TRENDS IN MANNED SPACECRAFT SUBSYSTEMS

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I. INTRODUCTION

The purpose of this paper is to provide the system oriented space engineer and scientist with a perspective view of the growth of manned spacecraft subsystems from first flight to future requirements and the techniques for accomplishing these requirements. Rather than attempt to describe each requirement and development which has been achieved or will be achieved for the many subsystems on modern spacecraft, the accomplishments, growth and future of a selected set of subsystems is traced to develop trends. The vehicle attitude control and life support systems whose design is usually very dependent on vehicle and mission requirements are not treated. Likewise the mission subsystems for rendezvous and rescue and the military mission subsystems for rendezvous, docking, inspection, reconnaissance, recovery and all weather landing are not discussed. Subsystem trends are developed for the following subsystems:

- Guidance
- Pilot Display and Control
- Communications
- Power Generation
- Environmental Control

The first part of the paper is devoted to describing the requirements and capabilities of these subsystems for the currently contracted manned spacecraft programs.

What we have learned from Mercury flights, analytical work and ground tests on the programs yet to fly is then described by choosing examples to illustrate trends.

Finally, the remaining portion of the paper is devoted to what future subsystems need to do and techniques which may be employed to achieve these more stringent requirements.

The manned spacecraft subsystem trends as developed by this paper can be summarized as follows: the subsystems must do more for longer times with increased reliability and at less weight and power. The most useful concepts developed to accomplish these increased objectives are further exploitation of the use of man as an active element in the subsystems; the use of backup systems on the vehicle, or ground based, which permit partial, safe mission completion, and the implementation of the best combinations of reliability improvement techniques for the specific mission and subsystems involved since reliability is the biggest single problem facing future manned spacecraft subsystems.

II. THE SUBSYSTEMS IN CURRENT MANNED SPACECRAFT PROGRAMS

II.A. Guidance Subsystems

Figure 1 compares the guidance subsystem requirements and capabilities for the currently programmed manned spacecrafts.

Mercury employed ground based guidance for the simple reason that successful manned flights were a prerequisite for introduction of the man and man's capabilities in the zero g environment of space were too unknown to place primary dependence on him.

Little use was made of man to guide the Mercury vehicle. An override on the retro function was provided to permit firing the retro rocket manually if ground control failed so that the pilot could at least return himself to earth.

Attitude control involving modes from ground controlled automatic, automatic under pilot control, to strictly manual control were provided and utilized to good effectiveness when failure occurred but this was attitude control not guidance. Man lived up to our highest expectations and proved to be dependable and adaptive.

The X-20, planned from the start as a system to demonstrate self-contained capability, is equipped with an inertial system and ground tracking information is not required or normally employed. The guidance system although designed to provide, as in Mercury, for unmanned flights is designed primarily for pilot usage. The pilot may choose automatic flight to a selected destination within a 5,000 by 3,000 mile footprint or may direct the vehicle manually by means of a continuously corrected energy management display to any one of ten destinations or abort sites which can be reached from almost every point on the trajectory. With the large footprint provided by the high lift to drag ratio capability of the vehicle, selection of alternate landing sites located several hundred or thousand miles apart is possible after the retro rocket has been fired.

Because of the large forgiveness factor provided by the large variation in lift to drag, an emergency re-entry system utilizing directly measured values of perigee acceleration and temperature can be used by the pilot to manage vehicle energy to reach a planned destination when a primary guidance failure occurs. In some cases landing at this destination will be possible

With minor emergency re-entry equipment changes landing at the destination will always be possible. As can be seen from further examination of Figure 1 the weight is considerable for this self-contained system as compared to that on Mercury. The reliability of the primary mode guidance system is expected to be inadequate for the initial ten flights. There is therefore a requirement for a backup system of some sort. The emergency re-entry system or an extremely simple backup system (like the one described later in this paper as an example of a way to achieve mission reliability) is required.

The Gemini guidance system employs a ground updated inertial system with the additional feature of a horizon scanner to permit shutdown of the system in space thereby achieving a major saving in electrical energy and hopefully an improvement in overall guidance reliability. With ground updating of position and velocity from a ground tracking network the landing area footprint is in the order of 450 x 150 miles. Should self-contained operation be required (no position and velocity updating) the footprint for mission planning purposes is reduced to the point where only the destination selected at retro firing can be reached. In the case of the X-20 the effect of position, altitude, and velocity guidance uncertainty at retro-rocket firing is to reduce the footprint from 5,000 x 3,000 miles to 4,400 x 3,000 miles.

As in X-20 extensive use will be made of the crew as mode selector and to provide backup capability.

Both X-20 and Gemini systems are provided with sufficient computer capability to permit incorporation of rendezvous and other mission capabilities.

The Apollo command module is called upon to perform a much more exotic guidance mission than the orbital systems described above. The primary system is inertial with a second inertial system installed to enhance reliability. Manual triangulation by the crew and command information from the Deep Space Tracking System can be employed as additional backup for primary guidance failure. Because of the long mission, completion of the mission becomes more practical than abort in many cases. The guidance system therefore needs to be designed to sustain multiple failures and still permit mission completion.

Reliability is therefore the biggest single guidance problem for lunar and, to an even greater degree, for planetary missions.

II.B. Pilot Display and Control

As mentioned earlier, on Mercury man's capabilities in the then unknown environment of space were to be tested, not depended upon from the first. A monitoring capability was provided, therefore, wherever possible and emergency control capability was provided as backup primarily for reliability purposes on important functions such as de-orbit and attitude control as shown on Figure 2 . As we can also see from this figure all other mission functions were controlled from the ground on Mercury.

X-20, with potential military use as a design criteria employed a self-contained rather than a ground controlled concept. Boost is monitored by the pilot and since guidance law gains have been set low, several seconds of warning are available before critical booster angle of attack can be reached. The pilot could take over, in such an emergency, and control the booster.

Automatic and manual primary control and manual backup subsystem control are provided for the injection, de-orbit and re-entry functions. The pilot is always the mode selector and after selecting the mode to control the vehicle he will monitor this system with the remaining modes available. As with Mercury several flight control modes are available.

On the X-20, vehicle attitudes to reach landing choices available are shown on an energy management display. The display mechanizes the concept shown on Figure 3 . Here we see a completely manual technique wherein the pilot selects, based on vehicle energy (velocity and altitude), the proper overlay for the particular path over the flat projection (map) of the earth. With position and course obtained from the inertial system he can position the overlay on the map and determine what landing sites can be reached by reading through the overlay.

The completely automatic system wherein guidance law equations are mechanized within the digital guidance computer to accomplish the same result is also illustrated.

Figure 4 illustrates a laboratory model of an energy management display which mechanizes the manual technique just described in such a way that only one set of symmetrical overlays are required for any path around the earth. Here, a range to go subroutine and a cross range to go subroutine are utilized to generate the range to go (Y AXIS Voltage) and the cross range (X AXIS Voltage) sequentially for 10 landing sites and this is repeated 20 times a second. The result is 10 landing sites appearing as dots on the cathode ray display. Since the sites are plotted relative to the instantaneous velocity vector of the vehicle, symmetrical overlays can be employed. The overlay selected to match the current velocity of the vehicle as indicated by the inertial guidance system is automatically pulled into place in front of the cathode ray display.

The pilot can select his landing site, read off the angle of attack and bank angles to fly and then control the vehicle to these angles or others he may choose to "over" or "under" control the vehicle. In a more recent version of this system the safe flight limits of the vehicle are also plotted on the overlay and another distinctively different symbol is generated on the cathode ray display to denote the vehicles current status relative to this display.

"Backup" energy management displays on the pilots instrument panel permit yet another mode of piloted energy management.

Gemini, as can be seen by referring again to Figure 2 , makes more extensive use of man in control of the vehicle than was done in Mercury. Since range is controlled by rolling the vehicle to modulate L/D , range control is a function of roll regime. With the inertial guidance system aboard the vehicle this systems measurements can be displayed to the pilot for his direct use. Since man was shown to be capable of normal pilot responsibilities by the Mercury flights, Gemini plans are to greatly increase his role in control of the vehicle. Decisions such as utilization of ground based tracking data or self-contained operation to determine retro-rocket firing can be made on board. The pilot will do the guidance shut down and assist in restart of the system. Extensive mode selection to be performed by the pilot is being incorporated into the primary guidance system to enhance reliability. A backup or secondary guidance system may be evolved to enhance mission reliability.

Apollo, with a much more complex mission, even for just the command module, and for a longer mission duration is planned to employ both automatic and manual control and through the crew utilize, as a backup, guidance information from the Deep Space Tracking Facilities. Details of displays and controls were not available since they had not been finalized. Use of the redundant inertial system in the LUNAR EXCURSION MODULE or parts of this subsystem is being studied for example.

Although abort modes will be incorporated, the current NASA concept is to provide sufficient backups to make mission completion reliable.

II.C. Communication Subsystems

Ultra High Frequency (UHF) in the order of 300 Megacycles and High Frequency in the order of 15 Megacycles/s communication was provided on Mercury to provide voice and 75 KC bandwidth of telemetry. The world wide Mercury tracking network was provided with receivers and transmitters for these frequencies. Essentially horizon to horizon coverage is possible except when re-entry blackout lasting in the order of several minutes is encountered at the end of the flight. See Figure 5 .

A requirement for the X-20 communication system was to provide voice and 750 channels of telemetry during the 30 minute re-entry period of the vehicle. Satisfactory communication during the hottest portion of the re-entry flight was considered of utmost importance since telemetry data would be invaluable in determining causes of failure should a vehicle be lost during this portion of the flight. Studies of the flow fields led to choices of low electron density, thin shock locations for the antenna outboard on the under side of the wings and on the top centerline. To minimize the number of ground stations for vehicle angle of attack varying from 15 to 55 degrees, top and bottom antennas were provided. Two transmitters each modulated by the total telemetry and voice information and operating at slightly different frequencies feed top and bottom antennas respectively thereby avoiding pattern lobing by frequency diversity.

Ten to 13.5 kilomegacycle frequencies were found to be the lowest frequencies which remained above the plasma resonant frequency (fp) for all but a few seconds of flight. Attenuations in the order of 60 db corresponding to power levels one million above levels required for free space transmission would be required for transmission at frequencies below fp . The 10 - 13.5 kmc range was also the highest frequency at which sufficient airborne transmitter power could be obtained from available tubes to provide horizon to horizon coverage and thereby reduce the number of ground and ship borne stations. Blackout or unexpected coverage gaps for periods of no more than a few seconds are expected.

II.D. Power Generation Subsystems

Power generation subsystems for specific spacecraft and missions are selected in early vehicle design development phases through comprehensive "trade" studies. These studies assess the relative advantages and disadvantages of alternative system concepts considering factors such as system weight, volume, reliability, servicing and maintenance requirements, compatibility with vehicle configuration limitations, and the several factors associated with system development risk, including the state-of-the-art of the technologies associated with a particular concept and system development schedules and cost.

Figure 6 shows the results of such studies by noting selected systems for existing spacecraft programs. In addition, the curve depicts an estimate of the trend in manned spacecraft power requirements.

Figure 7 depicts a rather conventional method of illustrating the applicable power/time regime for alternative space power systems. The system area boundaries are determined primarily on the basis of system weight and must be treated as broad gray bands rather than firm lines of demarcation due to the significant influence on system selection of factors other than weight as mentioned above. The Mercury, Gemini, and Apollo spacecraft all depend on zinc/silver oxide batteries as a source of power during the re-entry phase of their missions. This selection is consistent with reliability needs (batteries being "static" in operation with long history of reliable operation) and minimum system weight objectives (the re-entry phase for ballistic re-entering shapes being of short duration with relatively low power requirements). Battery power was also found suitable for the Mercury mission orbital phase. However, for orbital duration up to fourteen days as specified for Gemini and Apollo, it was necessary to develop a more suitable power source. Recent developmental emphasis on fuel cells will result very soon in power systems fully qualified to fit the needs of Gemini and Apollo and with continued development, should fill an ever-expanding area in the Figure 7 power/time regime.

With the significantly higher power required for flight control surfaces actuation in exploration of controlled re-entry flight, it was found that a cryogenic chemical fueled dynamic engine best met X-20A mission requirements. Advantage is also taken in this application of integration with the environmental control system to allow the cryogenic hydrogen to serve as a sink for waste heat before it is passed into the power unit combustor.

Space power system application studies have shown the need to emphasize reduction of load demands because of the significant penalties associated with placing large power generation systems and waste heat rejection systems into space. The present high premium placed on space vehicle subsystem weight is expected to continue. Although boosters are in development that will be capable of launching much larger payloads than at present, this increased capability will and should be reserved largely for accomplishing expanded mission objectives rather than vehicle supporting subsystems. For relatively short missions (under 24 hours) and a given power demand, emphasis must be placed on design concepts that minimize the fixed weight of the power system. As mission time requirements increase, ever increasing attention must be given to methods that minimize or eliminate the need for expendable energy sources such as chemical fuels. The high efficiency of chemical to electrical energy conversion exemplified by hydrogen and oxygen fuel cells and the use of solar and atomic energy sources, permit extended duration space missions with reasonable system weight penalties.

II.E. Environmental Control Systems

Figure 8 shows the magnitude of the heat load that must be accommodated in currently programmed space vehicles. The significantly higher heat load of the X-20A vehicle reflects the high electric load requirements for self-contained guidance capability, a reserve for mission subsystems, a large test instrumentation system, and the hydraulic system which remains in operation, although at reduced pressure, throughout the presently planned missions. Cryogenic hydrogen provides the heat sink for metabolic heat, equipment waste heat, and for aerodynamic heat that passes through the structure, insulation, and water wall. The cryogenic hydrogen that is used as a heat sink is subsequently routed to the combustor of the APU's and the excess, if not required by the power unit, is vented overboard. The power requirements, and thus the waste heat load, of Mercury, Gemini and Apollo are considerably reduced from the X-20A requirements. The thermal loads are controlled through water boiling on the Mercury vehicle. Radiators are used on the Gemini and Apollo to reject waste heat to space.

Figure 9 indicates that for space or orbiting missions of approximately six hours or more, radiation of waste heat to space during the orbital phase of a mission provides a weight advantage over the use of stored expendables. For space missions of a week or more duration, the weight of expendables becomes prohibitive whereas radiator weights are reasonably low. The increase in radiator weight with mission duration is due to required protection from meteoroid penetrations and the longer life required of heat transport pumping systems. Improvement in the efficiency of heat radiation to reduce radiator area and weight requirements must be made as spacecraft heat loads increase. Since heat rejection by radiation is not feasible during the re-entry phase, the need for expendable heat sink fluids for this mission phase will continue.

Figure 10 shows estimated weight ranges of both thermal and atmosphere control systems as related to the estimated increase in future spacecraft power requirements shown in Figure 6 and with anticipated increases in crew size and mission duration.

It appears that heat pump concepts to raise the radiation temperature, light weight materials, and high emissivity/absorptivity coatings will be required to maintain low radiator weights for the higher power missions envisioned for the next decade. Atmosphere control will require extremely low vehicle leakage and noxious gas removal methods as well as reclamation of human wastes in the longer duration, larger crew missions. Some increase of expendables will be required even with atmosphere reclamation processes in order to make up leakage and losses due to inefficiencies of reclamation systems.

III. WHAT WE HAVE LEARNED

III.A. Introduction

The subsystems of the currently programmed manned space craft have been described. What have we learned from the flights of Mercury and the development work accomplished to date on X-20, Gemini and Apollo?

Mercury flights have shown that: (1) Worldwide real time ground control is workable but unwieldy and expensive. (2) Man can be depended on in Space.

Since man can be depended on within limitations an operational manned space system with Worldwide flexibility can be achieved at less expense and complexity by providing a self-contained capability so man can make his own decisions in Space. The X-20 and Gemini designs are based on this concept.

Reliability data from the foregoing programs projected to the Apollo and orbital missions of similar duration show that reliability is the spacecraft designers biggest problem.

As an example of what has been learned the communication studies and tests on the several programs are described in the following section.

III.B. Re-entry Communications

Near space communications is similar to conventional aircraft and missile experience when the standard line of sight UHF frequencies are employed. An exception occurs during that part of re-entry when sufficient energy is transferred to the air surrounding the vehicle to cause thermal ionization. This phenomena becomes extremely pronounced for a period in the order of a few seconds for ballistic or near ballistic re-entry and although less pronounced in the case of a higher L/D vehicle may last for minutes. Figure 11 illustrates the white hot shock layer surrounding an X-20 model undergoing test. Note the much stronger effect on the lower surface.

Electromagnetic energy propagates through the plasma surrounding the vehicle when the operating frequency exceeds the plasma resonant frequency (fp). Below this frequency attenuation in the order of 60 db (transmission of only a millionth of the energy) is experienced. fp is a function of the electron density and collision frequency and is defined here by the following equation:

$$fp = 8.98 \times 10^3 \sqrt{Ne}$$

$$Ne = \text{Electrons/cm}^3$$

Plane wave analysis, confirmed by a more exact model for a specific case has shown that the operating frequency must exceed plasma frequency by a factor related to the angle of incident as shown in Figure 12 . To achieve appreciable propagation at incidence angles of 70° an operating frequency in the order of four times the plasma frequency is required.

The plasma frequency for several vehicles Lift to Drag (L/D) values is shown in Figure 13 as a function of re-entry velocity. Here, for simplicity, equilibrium glide at the noted L/D is assumed. From the fp values shown and the angle of incident factors which must be employed it is clear that frequencies in the order of 10, Kilomegacycles (SHF Band) are required for "glide" vehicles and frequencies several times this are required for near ballistic vehicles. Fortunately the plasma exists for a shorter time for the low L/D vehicles thereby requiring only one, or at most a few stations. For vehicles such as the X-20 the plasma exists for some time requiring several stations. By choosing a frequency such as SHF close to the plasma frequency it has been possible to get sufficient airborne transmitter power (50 watts) to permit horizon to horizon coverage using reasonable antenna gains on the ground. Higher frequencies would require higher powers, which are not available, and thus a greater number of stations at increased cost.

For the near ballistic vehicles the solution is to use some standard, lower frequency system, such as UHF and either ignore the blackout (as in Mercury), employ a frequency higher than fp at the next atmospheric window $\simeq$ 35 Kmc/s or employ an exotic technique to punch a hole in the plasma as discussed in a later section.

The antenna voltage breakdown or power handling capability of an antenna in the presence of a plasma has been determined from thermally and radio frequency generated plasmas with results as shown in Figures 14 and 15 . Note that the currently available airborne power levels at SHF are less than the breakdown levels. It is only when one goes to UHF that the airborne transmitter power must be limited to a few watts. Although blackout will normally occur before antenna voltage breakdown at SHF, this is not expected at UHF and the UHF power limitations can be serious.

Coupling between antennas can usually be prevented in the no plasma case by spacing the antennas far enough apart. Antenna coupling in the presence of a plasma is less than for free space for the useable frequencies above fp as shown in Figure 16 . Plasma noise may be a problem in some cases where extremely sensitive receivers are employed but is not expected to be a limitation on currently proposed UHF and SHF systems.

Signal intermodulation can occur when a desired signal is transmitted thru a path illuminated by a high power (such as pulsed) local transmitting antenna. If amplitude modulation is utilized this may at times present a problem. If frequency modulation is used as in most telemetry links the amplitude intermodulation which occurs has been shown to produce negligible effect in the telemetered signal.

To put the several parameters discussed above into proper perspective a system analysis has been performed to determine the relation between the number of stations required, vehicle L/D, operating frequency, available power and signal levels achievable relative to system threshold.

Figure 17 illustrates the number of stations required as a function of vehicle L/D assuming coverage within 2° of the horizon. Figure 18 summarizes the study showing signal margin in db above system threshold as a function of range to go for several L/D vehicles employing UHF and SHF frequencies. It can be seen from this figure that SHF will be adequate for L/D $\cong$ one but a higher frequency and thus more ground stations per mile of coverage may be required for the L/D $\cong$ 0.5 vehicles. The next atmospheric window is at $\cong$ 35 Kmc/s. Because of the higher speeds, shorter effective ranges and narrower antenna beams required to get adequate signal strengths acquisition and tracking problems are accentuated with 35 Kmc/s systems.

It is apparent that UHF should be employed because of its freedom from acquisition and tracking difficulties and reduced cost wherever blackout will not preclude its use or where blackout may be tolerated.

IV WHAT NEEDS TO BE DONE AND WAYS TO DO IT

IV.A. Introduction

Figure 19 illustrates the increasing complexity and longer duration of manned space missions. The mission subsystems employed on peaceful missions such as rescue and the military missions will further increase subsystem complexity.

These future requirements as a function of some typical missions are shown in Figure 20 .

From examination of these figures the future trends in manned spacecraft subsystems can be summarized as follows:

DO AN INCREASINGLY BIGGER JOB FOR LONGER TIMES AT SAME OR BETTER RELIABILITY FOR LESS POWER AND AT LESS WEIGHT.

There are a number of techniques which may be employed to achieve these requirements. Some of the more universal techniques are illustrated in Figure 21. Note for example that greater dependence on the crew and employment of simple manual backup systems are two effective techniques in that they permit some improvement in most of the objectives.

The matrix proposed is by no means all inclusive but is offered as an approach worthy of consideration.

In an actual subsystem trade study, quantitative values must be used to provide meaningful trends.

IV.B. Example of a Simple Backup Guidance Subsystem

A simple backup guidance system has been devised which because of its simplicity is an order of magnitude more reliable than conventional inertial systems. The system is capable of providing re-entry control to a pilot selected landing site after a number of orbits.

This particular system is suitable for re-entry vehicles with maximum lift to drag ratios in the order of 0.5 or larger.

Figure 22 shows the equipment required and the guidance law for angle of attack (α_c) which it generates.

A single stored nominal acceleration program (A_{NP}) corresponding to a nominal flight trajectory is programmed versus time, see Figure 23 . The vehicle normal acceleration (A_N) is measured with a body mounted accelerometer with its sensitive axis mounted perpendicular to the wing. The measured normal acceleration is subtracted from the programmed acceleration and integrated to generate the commanded angle of attack (α_c) as shown by the guidance equation. The pilot flys the vehicle based on this commanded angle of attack. For brevity, operation of the system only after it has established equilibrium glide will be explained. The detailed development, theory of operation, and six degree of freedom simulator evaluation of the system is contained in Reference 1 .

$$\ddot{h} = -go + \frac{V^2}{r_\circ} + A_L \simeq 0 \quad \text{-------(1)}$$

$$\therefore A_L = go - \frac{V^2}{r_\circ} \quad \text{--------------------(2)}$$

Where:

h = Altitude
go = Gravitational constant
$r_\circ$ = Radius from center of earth
A_L = Lift acceleration

The lift acceleration is the primary reason the accelerometer system works which also explains why the system is useful only when vehicle max L/D is in the order of 0.5 or more.

Since the lift acceleration (A_L) is uniquely related to the velocity, velocity can be controlled by controlling A_N (and thus A_L) .

This can be seen qualitatively in Figure 24 . Consider the case where the velocity of the vehicle is excessive for the desired trajectory and corresponding landing site. If the velocity is higher than the nominal then by virtue of equation (2) A_L is less than the programmed lift (A_{LP}) and hence A_N is less than A_{NP} . This difference in A_N will cause the angle of attack to increase until $A_N = A_{NP}$. Increased angle of attack increases the drag which causes the vehicle to slow down until A_L equals A_{LP} at which time $\alpha = \alpha_N$ and A_N also equals A_{NP} .

Total performance of the system for booster cut off overspeed and underspeed conditions for a typical one orbit flight are shown in Figure 25. The generated commands are engaged at a time corresponding to nominal re-entry time thus it is possible to employ the system for multiorbit use. For several orbit use clock time since boost has been found to be a sufficient criteria to start the programmer.

Cross range is controlled by banking to a fixed angle.

Performance of this system when nominal L/D is in the order of one is shown in Figure 26 .

The reliability of this 30 pound system consisting of two attitude gyros, one airframe mounted accelerometer, an acceleration programmer and an integrator is in the order of a magnitude better than that of a complete inertial guidance system with a digital computer.

Performance of the system as a function of L/D is shown in Figure 27 . As explained above the system depends on measurement of lift acceleration which explains the reduced performance for low L/D vehicles.

Multi orbit operation is achieved by the pilot re-aligning the attitude reference and engaging the programmer based on time from cut off with results as shown in Figure 28 .

If tracking data from the ground is employed to establish de-orbit time and program start, performance becomes independent of the number of orbits, as shown in Figure 27 .

IV.C. Manual Backup Lunar Landing

An example of increased dependence on man and employment of simple backup equipment to do manual landing follows:

A manual backup of the primary automatic lunar guidance is practical with a minimum amount of equipment and greater dependence on man particularly in the lunar de-orbit, braking, hover, and landing phases. A sufficient set of equipment consists of three body-mounted rate gyros as part of the rate stabilized control system, three body-mounted integrating gyros as a medium-term attitude reference, a low-magnification telescope body-mounted to permit horizon scanning, determination of star azimuth and landing area study before descent from low orbit.

With the above equipment, simple charts and nomographs and a clock to drive function programs corresponding to nominal descent pitch rate and thrust acceleration the vehicle can be controlled down to initiation of the braking maneuver.

The braking maneuver, hover, and landing can be accomplished by the man controlling attitude and thrust employing only visual cues.

Figure 29 illustrates a simulator built to evaluate the manual braking, hover, and landing phases by man using only visual cues. A TV pickup tube is gimballed and controlled by the pilot's attitude control to represent vehicle attitude. Vertical descent is controlled by an analog computer to represent the descent trajectory established by manual lunar descent guidance and is modified by the thrust and attitude actions of the pilot. This is represented by driving the TV pickup down toward the simulated lunar surface which in turn is driven horizontally to represent vehicle horizontal velocity over the surface of the moon. Figure 30 shows the display provided to the pilot. The technique employed to generate these displays is shown in Figure 31 . A horizon line is established by one projector and a star background by another. Both are coordinated with the pilot's attitude control so that realism in attitude is achieved.

To evaluate a particular landing guidance concept the total fuel used, landing impact velocity, and landing location are recorded for each flight. Total manual lunar de-orbit and landing fuel expenditures are in the order of 1.07 times that required for a crew controlled primary system employing inertial guidance.

IV.C. Space and Re-entry Communication at UHF

UHF is an ideal frequency for space communications because it is currently universally employed, line of sight ranges can be achieved with non-directional or at worst low gain antennas and therefore system costs are nominal.

Advanced techniques show great promise of permitting UHF use during re-entry. For near ballistic shapes techniques for local cooling of the plasma surrounding an antenna by means of evaporative techniques appear feasible. Advantage can also be taken of the fact that while the plasma attenuation per wavelength is large the plasma thickness for vehicles such as this is small in terms of a wavelength at UHF.

For the higher L/D vehicles in the 0.5 to 2 range although the plasma intensities never reach the values experienced by the near ballistic vehicles the air flow is complicated by the much larger range of angles of attack and the plasma layer is apt to be thicker. For these vehicles a survey of locations where electron densities are lower and the flow can be further cooled by gas ejection into the flow shows promise. Further work of this type is recommended.

Considerable effort employing these techniques is currently being sponsored by NASA.

V SUMMARY AND CONCLUSIONS

The growth of requirements placed on manned spacecraft subsystems with time resulting from demands for doing more for longer duration missions has been examined. Although the corresponding weight, volume, and power consumption penalties associated with these increased requirements could possibly be accepted, the increased mission requirements place an even higher cost on weight, volume, and power consumption. For these reasons the natural trends of increased equipment complexity, operating time and the corresponding growth in weight, volume and energy consumption which would result in lower mission reliability need to be reversed.

Some of the techniques described in this paper which are capable of effecting a reversal in these trends are maximum utilization of the crew and improved mission reliability through the best combinations of:

- Redundancy
- In flight maintenance
- Simple backup subsystems
- Turning equipment off when possible
- Dependence on ground based systems

Because of the many conflicting interests (for example the requirement to do more at less weight and power yet self-contained) the concepts of greater dependence on the crew, utilizing simple backup systems and equipment turned off when possible to save energy appear to be the most universally applicable techniques.

The purpose of this paper has been to give the Space Systems engineer an overview of the trends in manned spacecraft subsystem requirements and to suggest some of the approaches which need to be evaluated in designing optimum subsystem combinations for the particular missions contemplated.

REFERENCES

1. - X-20 Guidance Backup Equipment Study, Dr. A. A. Frederickson, The Boeing Company, September 13, 1962, Classified Confidential

PROGRAM	TYPE PRIMARY	TYPE BACKUP	USE OF MAN	CAPABILITY SELF-CONTAINED	PERFORMANCE VS MISSION TIME	WT
MERCURY	GROUND BASED	PILOT FIRE RETROROCKET	LITTLE • BACKUP	LITTLE	INDEPENDENT	30 LBS
X-20	INERTIAL	PILOTED EMERGENCY RE-ENTRY SYSTEM	EXTENSIVE • LANDING SITE SELECTION • LANDING • ENERGY MANAGEMENT • ERS OPERATION	• PRIMARY MODE	5000 x 3000 MI. FOOTPRINT • LAND AT PILOT SELECTED SITE THREE ORBITS • REDUCED CAPACITY FOR MORE ORBITS	260 LBS
GEMINI	RADIO BOOST + GROUND UPDATED INERTIAL RE-ENTRY	REDUNDANT MODES WITH PRIMARY EQUIPMENT	EXTENSIVE • MODE CONTROL • SPACE RE-START	• BACKUP MODE FOOTPRINT REDUCED GREATLY	UNLIMITED DURATION WITH GROUND BASED SUPPORT 450 x 150 MI. FOOTPRINT	220 LBS
APOLLO (COMMAND MODULE)	INTERTIAL & GROUND RADAR	SECOND INERTIAL MANUAL STADIA-TRIANGULATION DSIF	EXTENSIVE • MODE SELECTION	• PRIMARY MODE	• SYSTEM DESIGNED TO COMPLETE MISSION WITH MULTIPLE FAIR MULTIPLE FAILURES • ABORT CAPABILITY	270 LBS

Figure 1.- Current guidance subsystems.

PROGRAM FUNCTIONS MONITORED OR CONTROLLED BY PILOT

	BOOST	INJECTION	DEORBIT	RE-ENTRY	FLT ATTITUDE
MERCURY	MONITOR	MONITOR	MONITOR (EMERGENCY CONTROL)	BALLISTIC	CONTROL
X-20	MONITOR (EMERGENCY CONTROL)	CONTROL	CONTROL	CONTROL	CONTROL
GEMINI	MONITOR		CONTROL	CONTROL	CONTROL
APOLLO	MONITOR	EARTH / MOON CONTROL	MOON / EARTH CONTROL	CONTROL (PRIME & BACKUP)	CONTROL

Figure 2.- Pilot display and control.

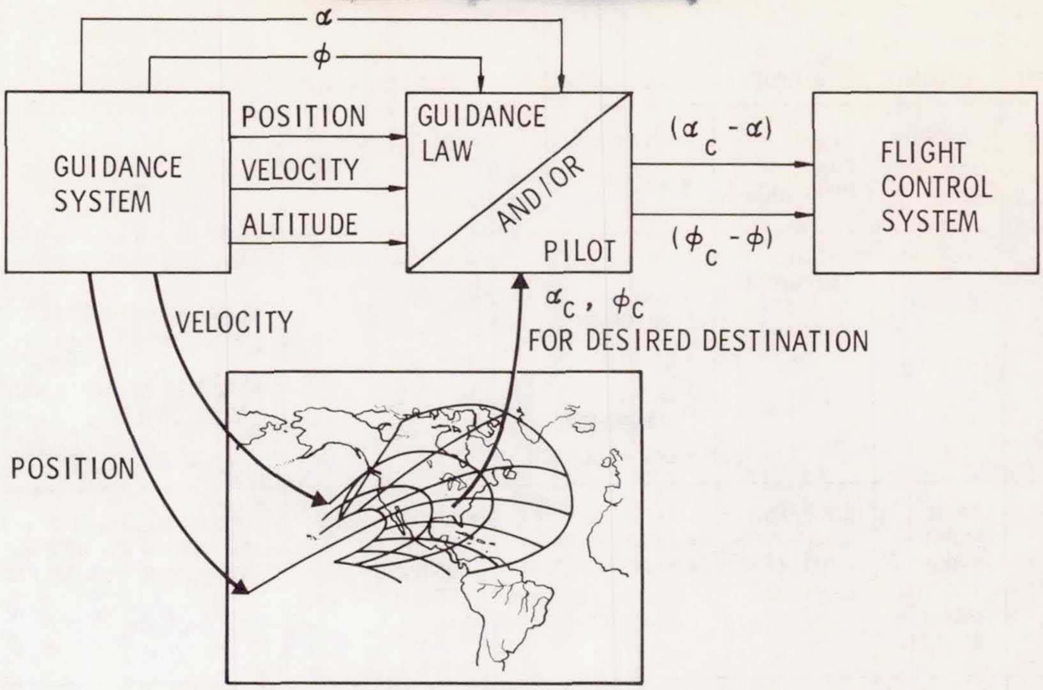


Figure 3.- Conceptual energy management system.

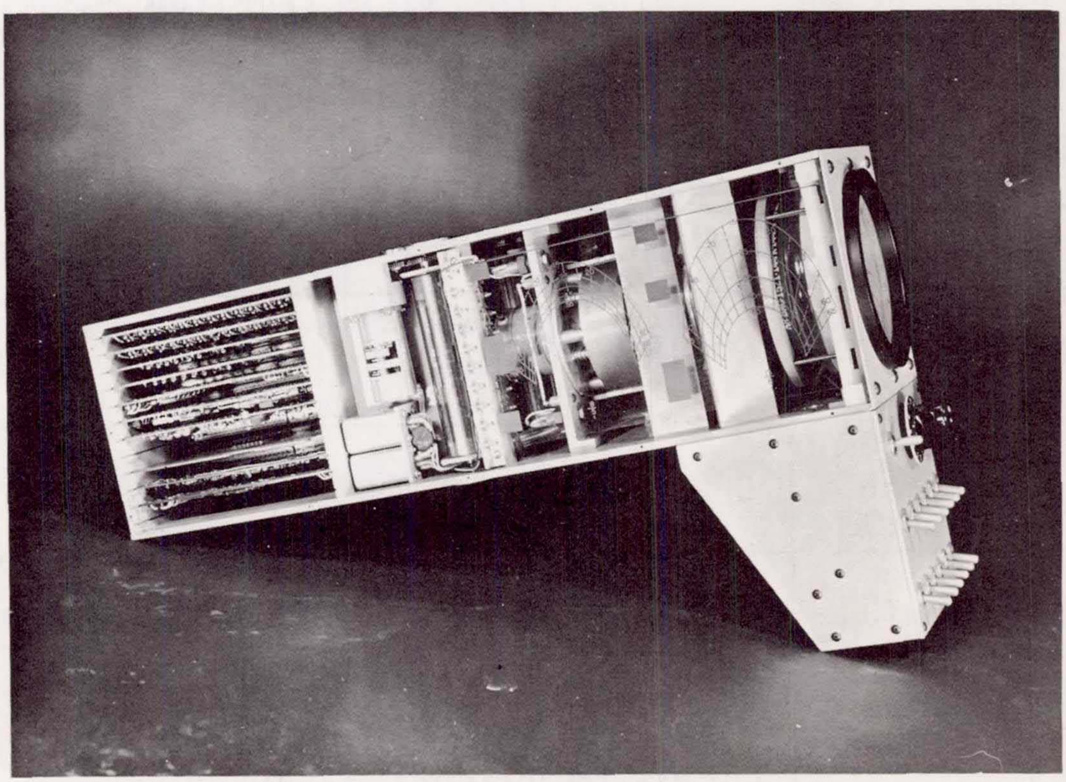


Figure 4.- Energy management display.

PROGRAM	FUNCTION	FREQUENCY	T/M BANDWIDTH	RE-ENTRY BLACKOUT	
MERCURY	COMMAND	406-450 MC			
	TELEMETRY	228,260 MC	70 KC		
	UHF VOICE	299 MC		300 TO 50K FT ALTITUDE	4 MIN 20 SEC
	HF VOICE	15 MC			
X-20	SHF COMMAND/ VOICE	10.4 GC			
	SHF VOICE/ TELEMETRY	13.5 GC	300 KC	DEPENDS ON RE-ENTRY ANGLE	5 SEC
	UHF VOICE	395 MC		ENTIRE RE-ENTRY	20 MIN
GEMINI	COMMAND	450 MC			
TENTATIVE DATA	TELEMETRY	225.7, 259.7 MC	70 KC	350 TO 190K FT	10 MIN
	UHF, HF VOICE	296.8, 15. MC			
APOLLO (COMMAND MODULE)	COMMAND	72, 450, 982 MC			
TENTATIVE DATA	TELEMETRY	225-260 MC 2.2-2.4 KMC	70 KC	350 TO 190K FT	20 MIN
	UHF, VHF, HF VOICE	299, 108, 15 MC			

Figure 5.- Earth orbit and re-entry.

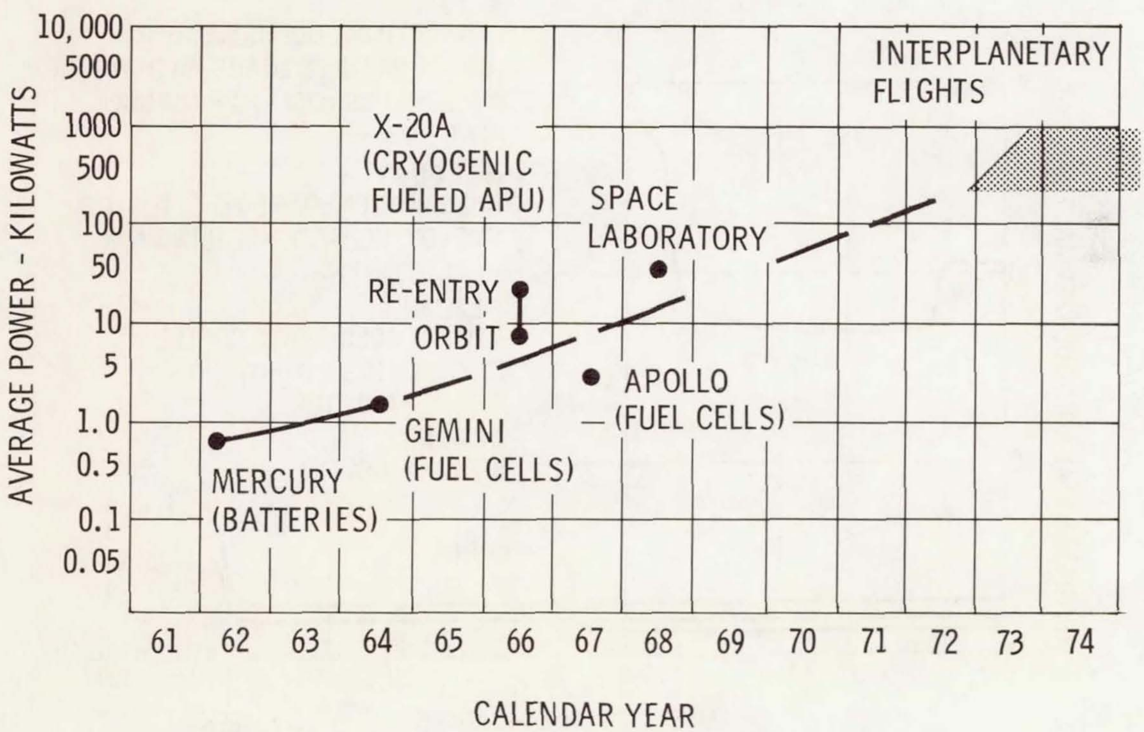


Figure 6.- Power requirements.

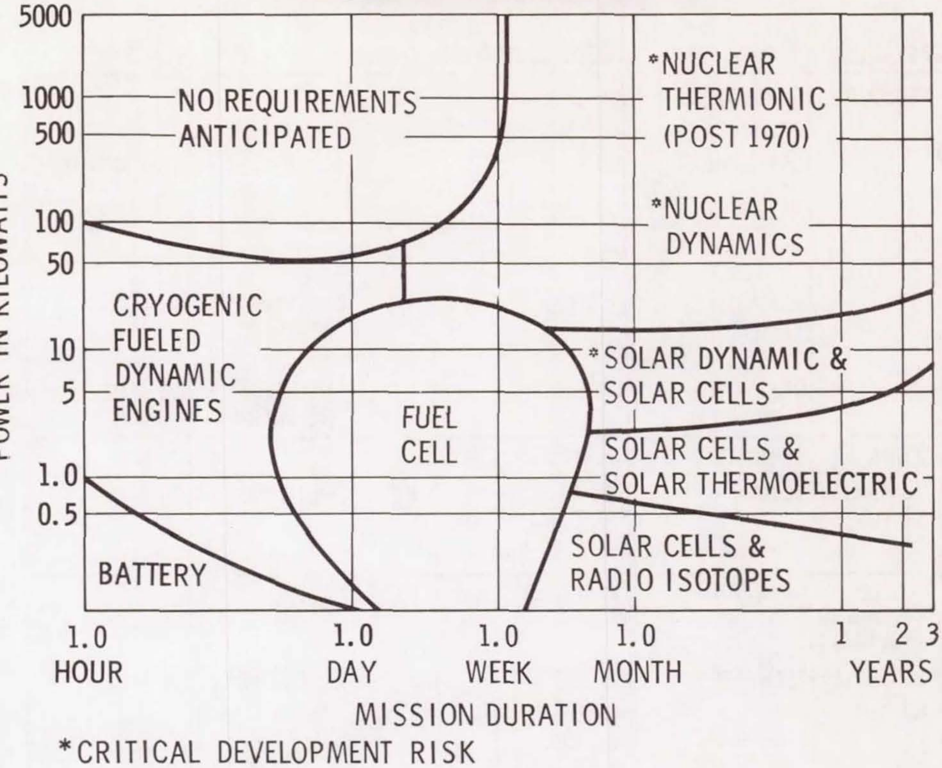


Figure 7.- Estimated optimum power systems, 1966.

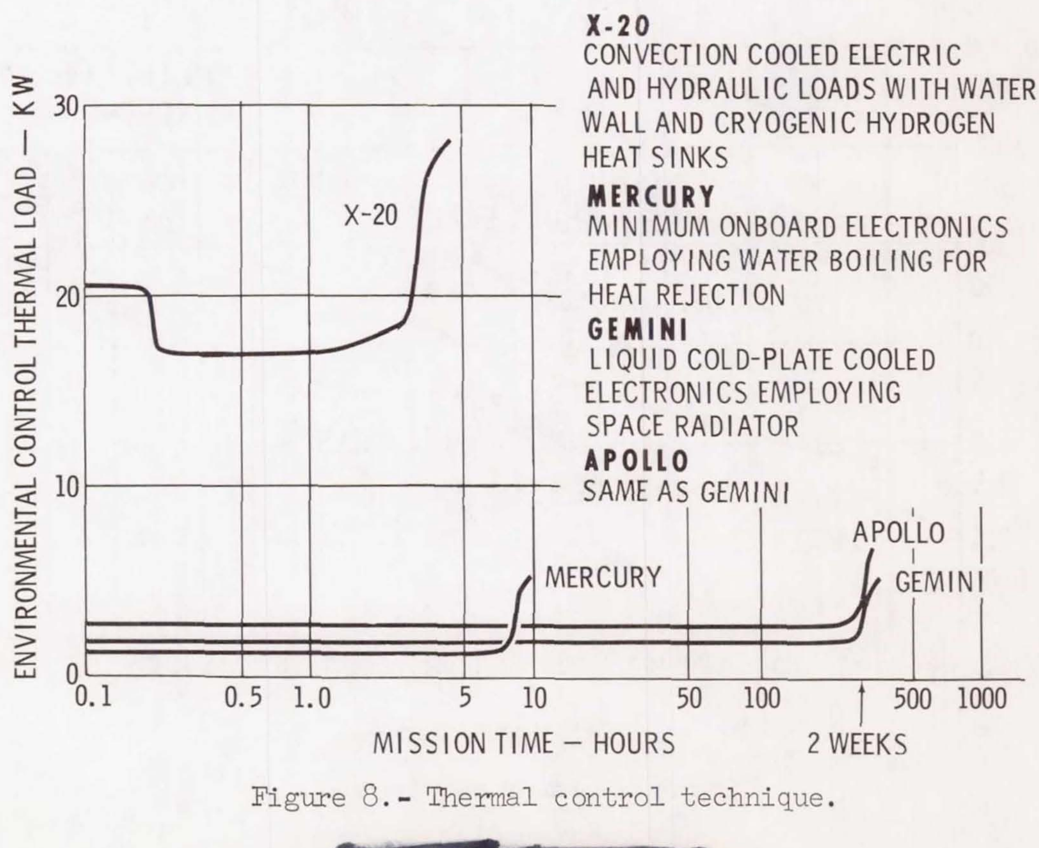


Figure 8.- Thermal control technique.

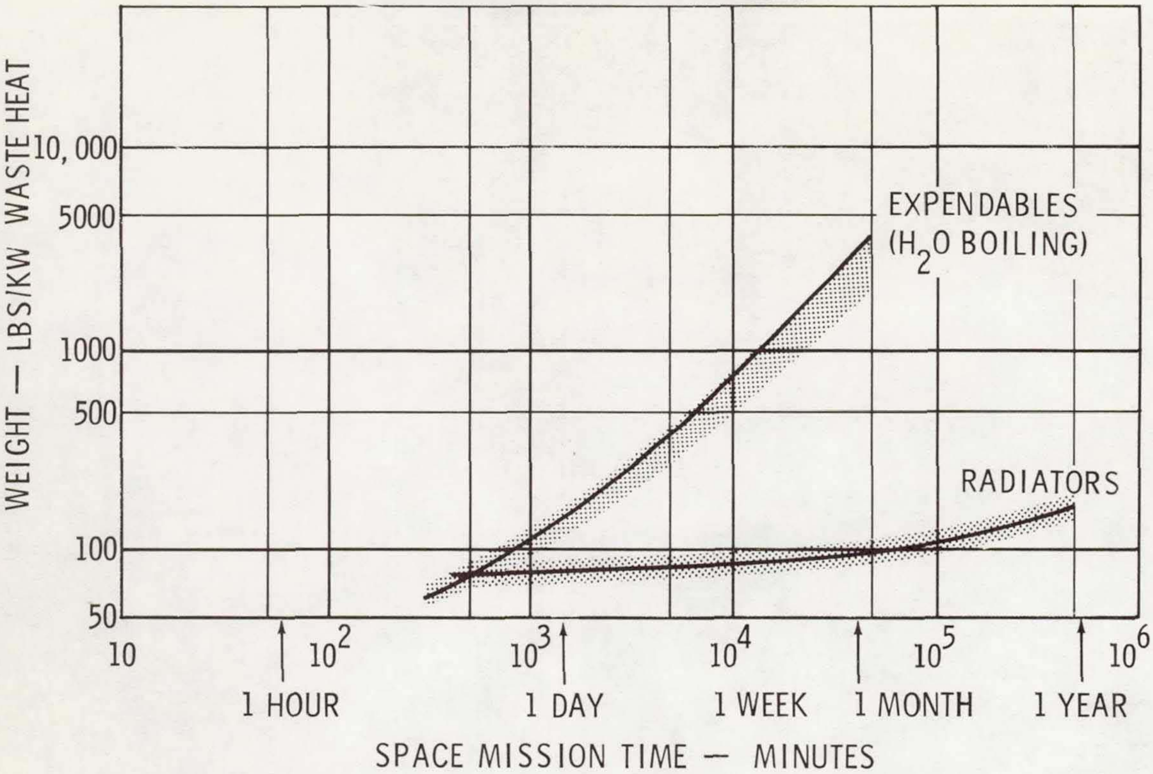


Figure 9.- Methods comparison for manned spacecraft thermal control.

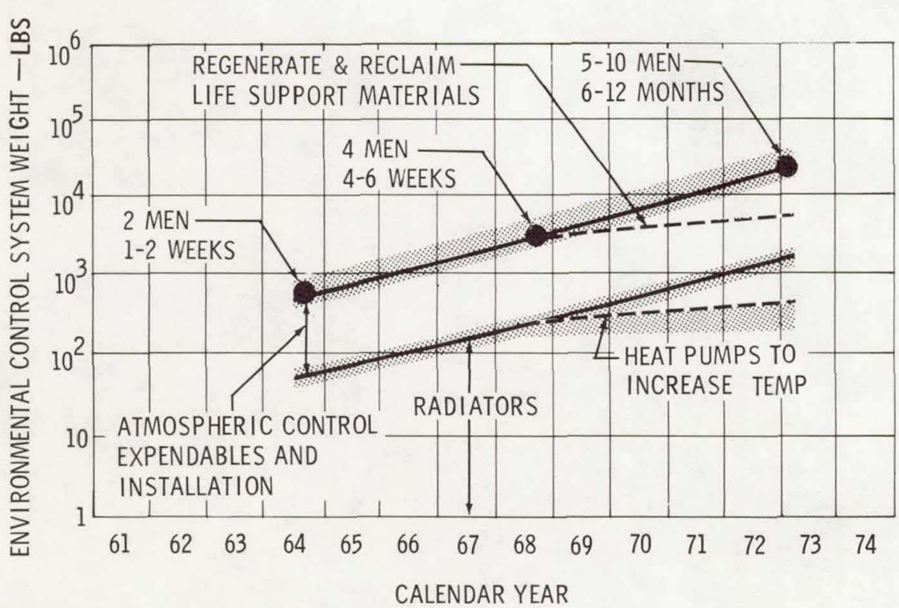


Figure 10.- Environmental control systems weight.

Figure 11. - White hot shock layer surrounding an X-20 model undergoing test.

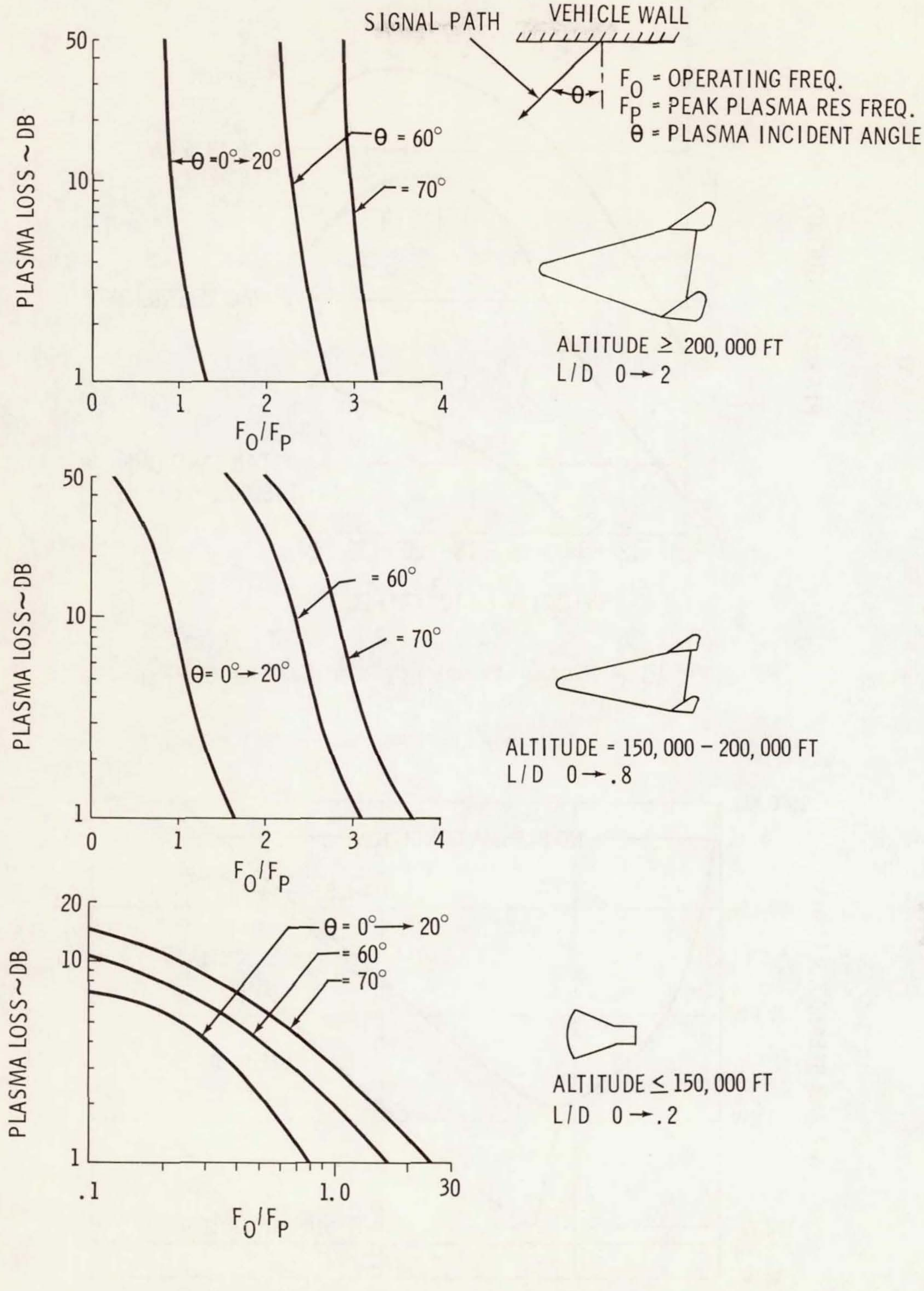


Figure 12.- Plasma attenuation versus plasma incident angle.

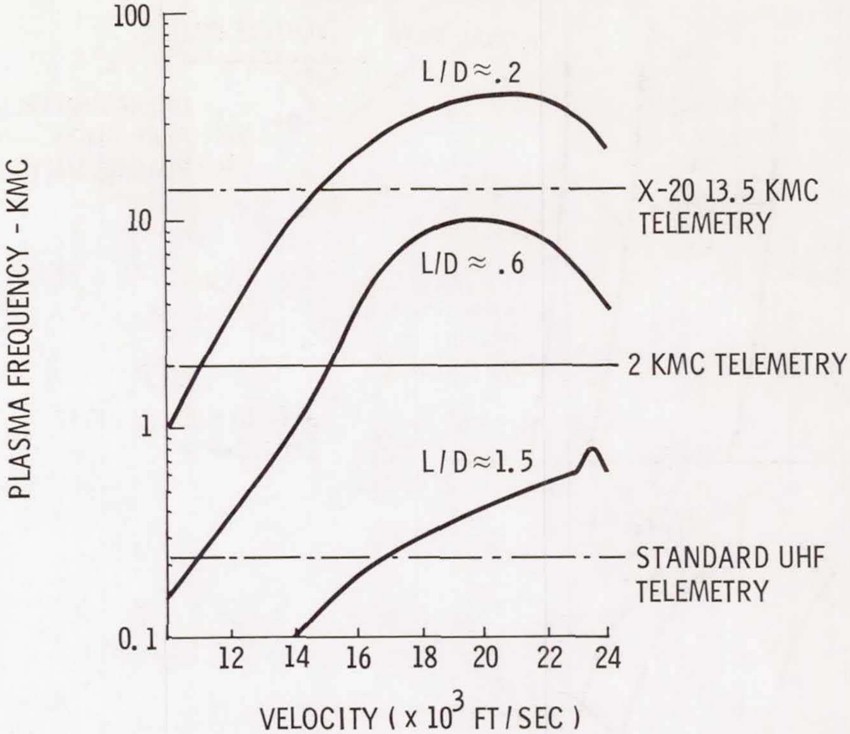


Figure 13.- Plasma frequency for L/D ranges.

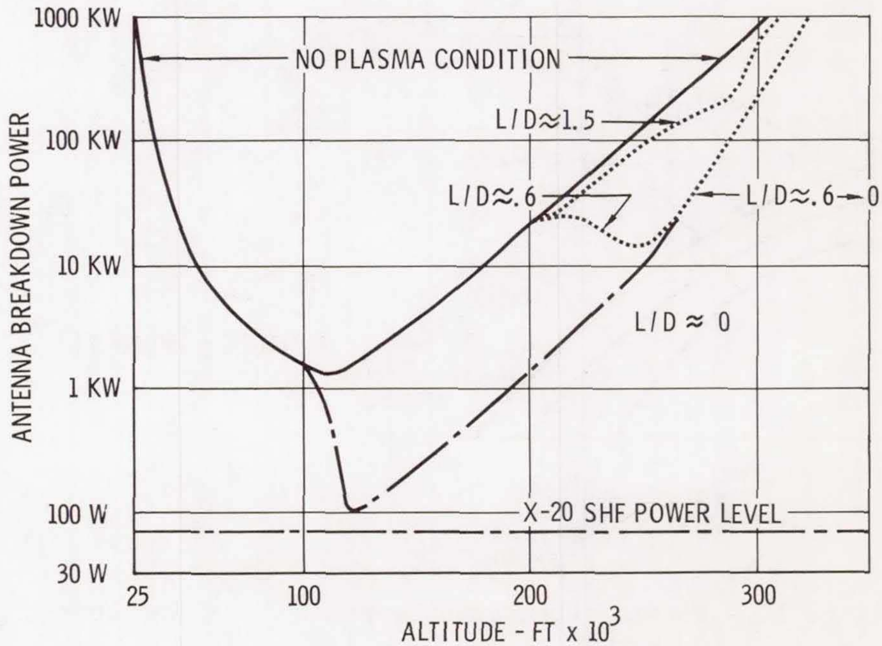


Figure 14.- X-band telemetry antenna power-handling capability.

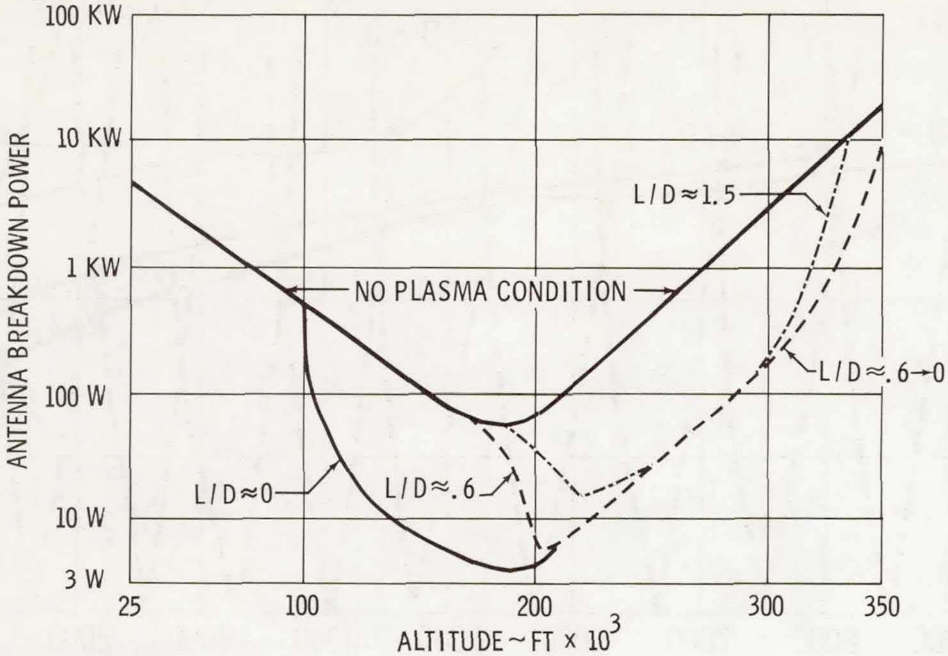


Figure 15. - UHF antenna power-handling capability.

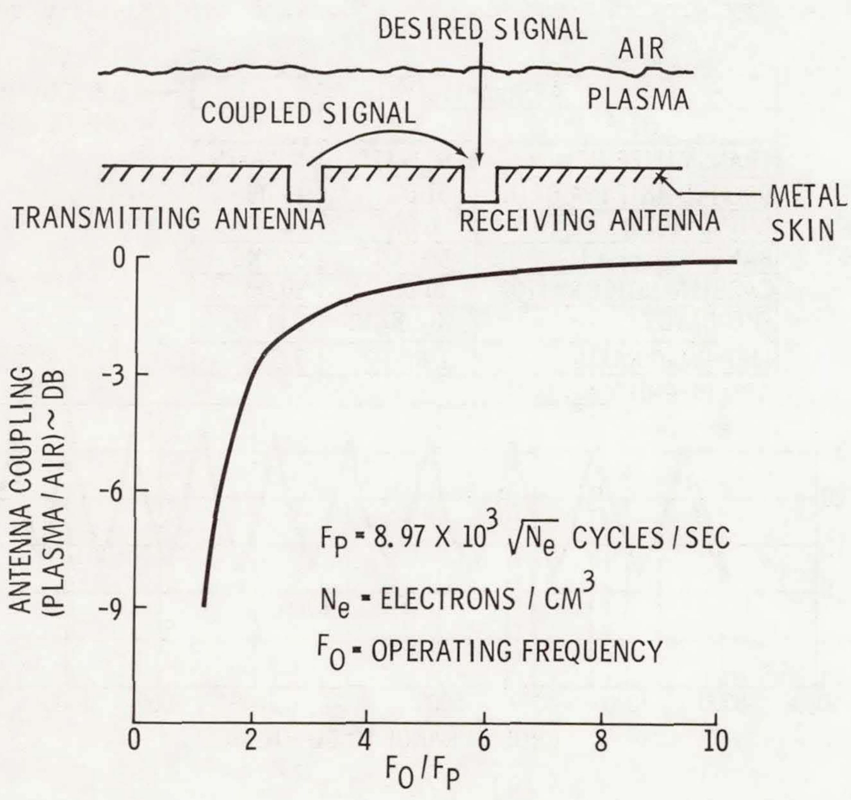


Figure 16. - Antenna coupling in plasma.

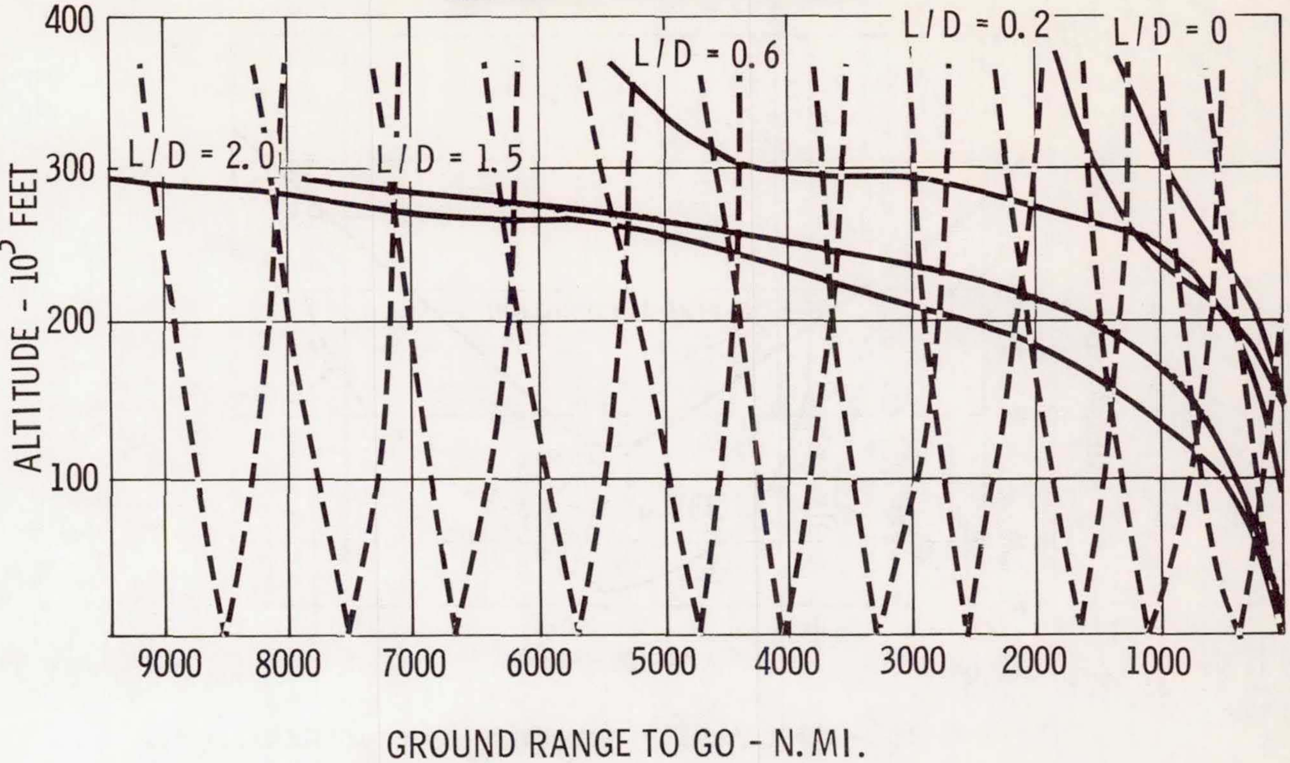
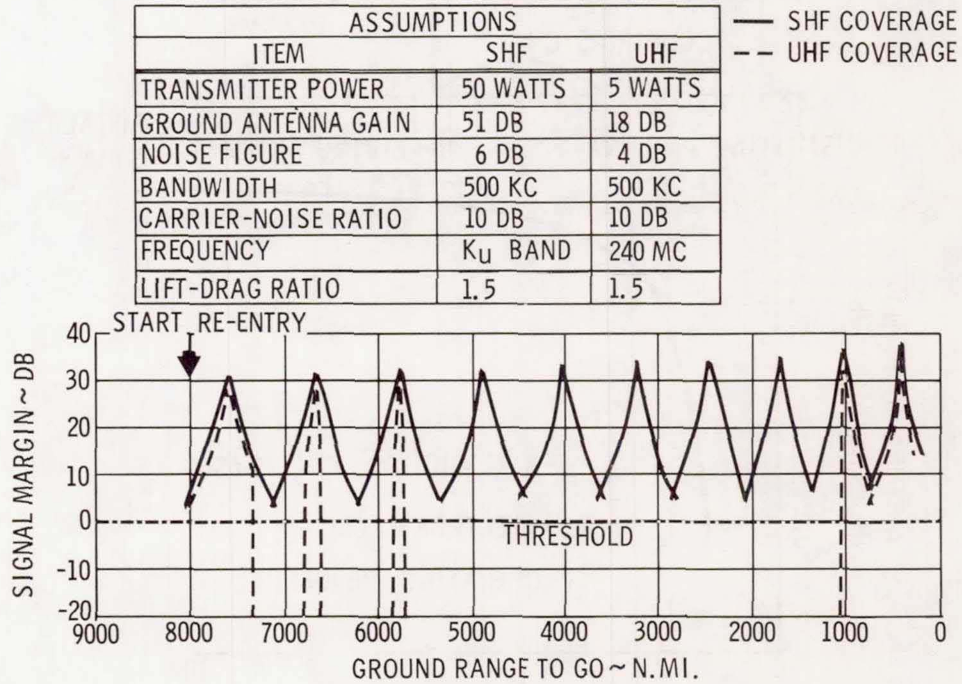


Figure 17.- Re-entry ground stations for L/D, with continuous coverage assumed.

ASSUMPTIONS		
ITEM	SHF	UHF
TRANSMITTER POWER	50 WATTS	5 WATTS
GROUND ANTENNA GAIN	51 DB	18 DB
NOISE FIGURE	6 DB	4 DB
BANDWIDTH	500 KC	500 KC
CARRIER-NOISE RATIO	10 DB	10 DB
FREQUENCY	K_u BAND	240 MC
LIFT-DRAG RATIO	1.5	1.5

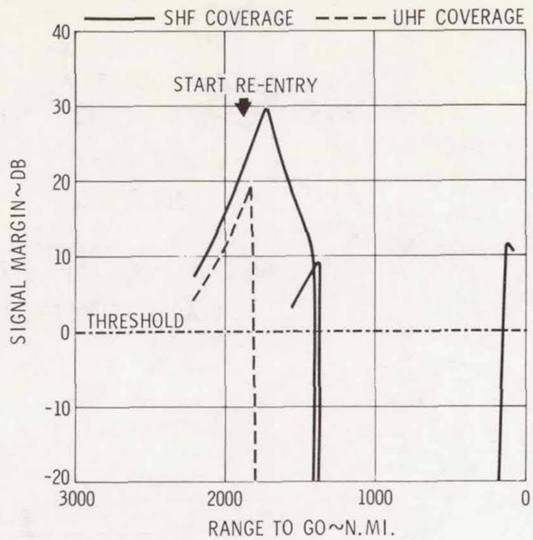


(a) L/D = 1.5

Figure 18.- Communications capability.

ASSUMPTIONS

ITEM	SHF	UHF
TRANSMITTER POWER	50 WATTS	5 WATTS
GROUND ANTENNA GAIN	51 DB	18 DB
NOISE FIGURE	6 DB	4 DB
BAND WIDTH	500 KC	500 KC
CARRIER/NOISE RATIO	10 DB	10 DB
FREQUENCY	Ku BAND	240 MC
LIFT/DRAG RATIO	0.2	0.2

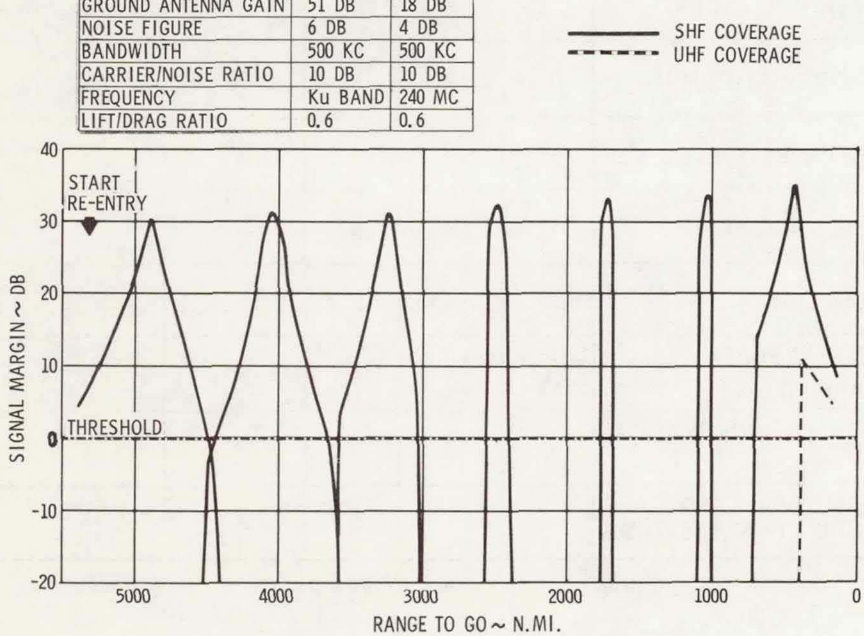


(b) L/D = 0.2

Figure 18.- Continued.

ASSUMPTIONS

ITEM	SHF	UHF
TRANSMITTER POWER	50 WATTS	5 WATTS
GROUND ANTENNA GAIN	51 DB	18 DB
NOISE FIGURE	6 DB	4 DB
BANDWIDTH	500 KC	500 KC
CARRIER/NOISE RATIO	10 DB	10 DB
FREQUENCY	Ku BAND	240 MC
LIFT/DRAG RATIO	0.6	0.6



(c) L/D = 0.6

Figure 18.- Concluded.

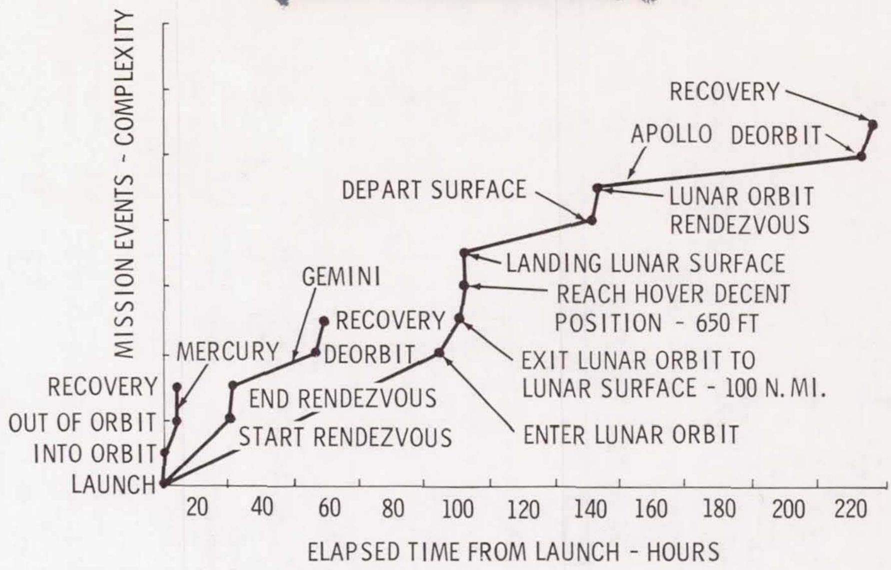


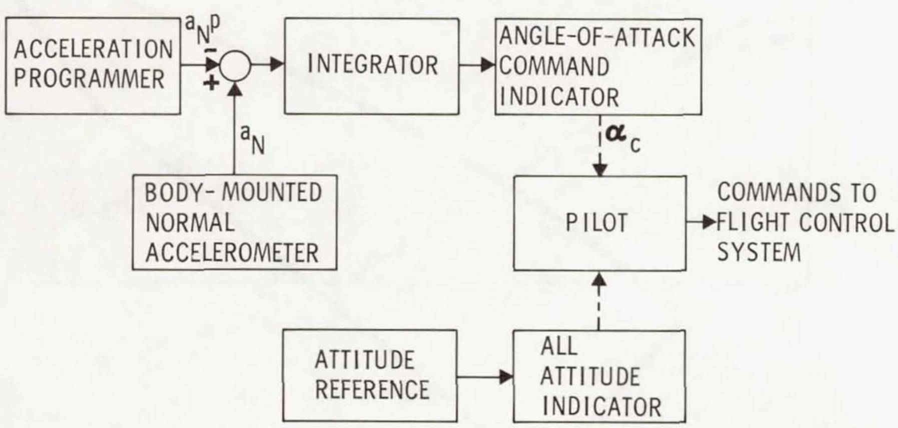
Figure 19.- Mission complexity and duration.

MISSIONS / REQUIRED FUNCTIONS	PEACEFUL				MILITARY OPERATIONS
	• FERRY	• RESCUE	• LUNAR	• PLANETARY	
CHOICE OF LANDING SITE		X			X
ALL WEATHER LANDING		X			X
SELF-CONTAINED GUIDANCE	X	X	X	X	X
LONGER DURATION	X	X	X	X	X
REDUCED WEIGHT	X	X	X	X	X
LARGER CREW		X	X	X	
INCREASED POWER CAPABILITY	X	X	X	X	X
INCREASED ENVIRONMENTAL CONTROL	X	X	X	X	X
RELIABLE SPACE AND RE-ENTRY COMMUNICATIONS	X	X	X	X	X

Figure 20.- Future subsystem requirements.

OBJECTIVES / ACHIEVE BY:	DO A BIGGER JOB	LONGER TIME AT HIGHER RELIABILITY	LESS POWER	LESS EQUIPMENT WEIGHT	BEST TECHNIQUES
INCREASED COMPLEXITY ON-BOARD	X				
DEPENDENCE ON GROUND-BASED SYSTEMS	X	X	X	X	X
GREATER DEPENDENCE ON CREW	X	X	X	X	X
BETTER COMPONENT RELIABILITY		X			
REDUNDANCY		X			
IN-FLIGHT MAINTENANCE		X			
OPERATION ONLY WHEN NEEDED		X	X		
MICROMINIATURIZATION		X		X	
SIMPLE MANUAL BACKUP SYSTEMS		X	X	X	X

Figure 21.- Techniques to accomplish objectives.



$$\alpha_c = \left[\alpha_n + K\int (a_{np} - a_n)\ at\right]^{\alpha = C_{L_{MAX}}}_{\alpha = L/D_{MAX}}$$

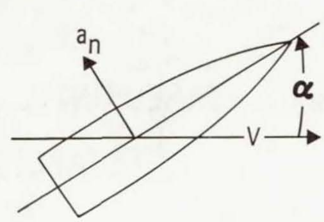


Figure 22.- Backup acceleration guidance subsystems.

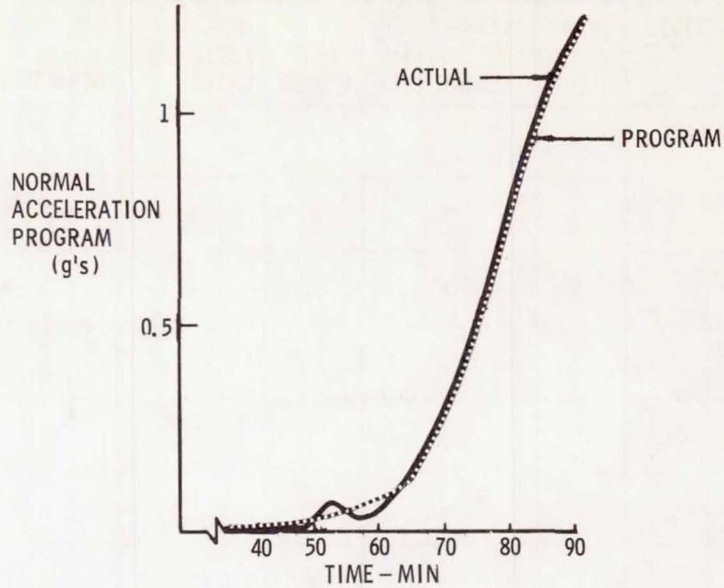


Figure 23.- Point slope acceleration program.

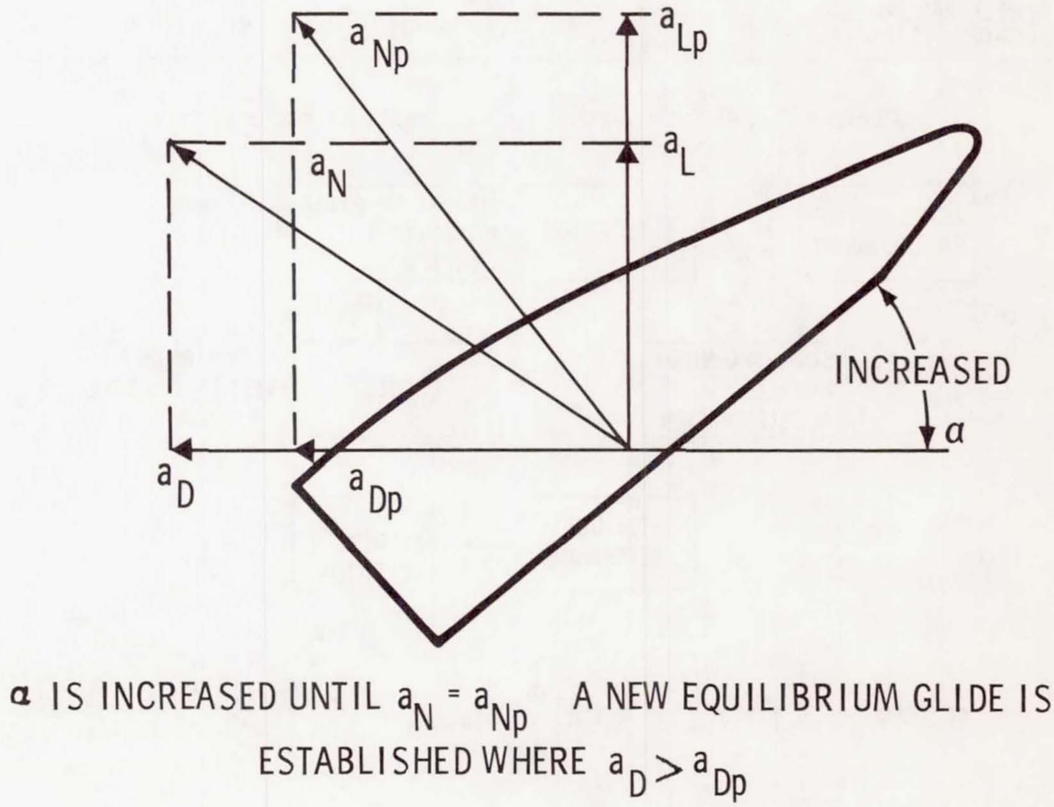


Figure 24.- Acceleration comparator subsystem operation with excess initial velocity.

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ACCELERATION COMPARATOR SUBSYSTEM OPERATION WITH INITIAL VELOCITY ERRORS

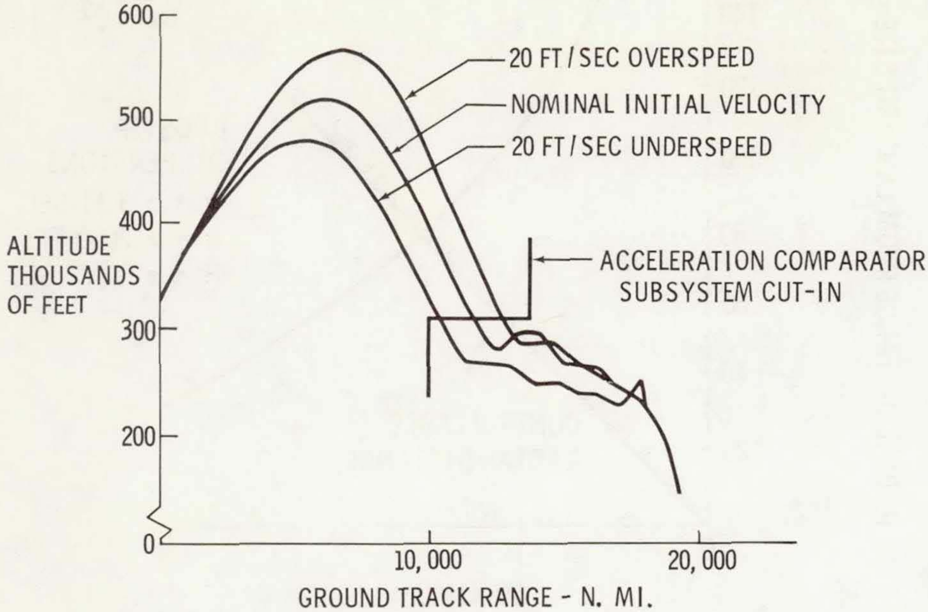

Figure 25.- Acceleration comparator subsystem guidance capability.

DOWN-RANGE SOURCE	(3σ) (N.MI.)	CROSS-RANGE SOURCE	(3σ) (N.MI.)
ACCELERATION SYSTEM IMPLEMENTATION	28	BOOST BURNOUT	21
COUPLING WITH CROSS-RANGE STEERING	21	BIAS DUE TO CONTROLLABILITY	48
BOOST BURNOUT	28	BIAS DUE TO DISPLAY AND READABILITY	24
DENSITY UNCERTAINTY	10	BIAS DUE TO ATTITUDE FIX	48
DRAG COEFFICIENT UNCERTAINTY	20		
OVERALL DOWN-RANGE	50	OVERALL CROSS-RANGE	75

Figure 26.- Acceleration comparator subsystem guidance capability.

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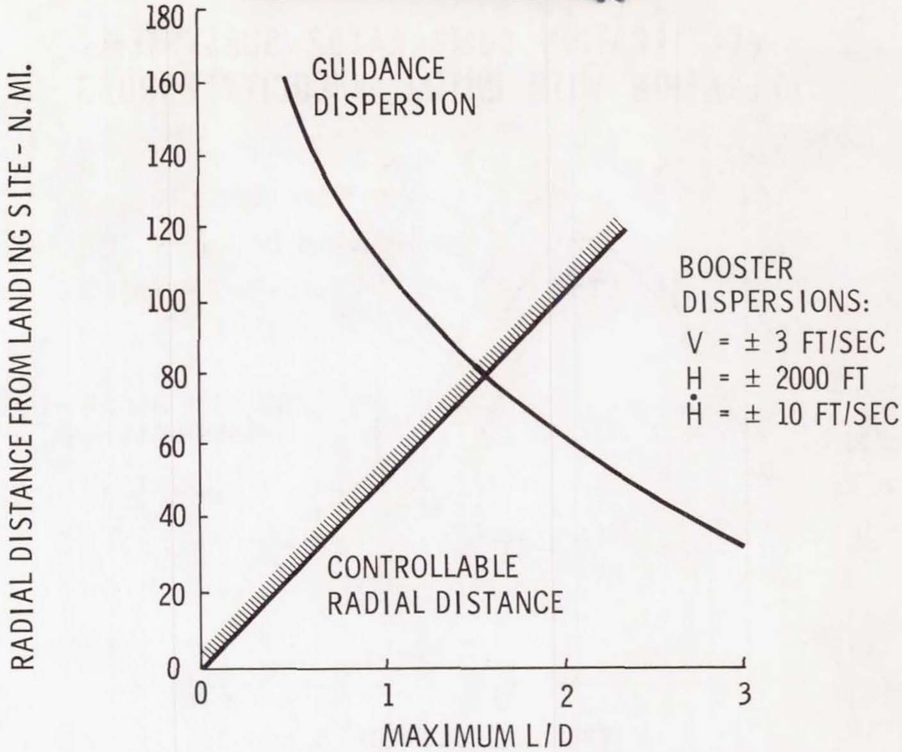

Figure 27.- 3σ performance versus vehicle lift capability.

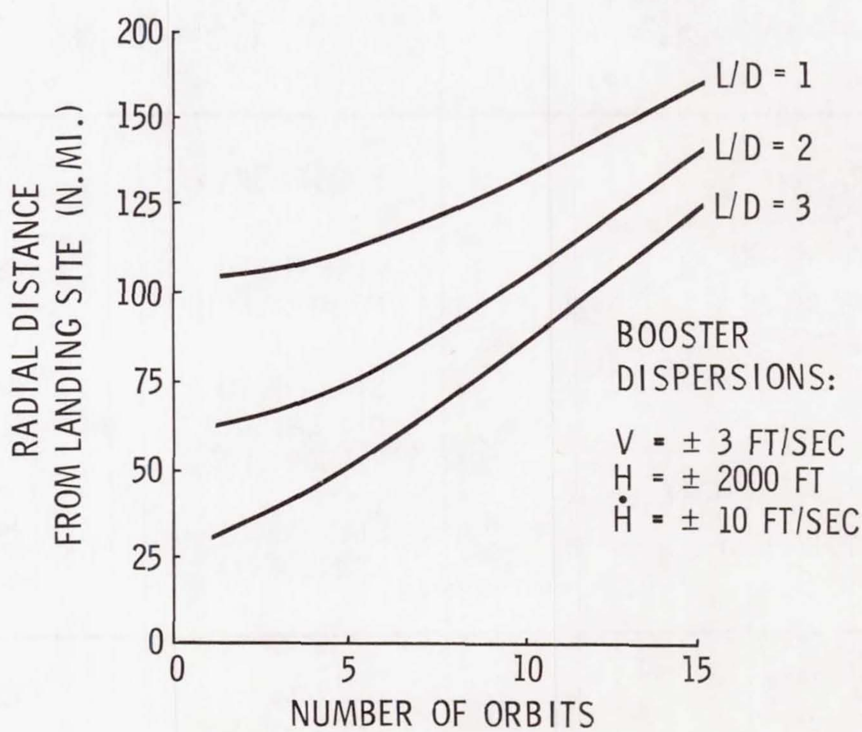

Figure 28.- 3σ dispersions at landing approach.

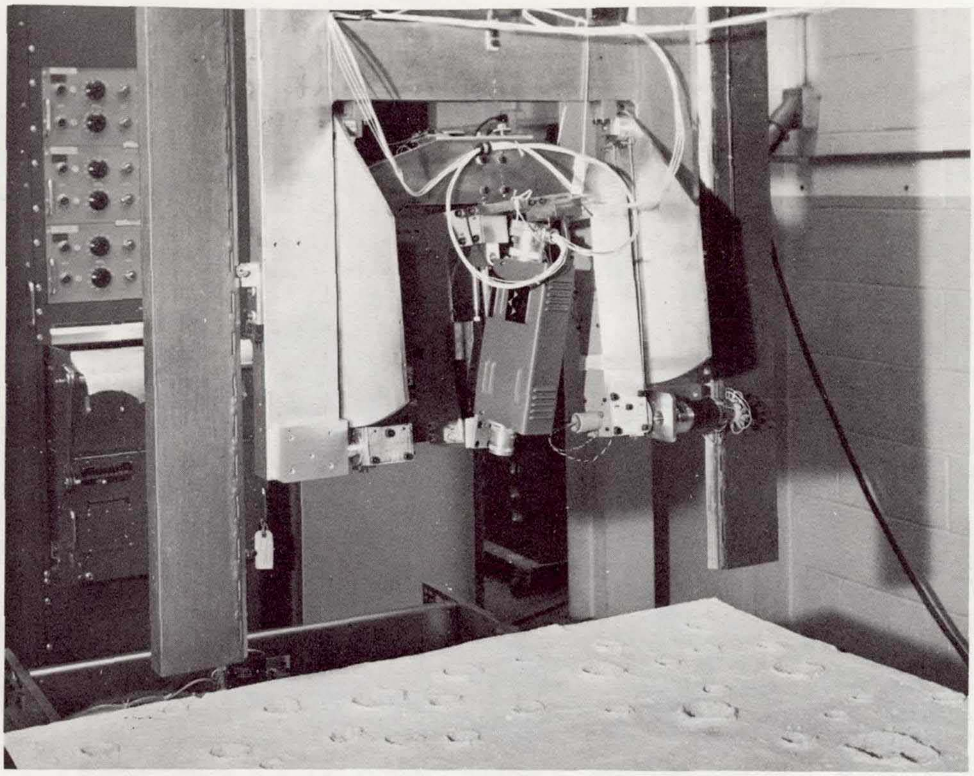

Figure 29.- Lunar-visual landing simulator.

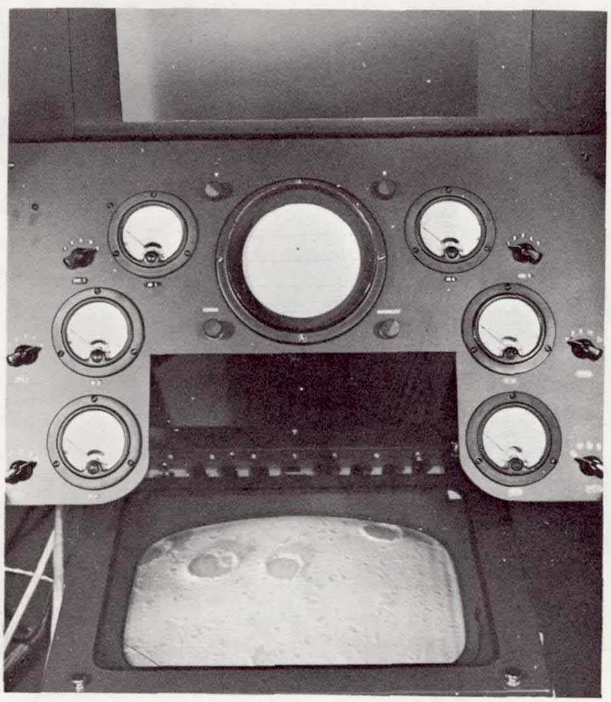

Figure 30.- Lunar-visual landing simulator.

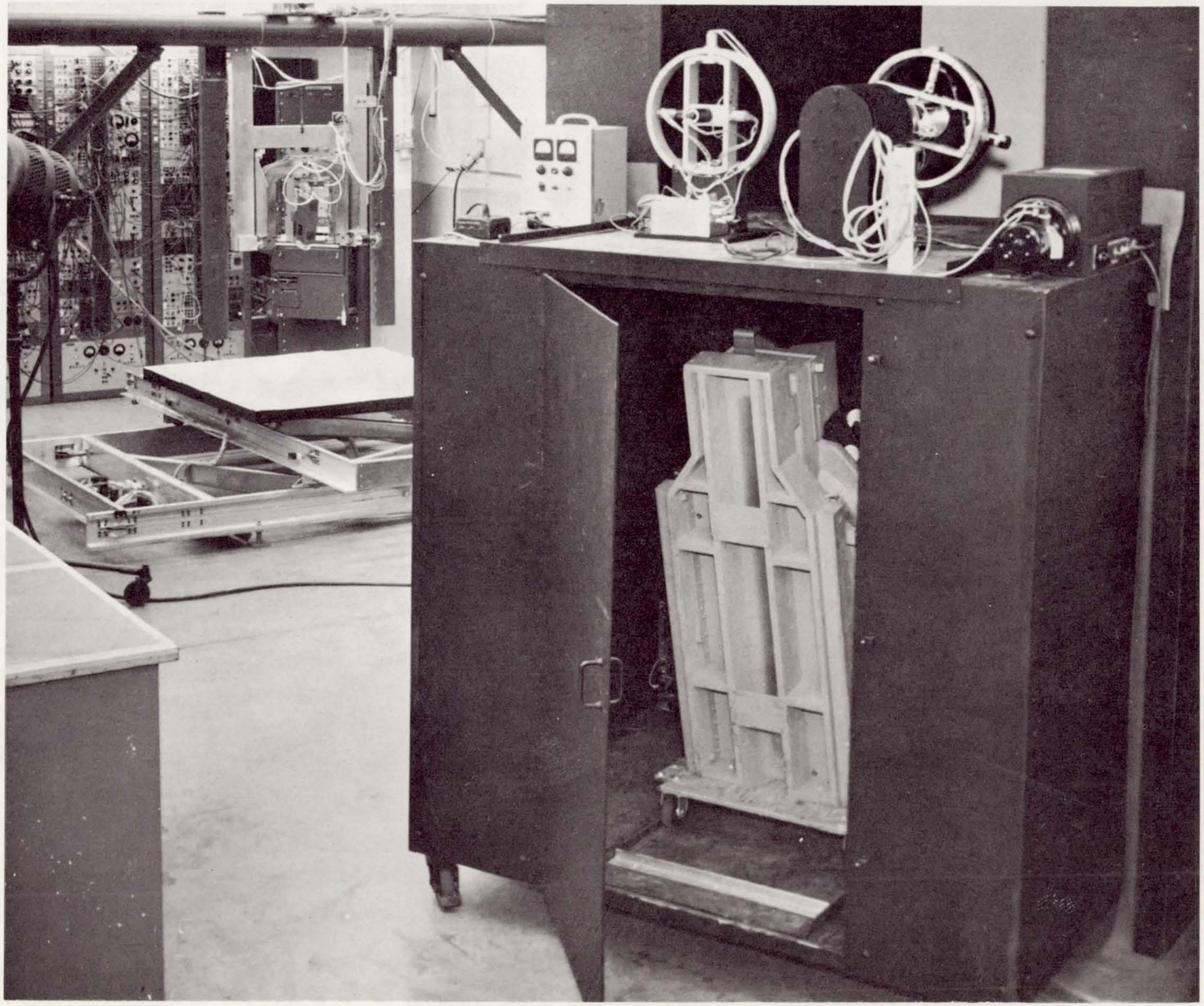

Figure 31. - Lunar-visual landing simulator.