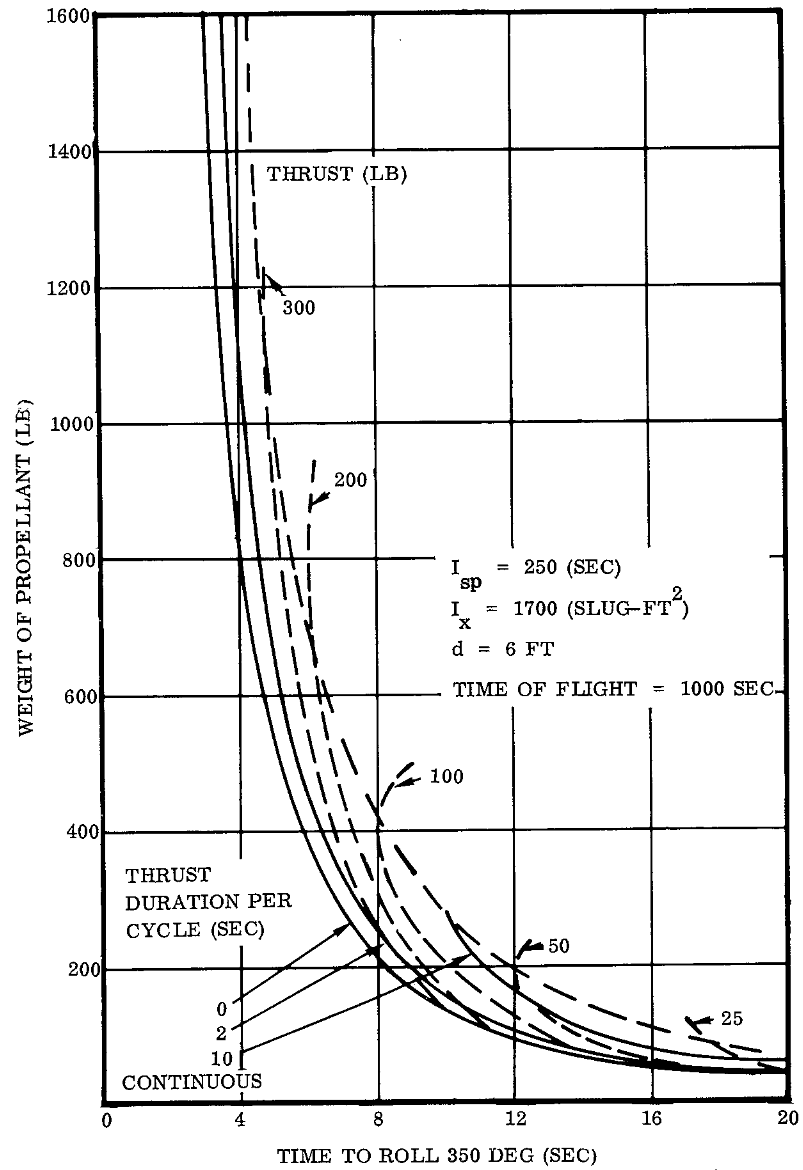


Figure 5. Weight of Propellant Required to Roll M-1 Vehicle at Various Rates

c. For the flight above 250,000 feet a roll rate of 360 degrees in 15 seconds is required.

Under these requirements a thrust level of 200 pounds will give the required performance at pull-up by continuous operation; below 250,000 feet by a maximum impulse duration per cycle of 2 seconds; and above 250,000 feet by an impulse of about 0.6 second maximum. For a maximum range trajectory this corresponds to a fuel requirement of only 140 pounds, with the flight ending at 50,000 feet. Using a factor of safety of 50 percent gives a propellant requirement of 210 pounds.

Since the vehicle is trimmed to fly at a constant angle-of-attack, any unintentional out-of-trim condition in yaw will give a resultant rolling moment. In order to be sure that this will not decrease the roll rate below those desired, an extra margin of performance must be provided. This is estimated to require an additional 40 pounds of fuel.

Finally there is the fuel required for pitch and yaw stabilization. Since this is required only to overcome disturbances, very little in addition to the present attitude control system is needed. The fuel weight required is estimated to be 10 pounds.

Table 1 gives the weight summary, and shows a total fuel weight as detailed above, of 260 pounds. The weight of fuel tanks, lines, thrust chambers, and miscellaneous plumbing is estimated to be 120 pounds, giving a total system weight of 380 pounds.

Table 1. Reaction Control System for Roll Modulation

ITEM	WEIGHT (LB)
Fuel For Roll Modulation	140
Fuel For Factor of Safety	70
Fuel to Overcome Untrimmed Moment	40
Fuel For Pitch and Yaw Stabilization	10
Total Fuel	260
Fuel Tanks, Plumbing, etc.	120
	380

2.1.4 Paraglider Recovery System for the M-1. A paraglider system of recovering the M-1 command module after re-entry has been investigated using two basic configurations. These configurations are shown in Figure 6.

Configuration A is based on a wing loading of 16 lb/ft^2 and has a total surface area of 375 ft^2. Basic components include a Rogallo type wing of flexible, high-strength fabric membrane, inflatable keel and leading edges, and shroud lines. The shroud lines are attached to the primary structure at the rear of the M-1 vehicle. The fore and aft supports consist of one continuous line running through an internally mounted winch which has the ability to lengthen or shorten opposite shrouds, thus controlling paraglider angle of attack. Trim is necessary to establish a steady glide. The two main support lines are fixed in length and if projected would intersect the center of pressure of the wing and the center of gravity of the vehicle. The system is stowed at the rear of the vehicle within the confines of the abort tower.

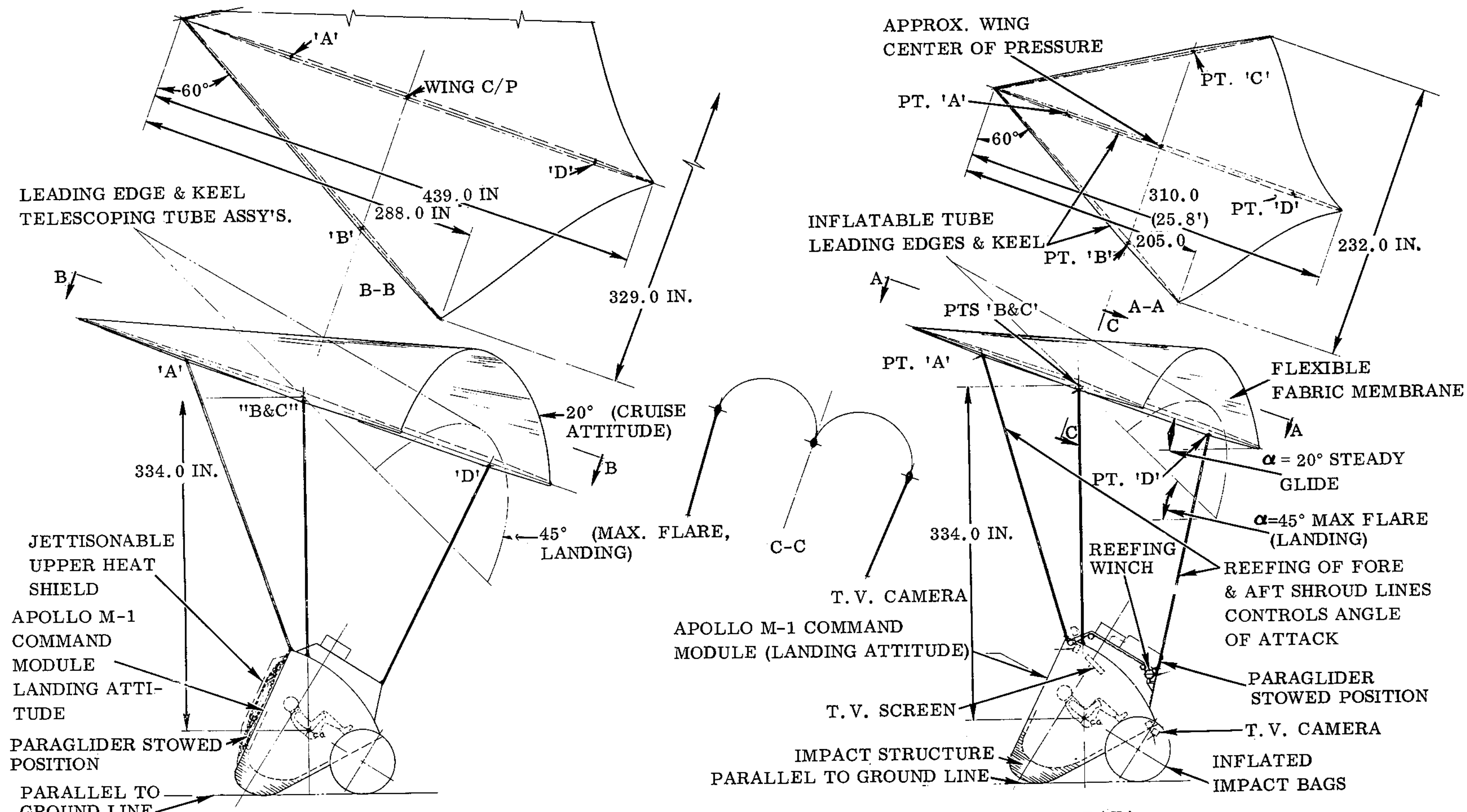
Configuration B has a wing loading of 8 lb/ft^2 and a total surface area of 750 ft^2. Wing fabric, shroud lines, and winching system are similar to configuration A. The keel and leading edges consist of telescoping aluminum alloy tubing which are pneumatically actuated. The stowed position for the 750 ft^2 glider is on the flat upper surface of the capsule. It is protected during launch and re-entry by a heat shielded, jettisonable fairing.

The paraglider concept of vehicle recovery is based on the Rogallo type design now under test at the Langley research center. Parametric studies of Apollo applications are being continued.

2.1.5 Guidance and Control

2.1.5.1 Growth Potential. The growth potential of the guidance and control system in this instance may be measured in at least three ways: 1) Its ability to realize the full capabilities of the Apollo vehicle, 2) Its ability to handle an extension of vehicle capabilities, and 3) Its ability to satisfy the requirements with less and/or lighter, more reliable equipment.

2.1.5.2 Maneuver Capability. The re-entry guidance and control system described in Volume III, Book 6 was studied against a landing footprint that was considerably reduced from the unguided footprint capability. There is nothing basic to the re-entry guidance system that will not permit the guidance system to realize the unguided footprint capability. As described in Volume III, Book 6, the major modification is the change in the point that the lift is cut off during the pull-up phase, for re-entry near the lifting undershoot boundary. This change does not alter the basic characteristics of the system.



DATA

WING LOADING	8 LB/FT^2
AREA	
TOTAL (FLAT PATTERN)	750/FT^2
PROJECTED	500/FT^2

DATA

WING LOADING	16 LB/FT^2
AREA	
TOTAL (FLAT PATTERN)	375/FT^2
PROJECTED	250/FT^2

Figure 6. M-1 with Paraglider Deployed

The altitude limit of 300,000 feet excludes skip-out type of trajectories. Without considering the desirability of such trajectories, the re-entry guidance system should be able to handle such maneuvers. Indeed, its predictive nature could be used to provide relatively accurate predictions of the point of exit. However, it seems likely that the equipment configuration would have to include additional componentry.

2.1.5.3 Vehicle Landing Capability. Future vehicles might have a landing capability. This extension should pass little problem to the re-entry guidance and control system as outlined. The change in concept might involve the way radio guidance is used during the terminal phase and the role the pilot will play.

2.1.5.4 Equipment Improvement. At present the initial conditions at re-entry are such that the platform indicated altitude is in error by about 30,000 feet near the parachute deployment point. This error, coupled with the indicated range error, requires the use of a terminal radio station as part of the reference guidance system. The altimeter described in Volume III, Book 6, might possibly be used to reduce these errors so that a terminal radio guidance station is not necessary. Essentially this would require better knowledge of density and density variations and better knowledge of the drag coefficient and its variation during re-entry flight.

The speed of the computer will improve. At present, however, computer speed does not present any problems. A faster computer would permit a somewhat more uniform operation, i.e., the predicted trajectory may run to impact over the whole range of actual re-entry trajectories.

2.1.5.5 Advanced Concepts. In the realm of advanced concepts one mode of vehicle control appears particularly attractive: pitch-trim, roll-modulation control scheme. A brief evaluation of this system is included in Section 2.1.3.

2.2 MISSION MODULE

2.2.1 Passive Thermal Control System. A unique and advantageous space heat control system for the Apollo mission module is presented as a possible growth thermal control system. The passive space thermal control system is schematically illustrated in Figure 7. A main blower, located at the bottom wall of the mission module, circulates air between the sides of a double skin wall. The top section of the double wall serves as a manifold from which the air is ducted to the command module and mission module electronic components. The air leaves the command module through the open hatch and is drawn through mission module equipment compartments and returned to the blower.

It has been shown that heat flux absorbed from the Sun and emitted by the vehicle walls can be substantially larger than the total internal heat load. Therefore, any variation of the internally generated heat load can have only a small effect on the internal temperature.

The bottom of the mission module will face the Sun and the outside surfaces will have a maximum absorbitivity/emissivity-ratio finish in order that maximum heat flux may be picked up by the circulating air in the bottom wall. The outside surfaces will have a value of emissivity that is calculated to emit a specified heat flux that is a function of the bottom solar heat input and the mean internal heat load.

2.2.1.1 Advantages of System. Distinct advantages of this system over the space radiator system are:

a. The double wall design is superior meteorite protection.

b. A substantial reduction in weight by removal of the space radiation and coolant circulation system can be realized.

c. A simpler and, therefore, more reliable system utilizing the large surface area of the mission module to best advantage.

2.3 PROPULSION MODULE

2.3.1 Propulsion System. The recommended propulsion system is easily adaptable to missions requiring an increase in total impulse. The only modification required by the present system is to increase the section diameter of the toroidal propellant tanks and to increase the capacity of the pressurization system.

Advanced propulsion systems which require complete development programs might take advantage of LOF_2 and LH_2 for main propulsion and LOF_2 and MMH or LOF_2 and LH_2 for attitude control and midcourse velocity corrections. These systems are discussed in Volume III, Book 4, as potential means for payload increases, but only after extensive experience with the selected propellants.

2.4 RENDEZVOUS OPERATION SEQUENCES

2.4.1 Rendezvous Operation Sequences--See Figure 8.

2.4.1.1 Initial Contact. Vehicles will come within range of each other, but will not attempt to make direct contact under their own power. This

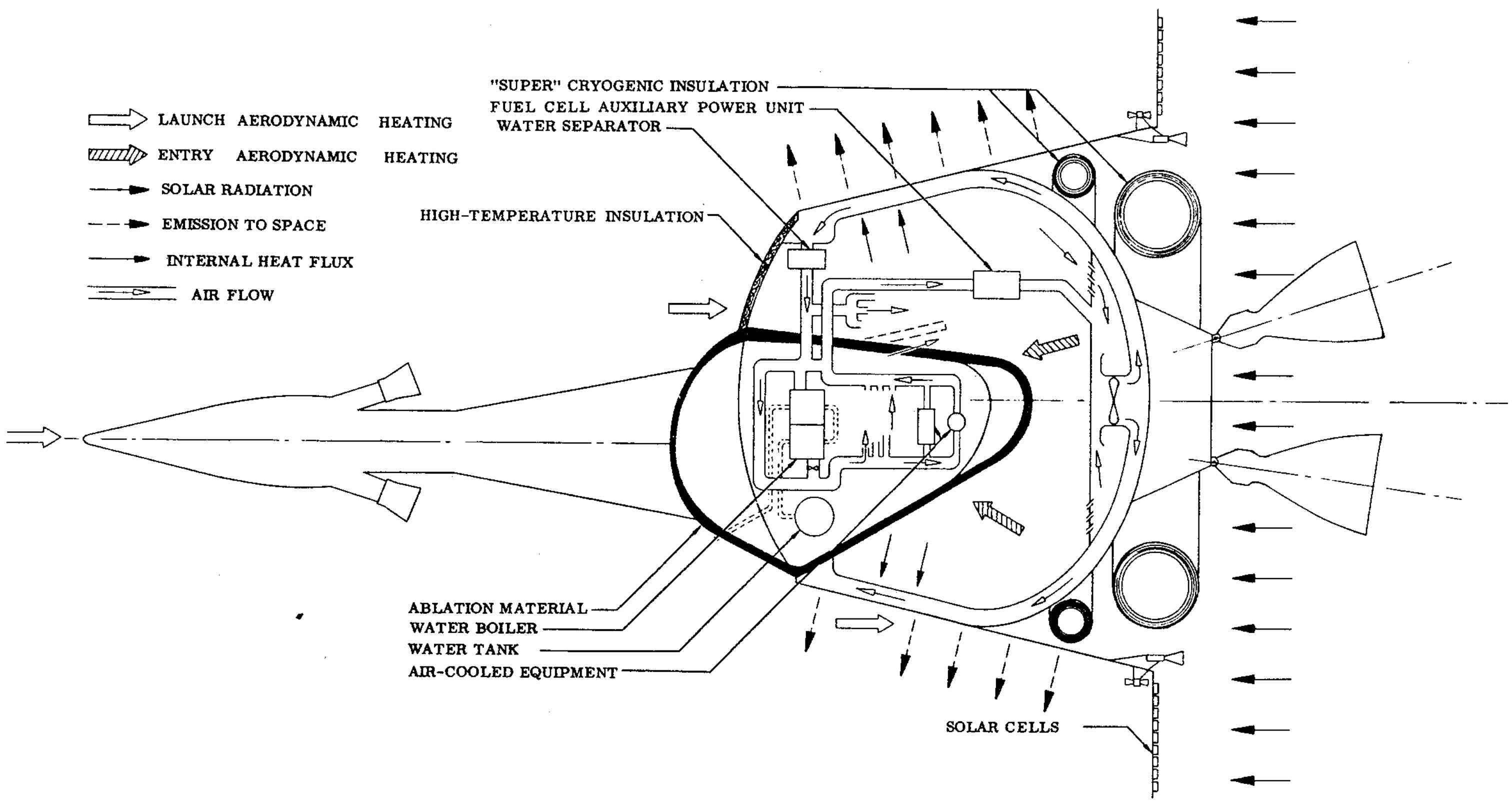


Figure 7. Passive Thermal Control Schematic

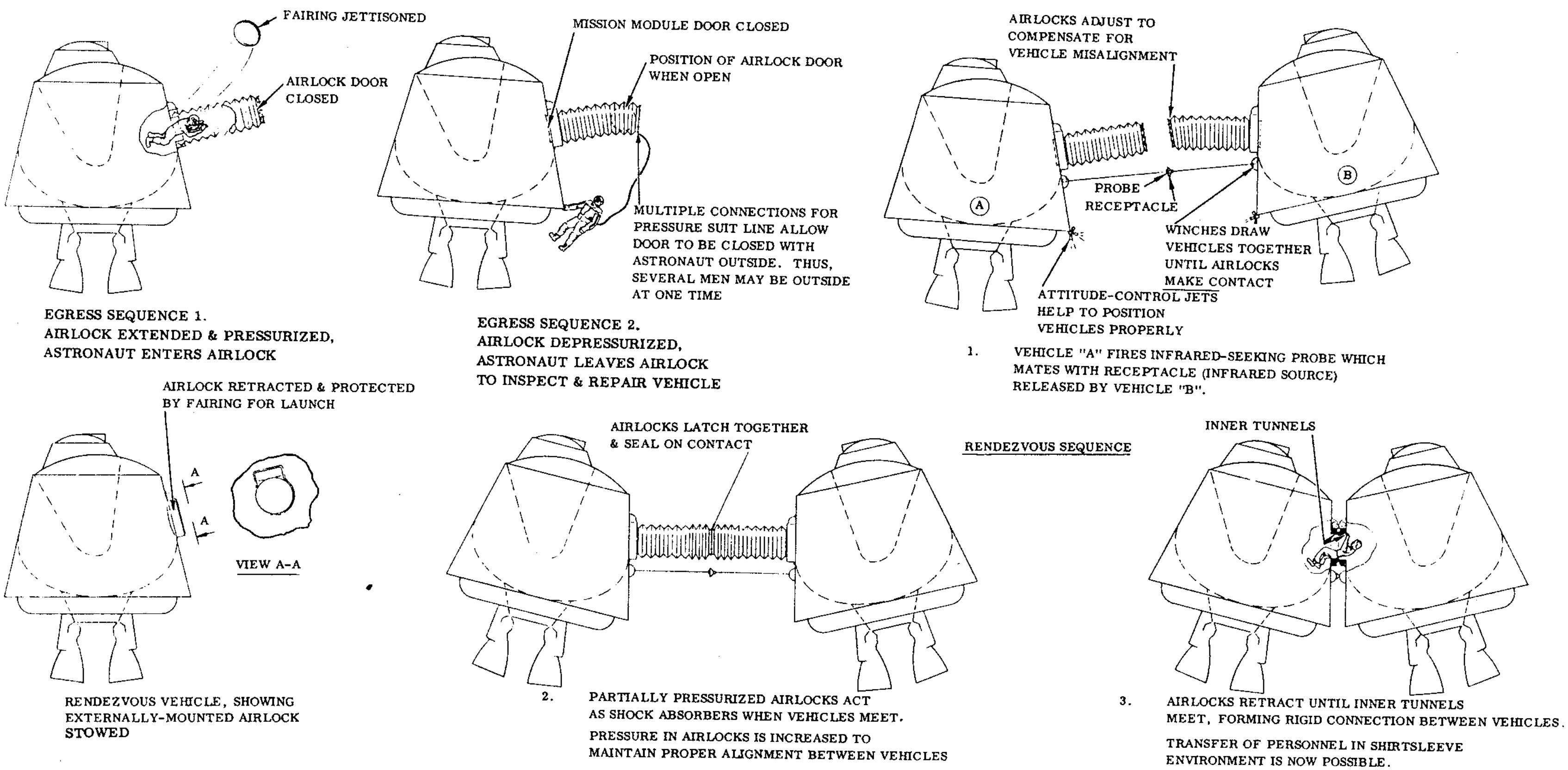


Figure 8. Rendezvous Provisions

is to eliminate possibility of damage due to collision and avoid the extreme precision. Each vehicle is equipped with one or more externally mounted probe and receptacle assemblies, whose function can best be illustrated by example:

a. Vehicle A ejects a probe attached to a cable which is unreeled from a drum mounted on the vehicle.

b. Vehicle B unreels a similar cable, on the end of which is attached a receptacle.

c. The probe, powered by a small solid propellant engine and containing an infrared seeking device, flies into the receptacle and latches to it, thus attaching the vehicles to each other.

d. Each vehicle reels in its cable, bringing the vehicles close enough together for their airlocks to make contact.

e. Each vehicle contains at least one probe and one receptacle, so that any vehicle may rendezvous with any other which is similarly equipped (If first firing misses target, probe could be reeled in, reloaded, and fired again, or, spare could be fired.)

f. Precautions must be taken that probes do not fly into and damage the vehicles. This is best accomplished by keeping the vehicles farther apart than the length of the probe cables.

2.4.1.2 Airlock Contact Procedure. Before vehicles begin the rendezvous sequence, each extends and pressurizes its airlock (mission module and airlock doors are closed), see Figure 9. As the probe and receptable cables reel the vehicles closer together attitude control jets are used to maintain cable tension and align vehicles for mating airlocks. Jets are vectored toward the mating vehicle to keep a slight separation force between vehicles and eliminate collision danger.

Additional misalignment compensation is provided by control of the angle of the airlock mating flanges (the astronaut has an omnidirectional control handle). Both pilots monitor positions of vehicles with bore-sighted TV, throughout the rendezvous procedure.

As airlocks make contact and latch together, low internal pressure allows airlocks to act as shock absorbers. When pressure seals in the mating rings are inflated, "safe" indication is registered on the vehicle instrument panel. If "unsafe" indication is noted, pressure suited astronauts enter the airlocks to inspect and realign the connection. Note that final latching and airlock functions are independent of each other. Airlocks are fully retracted by cable winchup until rigid inner tunnels make contact with door frames. The vehicles now are held together by the rigid airlocks, allowing personnel and equipment to be transferred in shirt sleeve environment, if desired.

2.4.1.3 Steps for Use of Airlock for Space Egress. Steps for space egress using airlock are:

a. Airlock extended and pressurized (doors closed) to mission module pressure. Leakage checked by pressure gage.

b. Astronaut enters airlock, mission module door is closed behind him.

c. Airlock depressurized, astronaut opens airlock door and goes outside to inspect and repair vehicle.

d. To permit the mission module and airlock doors to close with an astronaut outside, special connections must be provided for the pressure suit, and communication lines. These lines should contain a double plug, so that, before disconnecting from the umbilical inside the door, each astronaut can connect to the external umbilical.

e. In case of emergency, an unconscious astronaut could be retrieved by fellow crewmen following him outside. Rescuer could push distressed crewman into airlock and close the door from the outside. A third crewman inside the vehicle, would open the inner hatch and aid the injured man back into the vehicle.

2.4.1.4 Mechanical Advantages of Airlock. The mechanical advantages of the airlock are:

a. The minimum volume package has low drag during boost.

b. It requires no compromise of internal arrangement of either vehicle.

c. By being mounted on the mission module, it eliminates added weight and structural modifications to the re-entry vehicle.

d. External position precludes design for compressive pressure loads, thereby permitting use of a plastic-lined metal-weave bellows.

e. The estimated weight is 120 pounds.

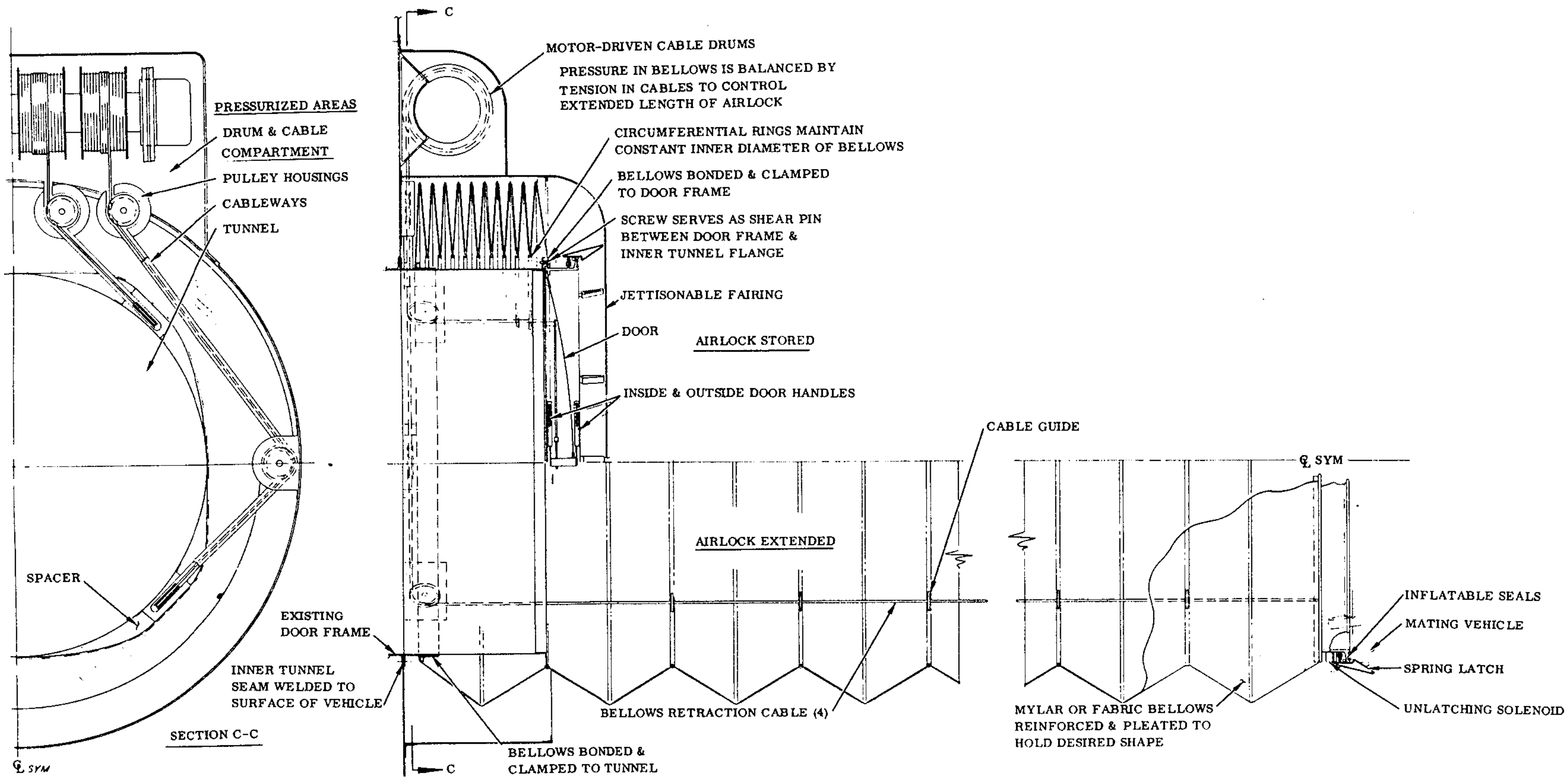


Figure 9. Rendezvous Airlock

f. Rendezvous could be performed, if only one vehicle were equipped with an airlock, providing the other vehicle had an external flange to receive airlock latches.

g. This airlock package can be used on any vehicle by mounting over the external hatch. It can be used, regardless of shape of the vehicle hatch so long as the maximum dimension fits within the airlock diameter.

2.4.2 Lunar Landing Concepts

2.4.2.1 Apollo Vehicle for Lunar Landing Design. The Apollo vehicle recommended is readily adaptable to lunar landing missions. The primary change is replacement of the circumlunar propulsion module with a larger, dual-purpose system, shown in Figure 10. The module concept is well suited to orbital assembly and merits continued study in this regard.

In paragraph 2.4.3.2, four configurations are discussed, in regard to propellant tank and engine arrangements. The design involves a multiple tank system with the lunar landing units jettisoned prior to takeoff. Advantages of the configuration selected are examined in the description which follows.

For lunar takeoff, propulsion is furnished by one 37.5K engine mounted on the central cylindrical tank. For landing, the center engine is used together with two other 37.5K engines, utilizing all tanks except the center one. Attitude control is provided by the existing engines mounted on the mission module during takeoff and by the two 4K engines during landing.

The clustered, pressure-stabilized, cylindrical tanks provide an ideal combination of light weight, reliability, and ease of manufacture. Dimensions of the cylinders and oblate spheroid bulkheads, are such that, for the tank pressures used, near minimum gage titanium alloy is adequate. For additional reliability, tanks are stiffened sufficiently so that loss of pressure will not cause vehicle damage under static ground loads.

Considerable adapter weight is saved by utilizing the H_2 landing tank to transmit vehicle loads to the booster stage below. Overall length is minimized by the internal conical tie-in structure inside the tank. These cones distribute the concentrated loads on the tank ends over the full pressurized areas, and are used both as main engine thrust structure, and to connect the lunar vehicle to the lower stage. Inter-tank shear structure is also shown.

Liquid oxygen used during landing is stored in two small tanks, to maintain weight symmetry. The spaces at 90 degree angles to the O_2 tanks, between the H_2 tanks, are used for storage of solar cells, antennae, and etc. The latter are attached to the mission module, and are available on the return trip to earth, after the landing tanks have been jettisoned.

The lunar launch tank is supported by the mission module. Conically arranged struts connect this tank to the forward adapter. Attachment of the forward adapter is the only structural change required in either of the manned modules for Apollo adaption to the lunar landing mission.

The aerodynamic fairing surrounding the tanks for earth launch provides an excellent meteorite shield. Note that the vital lunar takeoff tank, by its central position, is well protected with this arrangement.

The landing gear (see Figure 11 and Figure 12) serves as a shock absorber, leveler for sloping ground, and lunar launch platform. Unmanned lunar exploration will undoubtedly furnish information about the moon's surface which will be used to optimize landing gear design.

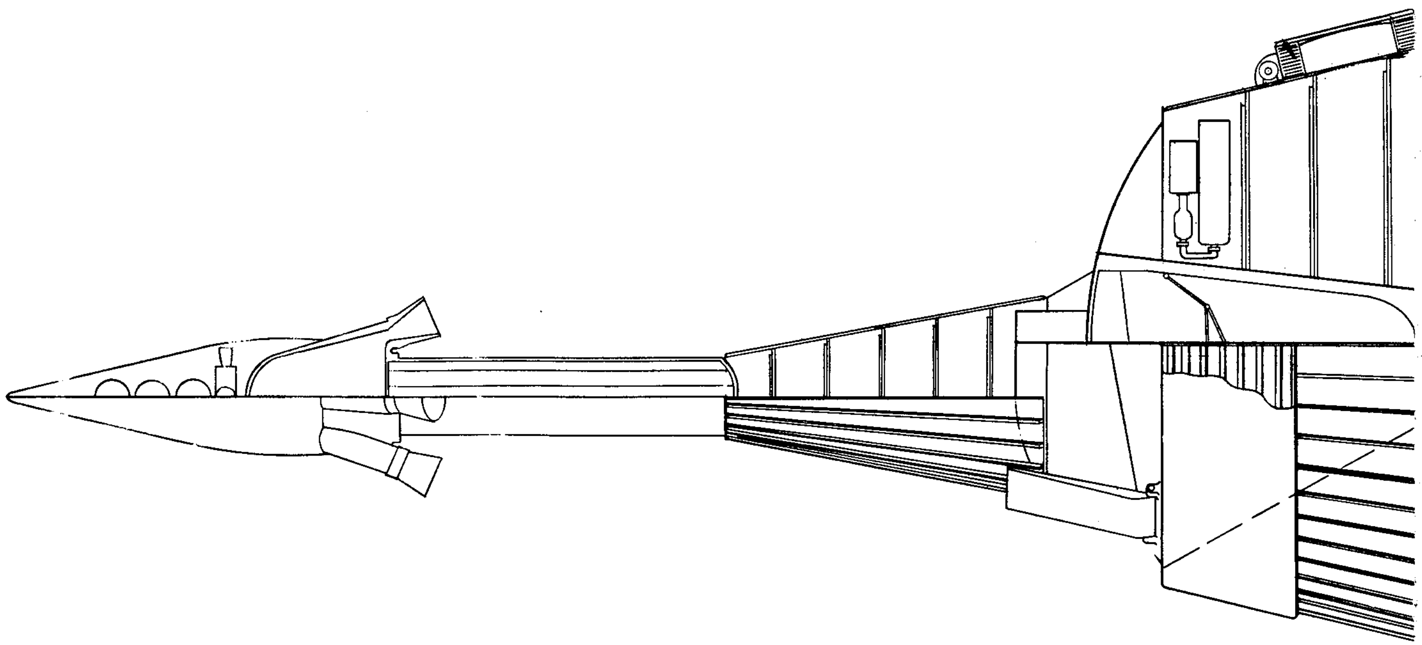
One other necessary addition to the Apollo vehicle for lunar landing is the airlock as described in section 2.4.2. The airlock may be jettisoned to lighten the return take-off load.

The weight trade-offs of a no abort versus full abort system are included. The earth launch abort capabilities of this vehicle are identical to those of the circumlunar Apollo vehicle. An abort tower for the lunar launch is not now justified because its use assumes an established lunar base.

2.4.2.2 Lunar Landing--Technique. The lunar landing considered here assumes an approach to landing from a low altitude, nominally circular orbit. Another possible approach technique is the direct approach to the moon's surface without any intermediate parking orbit. Although this latter method is somewhat more efficient, the use of a parking orbit allows greater flexibility in the choice of a landing site and allows more time for evaluation of the sites. Furthermore, the progressive approach to landing by the use of a parking orbit permits a complete evaluation of the state of the mission at several stages with the possibility of returning to the earth prior to the actual descent to the surface of the moon.

The choice of a lunar orbit will be made prior to launch and this choice will be reflected in the detailed constants of the launch guidance equations.

Similarly, midcourse guidance will correct for deviations in the earth-moon trajectory so that the spacecraft will arrive at the required location



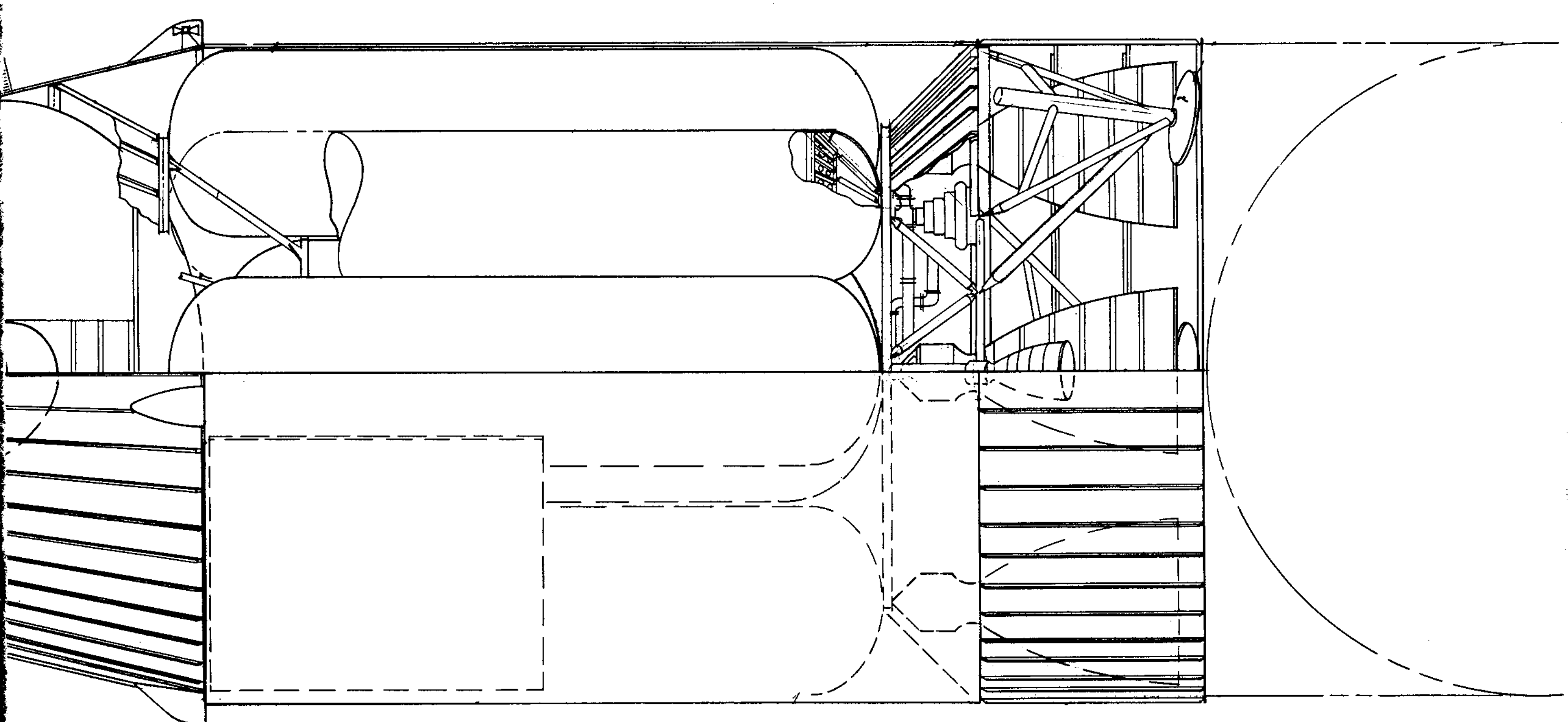
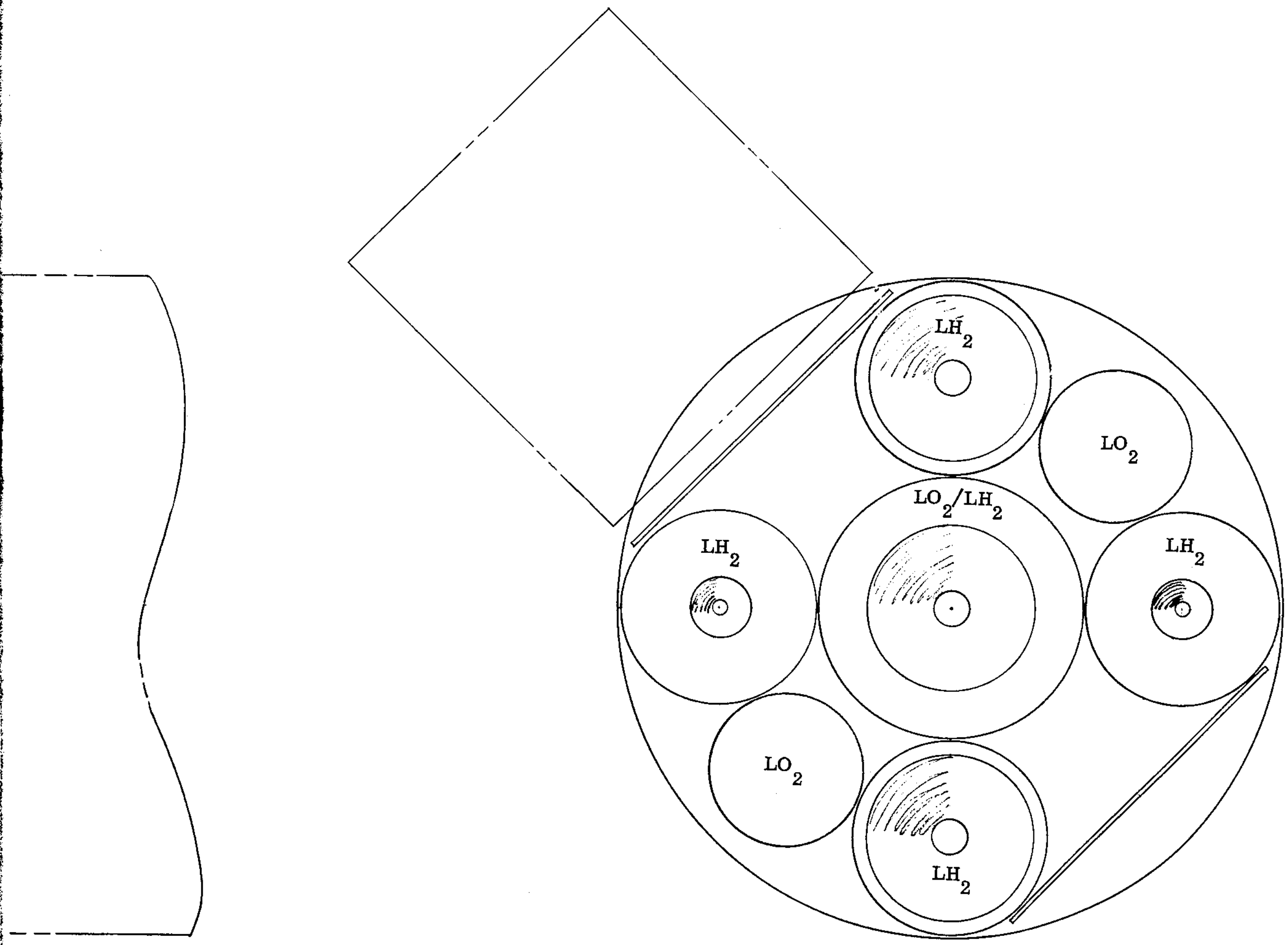


Figure 10. Apollo Lunar Landing Vehicle


LH_2
LO_2
LO_2/LH_2
LH_2
LH_2
LO_2
LH_2

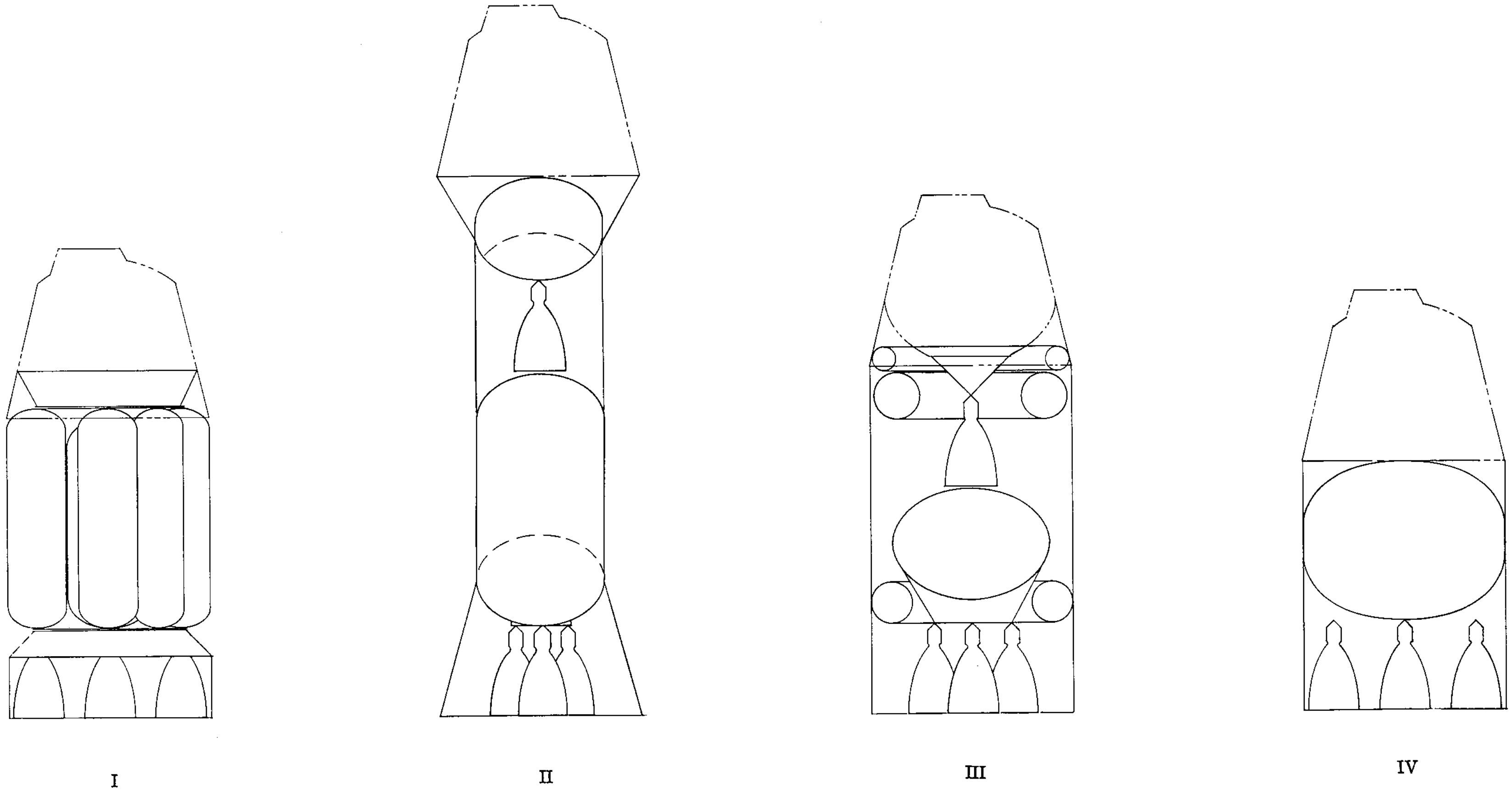


Figure 11. Apollo Lunar Vehicle on the Moon

with the required velocity for entry into the lunar orbit. In all significant aspects, the problems of launch and midcourse guidance are identical to those for a close pass orbit and the expected accuracies will also be identical. These topics are discussed in Volume III, Book 6. After entering the lunar orbit, prior to the descent to the surface, the orbit will be accurately established by observations with an altimeter and horizon scanner. These techniques will be supplemented by occultation observations and verified by visual observations of the surface of the moon. The techniques and accuracies of lunar orbit navigation are also discussed in Volume III, Book 6. During the moon orbiting phase site selection will be verified.

With an orbit accurately established, the descent to the surface of the moon is accomplished in three steps. The first step is a transfer ellipse to bring the spacecraft within a few miles (five or less) from the lunar surface. This ellipse is entered by applying a small decrement of velocity

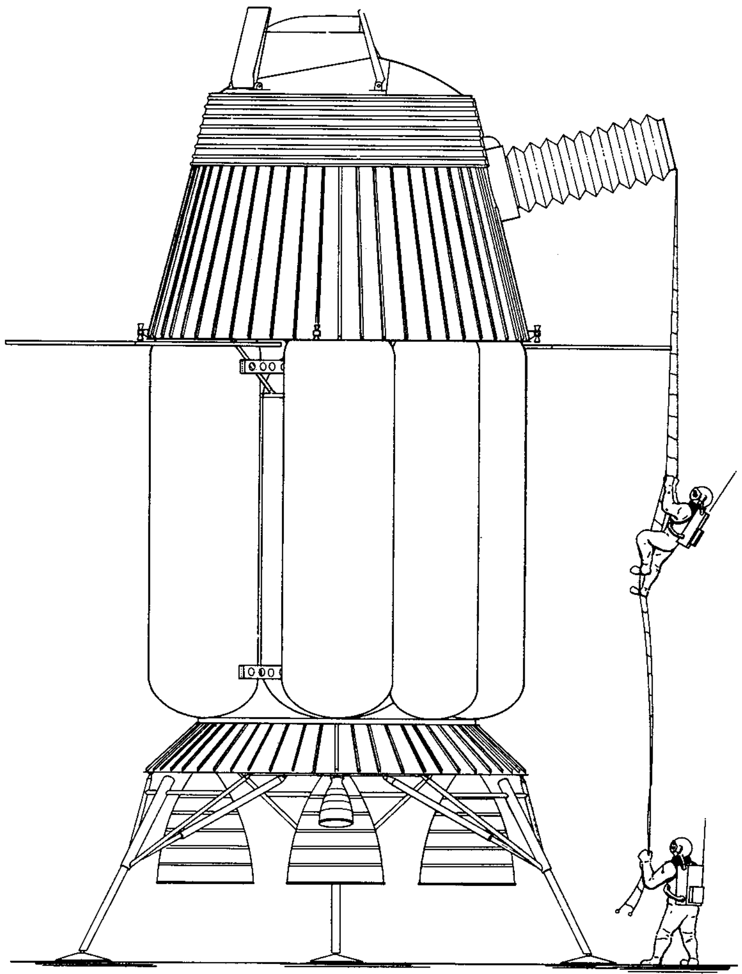


Figure 12. Apollo Lunar Vehicle Takeoff

somewhere between a quarter revolution and a half revolution from the landing area. Normally this decrement would include a small plane change to place the spacecraft at a site selected near, but not exactly on, the ground track of the lunar orbit. The exact point at which the velocity decrement is applied will be determined by the amount of plane change incorporated into the maneuver. Specifically for no plane change a Hohmann transfer would be used. For a relatively large plane change the velocity decrement would be applied closer to 90 degrees from the landing site.

The second step removes most of the velocity, which is largely horizontal, such that the vehicle descends almost vertically to the surface. For this phase and for the remainder of the landing maneuver an additional set of sensing equipment will be employed. This equipment consists of a low altitude altimeter and a multiple-beam doppler radar for measuring horizontal and vertical velocity with respect to the surface.

The third step is the final approach and the touchdown maneuver for a soft landing. During the final approach, the landing area is surveyed for possible obstacles, and any necessary maneuvering to avoid these is performed. Finally, all horizontal and vertical velocity is removed as the altitude approaches zero so that touchdown velocities of a few feet per second are achieved.

The problem of departure from the lunar surface involves nearly the same problems as departure from a lunar orbit. It is necessary to establish a reference orbit (as a function of time) which originates near the lunar landing site and terminates at a point near the earth such that re-entry and landing at a preselected site on the earth can take place. This departure trajectory has an equivalent hyperbolic excess velocity vector, $\vec{V}_{\infty}$, near the moon which is the quantity on which lunar launch guidance is based. This is discussed in much greater detail in Volume III, Book 6.

Table 2 shows the performance requirements for a soft lunar landing and return.

2.4.2.3 Comparative Configurations and Weights. The three lunar landing configurations studied, in addition to that described in the preceding section, are shown in Figure 13. Their weights are compared in Table 3.

The modular design of the Apollo vehicle makes it unnecessary to redesign it for the lunar landing mission. Only the lunar landing and launch propulsion modules need be added to the vehicle. While design of these modules is in process the basic Apollo vehicle will be proving itself in earth orbital and circumlunar flights. Larger boosters and, possibly, orbital refueling techniques will also be under development in the interim.

Configuration I, the one selected, provides the lightest practical lunar launch stage. An advantage which is reflected in earth launch propellant

Table 2. Performance Requirements for a Soft Lunar Landing and Return

ITEM	FT/SEC
Midcourse Corrections (Outgoing).	250
Enter 60-n. mi. Orbit (From 3.5-Day Trajectory).	2920
Transfer to 3-n. mi. Periselenium Orbit.	250***
Remove Horizontal Velocity at 3-n. mi. Altitude.	5370
Final Descent From 1-n. mi. Altitude.	670*
Ascent to Return Ellipse (Direct).	8850**
Midcourse Corrections (Incoming)	350
TOTAL	18,660

* Includes 100 ft/sec losses (gravity) + 150 ft/sec losses (maneuverable)
** Includes 350 ft/sec losses (gravity) + 100 ft/sec losses (maneuvering)
*** Includes small plane change

weights. Elimination of long inter-stage adapters; duplicate application of the lunar launch engine for landing, reliability inherent with multiple tanks, protection afforded the internally mounted take-off tank, and ease of manufacture all contributed to establishing this preference.

Configuration II is penalized by an unusable engine while landing, as well as long interstage adapters. Overall, it is also a much longer, less stable vehicle.

Configuration III carries the weight penalties of long interstage adapters and an unusable main engine during lunar landing.

Configuration IV was eliminated because of the weight penalty of the empty tankage carried unnecessarily at lunar launch, even though all but the center main engine could be jettisoned at this time.

A significant structural consideration is that the small diameter cylindrical tanks of configuration I allow lighter skin gages to be used than for the oblate spheroids.

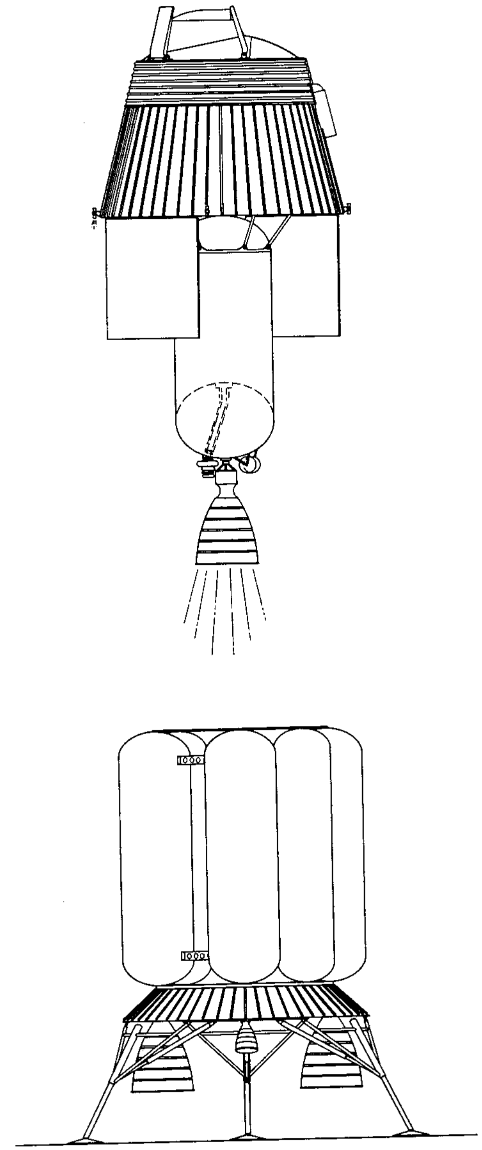


Figure 13. Apollo Lunar Landing Tank Configurations

Table 3. Lunar Landing - Apollo Weight Comparisons

DESCRIPTION	CONFIGURATION I (WEIGHT IN POUNDS)	CONFIGURATION II	CONFIGURATION III	CONFIGURATION IV
Lunar Launch Stage	(1723)	(2015)	(1865)	(3400)
Structure (includes tanks and insulation)	583	875	695	1239
Propulsion	550	550	550	550
Pneumatics	190	190	190	350
Hydraulics	41	41	41	41
Electrical	10	10	10	25
Propellant Utilization	35	35	40	35
Residuals	314	314	339	1160
Lunar Landing Stage	(5026)	(5537)	(5218)	(2163*)
Structure	1913	2037	1718	250
Propulsion	1470	1970	1970	1470
Pneumatics	380	380	380	300
Hydraulics	83	115	115	83
Electrical	25	25	25	10
Propellant Utilization	89	60	60	--
Residuals	1066	950	950	50

*Configuration IV shows 1116 lb less inert weight than Configuration I for lunar landing. Landing weight of Configuration IV would actually be higher than Configuration I, if recomputed propellant weights were included.

A mission weight profile is included for the Apollo lunar landing vehicle in Table 4. Table 5 defines weights nomenclature.

An Apollo lunar landing vehicle weight breakdown is included as Table 6.

2.4.2.4 Propulsion Aspects. The main objective of this phase of the study was to find combinations of lunar landing and launch modes which minimize stage sizes and guidance difficulties while providing a maximum in crew safety. Due to the absence of an atmosphere on the moon and its weak gravitational field the technical problems of manned lunar missions such as stage optimization, landing and launching, selection of trajectories safety aspects, etc., differ substantially from those for manned earth satellite missions. The reliability of manned lunar missions is of such importance that all resources and ingenuity must be applied to create safe operational concepts and systems.

Table 4. Weight Analysis - Configuration I

ITEM	APOLLO LUNAR LANDING VEHICLE (WEIGHT IN POUNDS)		
Based on	10,406 Pound Payload		
	1,653 Pound Lunar Launch Step		
	5,026 Pound Lunar Landing Step		
Entry Vehicle:	W_{ev}	=	5,700
Mission Module Jettison	W_j	=	3,606
Stage Burnout	W_b	=	9,306
Moon-Earth Midcourse Correction	W_p	=	340
Return Vehicle Initial	W_i	=	9,646
Lunar Launch Jettison	W_j	=	1,723
Lunar Launch Burnout	W_b	=	11,369
Lunar Launch Propellants	W_p	=	10,714
Lunar Launch Initial	W_i	=	22,083
Propellant Boiled-off	W_{pb}	=	700
Landing Weight of Lunar Launch Stage	W_{ll}	=	22,783
Jettison on Moon	W_j	=	5,026
Lunar Landing Vehicle Burnout	W_b	=	27,809
Propellant Boiled-off	W_p	=	25,982
Lunar Landing Vehicle Initial	W_i	=	53,791

Table 4. Weight Analysis - Configuration I (Cont)

ITEM	APOLLO LUNAR LANDING VEHICLE (WEIGHT IN POUNDS)		
Propellant Boiled-off	W_{pb}	=	2,500
Earth-Moon Vehicle Burnout	W_b	=	56,291
Jettison	W_j	=	0
Earth-Moon Midcourse Correction	W_p	=	2,234
Earth-Moon Vehicle Initial	W_i	=	58,525
Earth Escape Weight	W_{ee}	=	58,525

It is most efficient to use liquid hydrogen as fuel for both the launch and landing stages because:

a. The propellant loss caused by boiloff is small.

b. The powered flight time is relatively long which eases terminal guidance problems.

The optimum initial thrust-to-weight (earth pounds) ratio for lunar landing lies in the range of 0.5 to 0.6. The powered descent time is the order of six minutes. The starting altitude is sufficiently high to permit a circumlunar abort trajectory in case of landing propulsion failure. The reduction in gravity losses by increasing thrust is offset by the higher weight of the propulsion system. Lowering the thrust-to-weight ratio below 0.5 will increase gravity losses.

It appears that constant thrust descent rockets have advantages over variable thrust engines (higher specific impulses, increased reliability, simpler design, etc.). One or two high thrust engines with restart capability provide the main retrothrust for the descent phase. Thrust control for landing is obtained by swiveling two or more smaller engines. Their combined thrust is approximately one-sixth of the total thrust of the landing stage. They would also be used for midcourse orbit corrections on the earth to moon trip. The proposed method of terminal thrust control would minimize whirling up dust from the moon's surface.

Table 5. Weights Nomenclature

W_j	=	Jettison (inert) weight of stage in pounds.
W_b	=	Burnout weight of stage in pounds.
W_p	=	Propellant weight of stage in pounds.
W_i	=	Initial weight of stage in pounds.
W_{ev}	=	Entry vehicle weight after lunar escape (includes 6 astronauts)
W_{pb}	=	Propellant boiled off during transfer and/or on the moon. (Boil off of lunar launch stage propellant during earth to moon transfer is very small and is included in the inert weight of the landing stage.)
W_{ll}	=	Landing weight of the lunar launch stage (includes a crew of 3)
W_{ap}	=	Weight of lunar abort propulsion system, which is landed on the moon by a cargo rocket and therefore is not included in the payload of the manned lunar landing stage.
W_{3m}	=	Weight of 3 crew members with space suits.
W_{ee}	=	Earth escape weight of lunar vehicle.

NOTES: The lunar landing stage uses its low thrust rockets for midcourse orbit corrections during the transfer from earth to moon (the abort propulsion system is used for the return trip). All calculations are based upon realistic estimates of specific impulses, trajectory losses, guidance requirements and vehicle empty weights.

The powered descent flight considered can be approximated closely by a gravity turn trajectory. It is most efficient to remove horizontal position errors as early as possible. For the considered launch date, flight time, and landing point the landing trajectory is almost 20 degrees with respect to the local vertical.

The lunar launch stage has an adequate performance potential to return the whole re-entry vehicle to the earth in the event of landing propulsion failure occurring at the start of landing.

The optimum lunar launch stage thrust-to-weight (earth pounds) ratio lies in the range of 0.95 to 1.1. Some rocket engines used for the landing stage, can be used for the launch stage.

Lunar take-off abort is included to indicate the high weight penalty involved. The critical velocity for the full abort capability is approximately 5800 ft/sec. At this time the abort propulsion system can either return the whole re-entry vehicle to the earth on a four-and-a-half day transfer orbit or land the crew capsule on the moon. Or it can orbit the vehicle around the moon. Prior to the critical point the crew capsule will be landed back on the moon. It could also be propelled into the vicinity of the earth and brought into a high-altitude geocentric elliptic orbit (with a period of less than half a day) from which the crew could be rescued by a rendezvous operation. With approximately 25 percent more abort system propellant the crew capsule could be returned to the lunar base after an abort even at the critical point on the launch trajectory.

2.4.2.5 Lunar Landing Trajectories. The lunar landing problem is interrelated with other aspects of lunar missions such as earth-moon transfer orbits, navigational and guidance aspects, landing-location terminal-guidance methods, vehicle designs, propulsion systems, etc.

The transfer orbit and the prescribed landing location are the most important parameters of the landing trajectory. Both decide the energy requirements of the landing propulsion and the character of the flight path. For the considered landing point near the center of the moon's visible part the hyperbolic arrival trajectory of the two-and-a-half day transfer orbit is presented.

The hyperbolic arrival and departure orbits at the moon vary with the calendar date as discussed elsewhere in this report. The launch and landing vehicles of the operational system must be designed for the most unfavorable constellation conditions. Presently, detailed studies are in progress which will determine these conditions. The proposed system has been designed for a typical launch date in September 1968 with an ample performance reserve for unfavorable conditions. The selenocentric approach and departure orbit energies are not expected to vary significantly with launch date, but the flight path angles at the start of powered descent or ascent flight may change, noticeably affecting the powered flight program.

2.4.2.5.1 Maneuverability. The required maneuverability of the lunar landing stage depends on the nature of the approach orbit, midcourse

Table 6. Apollo Lunar Landing Vehicle Weight Breakdown

ITEM	(WEIGHT IN POUNDS)	
Lunar Landing		(5026)
Structure		(1913)
Tank Skins	394	
Doublers	20	
Forward Rings	30	
Access Doors	18	
Eng Thrust Build-up Bulkhead	100	
Thrust Cylinders	100	
Rings - Engine	40	
Fill and Drain System - Fuel	60	
Fill and Drain System - Oxidizer	26	
Mountings	90	
Insulation	215	
Tank Tie In Structure	150	
Aft Adapter	210	
Forward Adapter	160	
Landing Structure	300	
Propulsion		(1470)
Engines - Main	1000	
Engines - Roll	340	
Lines and Valves	130	
Pneumatics		(380)
Bottles and Controls	300	
Pneumatic Hardware	80	
Hydraulics		(83)
Actuators	45	

Table 6. Apollo Lunar Landing Vehicle Weight Breakdown (Cont)

ITEM	(WEIGHT IN POUNDS)	
Power Package	20	
Fluid and Accumulator	10	
Plumbing	8	
Electrical		25
Propellant Utilization System		(89)
Sensing Assembly	40	
Computer	7	
Plumbing	30	
Miscellaneous	12	
Residuals		(1066)
Oxidizer - Gaseous	300	
Liquid in Lines and Sump	150	
Fuel - Gaseous	200	
Liquid in Lines and Sump	100	
PU Error	300	
Helium in Tanks	12	
Helium in Bottles	4	
Lunar Launch		(1723)
Structure		(583)
Tank Skins	139	
Doublers and Splices	12	
Fluid Ring	5	
Tank Access Door	3	
Engine Thrust Structure	50	
Thrust Cylinder	50	
Ring - Engine Mounting	20	

Table 6. Apollo Lunar Landing Vehicle Weight Breakdown

ITEM	(WEIGHT IN POUNDS)	
Fill and Drain System - Fuel	15	
Fill and Drain System - Oxidizer	13	
Mountings	15	
Insulation	69	
Forward Adapter	122	
Intermediate Bulkhead Insul Ret	40	
Jettison Rails	30	
Propulsion		(550)
Engine	500	
Lines and Valves	50	
Pneumatics		(190)
Bottles	150	
Pneumatic Hardware	40	
Hydraulics		(41)
Actuators	20	
Pwr Package	12	

Table 6. Apollo Lunar Landing Weight Breakdown (Cont)

ITEM	(WEIGHT IN POUNDS	
Fluid and Accumulator	5	
Plumbing	4	
Electrical		(10)
Harnesses	10	
Propellant Utilization System		(35)
Sensing - Assembly	13	
Computer	7	
Plumbing	9	
Miscellaneous	6	
Residuals		(314)
Oxidizer - Gaseous	100	
Liquid In Lines and Sump	50	
Fuel Gaseous	60	
Liquid In Lines and Sump	25	
PU Error	75	
Helium	4	

guidance accuracy, vehicle design, propulsion capability and performance potential, and the terminal guidance accuracy. Maneuverability requirements are lower if:

a. Position error prior to powered descent is small.

b. Powered flight time is long.

c. Efficient terminal guidance system is used.

d. Accurate position and velocity information is available.

e. The propulsion system responds rapidly to correction signals.

f. A reasonable impact velocity is permissible.

Most of these desired mission features are not too difficult to realize. A position error of 5 to 10 miles at an altitude of several hundred miles above the moon which is not too stringent on midcourse guidance requirements can be tolerated. The hydrogen-oxygen propelled-landing stage has a relatively long burning time (approx. six minutes) in the range of optimum thrust-to-weight ratios. Terminal guidance problems may not be serious if one or more moon-based radio transmitters are used. The long descent time eases the problem of rapid propulsion system response so that efficient and reliable constant-thrust rocket engines can be used.

A few on-off thrust operations of the main propulsion unit appear to be sufficient. The vernier thrust control can be provided by one or two pairs of gimbaled or swiveled constant-thrust engines. These low-thrust auxiliary engines serve simultaneously for attitude and roll control during landing and provide midcourse guidance correction. In the event of failure of the main landing engines during the transfer to the moon (determined by monitoring devices and occasional testing) the auxiliary engines could be used for starting an early circumlunar abort orbit, so that the launch stage would have a large propellant reserve for the return trip to earth. This reserve could be used for reducing the approach velocity and also for obtaining desirable re-entry and landing conditions. The same statement can be made for failure of the auxiliary engines.

The vehicle is assumed to arrive at the moon on an oblique free-fall trajectory. At the proper altitude given by the optimum landing propulsion the previously checked-out engines apply retrothrust either for lunar orbit or to lower the vehicle to the landing site along a trajectory which in the ideal case is a gravity turn. In practice it will not be possible to follow an exact gravity turn because of the presence of initial position-velocity and timing errors. Also the variation in arrival trajectories with launch time and the pre-selected thrust levels must be considered. Therefore the gravity turn trajectory will be replaced by a powered flight program involving pitch rates, coasting phases, horizontal velocity increments, etc. For the determination of landing propulsion requirements, a gravity turn launch can be used with good approximation if the initial flight path angle is large. The thrust misalignment losses can be estimated and added to the gravity losses.

Horizontal position errors should be corrected at a high altitude. Figure 14 shows the velocity increments needed for horizontal displacements as a function of altitude, assuming a simplified model which does not consider misalignment losses.

Figure 15 shows the velocity required for horizontal displacements of a hovering vehicle near the moon's surface as a function of flight time.

2.4.2.5.2 Thrust-Weight Considerations. The optimum thrust level and variability of the lunar landing stage depends on the initial trajectory conditions constrained by terminal guidance aspects which prefer a long powered descent time.

It has been found that a thrust variability of ± 15 percent is adequate if other guidance criteria are met. This variability can be replaced by on-off operations of a constant-thrust main engine in combination with smaller variable thrust auxiliary engines.

The landing trajectory was simulated on a digital computer after making these simplifying assumptions; all forces on the vehicle are coplanar, and the thrust vector always points to the vehicle center of gravity. Constant retrothrust was applied to the vehicle flying a gravity turn trajectory until it reached a velocity of zero at the prescribed landing point. In practice, one would aim to reach zero velocity at a certain altitude above the landing point, which is a function of the terminal guidance accuracy, timing errors, in thrust application, errors in thrust levels, etc. From this altitude the vehicle would then hover and let down to the landing point. Realistic assumptions for the velocity losses of this landing method have been included. The gravitational effects on the landing trajectory are similar for higher thrust-to-weight ratios. For a vertical touchdown with low retrothrust, a gravity turn approach with or without superimposed small pitch rates is more efficient than a vertical landing trajectory. (If the thrust is high, the opposite is true.) Errors in arrival velocity and altitude of 5 percent correspond to a 5 percent change in landing weight.

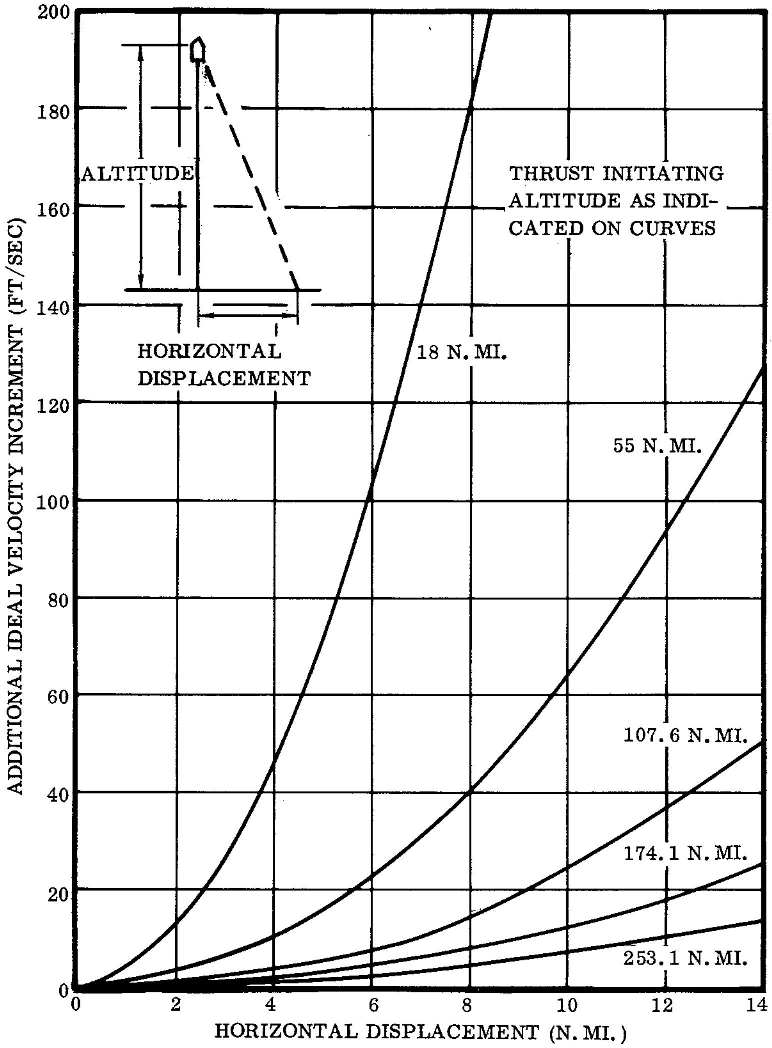


Figure 14. Velocity Increments Required for Horizontal Maneuvers Starting at Different Altitudes

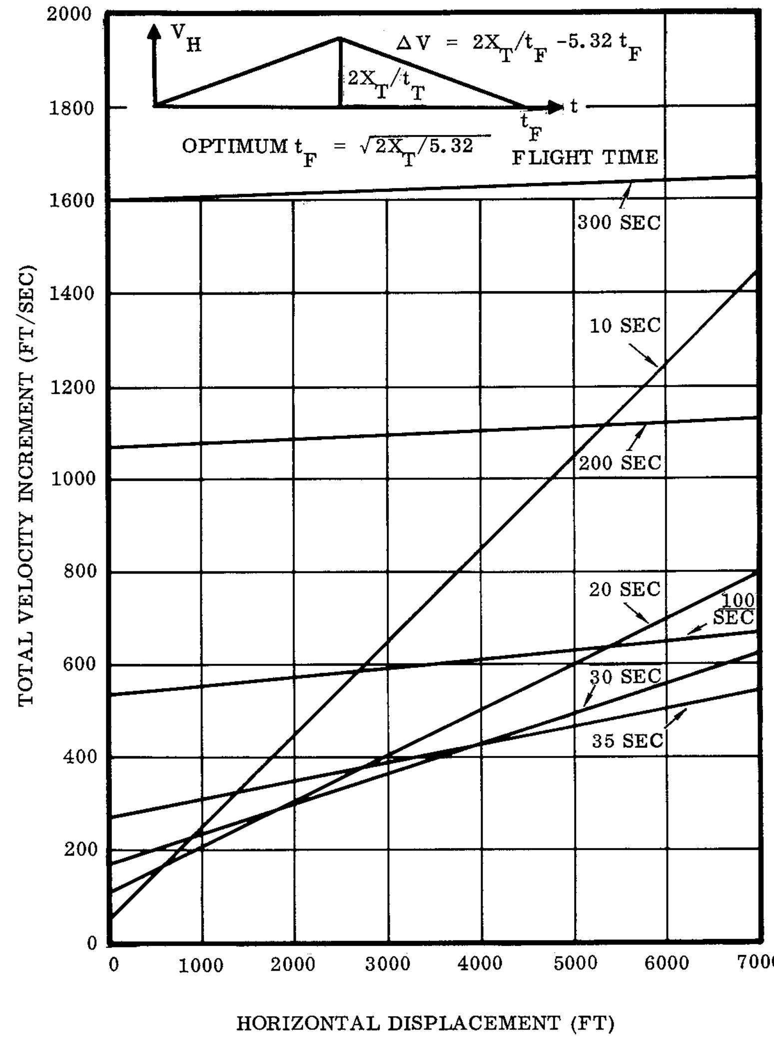


Figure 15. Velocity Requirements for Horizontal Motion Near the Moon's Surface

Figures 16 and 17 show vertical landing trajectory parameters for two specific impulses for reference purposes. The desired burning time of 6 to 7 minutes is obtained by an initial thrust-to-weight ratio of 0.5 to 0.6 for the case of hydrogen-oxygen propellant with a specific impulse of I_{sp} = 420 seconds. The gravity losses are approximately 1400 to 1600 ft/sec. The initial powered flight velocities are a function of altitude as given by the hyperbolic approach velocity shown in Figure 18.

Figure 19 shows gravity turn landing trajectory histories for initial thrust-to-weight ratio of 0.58. For this ratio the gravity loss is 1600 ft/sec. To this a misalignment loss V_{ml} must be added, which can be estimated by:

$$V_{ml} = \frac{T}{W_{AV}} t_B \; 1\text{-}\cos A_V \; (A_V)$$

Where:

T = Thrust,

W_{AV} = Average weight in earth pounds,

t_B = Burning time, and

A_V = Average swivel angle

It can be seen that a combined angle of attack and gimbal angle of 30 degrees and an average thrust-to-weight ratio of 0.9 and a burning time of 6 minutes cause a misalignment loss of only 50 ft/sec.

2.4.2.4.3 Mass Ratios for Lunar Landing. Hydrogen-oxygen propellant is the combination which will most likely be used for lunar landings. A conservative estimate of an I_{sp} of 415 seconds has been used in computing the mass ratios. The ideal velocity requirement for the landing operation is shown below.

V_i = hyperbolic arrival velocity,

V_{g1} = gravity loss,

V_{m1} = misalignment loss,

V_{guid} = velocity needed for terminal guidance corrections,

V_{hov} = velocity needed for hovering

V_{res} = small velocity reserve.

For a thrust-to-weight ratio of 0.55 the corresponding altitude for starting a powered-descent gravity turn is approximately 1,700,000 feet (see Figure 19). At this altitude the hyperbolic approach velocity is approximately 7,800 ft/sec (Figure 18).

The gravity loss is V_{g1} = 1,600 ft/sec (Figure 19). Misalignment losses $V_{m1} \cong$ 50 ft/sec. A terminal guidance velocity reserve of 500 ft/sec appears to be adequate; 500 ft/sec are reserved for hovering and horizontal maneuvers. A 250 ft/sec velocity reserve should cover increases in arrival velocities at more unfavorable launch dates and higher losses.

The total ideal velocity needed is:

$$V_i = 7,800 + 1,600 + 50 + 500 + 250 = 10,200 \text{ ft/sec}$$

The mass ratio is m = 2.23 for an I_{sp} = 415 seconds.

This mass ratio has been used to compute the landing vehicle stage size of example configurations shown.

The maximum deceleration during landing is of the order of one earth g.

2.4.2.5.4 Lunar Landing Abort Considerations. The lunar landing-stage propulsion system will be monitored continuously for signs of trouble during the last few hours of the transfer orbit. About 300 miles above the moon's surface a short thrust period checks proper operation. If one or more of the landing engines does not work, the landing stage could be jettisoned immediately. The remaining configuration could then be reoriented and the launch stage would propel the vehicle into a circumlunar orbit and then back to earth. As discussed previously, either two or three velocity impulses must be used to get into the proper moon-earth transfer orbit.

In order to convert an impact trajectory to a circumlunar trajectory, several orbit maneuvers are possible. The immediate concern of the crew is to clear the moon at a safe distance. The economical method with respect

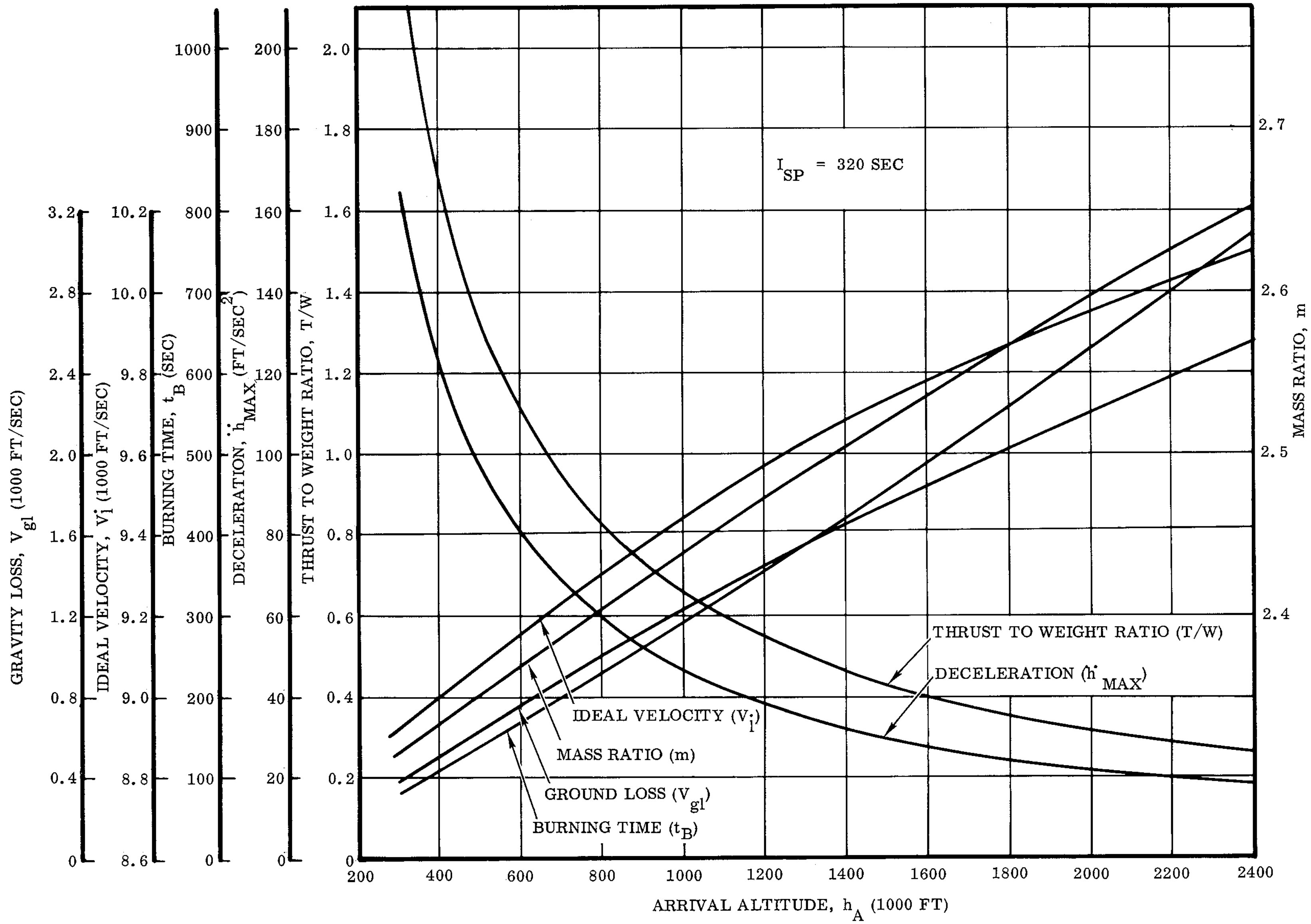


Figure 16. Vertical Lunar Landing Trajectories for Example Configurations

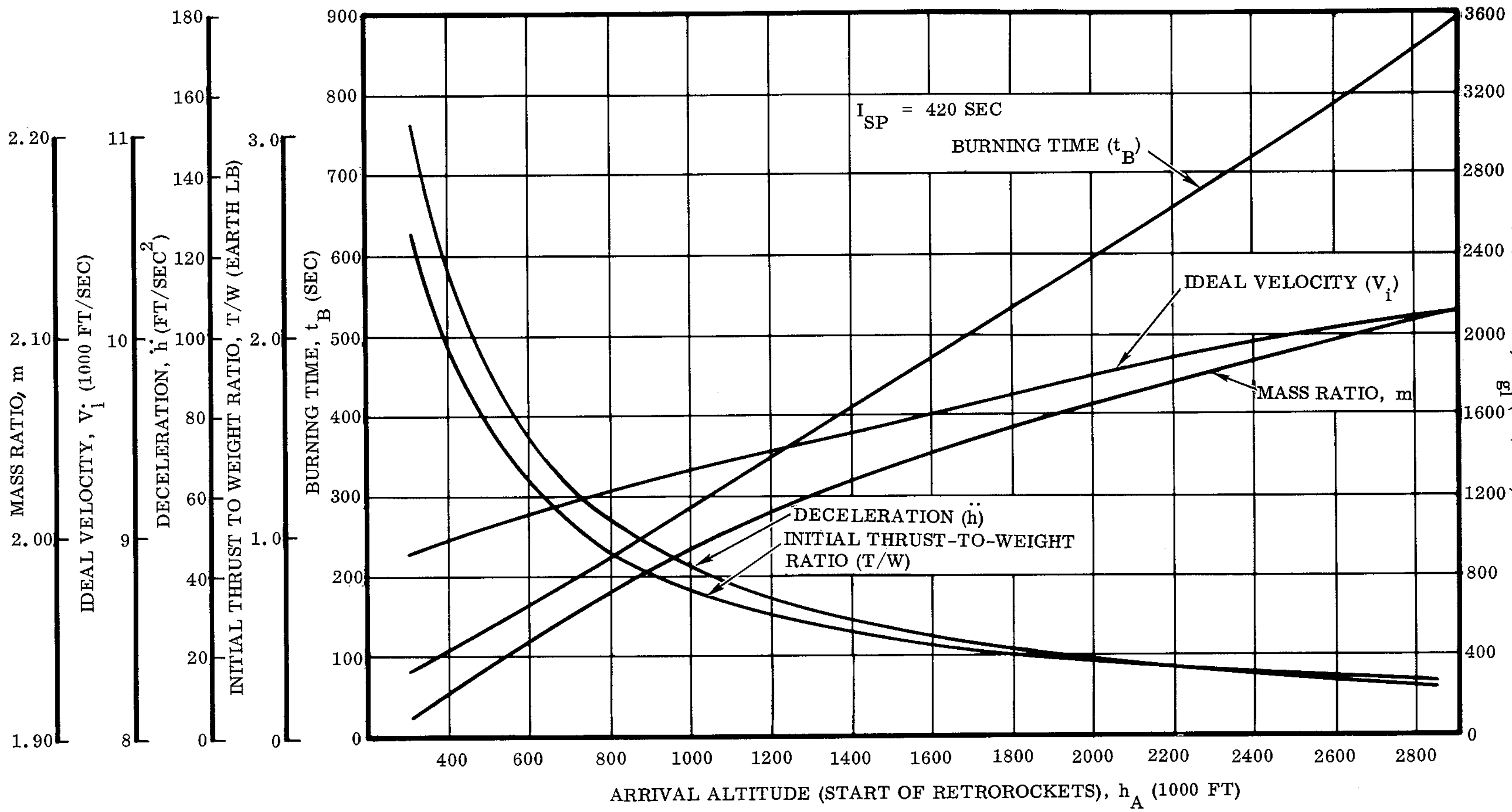


Figure 17. Lunar Landing Trajectories for Example Configurations

to propulsion, is to increase the angular momentum of the approach orbit while leaving its energy unchanged, i.e., by rotating the velocity vector. The semi-major axis of the hyperbolic pass orbit is the same as that for the collision trajectory.

The velocity increment for this instant maneuver is given by

$$\Delta V_I = 2V_A \sin \frac{\Delta \gamma}{2}$$

where:

V_A = arrival velocity.

$\Delta \gamma$ = change in flight path angle (with respect to the local horizontal).

$\Delta \gamma$ is given by $\Delta \gamma = \gamma_A - \gamma$

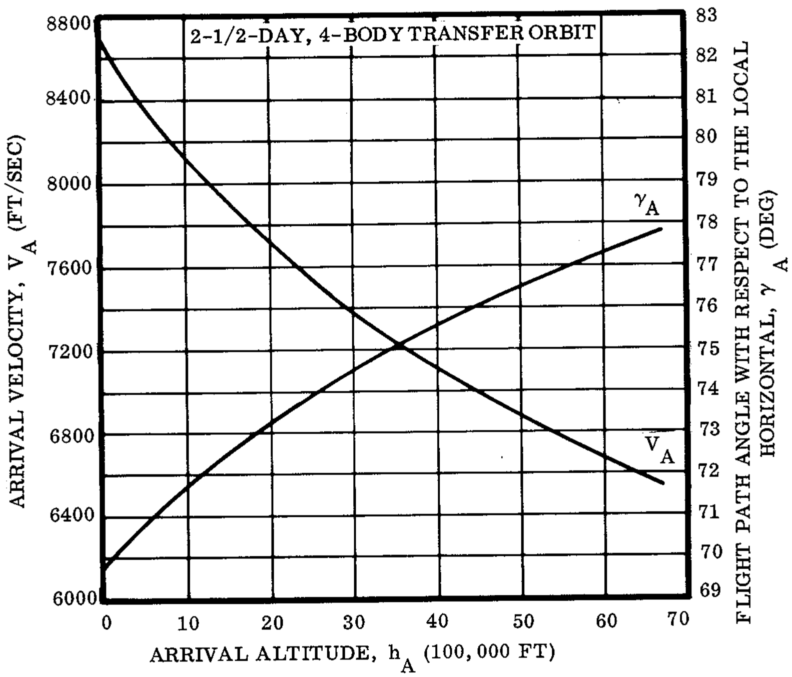


Figure 18. Earth-to-Moon Ballistic Flight Impact Trajectory for Example Configuration

where γ_A is the arrival flight path angle and γ the new flight path angle.

The new flight path angle is given by

$$\cos \gamma = \sqrt{\frac{r_p}{r} \quad \frac{2a - r_p}{2a - r}}$$

where r_p if the desired perselenion distance (lunar radius plus perselenion altitude), r the radial distance at the time of abort and 2a the major axis of the hyperbolic approach orbit determined by the Vis-Viva equation.

For a typical abort at 200 n. mi. altitude with a velocity of approximately 8050 ft/sec for a perelenion altitude of 100 n. mi. about 10 percent must be added to cover gravity and misalignment losses. The higher the abort altitude, the smaller the velocity impulse. The above calculation assumes an instantaneous impulse but the burning time is large enough to lose a substantial amount of altitude during burning time (107 seconds for $\frac{T}{W} = 1.0$, $I_{sp} = 420$, $V_1 = 4,000$ ft/sec, and the loss in altitude is 20 to 25 n. mi.)

At the perselenion point a retro-impulse of 3,600 ft/sec would allow the vehicle to achieve circular speed. After half a revolution a third impulse of 3,000 ft/sec propels the vehicle into a three-day moon-earth transfer orbit. The total velocity required is therefore approximately 10,900 ft/sec, which is 10 percent above the capability of the lunar launch stage. Therefore the perselenion distance must be decreased and the circular orbit portion must be replaced by an elliptical orbit. An error analysis for the calculation of the required abort maneuver accuracy has been carried out.

2.4.2.6 Lunar Launch Trajectories. The powered lunar-launch trajectory must approximate as closely as possible the initial conditions for the unpowered transfer trajectory from moon to earth. Figure 20 shows the departure conditions in terms of velocity and flight path angles as a function of altitude for 2-1/2 day transfer orbit beginning on 9 September 1968 at 2:28 am at an altitude of approximately 100 n. mi. above the lunar base.

The departure conditions vary with the launch date. A detailed study should be carried out to determine the most unfavorable return conditions from a propulsion point of view.

2.4.2.6.1 Thrust-to-Weight Ratio Considerations. The selection of thrust-to-weight ratios for launching is far less critical than for landing and therefore it is possible to use a pair of engines identical to the main landing engines. The thrust-to-weight (earth pounds) ratio is then close to one which keeps the gravity losses sufficiently low, and the burning time is acceptable. Figures 21 and 22 show the gravity loss burnout time, altitude and acceleration of a vertical launch trajectory as a function of initial thrust-in-weight ratios for storable propellant and oxygen-hydrogen launch stages.

According to Figure 20 the velocity and flight path angle for the 2-1/2 day return trip at the stated launch date is 8600 ft/sec and 54 degrees, respectively, for a burnout altitude of approximately 700,000 feet.

Figure 23 shows the time history of a launch trajectory for a hydrogen-oxygen propelled vehicle which follows a gravity turn program.

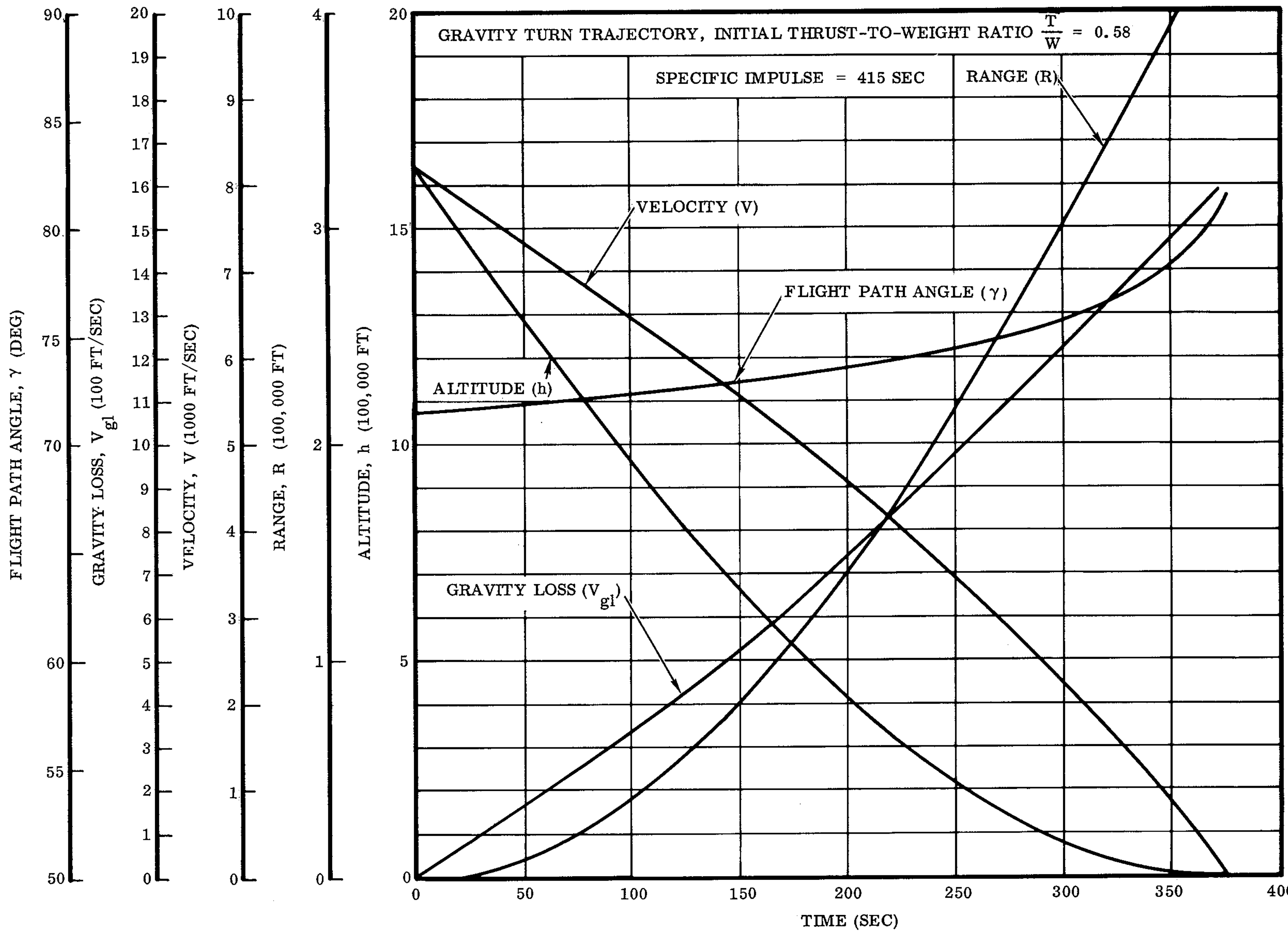


Figure 19. Lunar Landing Trajectory--Powered Flight Time History for Example Configuration

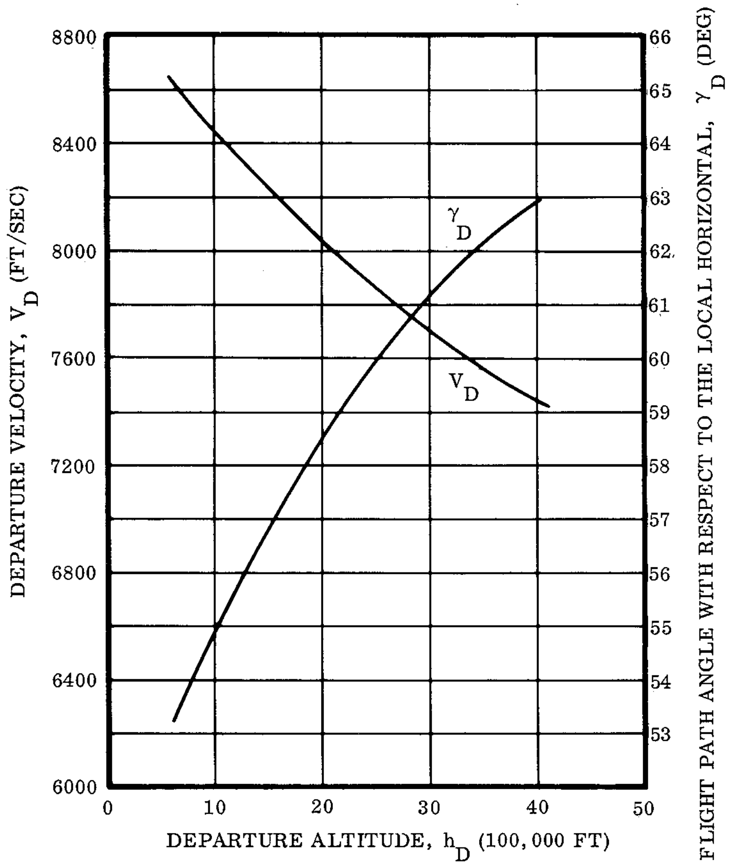


Figure 20. Moon-to-Earth Ballistic Flight Departure Trajectory

1 STORABLE PROPELLANT: (UDMH-HYDRAZINE-N_2O_4), I_{SP} = 320 SEC

2 BURNOUT VELOCITY, $V_B \approx$ 8900 FT/SEC

3 VERTICAL LAUNCH TRAJECTORY, γ_B = 90°

4 h_B = 31 x 10^5

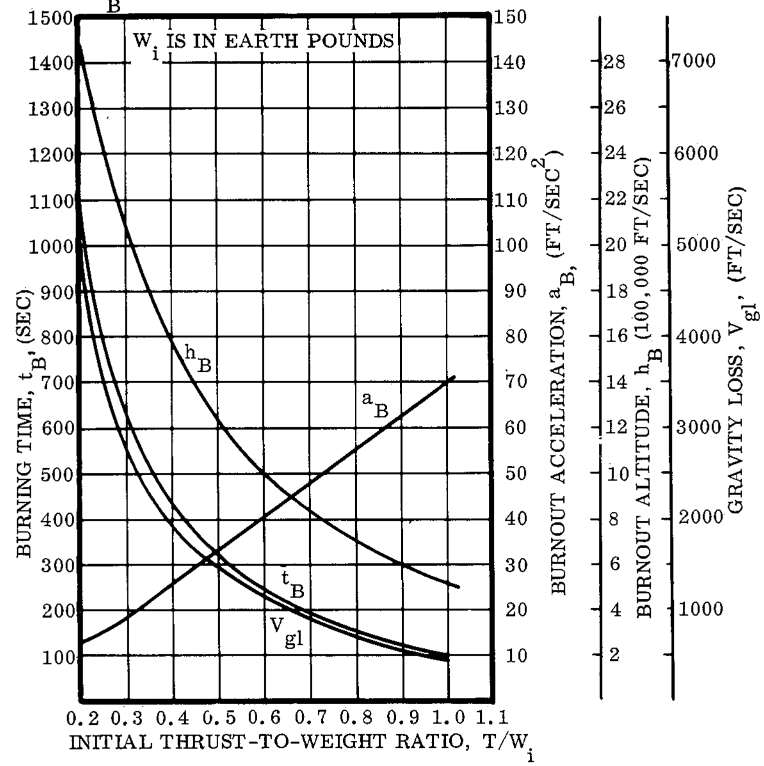


Figure 21. Lunar Launch Trajectories, I_{SP} = 320 Seconds

The ideal velocity is given by

$$V_i = V_b + V_{gl} + V_{m1} + V_{guid}$$

where the symbols have the same meaning as for the landing trajectory.

The gravity loss is 900 ft/sec, the misalignment loss 50 ft/sec and the velocity reserve for an initial guidance correction of 500 ft/sec.

The total ideal velocity therefore is 10,300 ft/sec including a 250 ft/sec reserve for less favorable launch conditions. With a specific impulse of I_{sp} = 420 seconds the mass ratio for the launch stage is m = 2.145. (For storable propellants with I_{sp} = 320, m = 2.707).

All launch vehicle configurations listed in Section B have this mass ratio of m = 2.145.

HYDROGEN - OXYGEN PROPELLANT, I_{SP} = 420 SEC

VERTICAL LAUNCH (γ_B = 90°)

BURNOUT VELOCITY, V_B = 8900 FT/SEC

W_i = EARTH POUNDS

A_B = BURNOUT ACCELERATION

h_B = BURNOUT ALTITUDE

V_{gl} = GRAVITY LOSS

t_B = BURNING TIME

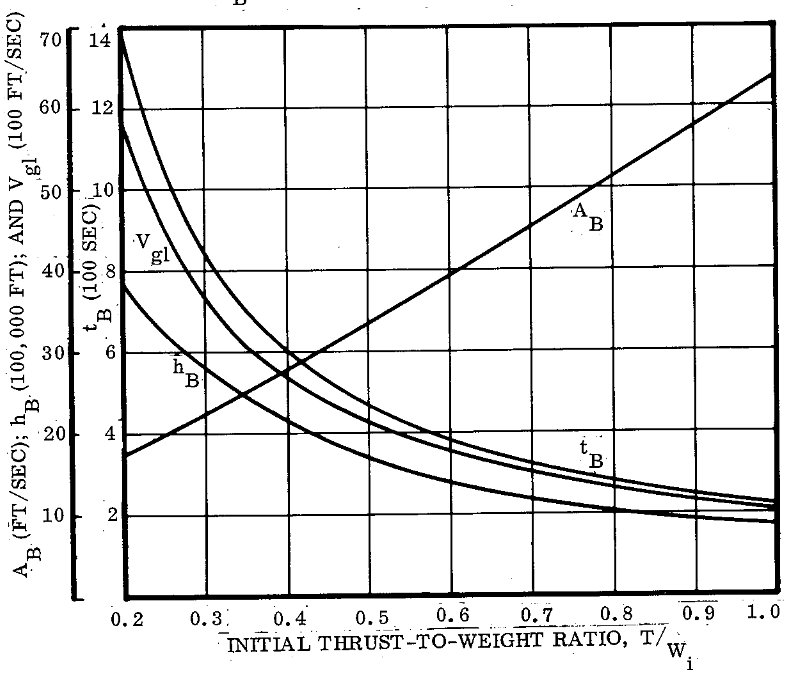


Figure 22. Lunar Launch Trajectories, I_{SP} = 420 Seconds

2.4.2.6.2 Lunar Launch and Landing Vehicle Configurations. The following Tables 7 and 8 list the stage, inert and propellant weights of four lunar launch- and landing-rockets whose performance potential has been analyzed in detail. The example configurations were selected to vary size, mission safety, type of re-entry vehicle, and type of propellants used for the lunar launch vehicles. All lunar landing vehicles considered use hydrogen-oxygen propellant. Calculations were independent of booster system's capability.

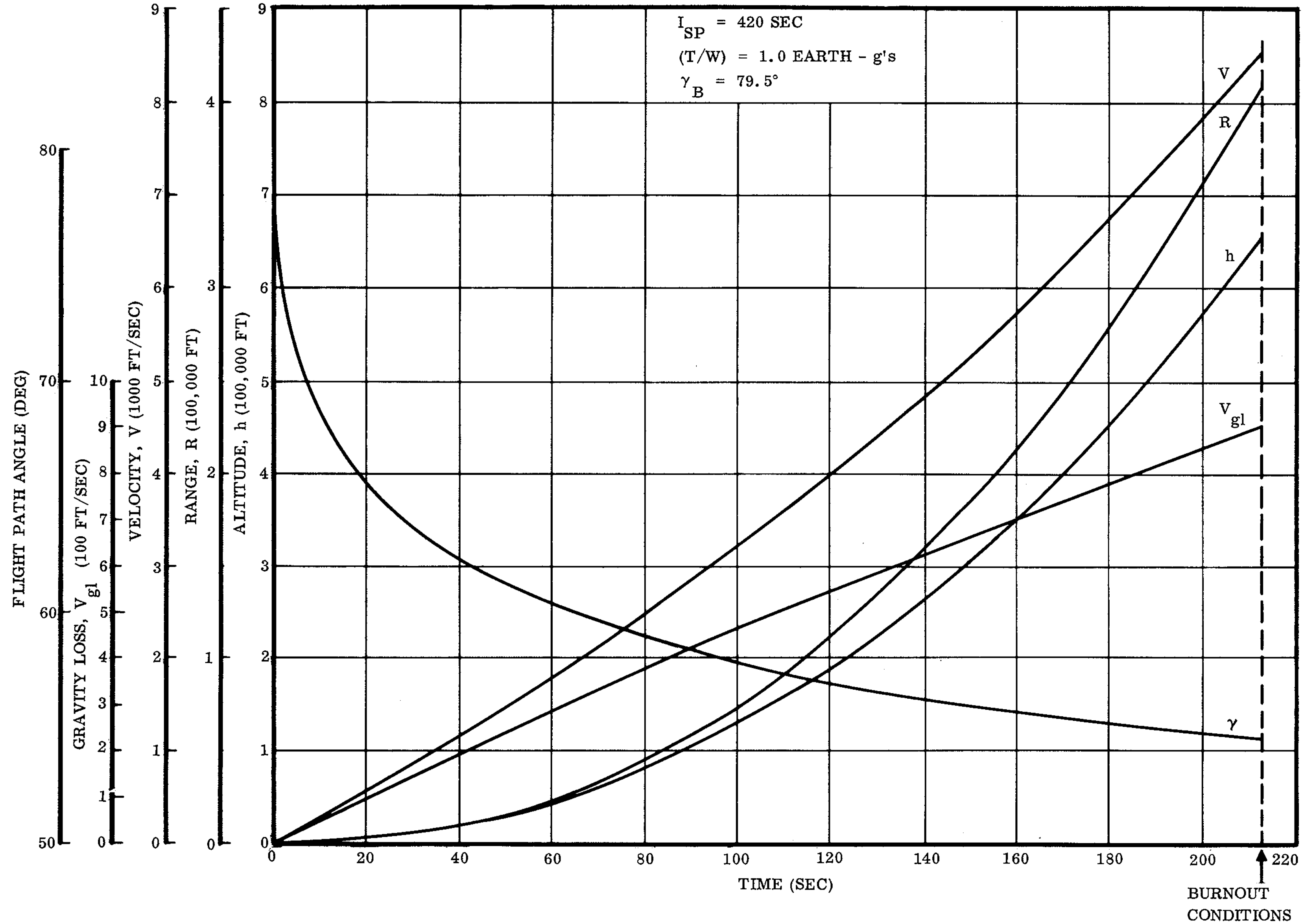


Figure 23. Example Configuration Lunar Launch Trajectory (Gravity Turn)

Table 7. Example Configuration Comparisons for Lunar Landing

NOMENCLATURE	SYMBOL	CONFIGURATION WEIGHTS (POUNDS)			
		A	B	C	D
Entry Vehicle After Lunar Escape	W_{ev}	20,445	16,240	23,615	23,615
Jettison Weight	W_j	95	70	1,535	1,535
Burnout Weight	W_b	20,540	16,310	25,150	25,150
Propellant Weight	W_p	660	530	7,850	7,850
Initial Weight	W_i	21,200	16,840	33,000	33,000
Lunar Launch Vehicle					
Jettison Weight	W_j	5,860	5,060	6,650	7,900
	W_b	27,060	21,900	39,650	40,900
	W_p	31,000	25,240	45,400	69,900
	W_i	58,060	47,140	85,050	110,800
Propellant Boil-off Weight	W_{pb}	470	380	700	---
Less 3 Men	W_{3m}	-840	-840	-840	-840
Less Propellant Transferred	W_{ap}	---	---	-9,385	-9,385
Landing Weight	W_{ll}	57,690	46,680	75,535	100,610
Lunar Landing Vehicle					
	W_j	10,110	9,800	11,445	12,270
	W_b	67,800	56,480	86,980	112,880
	W_p	83,500	69,500	107,020	138,720
Initial Weight of Stage	W_i	151,300	125,980	194,000	251,600
	W_{pb}	420	350	540	700
Earth Moon Stage					
	W_b	151,720	126,330	194,540	252,300
	W_p	3,800	3,170	4,860	6,300
	W_i	155,520	129,500	199,400	258,600
Earth Escape Weight	W_{ee}	155,520	129,500	199,400	258,600

Table 8. Example Configuration Identification

A	=	Delta Wing Re-Entry Vehicle No Abort Capability From Lunar Take-Off Hydrogen-Oxygen Propellants--All Earth Escape Stages Orbit Thrust = 2000 pounds Lunar Launch Thrust = 2 × 35,000 pounds Lunar Landing Thrust = 2 × 25,000 + 2 × 7,000 pounds Midcourse Thrust = 2 × 7,000 pounds
B	=	Lenticular Shape Re-Entry Vehicle No Abort Capability From Lunar Take-Off Hydrogen-Oxygen Propellants--All Earth Escape Stages Orbit Thrust = 1,000 - 2,000 pounds Lunar Launch Thrust = 2 × 27,000 pounds Lunar Landing Thrust = 2 × 27,000 + 2 × 6,000 pounds Midcourse Thrust = 2 × 6,000 pounds

Table 8. Example Configuration Identification (Cont)

C	=	Delta Wing Re-Entry Vehicle Abort Capability From Lunar Take-Off (With Tower Attached After Landing) Abort Propulsion System Used for Moon Earth Midcourse Corrections Hydrogen-Oxygen Propellants--All Earth Escape Stages Orbit Thrust = 4 × 7,000 + 4 × 500 pounds Lunar Launch Thrust = 2 × 45,000 pounds Lunar Landing Thrust = 2 × 45,000 + 2 × 10,000 pounds Midcourse Thrust = 2 × 10,000 pounds
D	=	Delta Wing Re-Entry Vehicle Same as Configuration C Except UDMH + Hydrazine - N_2O_4 Propellants Orbit Thrust = 4 × 8,000 + 4 × 500 pounds Lunar Launch Thrust = 2 × 60,000 pounds Lunar Landing Thrust = 2 × 60,000 + 2 × 12,000 pounds Midcourse Thrust = 2 × 12,000 pounds

DATA	
PLANFORM AREA	
TOTAL	215 FT^2
SURFACE AREA	
BODY	556 FT^2
CONTROL SURFACES	73 FT^2
TOTAL	629 FT^2
VOLUME	
TOTAL	884 FT^3
HABITABLE VOLUME	547 FT^3
WEIGHTS FOR M-2 CONFIGURATION	
COMMAND MODULE	8,085 LBS
MISSION MODULE	3,719 LBS
PROPULSION MODULE	1,476 LBS
TOTAL ORBITAL WEIGHT	22,047 LBS

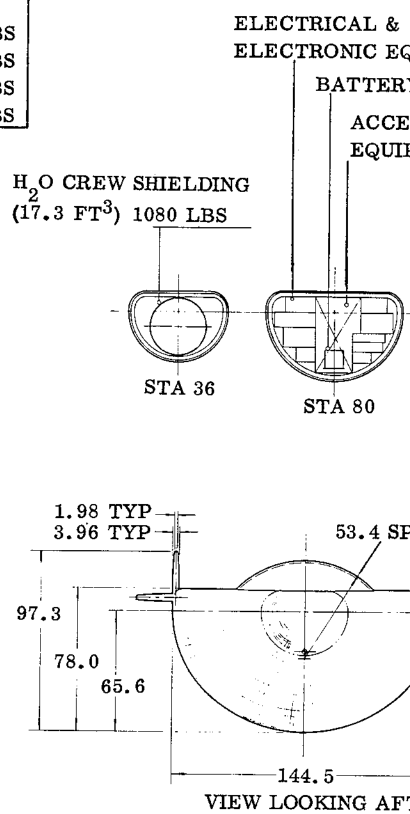
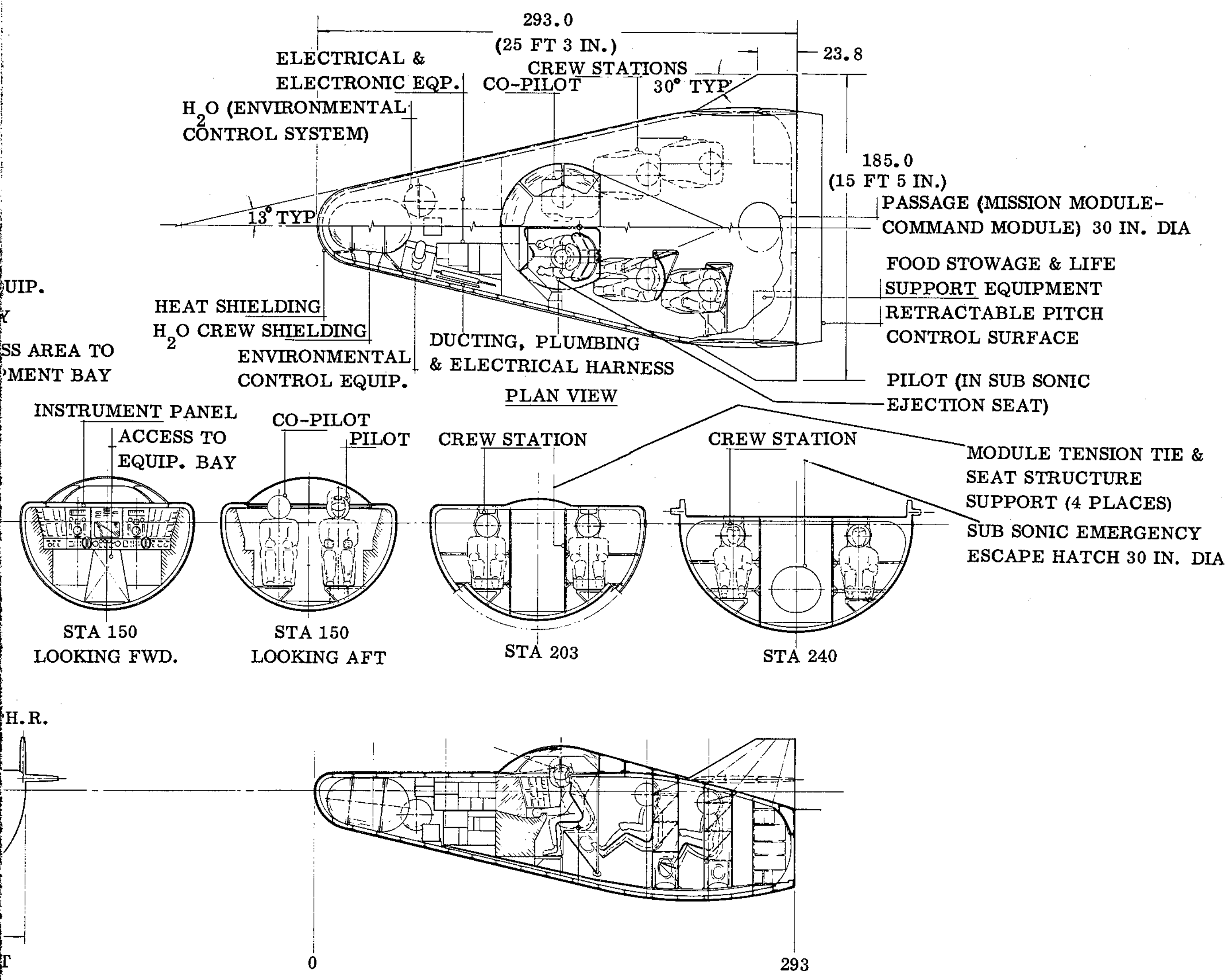


Figure 24.



M-2 Re-Entry Vehicle--6-Man Configuration

4 M-2 LAUNCH CONFIGURATION

The M-2 launch configuration (Figure 25) consists of the following primary components: 1) Abort tower, M-2 re-entry vehicle, pressure stabilized titanium fairing (to provide summetry to forward portion of M-2 vehicle). 2) Mission module. 3) Propulsion module. The symmetrical shape coupled with a slim forward portion gently increasing in diameter to the desired 220.0 inch diameter (Saturn adaption), affords an excellent aerodynamic shape for initial boost. This gives complete protection to the upper body surface and control surfaces of the M-2 vehicle.

4.1 MISSION MODULE. The M-2 re-entry vehicle is attached to and partially enclosed by the mission module during all phases of the mission, except during re-entry or abort.

The mission module is comprised basically of four petal-like doors made of honeycomb paneling, frames, and inner skin. When closed, and tension band sealed, these doors provide a pressurized and habitable area.

The conical doors are opened by the release of explosive tension bolts (located at the upper tension ring) and the simultaneous actuation of the door opening mechanism. The open position provides clearance for escape of the M-2 vehicle during abort or normal mission. The floor is a combination of aluminum-alloy honeycomb paneling and a radial system of beams located and sized compatible with loads being applied by the M-2 vehicle, equipment above the floor, and thrust forces below the floor.

The interior of the mission module contains electronic equipment for guidance and communication, life support, and scientific research equipment. All items are located to optimize CG position of entire launch configuration for stability and control.

4.2 PROPULSION MODULE. The propulsion module is similar in purpose, components, and structural design to the M-1 configuration. Differences occur in sizing of the propellant tank and method of supporting main engines. These differences are commensurate with configuration weight and floor structure of the mission module respectively.

4.3 ABORT SYSTEMS. An abort of the M-2 vehicle (on launch pad or during boost phase) is accomplished by firing 4 main abort tower rockets, simultaneous opening of 4 mission module doors, and release of 4 explosive bolts securing the M-2 vehicle to mission module floor. At the prescribed altitude the abort tower with the conical fairing is jettisoned from the M-2 vehicle by means of shaped-charge attachments.

The M-2 abort tower closely resembles the M-1 tower configuration with the exception of means of attachment to the M-2 shape. Main abort rockets could have approximately the same thrust rating as the M-1 since the additional weight of the M-2 vehicle is offset by its cleaner aerodynamic shape.

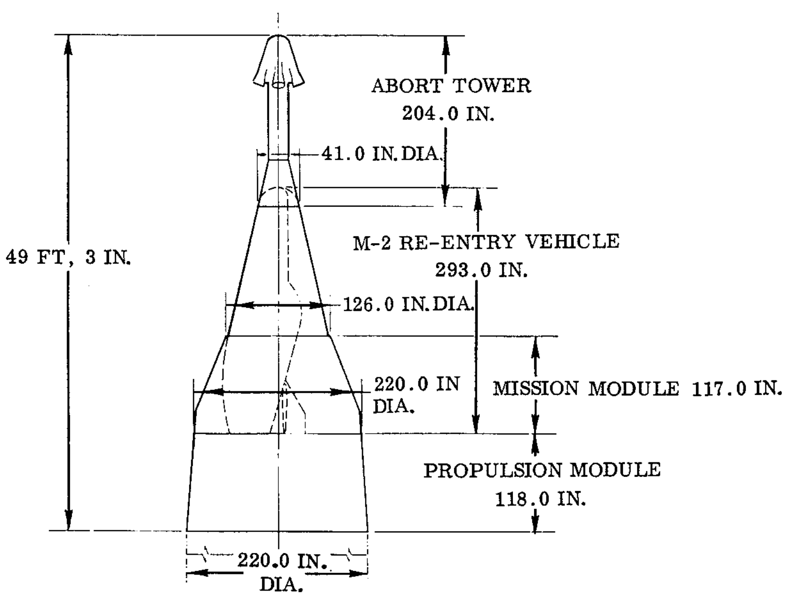
4.4 RE-ENTRY. The M-2 vehicle satisfies two major conditions in returning to the earth's atmosphere. First, it is an excellent shape for heat shield protection including radiation heating, and secondly, it possesses sufficient maneuverability to permit landing at a desired location. The M-1 re-entry experience will provide a high confidence level for M-2 design.

4.5 LANDING. The pitch, yaw, and roll control surfaces of the M-2 vehicle along with its glide capability (equivalent to a lift to drag ratio of around 5) permit subsonic maneuvering and landing characteristics comparable to a dead stick landing in a current jet intercepter. The flared-ground effect touchdown speed for a wing loading of 38 lbs per sq ft shoudl not exceed 150 knots.

An subsonic emergency escape hatch has been provided for low altitude bail out of crew with ejection seat escape provided for the pilot should unavoidable hazards occur.

Several landing gear configurations have been considered; they vary from glider type landing (no external assist) to a tricycle landing gear system. This design is subject to further study.

4.6 WEIGHTS. The weight summary for the M-2 vehicle is presented in Table 9. Table 10 shows the weight breakdown.



LAUNCH CONFIGURATION

CONICAL DOOR, FULLY OPEN
RE-ENTRY VEHICLE SEPARAT

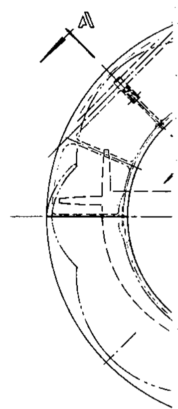
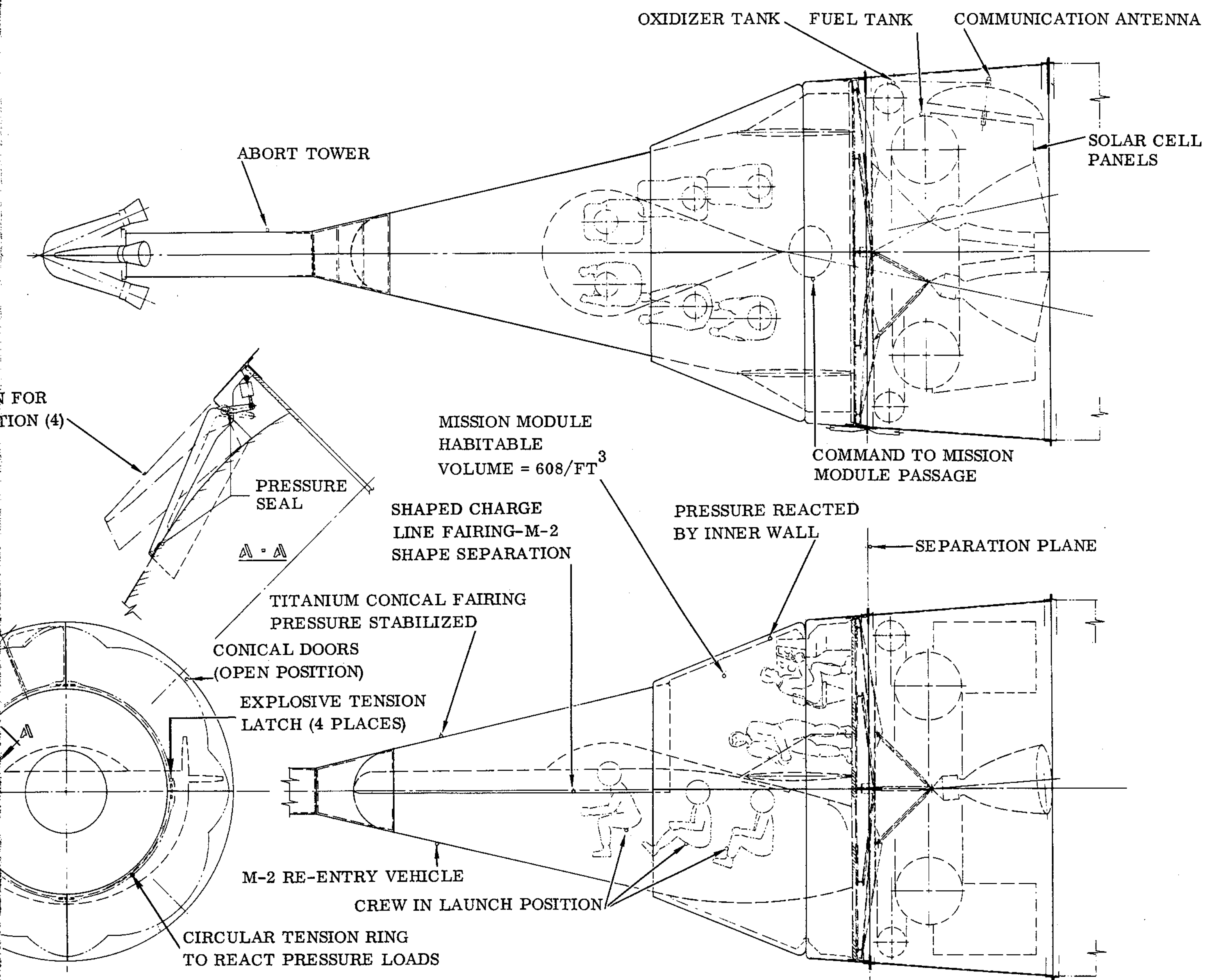


Figure 25.



M-2 Re-Entry Vehicle--6-Man Launch Configuration

Table 9. Weight Summary M-2 Vehicle

ITEM		(WEIGHT IN POUNDS)
Command Module-Dry		(8085)
Structure	4700	
Hydraulic System	150	
Electronics	542	
Power Supply System	280	
Life Support	1098	
Crew	1200	
Locating Aids	20	
Attitude Control System	95	
Mission Module-Dry		(3719)
Structure	1705	
Power Supply System	380	
Life Support	1171	
Scientific and Research Equipment	170	
Electronics	293	
Propulsion System-Dry		(1476)
Main Propulsion System	1246	
Midcourse and Attitude Control	175	
Altitude Abort System	47	
Module Separation Unit	8	
Adapter to S-IV		350
Abort System		2378
Total Weight at Launch-Dry		(16,008)
Liquids and Gases		(8520)
Command Module	140	
Mission Module	8380	
Launch		(24,528)
Escape System Jettisoned		2378
Effective Weight		218
Orbital Weight		(22,368)

Table 10. Estimated Detailed Weight Breakdown - M-2 Configuration

ITEM		(WEIGHT IN POUNDS)
Command Module		
Structure		(4700)
Heat Shielding	2400	
Insulation	205	
Nose Cap	20	
Structure-Honeycomb	480	
Frames and Struts	400	
Rings	130	
Longerons	15	
Canopy and Structure	150	
Fins and Control Surfaces	500	
Structural Supports	300	
Equipment Mounting	100	
Hydraulic System		(150)
Accumulator	36	
Reservoir and Fluid	31	
Valves	6	
Actuators	58	
Pump and Electric Motor	19	
Electronics		(542)
Guidance	247	
Communications, Tracking and Telemetry	195	
Harnessing	100	
Power Supply System		(280)
Batteries	250	
Power Change-over Switch	10	
Voltage Regulator	20	
Life Support		(1098)
Atmospheric Control	116	
Shielding System	390	
Control Consoles and Displays	140	

Table 10. Estimated Detailed Weight Breakdown - M-2 Configuration (Cont)

ITEM	(WEIGHT IN POUNDS)	
Crew Supports and Restraints	270	
Survival Equipment	72	
Auto and Manual Controls	50	
Waste and Sanitation Equipment	22	
Water System	24	
Thermal Control	14	
Liquids and Gases		140
Crew		1200
Landing and Recovery		(20)
Locating Aids	20	
Attitude Control System		95
Mission Module		
Structure		(1705)
Bulkhead	90	
Breakaway Doors	450	
Floors and Supports	100	
Rings	200	
Frames	90	
Fairing	350	
Cylindrical Structure	200	
Internal Structure	150	
Equipment Mountings	75	
Electronics		(293)
Guidance	33	
Communications, Tracking and Tlm	160	
Harnessing	100	
Power Supply System		(380)
Fuel Cells and Controls	150	
Solar Cells (including Struct)	210	
Solar Cells Deployment Mech	20	

Table 10. Estimated Detailed Weight Breakdown - M-2 Configuration (Cont)

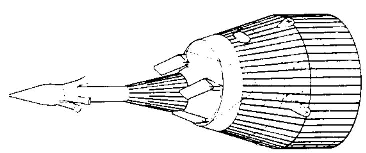
ITEM	(WEIGHT IN POUNDS)	
Life Support Equipment		(1171)
Food System	254	
Medical and Hygiene Equip	20	
Waste and Sanitation Equipment	70	
Water System	39	
Space Radiator	200	
Crew Supports and Restraints	30	
Control Consoles and Displays	75	
Auto and Manual Controls	35	
Personal Belongings	30	
Recreation and Rest Equipment	50	
Atmosphere Control	300	
Thermal Control	6	
Module Insulation	62	
Scientific and Research Equipment		170
Camera	150	
Misc Equipment	20	
Liquids and Gases		8380
LO_2	4495	
LH_2	865	
Attitude Control	1100	
He for Pressurization	20	
O_2 for Environmental Control	150	
N_2 for Environmental Control	57	
H_2O for Crew Shielding	1010	
Air in Module	100	
LO_2 Residuals	311	
LH_2 Residuals	197	
Glycol	75	
Adapter to S-IV		350

Table 10. Estimated Detailed Weight Breakdown - M-2 Configuration (Cont)

ITEM		(WEIGHT IN POUNDS)
Propulsion Module		
Main Propulsion System		1246
Main Engines	570	
Propellant Ducting	24	
Fill and Drain Valve	6	
He Containers	200	
Pneumatic Hardware	80	
Gimbal Actuators and Drive	36	
Propellant Tanks	330	
Midcourse and Attitude Control System		175
Oxidizer Tank	40	
Fuel Tank	30	
Plumbing	30	
Manual Controls	10	
150 Pound Thrust Chambers (4)	40	
7.5 Pound Thrust Chambers	25	

Table 10. Estimated Detailed Weight Breakdown - M-2 Configuration (Cont)

ITEM		(WEIGHT IN POUNDS)
Abort System		2378
Abort Tower	111	
Rocket Casing and Grain	2000	
Attitude Control Systems	225	
Separation Rockets	12	
Fairing	30	
Altitude Abort System		47
Propellant	33	
Igniter and Nozzles	4	
Case and Liner	10	
Module Separation Unit		8
Propellant	6	
Nozzle and Casing	2	



5 M-3 RE-ENTRY VEHICLE CONFIGURATION

5.1 DISCUSSION. The M-3 (proposed version of the NASA concept) re-entry vehicle, Figure 26, is a direct outgrowth of the M-2 design. Improvements are aimed toward increased aerodynamic performance, especially subsonically, through a fold out wing arrangement, reducing the sink rate, and improving the L/D ratio.

This feature plus the advantages of a totally integrated space vehicle make the M-3 design worthy of investigation.

5.2 STRUCTURAL DESIGN. The M-3 vehicle, having basically the same design criteria as the M-2, incorporates a similar system of frames covered with titanium alloy honeycomb and ablatable heat shielding. Structural components of the wing include spanwise spars located stationwise at the wing folding mechanism points, chord-wise ribs covered with honeycomb paneling, and heat shielding on lower surfaces only. This is permissible because of the folded wing configuration during re-entry.

Placing personnel close to contour allows a more efficient structural tie between seat structure and the adjacent outer shell. This affords a reasonable amount of access between seats along the longitudinal center line.

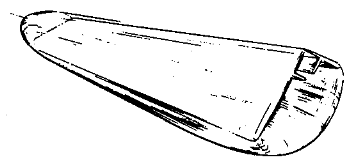
5.3 RE-ENTRY SYSTEM. The M-3 vehicle, during re-entry phase, embodies all the desirable heat resisting features of the M-2 design with its blunt nose and lower surface. It also has the inherent ability to protect its upper body surface, canopy, upper wing surface, and vertical tails from the effects of aerodynamic heating. The re-entry configuration consists of the M-3 vehicle with wings folded together and secured by means of explosive latches. A small jettisonable fairing covering the area immediately forward of the folded wing and extending forward to the nose of the vehicle completes the launch exterior.

5.4 SUBSONIC FLIGHT. With its unique wing design unfolded, the M-3 vehicle closely resembles the highly swept delta aircraft of today. Having, in a matter of seconds, attained a lift to drag ratio of greater than 10, along with the excellent control potential, the re-entry vehicle becomes an aircraft with extended glide maneuver and landing capabilities.

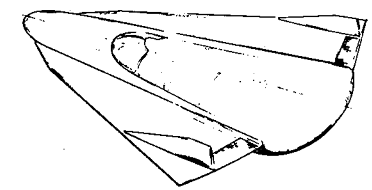
These flight characteristics could be increased to powered flight by a growth version of the M-3 which would house small turbojet engines (approx 5000 pound thrust each), located at the aft outboard corners of the body. This vehicle would have an increased length to allow for additional weight and space for propulsion systems. A detachable engine and tankage system has also been considered.

5.5 LANDING SYSTEM. The excellent flight characteristics, unimpaired vision, and structural integrity of the M-3 vehicle assure maximum protection for crew and equipment during landing.

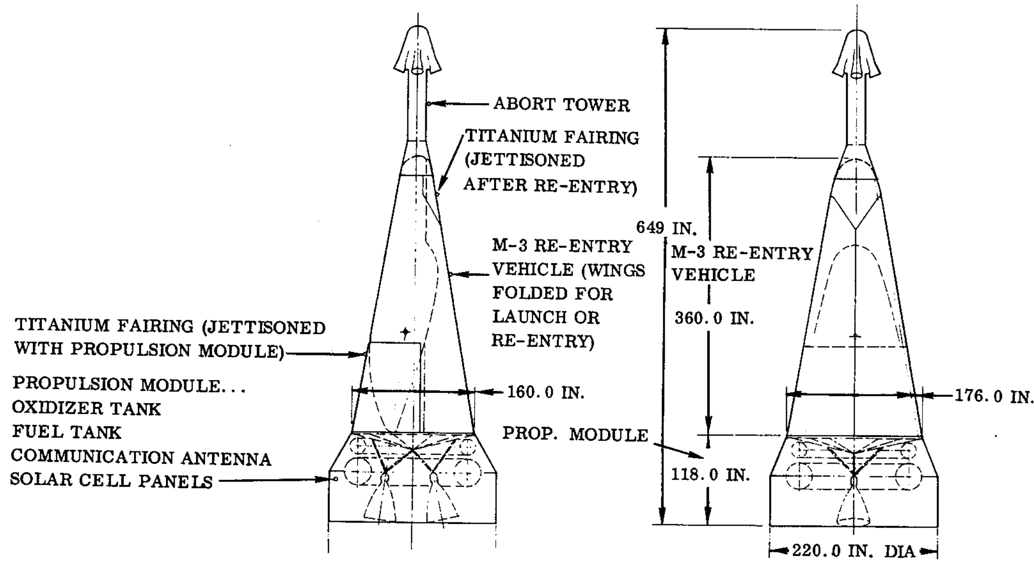
The M-3 design with its high L/D ratio and low wing loading would be capable of touchdown speeds approaching 150 knots. Use of extendable skids would permit reuse of the vehicle after refurbishing.



RE-ENTRY CONFIGURATION



LANDING CONFIGURATION



LAUNCH CONFIGURATION

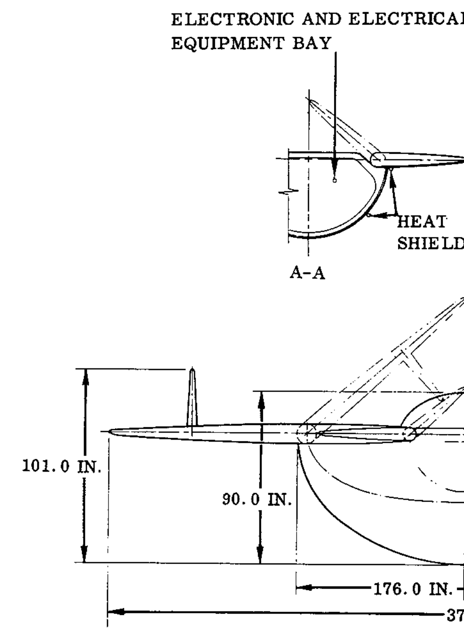
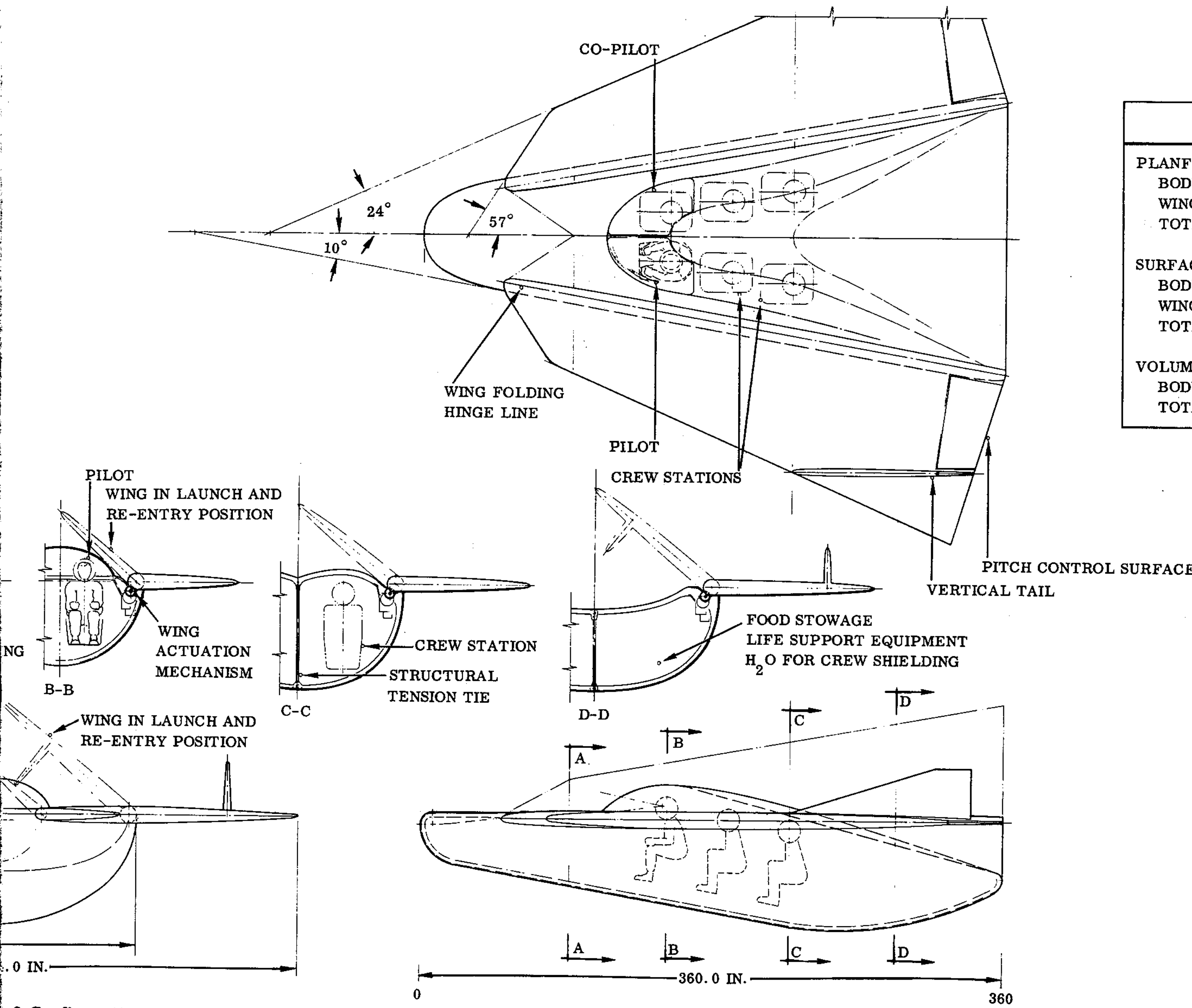


Figure 26.



DATA	
PLANFORM AREA	
BODY	220 FT^2
WING	300 FT^2
TOTAL	520 FT^2
SURFACE AREA	
BODY	628 FT^2
WING & TAIL	656 FT^2
TOTAL	1684 FT^2
VOLUME	
BODY	724 FT^3
TOTAL	956 FT^3

-3 Configuration

6 M-3 LAUNCH CONFIGURATION

6.1 DISCUSSION. The launch configuration for the M-3 design is geometrically attractive because of its clean aerodynamic shape on the booster. It consists of an abort tower and the totally integrated M-3 vehicle with wings in folded position completing the drag fairing. A jettisonable nose and afterbody fairing are required to give external symmetry. A transition section (about 47 inches long) adapts the trailing edge of the M-3 to the Saturn booster.

6.2 ABORT SYSTEM. An M-3 Configuration abort is accomplished by the release of the M-3 vehicle from the afterbody fairing and the propulsion module by means of energized shaped charge attachments, and the firing of the abort tower rockets. At the prescribed altitude the abort tower and the nose fairing are jettisoned, allowing the M-3 vehicle to unfold its wings when at apogee and glide to a safe landing.

6.3 PROPULSION CONCEPTS. The propulsion module for the M-3 configuration parallels that of the M-3 and M-2 designs. Its main components are: 1) outer structural shell adapting the M-3 shape to the Saturn booster, 2) two toroidal propellant tanks, 3) 2 main rocket engines supported by truss structure and vectored thru the CG of the launch configuration, 4) communication antennas and solar cell panels. Midcourse and altitude control rockets are mounted externally.

6.4 WEIGHTS. The M-3 weight summary is presented in Table 11. Table 12 shows the M-3 weight breakdown.

Table 11. Weight Summary: M-3 Vehicle

ITEM	(WEIGHT IN POUNDS)	
Command Module - Dry		(11,067)
Structure	7150	
Hydraulic System	280	
Electronics	542	
Power Supply	660	
Life Support	1123	
Crew	1200	
Locating Aids	17	
Attitude Control	95	
Propulsion Module - Dry		(1,437)
Structure	645	
Main Propulsion System	570	
Mid-course and Attitude Control	175	
System	47	
Adapter to S-IV		350
Abort System		2,378
Total Weight @ Launch-Dry		(15,232)
Liquids & Gases		7,935
Launch		(23,167)
Abort System Jettisoned		2,378
Effective Weight		218
Orbital Weight		(21,007)

Table 12. Weight Breakdown: M-3 Configuration

ITEM		(WEIGHT IN POUNDS)
Command Module		
Structure		7150
Heat Shielding	3300	
Insulation	250	
Nose Cap	20	
Structure-Honeycomb	600	
Frames & Struts	500	
Rings	200	
Longerons	300	
Canopy & Structure	180	
Aerodynamic Surfaces	1400	
Structural Supports	400	
Hydraulic System		280
Accumulator	70	
Reservoir & Fluid	60	
Valves	10	
Actuators	115	
Pump & Motor	25	
Electronics		542
Guidance	247	
Communications, Tracking & TLM	195	
Harnessing	100	
Power Supply System		660
Batteries	250	
Power Change Over Switch	10	
Voltage Regulator	20	
Fuel Cells & Controls	150	
Solar Cells & Deployment Mach	230	

Table 12. Weight Breakdown: M-3 Configuration (Cont)

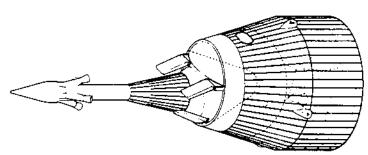
ITEM		(WEIGHT IN POUNDS)
Locating Aids		17
Crew		1200
Life Support		1123
Atmospheric Control	116	
Shielding System	390	
Control Consoles & Displays	140	
Crew Support & Restraints	270	
Survival Equipment	72	
Auto & Manual Controls	75	
Waste & Sanitation Equipment	22	
Water System	24	
Thermal Control	14	
Attitude Control System		95
Propulsion Module		
Structure		645
Bulkhead	190	
Rings	150	
Frames	60	
Tapered Structure	140	
Propellant Tank Support	30	
Miscellaneous Structure	75	
Main Propulsion System		1246
Main Engines	570	
Propellant Ducting	24	
Fill and Drain Valve	6	
He Tanks	200	
Pneumatic Hardware	80	
Gimbal Actuators and Drive	36	
Propellant Tanks	330	

Table 12. Weight Breakdown: M-3 Configuration (Cont)

ITEM		(WEIGHT IN POUNDS)
Mid-course and Attitude Control System		175
Oxidizer Tank	40	
Fuel Tank	30	
Plumbing	30	
Manual Controls	10	
150 Pound Thrust Chambers (4)	40	
75 Pound Thrust Chambers	25	
Attitude Abort System		47
Propellant	33	
Igniter and Nozzles	4	
Case and Liner	10	
Liquids and Gases		7935
LO_2	4160	
LH_2	830	
Air in Module	100	
Attitude Control	1100	
He for Pressurization	20	
H_2O for Crew Shielding	1010	
O_2 for Environmental Control	150	

Table 12. Weight Breakdown: M-3 Configuration (Cont)

ITEM		(WEIGHT IN POUNDS)
N_2 for Environmental Control	57	
LO_2 Residuals and Boiloff	311	
LH_2 Residuals and Boiloff	197	
Adapter to S-IV		325
Escape System		2378
Abort Tower	111	
Rocket Casing and Grain	2000	
Attitude Control System	225	
Separation Rocket	12	
Fairing	30	
Power Supply System		380
Fuel Cells and Controls	150	
Solar Cells (Including Structure)	210	
Solar Cell Deployment Mechanism	20	
Fairings (Jettisoned with Propulsion Module)		310
Aft Fairing	250	
Fwd Fairing	60	



7 COMMUNICATIONS AND GUIDANCE

7.1 COMMUNICATIONS - GENERAL. The following must be considered in future growth:

a. The transfer of more information per unit time with less spacecraft primary power.

b. A decrease in volume and weight of spacecraft equipment.

c. Additional improvements in the reliability of electronic equipment by using improved components.

Several growth items that may improve the future performance of the prime UHF communication system are discussed in the ensuing sections.

7.1.1 UHF Transponders. Two major improvements can be made to this equipment. The first involves modifying the transmitter portion, using an all solid-state design, to increase the power output from the present 6 mw to 2-to-8 watts. This transmitter power would permit satisfactory operation to a slant range of 5000 to 10,000 n.mi. on the low gain omni-antenna.

Using a high gain antenna with a specialized beam pointing system, a slant range of 50,000 to 100,000 n.mi. before additional power amplification is needed. Two methods are considered feasible for this:

a. Develop an amplifier with the 6 mw transponder as a driver.

b. Modify the transponder with a new transmitter design using high power frequency multiplication techniques which are being developed.

The second improvement is that of decreasing the excess noise temperature of the receiver by adding a low noise preamplifier. An obvious method is that of using parametric amplifiers. An increase in sensitivity from 5500°K to 500-1000°K is feasible.

7.1.2 Spacecraft Antenna. To eliminate the beam pointing problem and yet have the capability of a higher gain antenna, two parabolic antennas can be mounted back-to-back on the extendable mast. One antenna would have a wide beamwidth while the other would be narrow and both would be proportioned to complement each other with regard to complete Earth illumination. Each antenna would be driven separately with a coaxial switch being used to make the systems transfer. A gain of 8.5 db over the present system can be realized. An inflatable antenna may be used during space flight to provide additional gain.

At lunar range an 8.5 db gain would allow the system bandwidth to be increased 600 percent with no degradation in performance. With the increase in bandwidth, the capability for additional information channels such as television becomes available.

7.1.3 Landing and Rescue Equipment. Both VHF, HF, voice, and rescue beacon transmitters have tubes as their power output stages. In addition to filamentary power drain, a dc-dc converter is required to supply the high voltage necessary for satisfactory operation. It is now possible to generate in excess of 100 watts in the HF range and 10 watts in the 100-200 mc range using solid-state devices. By converting the power output stage in each transmitter to solid state, a saving in power consumption could be realized.

7.1.4 Advanced Telemetry. The advantages of a variable communication-pattern telemetry are unlimited in the variety of data sampling rates possible as interest changes in various data during the course of flight. This also offers the capability of reduced data flow rate for onboard data reduction or emergency conditions. The disadvantages are increased size, weight, power requirements, and decreased reliability arising from the increase in complexity. In view of the anticipated strides in microminiaturization and improved reliability of components, here is an advanced system which should offer maximum growth capability.

This system (Figure 27), differs from conventional telemetry systems in that the sequence of addresses to be sampled is stored in a portion of the core memory. Thus, the sampling sequence can be altered by simply changing the stored program. The sequence of transmitted data words can be entirely different from, and asynchronous to, the sequence of sampled inputs. A fairly large memory is available to provide a buffer storage between units with unequal or asynchronous data rates.

The advance system will use a data format identical to the basic system. Hence, synchronization generation and recovery will also be accomplished in the same fashion as the basic system. Since the only changes over the basic system are in the vehicle, the entire ground system will be the same.

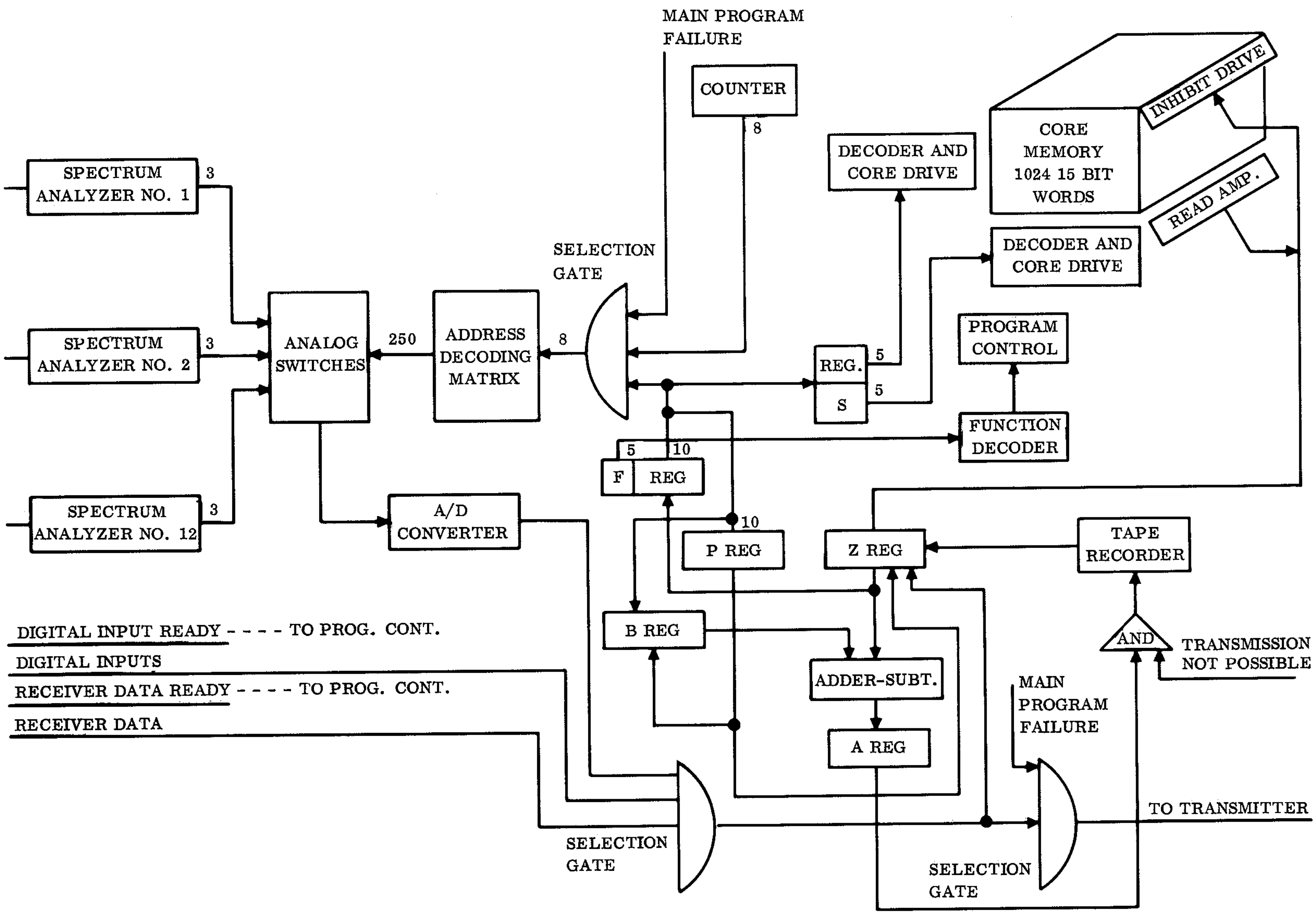


Figure 27. Stored Program Telemetry

The stored program accomplishes the functions of sampling the inputs, storing the data so obtained, and transferring data received from "Up" telemetry to its proper destination. Data transfer from the ground and guidance computer is readily accomplished. The stored data program can also be modified by the ground system.

When direct transmission of telemetry data is not possible, a tape recorder is provided for storage of a large amount of data. Should the main system not function correctly a back-up fixed program can be switched into operation. This telemetry system can also respond to a limited number of instructions.

7.2 GUIDANCE

7.2.1 Rendezvous. With several small modifications and/or additions, the Apollo vehicle will be capable of performing rendezvous in space. There are a number of guidance techniques that could be used to perform this task. The most promising appears to be an adaptation of the MIT range-rate system. This system is one in which a continuous variable thrust is assumed. Nevertheless, this guidance scheme is one which has the built-in feature that it will call for a constant thrust in the steady state. Such a system is one which implies that

$$\dot{R} \sim \sqrt{R}$$

where $\vec{R}$ is the vector from the Apollo vehicle to the target. It is therefore reasonable that a thrust proportional to $(\dot{R} + k\sqrt{R})$ should be applied in the $\vec{R}$ direction. The angular component of velocity is dealt with by applying a component of thrust proportional to this angular component to cancel out the angular momentum; that is to say, a component of thrust proportional to $\vec{\omega} \times \vec{R}$, where $\vec{\omega}$ is the angular velocity of the Apollo vehicle about the target. This gives rise to the guidance equation

$$\vec{T} = -S_1(\dot{R} + k\sqrt{R})\frac{\vec{R}}{R} - S_2(\vec{\omega} \cdot \vec{R})$$

where $\vec{T}$ is the thrust to be applied, R is a constant related to the steady-state thrust desired, and S_1 and S_2 are system gain constants. Since this equation was proposed by the MIT Instrumentation Laboratory, a system based on this guidance equation has been referred to as the MIT range-rate system.

Although this system assumes the use of a variable-thrust engine, a modification of it can be made to work satisfactorily with constant-thrust engines. This can be done by providing two or three constant-thrust levels and the appropriate switching logic to operate in conjunction with the range-rate guidance equation given above. A digital simulation of such a system has been devised and used to demonstrate its workability. The capability to perform the rendezvous computations, in addition to its present functions, could be achieved by the addition of only approximately 250 words to the Apollo computer.

The thrust levels that would have to be provided would be of the order of 2500 pounds, 1000 pounds, and 200 pounds. In addition, the attitude control jets would be used for fine trimming on final approach.

Provisions for sensing range, range rate, and angular rates must be provided as input to the range rate system. Angular rates would be obtained from rate gyros attached to the dish of a tracking radar. This radar could be the same system employed for the lunar and earth radar altimeter. Range and acquisition requirements are not major limitations with a friendly vehicle as lightweight transponders can be employed.

7.2.1.1 Altimeter Application as Rendezvous Guidance Radar. The high altitude altimeter has been investigated to determine its growth possibilities to a unit suitable for rendezvous guidance measurements. During rendezvous missions with manned spacecraft or unmanned craft of comparable size, the altimeter was found to be very useful in its present state. Only a search program for antenna angle and a more complex computer and inertial guidance unit tie-in are required.

In the space rendezvous mission two spacecraft are located in a certain vicinity but neither knows the precise location of the other. The first object of the rendezvous is to have one spacecraft, designated the patrol spacecraft, determine the range and bearing of the other spacecraft. Let the other spacecraft be called the object spacecraft. It is considered that the patrol spacecraft must be equipped with a search radar. The search radar should be a high performance unit with capabilities of collecting position information at a rapid rate. It is the object of the following discussion to consider the capability of the high altitude altimeter when operated as a search radar in a space rendezvous. The range capability depends on the radar cross section of the target. If a transponder can be employed the range can be extended considerably.

A model situation must be defined as a basis or reference condition for analysis. The model situation assumes that the patrol craft knows the location of the object craft within a 30 degree sector. The solid angle of uncertainty subtended is 0.22 steradians. The range interval of uncertainty is equal to the total range to the object craft. The object craft is not cooperative and the physical dimensions of the object craft are such that its radar cross section σ_T is equivalent to a 6 foot diameter sphere. The value of σ_T is then 28.3 ft^2.

The range capability and the time required to search the 30 degree sector can be determined for any search radar and these two figures describe the performance capability of the radar. Once computed, these two figures can be extrapolated to describe the range capability and search time values characteristic of the radar when the object craft cooperates to various degrees in the search. For instance, the object may have a steerable antenna of cross section A_o which can be pointed at the patrol craft, and a transponder with power gain G_{po} connected to the antenna.

If in the model case the radar was characterized by range capability R_m and search time capability T_m, then in the special case the new range capability R', provided the new time capability T' was maintained equal to T_m times (R'/R_m) would be

$$R' = R_m \left(1 + \frac{4\pi A_o^2 G_{po}}{\lambda^2 \sigma_T}\right)^{1/4}$$

For example, let $A_o = 4$ ft^2, $G_{po} = 100$, $\lambda^2 = 10^{-2}$, $\sigma_T = 28$ ft^2; then

$$R' = 16.4\ R_m \text{ and,}$$

$$T' = 16.4\ T_m.$$

With the 3 foot antenna and the 200 watt transmitter, the values R_m and T_m for the high altitude altimeter are:

$$R_m = 26 \text{ n.mi.}$$

$$T_m = 119 \text{ seconds or 2 minutes.}$$

For the cooperative target with the addition of the antenna and transponder the values R' and T' become:

$$R' = 427 \text{ miles.}$$

$$T' = 3.3 \text{ minutes.}$$

The high altitude altimeter has the growth potential to extend the foregoing basic capabilities R_m and T_m to new values R_{m1} and T_{m1} equal to:

$$R_{m1} = 82 \text{ n.mi.}$$

$$T_{m1} = 15.8 \text{ seconds or 0.25 minutes.}$$

by altering the system parameters as shown in the following:

PARAMETER	ORIGINAL VALUE	NEW VALUE	GAIN (db)
Transmitter Power	200 Watts	2000 Watts	10
Antenna Diameter	3 Feet	5.33 Feet	10
PRF	40	950	

With the new design parameters the altimeter could perform with new values of R' and T' for the special case of the cooperative object ship given by:

$$R'_1 = 1350 \text{ n.mi.,}$$

$$T'_1 = 4 \text{ minutes.}$$

7.2.2 Altimeter Application as Lunar Landing Guidance Unit. The altimeter may be used for providing altitude information at relatively low altitudes during soft lunar landing missions by incorporating a few modifications. The adaptation is possible since this is merely an extension of the normal operating range of the set and not a completely different use.

The low altitude performance of the altimeter is limited by the following design parameters: pulse width, receiver bandwidths, AFC characteristics (if pulse width is reduced), modulator characteristics and the transmit-receive unit response. With the present design parameters the low altitude capability of the altimeter could be extended to 1 n.mi. or 6000 feet. The principal limiting factor would be the pulse width. By incorporating an alternate set of parameters for the altimeter to be switched in when a low altitude operation is required, a low altitude capability of approximately 1500 feet could be achieved. To try to extend the low altitude capability below this value would necessitate extensive changes in the altimeter circuitry, particularly in the modulator circuits, AFC circuits and the receiver detection circuits. Since the transmitter tube for the altimeter is a klystron, it is considered that a CW mode of operation would be advisable for the altimeter in very low altitude operations. In this mode of operation nearly all the altimeter circuits would be replaced by alternates. The same transmitter tube and the same antenna would be used, however. Low altitude performance down to 10 or 20 feet could be achieved.

7.3 RE-ENTRY COMMUNICATIONS STUDIES

7.3.1 Introduction. The problems encountered with electromagnetic signal propagation through the plasma surrounding a re-entry body is described in Volume III, Book 5, Landing Guidance and Flight Test Telemetry. On a full velocity re-entry the M-1 Apollo vehicles will be in communications blackout for a large portion of the flight, when conventional frequencies are used. Telemetry equipment operating in the K_a-Band spectrum is now available and it will allow communications after 150 seconds from re-entry in all trajectories within the safe corrective limits. It is certain that K_a-Band will allow continuous all-direction communications for orbital re-entry of the Apollo M-1 vehicle.

Continuing research and development is being made to extend both the usefulness of K_a-Band telemetry (DRET) and to find techniques which will offer continuous communications during the severe pull-up conditions that accompany the Apollo re-entry from escape velocity.

A program to provide two-way communication link based upon present DRET equipment is in a position where such a system to provide acquisition, telemetry, command, and speech capability is believed possible within two years. A more sophisticated design utilizing continuous, rather than pulse, modulation is also under development.

The principles evolved for two-way communication are not restricted to particular DRET frequencies, so the findings are generally applicable.

The plasma penetration ability of higher EHF bands will provide some help, but at present it is not known whether 81 GC/S will completely eliminate the blackout for Apollo. The 81 GC/S frequency is in the atmospheric transmission window immediately above 35 GC/S. However, atmospheric attenuation is still great at low angles to the horizon, and the transmission "space loss" is much greater because of the small antenna sizes.

7.3.2 Two-Way Re-entry Communication, EHF

7.3.2.1 General. In addition to the work on the compact DRET telemetry system, preliminary investigations are underway concerning two-way re-entry communication system. This system would take advantage of the previous DRET work and use it for the air-to-ground transmission system. In addition, a new system operating from the ground to the space vehicle would be designed. The resulting combination would then be a simple two-way re-entry communication system. Addition of Speech modulation at both terminals of the two-way system would allow two-way Speech transmission between the re-entry vehicle pilot and the ground station that may be controlling his testing. Development of compatible Speech modulation equipment has been accomplished. In addition to the obvious psychological advantages of two-say Speech transmission, the pilot could make detailed observations of the re-entry maneuver that could not be telemetered, and transmit these observations at the time of occurrence.

The two-way capability would also be used to obtain radar position information concerning the vehicle. A suitable ground station would operate as a conventional radar system, and the vehicle equipment would act as a transponder. The two-way communication signals could also be used to command-control the vehicle during the critical re-entry period. This approach appears particularly attractive during the re-entry portions of an Apollo vehicle trajectory where all forces that act on the vehicle are controllable.

Results of the research and development work on two-way re-entry communication has indicated that at least a flight test experiment is necessary to demonstrate feasibility of vehicle receivers. Flight tests will establish the feasibility of a trailing antenna to reduce the frequency necessary for transmission. A re-entry vehicle receiver experiment is now being designed for flight. Testing will begin this year.

7.3.2.2 Design Study for 35 GC/S Two-Way Re-entry Communication Equipment

7.3.2.2.1 General. The configuration described below is based upon existing components and known electronic techniques. The configuration is one which may be designed immediately with a target date for model delivery in approximately 18 to 24 months.

From system studies, the following requirements appear appropriate. These are preliminary estimates, but variations may be made according to further establishment of range, power, and data values.

For space acquisition, a maximum range is required to conduct angle and frequency search. Two thousand nautical miles is anticipated. This will allow three to four minutes for lock-on, data gathering, computation, and command transmissions before re-entry.

7.3.2.2.2 Tracking. While a lower range is required for tracking, it is also true that geometric restrictions are imposed. For estimation purposes, it was determined that extremes of K_a-Band blackout in the forward direction can exist between 190,000 and 120,000 feet altitude on descent. It may be assumed that Earth Stations (E/S) are located such to just be able to see the vehicle at altitudes and low horizontal angles and ranges as computed. E/S on aircraft and at sea level are considered. Five degrees is used for the lower horizontal angle limit when operating at ground level because of atmospheric absorption. Thus, equipment ranges necessary for the best utilization of E/S may be estimated as follows:

a. Airborne Tracking Station at 40,000 feet (at 0° elevation). Maximum visible range to R/V at 190,000 feet is 450 n.mi. Maximum visible range to R/V at 120,000 feet is 361 n.mi.

b. Sea Level Tracking Station (at 5° elevation). Maximum visible range to R/V at 190,000 feet is 261 n.mi. Maximum visible range to R/V at 120,000 feet is 177 n.mi.

7.3.2.2.3 Basic Information Rates. Basic information rates are shown in Table 13. Equal re-entry vehicle and ground rates are assumed.

A wide variation requirement in the pulse ratio may be expected between the SPPM and PCM coding methods. A desire to keep emitted power from the vehicle to a minimum and other practical considerations indicates that synchronously addressed pulse position modulation (SPPM) is preferable to pulse code modulation (PCM).

7.3.2.2.4 Frequency Selection. For the purpose of establishing a preliminary configuration, 35 GCS frequency is selected because it is the highest direct re-entry frequency compatible with atmospheric absorption and also the availability of R-F generators which have appreciable power. Equipment of proven capability for re-entry telemetry, utilizing magnetrons at K_a-band, is in existence. Airborne tracking operations from 40,000 feet altitude could utilize higher frequencies for better plasma penetration, if such suitable generators become available, without essential alteration of the configurations. In this respect 81 GCS is of interest, since it is in the next highest atmospheric window. Generators at this frequency are becoming available at about 10 kw peak pulse power.

Table 13. Basic Information Rates

DATA	ANALOG	SPPS 0.2 PERCENT ACCURACY	PCM BITS/ SECOND
Inertial Platform	3 Axis 0.01° at 15 C/S	180	2000
Sensors	30 0.5% at 1 C/S	60	660
Actuators and Switches	10 Channels 500 Levels	100	1000
Subtotal for Control		340	3660
IF Speech	2500 C/S	1000	----
Vocoder	3000	650	3000
Telegraph Display	5 C/S 500 Characters	10	1000
R/V Pulse Totals	Control Plus Manned Equipment		
	IF Speech and Telegraph	1350	
	Vocoder and Display	1000	7660

7.3.2.2.5 Frequency Control. Plans for state of the art Elementary Configurations are to operate both vehicle and Earth Station on the same carrier frequency in order to eliminate standards in the vehicle. Advanced configurations could use frequency off-set and CW/FM.

The characteristics of a single frequency dual link are listed:

a. Vehicle transmitter runs without tuning.

b. The Earth receiver indicates the vehicle frequency (with Doppler error, 1 - 2 MC maximum).

c. The Earth transmitter tunes to the frequency of vehicle transmitter (subtracting twice the estimated or computed Doppler frequency).

The vehicle receiver is tuned to the frequency of its own transmitter as soon as it is in operation. Therefore, the vehicle receiver is prepared to receive earth signals, which arrive at the same frequency as the vehicle transmitter. Separate transmitter and receiving antennas and ferrite switches protect the receivers from airload. Diplexing and range measurement are easily accomplished for single frequency SPPM exchange as described in the next section.

7.3.2.2.6 Modulating Code, Multiplexing and Range Measurement. In the proposed use of a single frequency on both data links, it is possible, unless suitable provisions are made, that the transmitter at either station may pulse and cause interference in its companion receiver during a received message period causing false or lost information. In order to avoid this situation, several multiplexing methods have been devised. The simplest is related to synchronous pulse position modulation (SPPM) and assumes that 90 to 100 data channels will be sufficient in early versions as indicated in the data estimates. A receiver is isolated from its transmitter by use of separate antennas and ferrite switches controlled by blanking circuits.

Since studies have shown SPPM to be a data link adequate in capacity and lowest in average power (hence, vehicle loading) at the expense of inefficient transmission bandwidth utilization (which is a minor problem at millimeter wavelengths), SPPM is proposed for both re-entry vehicle and Earth Station data coding. For simplicity of explanation, 99 channels plus a synchronizing channel is the commutator capacity. The frame rate is 10 C/S. Thus, a synchronization pulse group (SP), unique in character, gives a timer reference at 10 C/S.

Present DRET telemetry use a doublet for SP in order to distinguish it from the single data pulses. The time period between SP's is divided into a sequence of equally spaced smaller intervals called commutator segments. The sequence number of each segment interval is assigned a channel number, and data addressing is by assigning channels. The synchronizing and data segment switching is controlled by input and output commutators, at transmitter and receiver respectively, running at a constant cycle rate (one to one correspondence with the SP reference signals). Within each segment, with identical subdivision, are intervals of three equal-length time bases occupying about one third of the interval period. An exact harmonic relationship of 1:100:300 is maintained with the frequency of synchronization pulse, commutator switching, and time bases phased at the instant of their initiation. The position of a signal pulse on the time base indicates relative magnitude of the channel quantity transmitted with the segment period. Figure 28 illustrates the proposed SPPM vehicle transmission format in signal-time domain.

The format for the Earth transmission is essentially the same, except that there are two time bases within the commutator segment and they occupy nearly two thirds of the available time.

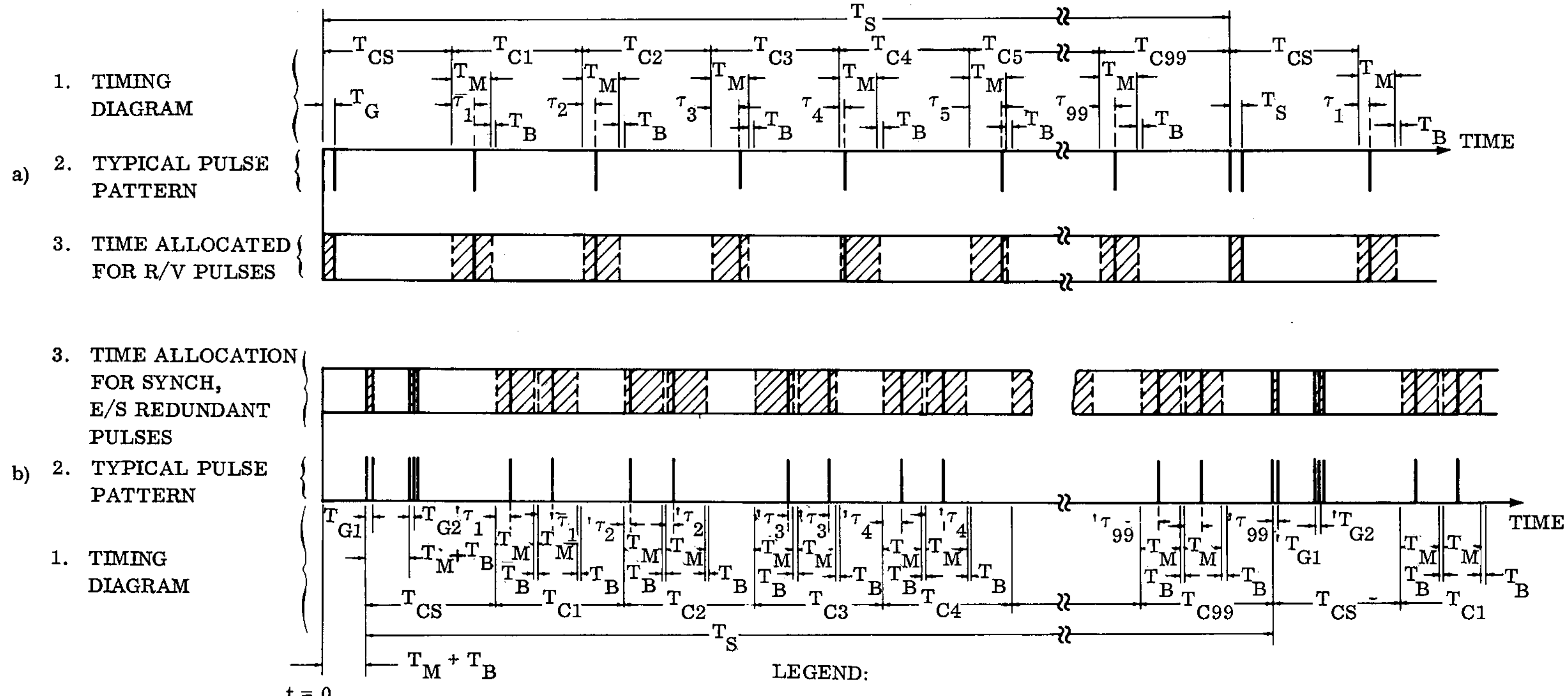
The fixed delay period is slightly longer than one time base; thus, it is assured that succeeding transmission will not occur during the period of vehicle signal expectancy. It is called to attention that the duration of three time bases does not sum to the period of one commutated channel - the difference being the time required to blank and unblank the receivers three times.

At the vehicle, because of constantly changing range, there can be no assurance as previously mentioned that an earth signal will not arrive during a vehicle transmission. This is the reason for the repetition of each ground synchronizing and data pulse. Because of the described time allocation, at least one signal will be received. If both earth synchronizations are identical in character, it is sufficiently adequate for synchronizing the vehicle decommutator if we say that the SP received is always treated as the second set. Then if the first set were lost, the commutator would be in the same correspondence with the input commutator with respect to the signal positions within the segment period. However, if the first SP were received, the vehicle commutator would be running advanced by one time-base period (plus blanking time). This would cause no difficulty since no information can occur in the advanced portion. Still the segment spans the two data carrying time bases.

Range measurement is made without extensive instrumentation on this multiplexing system. Since the Earth Station SP is slaved with a fixed time delay after receipt of the vehicle SP, the time measured at the vehicle between transmission of the master SP and receipt of the slave (beacon) SP, less known fixed delays, is the range period. Range measurement made at the vehicle, provides range information there, and is telemetered to the earth. If both earth SP groups are identical, there is an uncertainty of correct value by the time base range equivalence. This may be resolved if it is known which of the two pulses is used and if they are distinguishable.

Tentative values for vehicle SPPM telemetry are set out in Table 14.

7.3.2.2.7 Acquisition Plan. The acquisition plan depends upon 'a priori knowledge of possible frequency bandwidth of the vehicle transmitter and direction to target within a specified probable solid angle. A horn type antenna, for low noise, of beam coverage compatible with expected angle is used for acquisition. It is here that a maser could be used for 10 to 20



a) VEHICLE TELEMETRY SPPM FORMAT

b) EARTH STATION SPPM FORMAT WHEN SYNCHRONIZED TO R/V PATTERN ON RECEIPT OF SP AT t = 0

LEGEND:

T_S	-	SYNCHRONIZATION PERIOD (SP)
T_{CX}	-	COMMUTATOR PERIOD X = S = (SP) SEGMENT X = 1, 2, ---, 99 - CHANNEL SEGMENT
T_M	-	MEASUREMENT TIME BASE PERIOD
T_B	-	BLANKING PERIOD
τ_Y	-	VEHICLE INFORMATION PULSE POSITION T_M Y = 1, 2, ---, 99 - CHANNEL
T_G	-	VEHICLE SYNCHRONIZATION SIGNATURE
$'\tau_Y$	-	EARTH STATION INFORMATION P. P. ON T_M Y = 1, 2, ---, 99 CHANNEL
$'T_{G1,\ G2}$	-	EARTH STATION SP SIGNATURES

Figure 28. Two-Way SPPM Multiplex Data Format

Table 14. Vehicle SPPM Telemetry

TENTATIVE VALUES	
Synchronization Period (T_s)	100μs
Commutator Segment Duration (T_c)	1 ms
Blanking Period (T_B)	4μs
Time Base Period (T_m)	329μs
Analog to Pulse Standard Deviation (σ_t)*	0.25μs
($3.5\sigma_t \sim 0.9996$ Confidence)	0.87μs
Data Sample Rate	990 SPS
Total Pulse Rate	1000 PPS
Pulse Duration	0.25μs

*$\sigma_t = 0.376 \exp[-\text{db Dynamic Range}/10]\ T_m$, (assume linear sweep and amplifier), $\sigma_t = 0.25\mu s$ if Dynamic Range is 60 db.

db S/N improvement to gain time for frequency-space search. Simple delay type coherent detectors could also be used if vehicle transmits at constant known PRF for about 10 db. Once target is acquired, the tracking antenna searches the acquisition beam and locks on. Lobe switching of two orthogonal fan beams may be required for tracking by the acquisition system. By use of these methods, 20 to 30 times the solid angle of the tracking antenna may be searched for acquisition at the same S/N threshold.

7.3.2.2.8 Antennas. A research study into antenna pair polarization shows that fading due to random relative orientation is minimized when one antenna is circularly polarized and the other is linear. When orthogonal polarization is employed by the linear antenna and diversity combination is made of the receiver outputs, then a high average factor of the theoretical maximum is attained. Equal gain video combiners, although slightly lower in average reception than maximal S/N ratio combination, is suggested on grounds of equipment simplification and reliability.

The plan then, is to transmit circular polarization from both re-entry vehicles and Earth Station. The respective receivers will be Diversity Polarized.

7.3.2.2.9 Operational Features. Operational features are shown in Figures 29 and 30.

The telemetry system is, in general, similar to DRET-SPPM telemetry equipment. It is noted that the commutator is flexible and most any arrangement is possible, depending on logical circuitry and capacity limit. Blanking signals protect the mixers by ferrite switches and can cause elimination of any data received during a transmission interval, which has been previously discussed.

The single frequency two-way system has an important function of establishing reception at the vehicle at the earliest possible time by pretuning the receivers to the frequency sample provided by the vehicle magnetron. When re-entry vehicle transmission starts, the Blanker Action is suppressed until it is possible to receive the re-entry vehicle magnetron signal and the re-entry vehicle receivers are set on their frequency. By arrangement the E/S transmits on the frequency it receives. Thus, the re-entry vehicle receivers are prepared in advance, except for Doppler effects, to the proper frequency. Automatic frequency tracking then follows drifts in the system.

The instantaneous speech (IF) is a method of modulating pulse transmitters at low pulse rates for speech communication. This device is simple and highly intelligible and is compatible with the DRET Telemetry System. In the present system it is used on independent K_a-band frequency.

There is considerable reliability in this system, since both transmitters and diversity receivers are identical. Failure in the data transmitting channels may be remedied by switching to the speech magnetron. A diversity receiver will continue to work when one polarization section has failed, but with some loss of fading gain margin. The best receiving set would always be on the command data channel, since this is the more important link.

The E/S equipment operates in essentially the same manner as the re-entry vehicle equipment with the exception that its SP is slaved to receipt of the vehicle SP for range measurement and non-interference purposes. Two magnetrons with an average data rate over 500 SPS is needed to maintain sufficient average power with presently available tunable K_a-band

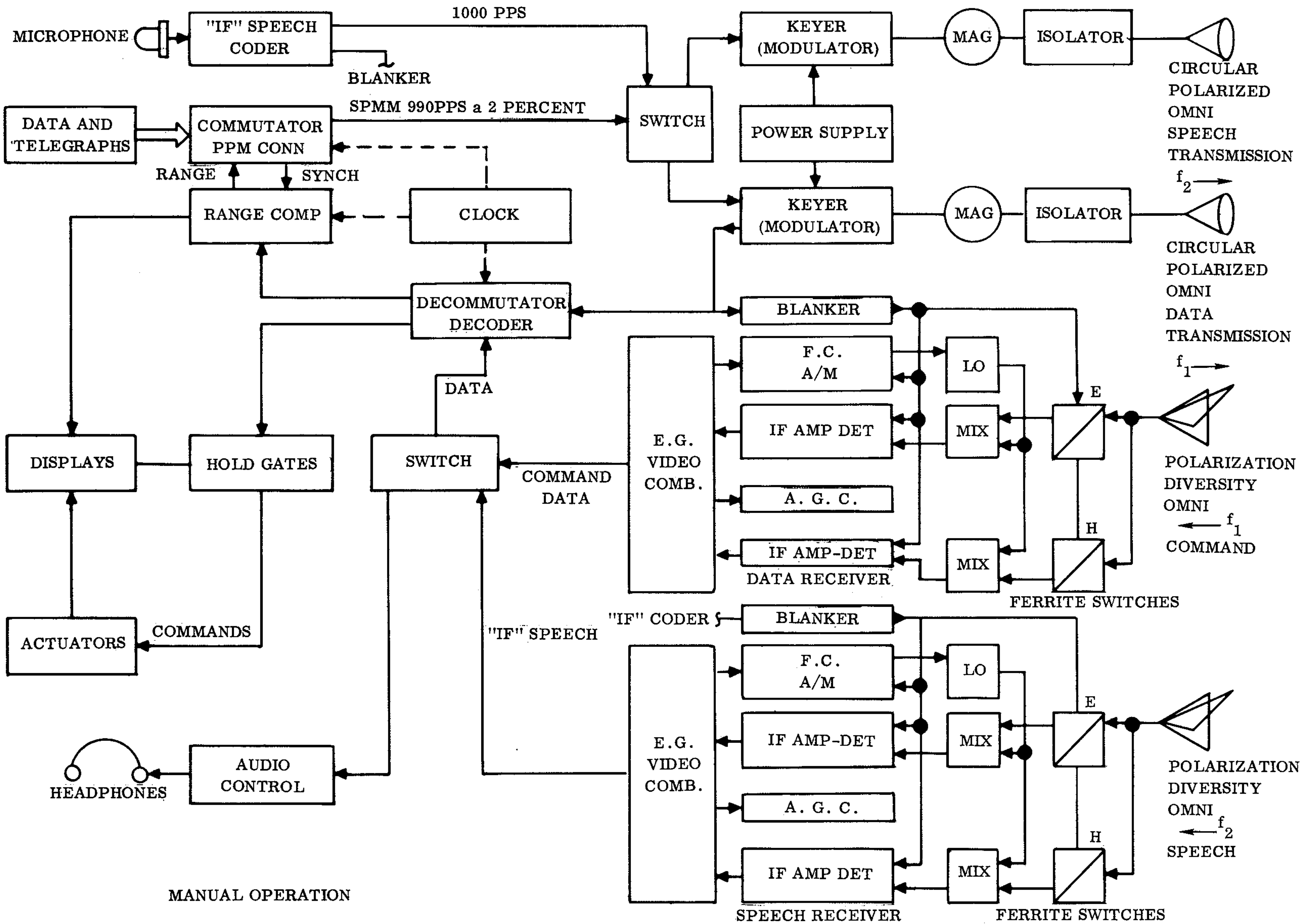
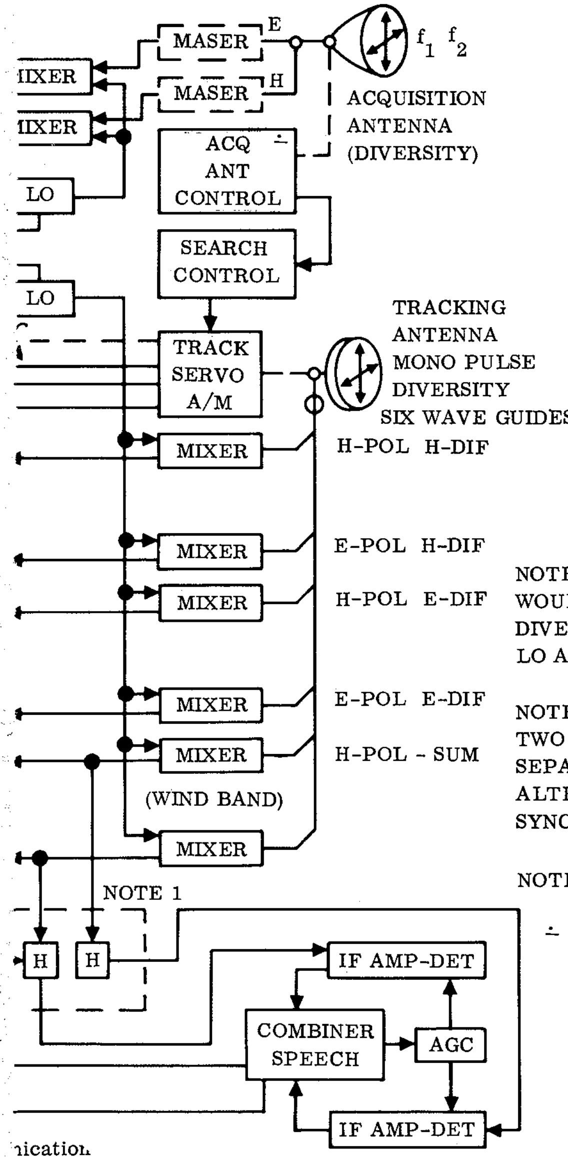


Figure 29. Configuration Vehicle Station Dret--SPPM Two-Way Communication Equipment



NOTE 1: ALTERNATE CONFIGURATION WOULD USE A SEPARATE SLAVED DIVERSITY ANTENNA, CONVENTIONAL LO AND TUNER FOR SPEECH.

NOTE 2: DATA TRANSMITTER CONTAINS TWO TUNABLE MAGNETRONS WITH SEPARATE OUTPUTS. THE KEYER IS ALTERNATELY SWITCHED AT A RATE SYNCHRONOUS WITH PPM FORMAT.

NOTE 3: BLANKING CIRCUITS NOT SHOWN.

∸ MANUAL CONTROL

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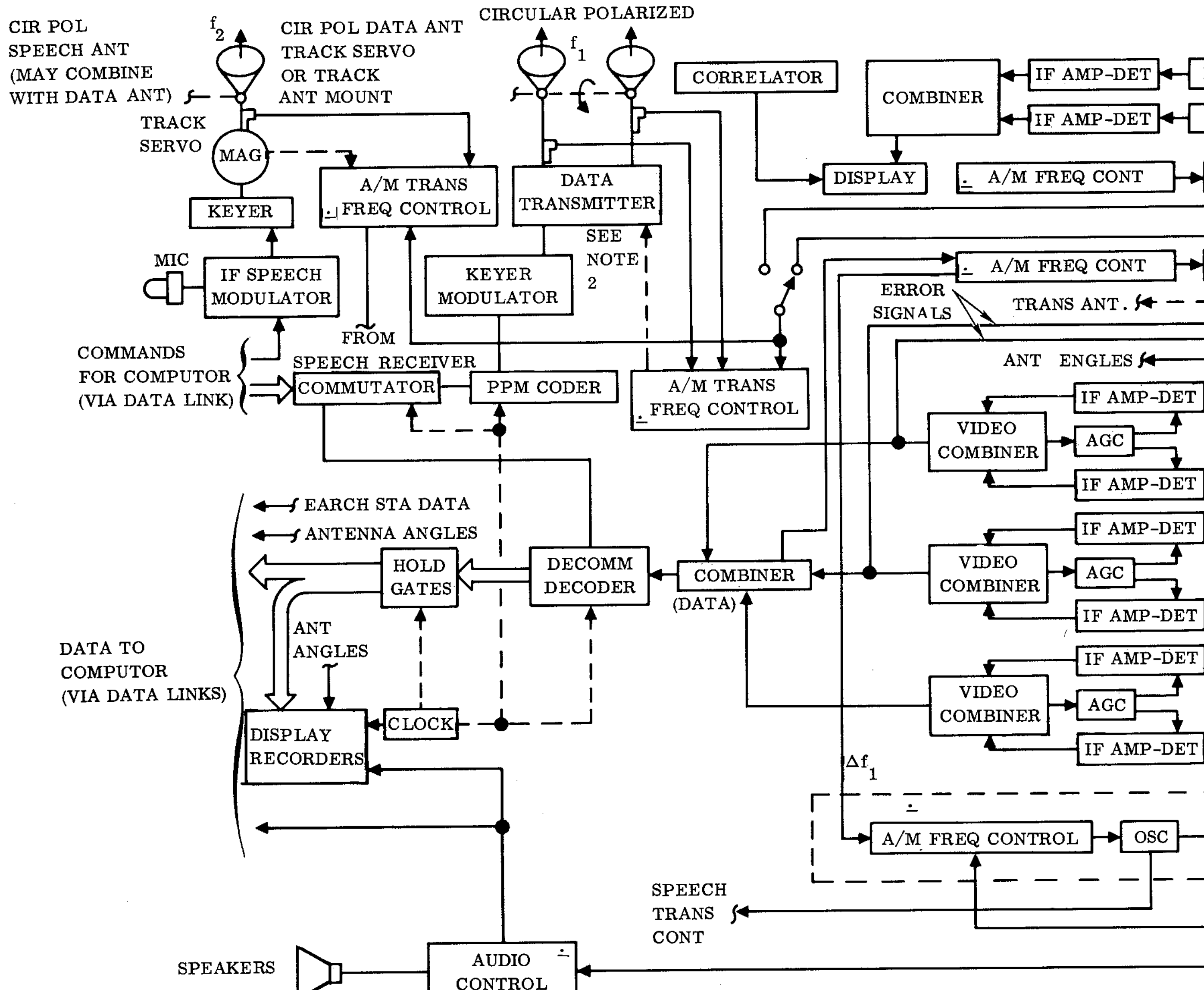

Figure 30. Earth Station Dret--SPPM Two-Way Commu
Equipment Configuration

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magnetrons. The array of receivers is necessary for the diversity monopulse antenna; the purpose of which is to assure reception from fading due to tumbling.

Acquisition equipment is independent of the data-speech system. Once re-entry vehicle frequency and direction are determined, the proper L.O. frequency is given to the tracking receiver and the angles are given to the tracking antenna servo. Once tracking is established, the acquisition system follows along furnishing new information in case of track break, or furnishes an alternate information channel in case of better reception.

7.3.2.2.10 Weight and Power Estimates. On present knowledge of DRET equipment and experimental receiver design, the following is estimated for the vehicle in Table 15.

7.3.2.3 Re-Entry Vehicle Receiver Test Program. Re-entry communication systems are designed to provide a data link for handling re-entry vehicle data in real time while the vehicle is surrounded by highly ionized plasma sheath. In addition to the one-way capability, a two-way communication system is being designed. One of the major problems associated with the two-way communication system is the provision of a suitably integrated re-entry vehicle receiver. Through use of this receiver and associated ground transmitter, a two-way re-entry communication capability may be maintained.

Flight testing associated with an actual re-entry vehicle flight is necessary to determine the intensity, duration, and gradients of the plasma. At the present time there is no conceivable way of duplicating this information other than flight test. This has been planned as an auxiliary-type experiment in the Titan re-entry vehicle program for October and November, 1961.

7.2.3.1 Objectives. The objective of the receiver program is to design and establish the degree of feasibility of K_a-band reception within a re-entry vehicle during re-entry. The experiment will indicate the noise level and received signal strength before, during, and after re-entry.

The analytical work necessary to establish the theoretical feasibility of the re-entry vehicle experiment is complete and is discussed below.

There is no reason to believe that the receiving system, antenna patterns, transmission path, etc., will not be the same as the transmitting path condition and, therefore, reciprocity holds concerning these factors. However, the plasma temperature is not expected to exceed 5000 degrees Kelvin regardless of the vehicular type, and will, therefore, not seriously effect the receiver. The receiver noise temperature is 7000 degrees Kelvin. The total effect that plasma noise can have on the receiver performance is less than 2.3 db. If the plasma temperature exceeds 5000 degrees Kelvin, the plasma attenuation will be excessive and will black out the received signal. An analysis of the temperature distributions around a re-entering vehicle show that antenna patterns will not be seriously affected, unless the system is already blacked out. The radiometric emissivity of the plasma is equal to the attenuation factor. It can be seen that it is clearly feasible from a strictly theoretical viewpoint to provide K_a-band receiver in the re-entry vehicle and integrate this receiver into a vehicular system.

7.2.3.2 Trailing Antenna Experiment

7.2.3.2.1 Introduction. Current efforts on direct re-entry communications have been limited to unmanned vehicles. In the future, it is expected that direct re-entry communications will be practically essential for manned space vehicles. Past success achieved in the 35 GC/S development and flight test program indicates that it is reasonable to extend the knowledge of plasma attenuation into areas about the vehicle which are believed to be considerably cooler than the locations presently utilized for telemetry antennas. After verification by actual experiment of the theoretical analysis for a trailing antenna, conventional frequencies such as 5 GC/S might be used in re-entry communications instead of 35 GC/S. Otherwise, for high velocity re-entries, where a cylinder-mounted antenna requires 80 GC/S, a trailing antenna should be suitable at 35 GC/S.

As a continuation of the direct re-entry communication development program, the proposed trailing antenna experiment is designed to determine, by re-entry flights, the extent of electromagnetic transmission, plasma conditions, and heat inputs for an antenna system located at the rear of a vehicle. A trailing antenna position is sought from which the radiating element will transmit through as cool a plasma as possible. In this experiment, the radiating elements will be positioned to the rear of a re-entry nose cone within the limitations of structural and aerodynamic considerations. The tests are aimed at plasma penetration in a location where electromagnetic absorption is expected to be less severe than along the sides of the vehicle.

The equipment development to support this program is proceeding toward a last quarter 1961 flight date.

7.2.3.2.2 Plasma Considerations. Theoretical analysis to date shows that it may be possible to gain several advantages with a trailing antenna

Table 15. Estimate of Weight and Power, Elementary Vehicle, Two-Way 35 GC/S Communication Equipment

ITEM	UNIT POUNDS	UNIT WATTS	QUANTITY	TOTALS POUNDS	TOTALS WATTS
Transmitting Antenna	3	-	3	8	-
Receiving Antenna	3	-	4	12	-
Waveguide	0.15/foot	-	100 feet	15	-
MA-200 Magnetron	10	-	2	20	(in keyer)
Keyer (75 Percent Utilization)	10	300	2	20	400
Power Distribution	5	2	1	5	2
Diversity Receiver	13	60	2	26	120
Comm -Converter	5	15	1	5	15
Recom -Decoder	5	15	1	5	15
Speed Modulator	1	10	1	1	10
Audio Control	1	5	1	1	5
Hold Gates	6	10	1	6	10
Range Comp	3	15	1	3	15
Blanker	1	5	2	2	10
RF Switch	5	-	1	5	-
Components Subtotal				132	597
Structure at 30 Percent				40	
Cables at 20 Percent				16	
Equipment Total				188	597
Power Supply 0.025 pounds for 0.5 hour WH (Silver Cells)				8	300 WH
GRAND TOTAL				196 Pounds 350 Watt-Hours Electrical Energy	

transmitting system. However, most calculations make broad assumptions regarding the boundary-layer phenomena about the vehicle and the conditions inside the wake. These assumptions are made in the absence of previous experimental data. The temperatures and flow line assumptions are based upon an atmosphere around the vehicle free of any foreign objects. With the introduction of an external antenna pedestal, such as that which would be required in placing an antenna to the rear of the vehicle, additional disturbances are made to the existing temperature and shock layers. Therefore, to verify the validity of theoretical analysis and conclusions, actual feasibility tests will be made by implementing a vehicle for a trailing antenna experiment.

As a by-product of the trailing antenna experiment, additional data will be obtained which will enlarge the knowledge presently available concerning the plasma surrounding a re-entry vehicle. Most of the data to date are concerned with plasma surrounding the nose and the boundary-layer along the sides of the vehicle. With the advent of manned re-entry vehicles, possibly making skip-out maneuvers, additional data will be required concerning the conditions at all points about re-entry vehicle.

The initial approach to placing the antenna to the rear of a re-entry vehicle has been to study all of the factors which coexist in implementing such a system. Studies dealing with the plasma attenuation problem and transmission path losses for particular trajectories have been made. Preliminary studies were concentrated on transmissions on 5.8 and 9.265 GC/S. These frequencies were selected because they lay between 1 and 15 GC/S. Below 1 GC/S, plasma attenuation is objectionable, and above 15 GC/S atmospheric attenuation becomes a problem. Considerable analysis has been given to the antenna location, orientation, directivity, and gain problem, the atmospheric and plasma attenuation problem, and the transmitter-receiver relationships involved in free-space transmission. In addition, the new problems, and their effects and limitations on the design and placement of the antenna structure to the rear of the vehicle, have been explored. Thermodynamics, aerodynamics, and the use of high temperature materials have been investigated.

Consideration has been given to placing the trailing antenna in various positions behind the vehicle. These positions may be broadly classified as: inside the trailing wake on the fringe, and outside the trailing wake, as shown in Figure 31.

7.4 EHF APPLICATIONS. The preceding sections outlined the background and described work underway to give versatility to the EHF system. The communication barrier is a major hurdle in man's conquest

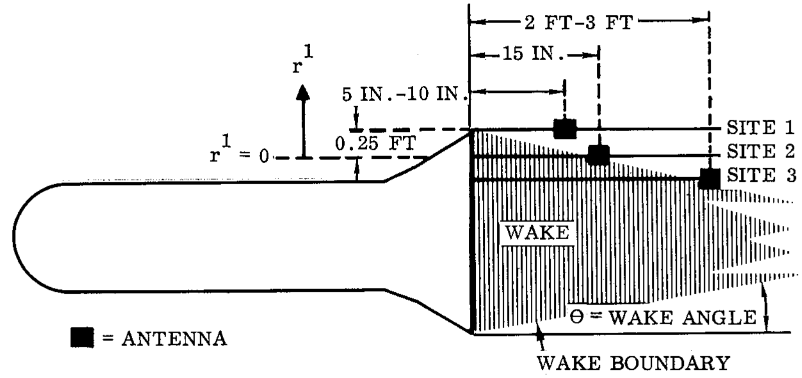


Figure 31. Plasma Analysis Geometry

of near orbital speed within the atmosphere; its attainment falls next beyond the thermal thicket. In this regime radio communication is possible by utilization of EHF to earth commuting vehicles. Specific application studies have been made for systems presently under design. Approaching maneuverable vehicle programs indicate a real need for command capability, especially in the flight test stages. Some applications are discussed.

7.4.1 ICBM Telemetry. The original 35 GC/S work was conceived to overcome the blackout phenomena of radio transmission during re-entry. The telemetry equipment has been developed and is being flight tested. To date, it substantiates theoretical predictions of superiority over VHF. Sets are now available for telemetry of ICBM data and could be easily modified for most ICBM re-entry telemetry requirements. This equipment is basic for the proposed Apollo test telemetry and the suggested command control system for re-entry navigation. Another application in design is a device for miss-distance indication for Nike-Zeus evaluation tests.

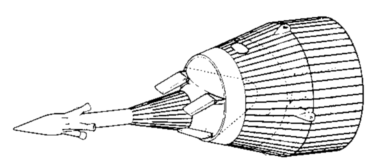
7.4.2 Maneuvering Re-Entry Missiles. Several ICBM's have been proposed which would correct impact errors by aerodynamic control after re-entry, or would re-enter short of the target and fly the remaining distance to avoid detection. An essential part of developing such missiles is the test flight program. Here command by DRET can be used to more accurately command responses than pre-programmed flight control. It is conceivable that improper pre-computed and stored commands could lead to instability and a lost vehicle. Two-way communication studies have indicated that the system outlined in Section 7.3.2.2 (without speech) could fully control the test flights, after ballistic pull-up, with the flight computer remaining on the ground.

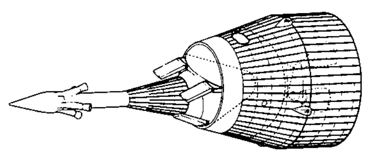
7.4.3 Sub-Orbital Vehicles

7.4.3.1 Ballistic. Studies on the aerodynamic conditions of sub-orbital vehicles have been made establishing the suitability of this type of vehicle to continuous communications with EHF equipment. The velocity-altitude trajectories, in conjunction with the low re-entry angles, are such that 35 GC/S is continuously useful providing some precuations about antenna locations are taken.

7.4.3.2 Maneuvering. The most significant aspect of communication with this type of vehicle is in connection with the maneuvering glider. When the W/C_DA is less than 100 and the L/D ratio is on the order of 0.5 ($C_D = 1$), the maneuvering flight regime is such that K_a-band energy may be propagated in all directions, even through the nose. No lower frequency compatible with atmospheric attenuation will do the same. The great advantage of forward propagation ability is that no maneuver boundaries are imposed on the flight. A vehicle described by the above parameters is limited to 3g deceleration and a human occupant could perform limited functions for a prolonged period. The ballistic parameter, W/C_DA, for the present Dyna Soar vehicle is in the order of 50. Considerable work has been done on continuous two-way communications for this vehicle to design independent equipment for acquisition, position tracking, command, and telemetry on 35 GC/S. A pulse-position-modulated single frequency link can be provided for position tracking, data, and command, when test and operational requirements indicate that continuous communication is desired.

7.4.3.3 Re-Entry Vehicles, Orbital and Space Probes. There are many considerations required before re-entry communications for orbital or space probe vehicles such as Apollo can be specified. These encompass trajectories, shapes, aero-physics, and plasma. The applications of EHF depend upon the communication requirements, antenna locations, ablation, and earth station locations. During the study phases of this program, all the aspects of this approach were considered. After the vehicle shape and re-entry trajectories were known, the plasma attenuation for the worst trajectories was computed for various antenna locations. Although 80 GC/S allows less interruption, 35 GC/S is selected as proper frequency because of hardware availability. At worst, only the first 150 seconds after re-entry would be lost. Aircraft are proposed to carry the earth station tracker because of their logistic maneuverability and the lessening of atmospheric attenuation (by being above the weather and much of the atmosphere). A link of two to five aircraft will be required to cover the track, depending on the range of coast out and landing point navigation.





8 RELIABILITY CONSIDERATIONS

8.1 DISCUSSION. Consideration can be given to Apollo as a manned carrier for moon surface landing as a practical development program based on its capability as a manned moon-orbital vehicle.

Reliability investigations in such a development program would be focused primarily on the required additional subsystems and secondarily on the modified existing Apollo moon-orbital subsystems. A study of reliability apportionment would be accomplished on additional subsystems for Apollo-moon landing requirements.

a. Transition from moon orbit to descent flight path.

b. Attitude control during descent with man as a control factor. factor.

c. Landing gear configuration.

d. Determination of moon surface adequacy to support a landing at the location selected.

e. Determination of rate of descent during landing.

f. Vehicle-installed subsystems checkout and launch control subsystem.

g. Life support for the crew external to the vehicle.

h. Extended operational life requirements.

i. Moon-launched flight control and guidance subsystems.

A study of reliability re-apportionment would be accomplished on the modified Apollo-moon orbital subsystems for Apollo-moon landing requirements.

a. Vehicle structural modifications to safely withstand landing stresses.

b. Additional propellant and pressurization storage.

c. Flight-landing-programing re-set and adjustment by crew.

d. Inter-crew communications on the moon's surface.

e. Stringent pressure integrity of all pressurized subsystems as a result of a more extensive and intensive mechanical shock and vibration environment.

f. Higher capacity vehicle electrical power system with a significantly extended operational life.

The apportionment of the new subsystems and re-apportionment of the modified subsystems would then be integrated to form a predicted mission reliability. Reliability growth, based on past experience with Apollo-moon-orbital, could be depicted at the subsystem and mission level.

The two major areas of reliability effort and contribution in a moon surface landing vehicle development program are man as a major vehicle control element and the fact that the Apollo moon landing vehicle is a natural outgrowth of the Apollo-moon-orbital vehicle.

8.2 APOLLO FLIGHT SAFETY SYSTEM CONCEPTS. Current experience with flight safety systems for manned space flight has led to the design of a system which monitors the performance of critical booster subsystems during booster powered flight and generates an abort command if a catastrophic failure is imminent. Due to the short time between malfunction and failure, automatic failure detection is required. After booster burnout an abort command can be generated by the pilot or by the ground control. For advanced space missions such as lunar landings, etc., a safety system with dual capability is called for. As in the case of the system mentioned above, it must be able to command an abort in case of an impending catastrophic booster failure, but there may be malfunctions which would not lead to catastrophic failure but would prevent completion of the scheduled space mission. This might not rule out the possibility of changing the mission to a useful probe or earth orbital flight. The flight safety system should have the capability of programing the flight into such an alternative trajectory to make full use of the reduced capability. The system must be capable of shutting down one or more engines, initiating staging sequences, re-computing trajectory guidance equations for the airborne computer and accepting crew inputs.

Such a system will assure a maximum of useful and safe missions by compensating for any booster subsystem malfunction in such a way as to make the best use of the vehicle's energy and capability.