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**PROCEEDINGS OF THE SECOND
MANNED SPACE FLIGHT MEETING**

CLASSIFIED PORTION

April 22-24, 1963

Dallas, Texas

*Manned Spacecraft Center,
Houston, Texas*

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Attached for insertion in the subject Technical Memorandum is a corrected title page. The title page has been corrected to include the espionage notice, the group category, and the classification.

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FOREWORD

The papers presented in this report represent the classified portion of the Second Manned Space Flight Meeting which was held in Dallas, Texas, on April 22-24, 1963. The meeting was co-sponsored by the American Institute of Aeronautics and Astronautics and the National Aeronautics and Space Administration. The following subjects are discussed in the report: Manned Space Flight Programs, Launch Vehicles, Spacecraft Design, and Guidance and Control.

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GEMINI DESIGN FEATURES

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The Gemini program, which was initiated by the NASA approximately 16 months ago, is being implemented by the McDonnell Aircraft Corporation as the next logical step in the nation's manned spacecraft program. The underlying concept of the Gemini design is to utilize the Project Mercury background to the fullest possible extent as a stepping stone to a practical operational spacecraft. The key word here is "operational." Project Mercury's basic objective was simply to put man in space and bring him safely back. Gemini, in contrast, aims at exploring and exploiting man's ability to function in space and to develop truly operational systems and techniques applicable to a variety of missions. Retention of the basic Mercury aerodynamic configuration and re-entry heat protection concepts has permitted development to proceed with a minimum of costly and time-consuming flight demonstration testing. This has allowed emphasis to be placed upon development of the various spacecraft systems. It is in the latter area that the real advances of Gemini over Mercury are evident.

Before going on to a more detailed description of the spacecraft and its systems, it might be well to first examine the Gemini mission objectives and consider how they have influenced the spacecraft design. Quoting from the Gemini contract:

"The objective of this contract is the research and development of a versatile general purpose spacecraft for the accomplishment of space missions of increasing complexity.

"Specific objectives are: (not in order of importance)

- a. Fourteen-day earth orbital flights.
- b. Controlled land landing as primary recovery mode.
- c. Demonstrate rendezvous and docking with a target vehicle in earth orbit as an operational technique.
- d. Develop simplified spacecraft countdown techniques and procedures for accomplishing the rendezvous mission which are compatible with spacecraft launch vehicle and target vehicle performance.
- e. Determine man's performance capabilities in a space environment during extended missions."

Additional major design considerations were the designation of the Titan II as a launch vehicle, the selection of the Agena as a rendezvous and docking target vehicle, and the requirement for a two-man crew.

Each of the stated mission objectives represents a significant step forward, and each dictates specific design requirements beyond those imposed by the Mercury mission. In Table 1, an attempt is made to categorize according to mission objective the many Gemini subsystems and design features which are new or significantly improved over corresponding features in the Mercury spacecraft.

Consider first the 14-day mission. This requires an order of magnitude improvement in mean-time-before-failure in many areas to achieve comparable mission reliability to the Mercury program. Added electrical energy requirements have dictated the selection of a fuel cell system for power while in orbit. Cryogenic storage of hydrogen and oxygen fuel cell reactants and breathing oxygen are used to conserve weight and volume. Heat generated by equipment and crew is rejected to space by a radiator using a liquid coolant which is circulated through cold plates and heat exchangers in place of the water boiling technique used for the shorter Mercury missions. Pulse code modulated telemetry gives the high data transmission rates needed to dump the stored information during the limited time over ground tracking stations. Due to the bulk of expendable supplies, the adapter between the launch vehicle and the spacecraft re-entry module is utilized as an equipment compartment and is retained with the spacecraft in orbit. It is jettisoned just prior to re-entry in contrast to the Mercury procedure of leaving the adapter attached to the launch vehicle.

Achievement of the second objective of a controlled land landing at a pre-selected point involves control of both the re-entry trajectory and the final touchdown maneuvers. By offsetting the center of gravity of the re-entry module approximately 1.75 inches from the longitudinal center line, an aerodynamic lift-to-drag ratio of approximately .22 is generated. The resulting lift vector is directed as needed to modulate the re-entry trajectory by controlling the roll attitude of the spacecraft. The roll attitude is adjusted in response to error signals generated by an on-board inertial guidance system consisting primarily of an inertial measuring unit and a general purpose digital computer. The inertial guidance system also performs orbit navigation functions to keep track of present position and compute the proper retrograde time to allow touchdown at any pre-selected site within the maneuvering capability of the vehicle. A digital command receiver permits periodic up-dating from ground tracking stations. Controlled landing is accomplished by means of a paraglider with final touchdown on a 3-skid landing gear. Ejection seats are provided as a backup for the paraglider. They also serve as a crew escape system during the early portion of the launch and pre-launch mission phases.

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The third mission objective is to rendezvous and then to dock with an Agena target vehicle. Target bearing, range, and range rate is detected by a rendezvous radar system installed on the nose of the spacecraft. An orbit attitude and maneuvering propulsion system (OAMS) utilizing hypergolic storable propellants is installed in the adapter module and permits three axis attitude and translation control. The previously mentioned inertial guidance system platform and computer units are utilized to convert the radar outputs into displayed thrust and attitude commands which enable the crew to accomplish the rendezvous maneuvers. The digital command system is used to receive ground commands to enable the crew to maneuver to within radar range of the target. Docking latches mounted in the nose of the spacecraft are utilized in conjunction with a docking adapter mounted on the target vehicle to accomplish the final docking operation. Storage of the propellants and thrusters again dictates use of the adapter as an equipment bay.

The fourth mission objective, accomplishment of simplified countdown techniques, has significantly affected the design of the spacecraft. A number of the major subsystems such as the radar, re-entry attitude control system, paraglider installation, fuel cell and reactant system, cooling pump package, environmental control system, and maneuvering propellant system, have been built into separate subassemblies. This modular concept allows systems to be checked out on the bench and quickly installed in the spacecraft. All of the electrical and electronics equipment in the re-entry module is installed in equipment bays easily accessible through doors in the outer mold line of the spacecraft. Test points are built into all systems with necessary leads brought to conveniently accessible connectors for tie-in to test equipment. Automatic checkout equipment is provided for rapid countdown operation, and all Aerospace Ground Equipment has been carefully integrated with the spacecraft and launch pad systems.

The final objective of establishing man's performance capabilities during extended periods in orbital flight has led to the basic concept of on-board mission command. Decisions and control capability are crew functions. The crew makes such decisions as to when to abort, when to initiate rendezvous and retrograde maneuvers, with the ground complex serving in a monitoring and advisory capacity. Attitude and translation maneuvers are manually controlled. All on-board systems are monitored and operated by the crew. The space suits have been designed such that helmets, arms, and legs may be removed to approximate a shirt-sleeve operating condition. Provisions have been made in the hatches and the pressurization system to allow egress from the cabin into space when properly suited.

This concludes the roll call of new features introduced by the specific Gemini mission objectives. To this list can be added those items which have direct counterparts in the Mercury spacecraft. These include the basic aerodynamics shape and re-entry heat protection concepts, the life support

system utilizing a 5 psia oxygen atmosphere, UHF and HF voice communications, S-Band and C-Band tracking beacons, solid propellant retrograde rockets, re-entry module attitude control thruster system, silver zinc batteries for re-entry and post-landing electrical power, and various recovery aids.

The remaining portion of this paper will discuss the integration of the foregoing features into the Gemini spacecraft design and will present a more detailed description of some of the major systems.

The full scale mockup photograph in Figure 1 serves to relate the overall size of the spacecraft to the crewmen standing alongside. Worthy of note in this view are the inset individual windshields for the pilot and crewman. In Figure 2, the mockup is arranged to illustrate the division points between the major structural assemblies. At the right of the photograph is a 5-foot diameter target docking adapter which is supplied by McDonnell and is bolted to the Lockheed Agena target vehicle. Next in line is the re-entry module. It consists of a conical cabin section housing the crew and most of the environmental control and electronics equipment, surmounted by a cylindrical section containing the re-entry attitude control thrusters to which is attached the rendezvous and re-entry section in which the paraglider rendezvous radar and docking provisions are stowed. The spherical surface of the ablative heat shield forms the base of the re-entry module. The overall length of the re-entry module is 144 inches, and its maximum diameter is 90 inches. For reference, corresponding Mercury dimensions are 90 inches and 74.5 inches, respectively. The adapter shown here in two sections is actually built as a single structural unit and severed during the course of the mission into the two parts illustrated. The complete adapter is 90 inches long and tapers from the 120-inch Titan II diameter at one end to the 90-inch re-entry module base diameter at the other. The part adjacent to the re-entry module, termed the retrograde section, contains the four solid propellant retrograde rocket motors. The equipment section houses the fuel cells and reactants, OAMS propellants, coolant circulating pumps, and miscellaneous electronics and instrumentation equipment. The equipment section is jettisoned just prior to the retrograde maneuver by severing the structure at the point shown in the photograph by means of a flexible linear shaped charge. A similar shaped charge at the 120-inch diameter base of the adapter is used to disconnect the spacecraft from the launch vehicle after insertion into orbit. Attachment of the adapter to the re-entry module is by 3 steel straps spaced about the periphery of the re-entry module. These straps, along with wiring and tubing, are cut simultaneously by shaped charges when the retrograde section is jettisoned.

Figure 3 shows the interior arrangement of the spacecraft. The crewmen sit side by side but with each seat canted outboard 12°. This eliminates any chance of contact during simultaneous seat ejection and also conserves space for equipment. The pressurized cabin area houses the crew and their

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directly associated equipment and supplies, such as food, water, and waste provisions, environmental control system, displays, and crew-operated controls. All other equipment needed for re-entry and post-landing is housed in the re-entry module in equipment bays which are outside the pressurized cabin. The major electrical and electronic bays are on either side of the cabin and are accessible through removable doors in the outer mold line. A third bay is located under the floor. The equipment in these compartments is so arranged that each unit may be removed and replaced without disturbing any other.

Equipment and supplies which are not needed for re-entry and post-landing phases of the mission are stowed in the jettisonable adapter. This results in a first-order reduction in the weight and size of the re-entry module, plus a second-order reduction in the weight and size of such items as retrograde rockets, heat protection, paraglider, and landing gear which are involved in recovery of the spacecraft from orbit.

As noted earlier, the modular concept has been adopted in a number of areas to facilitate fabrication as well as to expedite checkout and maintenance. The extent to which this principle has been applied is illustrated in Figure 4. In the re-entry module, the radar is installed as a unit in the nose of the rendezvous and radar section which, in turn, is installed as a module complete with the paraglider wing. The paraglider inflation bottle and fittings, along with its forward cable reel assemblies, are mounted on the nose landing gear, and the entire assembly is installed as a unit. The re-entry attitude control system is completely contained in a cylindrical module which bolts on to the front end of the cabin section. This module includes propellant and pressurization tanks, pressure regulators, valves, and thrust chamber assemblies. The greater part of the environmental control system is installed in a single unit through an access door beneath the crewmen's seats. It contains the cabin and suit atmospheric circulating fans, carbon dioxide and water removal systems, oxygen pressure regulators, and various control valves.

The primary breathing oxygen supply is in a separate module; the fuel cell assemblies, complete with cryogenic reactant supply system and all related controls are in another. A third module accounts for the orbit attitude control and maneuver propellant tanks, along with their associated pressurization valves and regulators. The coolant pumps and heat exchangers for the environmental control system cooling loop are mounted in a fourth unit, and all of the electronics equipment stowed in the adapter is mounted on a fifth unit. The retrorockets are installed individually in the retrograde section.

In the adapter, particularly, the modular arrangement provides for mission flexibility. Except for the outer adapter shell and supporting beams for the retrorockets, all of the structural supports for the tankage and equipment are contained within the modules. Two interchangeable

versions of the propulsion, fuel cell, and breathing oxygen modules, differing only in tankage capacity, are being built to allow for the different requirements of the 14-day mission and the 2-day rendezvous mission. As mission requirements change during the course of the program, it will be possible to modify the various modules as needed with a minimum of change to the basic spacecraft.

Structure and Heat Protection. Figure 5 shows the basic structural arrangement of the re-entry module. The design concept is to provide a basic load-carrying structure of titanium which is protected from the heat of re-entry by an outer sheath of high temperature material. The pressurized cabin walls are of a double layer of .010 inch titanium, reinforced by stiffeners and by the equipment shelves as shown. The cylindrical re-entry control module is bolted to a ring at the small end of the conical cabin section with 9 attachment bolts. The ablation shield is fastened to a ring at the opposite end of the cabin section. The jettisonable rendezvous and recovery section is fastened to the re-entry control module by a ring of bolts whose heads are blown off by a mild detonating fuse assembly to deploy the paraglider.

The main landing gear skids fold into two longitudinal bays which extend the length of the conical section immediately below the equipment bays on either side of the cabin. The main gear struts extend outward and downward under the action of a pyrotechnic actuator. Forged fittings transmit the trunnion loads into the cabin walls and the aft bulkhead. The landing gear doors are bolted on and are jettisoned pyrotechnically in a similar manner to the rendezvous and re-entry section. The nose landing skid is attached to the forward face of the re-entry control section and, when retracted, is covered by the rendezvous and re-entry canister. The heat shield which covers the face of the re-entry module consists of a fiber glass honeycomb structural dome which supports the ablative facing material. A typical section through the heat shield is shown in Figure 6. The DC-325 ablative material is a McDonnell-developed silicon elastomer now commercially available from Dow Corning. It has excellent ablative characteristics particularly with respect to the char layer formed during ablation, is stable in a vacuum, and is able to withstand the temperature ranges encountered in the space environment. It is retained in the open face cells of fiber glass honeycomb which is bonded to the structural dome. A Fiberite (MX 2625) ring is used around the outer edge of the heat shield where extra bearing strength is needed to withstand the launch loads transmitted from the adapter. This ablative heat shield concept represents a significant design improvement over the Mercury heat shield which utilized a phenolic resin impregnated, laminated fiber glass cloth ablative layer. Weight of the Gemini shield is 317 lbs. as compared to 303 lbs. for the Mercury shield - an increase of only 4-1/2% in spite of a 48% increase in area, a 25% increase in the ballistic loading parameter $W/C_D A$, and a 90% increase in the design total heating per square foot due to the more critical lifting re-entry required by the Gemini spacecraft.

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The afterbody heat protection used on Gemini is almost identical to that proven on the Mercury spacecraft. As shown in Figure 7, high temperature Rene 41 shingles .016" thick are used over the conical section. Withstanding temperatures of up to 1800°F, these shingles achieve a thermal balance by radiation to the atmosphere. The shingles are attached to the basic structure using bolts and washers through oversized holes to allow for thermal expansion. Small blocks of Min-K insulation are used at the support points, and a layer of Thermo-flex insulation is used between supports to keep substructural temperatures within limits. Over the cylindrical sections of the afterbody, heating rates are too high for efficient radiation cooling and, therefore, a heat sink principle is utilized. Beryllium shingles .24 inches thick on the windward side and .09 inches thick on the leeward side are installed over this area. Again, provisions for thermal expansion are included.

As a matter of interest, the heat distribution pattern over the afterbody, based upon wind tunnel model data at a Mach number of 10 and an angle of attack of 20°, is shown in Figure 8. The isotherms shown represent constant values of the ratio of the local to the stagnation heat transfer coefficients, where

$$h_{\text{local}} = \frac{q_1}{T_o - T_w}$$

$$h_{\text{stag}} = \frac{q_{\text{stag}}}{T_o - T_w}$$

q_1 = local heat transfer rate

q_{stag} = stagnation point heat transfer rate

T_o = free stream stagnation temperature

T_w = wall temperature at point considered

The more critical conditions on the cylindrical section are apparent.

The adapter structure illustrated in Figure 9 consists of a cylindrical shell of HK-31 magnesium skin .032" thick, stiffened by longitudinal stringers of HM-31 magnesium with stabilizing aluminum rings at several locations. As previously mentioned, the only other structural elements in the basic adapter are the retrorocket support beams shown in the Figure. Magnesium is utilized as the basic structural material in order to withstand launch temperatures of up to 600°F without further protection. A unique feature of the adapter, shown in the sectional view, is the manner in which the entire outer surface is used as a space radiator. The environmental control system coolant is circulated through .25" tubes which are extruded integrally with the longitudinal structural stringers. Fifty foot long extrusions are doubled back and forth to form redundant coolant loops with a minimum of connections. This arrangement not only saves weight, but results in a superior design from the meteoroid puncture standpoint, since the coolant tubes are protected both by the outer skin and the legs of the extrusions.

Environmental Control System. As noted in Figure 10, the basic concepts of the Gemini environmental control system are similar to those of Mercury. Points of similarity include the use of a 5 psia pure oxygen atmosphere, use of a space suit to back up the pressurized cabin, CO₂ removal by lithium hydroxide. Two significant departures from the Mercury system are incorporated in Gemini. These are the use of cryogenic rather than gaseous storage for primary oxygen, and the use of a coolant fluid and space radiator as the primary means of heat removal rather than water boiling.

With the exception of the radiator, the environmental control system is supplied by the AiResearch Division of the Garrett Corporation. AiResearch also supplied the Mercury system, and is thus able to draw upon this back-log of experience.

The suit and cabin are pressurized with oxygen supplied from either the primary cryogenic source or a secondary gaseous supply. The secondary supply is stored at 5,000 psi in two 7-lb. capacity bottles in the re-entry module, and serves both as an emergency supply in orbit and as a normal supply during re-entry. Either of the two bottles will permit at least one full orbit plus re-entry. The cryogenic supply, stored in the adapter, contains up to 10⁴ lbs. of super-critical oxygen.

The suit compressors circulate the oxygen through an odor and CO₂ adsorber, a heat exchanger, a water absorber, the pressure suits, and a solids trap. In the event the primary compressor becomes incapable of maintaining the required circulation rate, a second redundant compressor is activated. A cabin fan circulates the cabin atmosphere through a second heat exchanger.

A silicon ester coolant, Monsanto MCS 198, is circulated through the cabin and suit heat exchangers, equipment cold plates, fuel cells, and finally the space radiator to remove the heat absorbed. The coolant loop is completely redundant, and two pumps are provided in each loop. Normally, only one pump in one loop is required; however, for peak electrical and solar load conditions, two pumps in one loop, or one pump in each loop, are turned on. During launch, aerodynamic heating raises the temperature of the adapter surface to the point where the radiator is ineffective. Therefore, it is by-passed and the coolant is circulated through a water boiler during this phase of the mission. Approximately 30 minutes are required after launch for the radiator to cool back down. The total water requirement for this operation is less than 10 lbs.

The cabin is equipped with a dump valve to effect depressurization, and a high flow rate repressurization valve. These, coupled with the single point hatch unlatching mechanism, provide for egress experiments in space. NASA is currently developing the necessary portable life support kit for this application.

Electrical System. A block diagram of the electrical system is shown in Figure 11. Primary electrical power during launch and orbit phases of the mission is supplied from a hydrogen-oxygen fuel cell battery stowed in the adapter. This unit,

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currently under development by the General Electric Company, is of the ion exchange membrane type. Actually, the installation in Gemini consists of two separate, identical packages - or sections, as they are called - each of which has redundant coolant loops, its own reactant control valves, electrical controls, and instrumentation. Each section is made up of three stacks of 32 individual cells. Each stack has a rated output of 350 watts at 23.3 volts for a total rated power of 2100 watts. No load voltage is 28 volts. It is possible to shut down any single stack in the event a malfunction is detected. Peak power requirements for presently planned missions can be met by the fuel cell battery even with one stack inoperative.

The cryogenic hydrogen and oxygen reactant storage and regulation system is supplied by the AiResearch Division of the Garrett Corporation. As in the case of the breathing oxygen, two sizes of tanks are being developed with usage depending upon the mission length.

For retrograde, re-entry, and post-landing phases of the mission - which occur subsequent to jettisoning of the equipment adapter - power is supplied from a bank of four 16-cell silver zinc batteries rated at 40 amp hours each. These batteries are tied into the same main bus as the fuel cells, and serve as an emergency orbital power supply in case of a fuel cell failure. In the event of a partial fuel cell failure, the silver zinc batteries may be used to augment the fuel cells during the few hours when peak power is required. This will permit successful completion of the mission even if one complete section is lost. In the event of complete fuel cell system failure, the batteries will provide for at least one orbit followed by a normal re-entry and a minimum of 12 hours post-landing equipment operation.

A second battery system consisting of three 15 ampere-hour 16-cell silver zinc batteries is provided in the re-entry module to power pyrotechnic devices and various control relays and solenoids. Isolation of these systems from the main bus prevents feedback of voltage spikes, resulting from such devices, into critical electronics equipment. This design results from Mercury experience where such "glitches" proved to be a troublesome nuisance. As shown in the block diagram, diodes are used to isolate the two pyrotechnic squib batteries from each other so that complete redundancy in pyrotechnic systems is carried all the way back to the power source.

Another deviation from Mercury practice is the provision of individual inverters for each of the several AC powered devices such as the control system electronics, inertial guidance system, suit and cabin fans, and coolant pumps. This allows electrical characteristics of each inverter to be matched in its particular application. Off design operation with resulting penalties in conversion efficiency is thereby minimized.

Attitude and Maneuver Propulsion Systems. A total of 32 bi-propellant liquid rocket thrust chambers are used for controlling attitude and maneuvering the Gemini spacecraft. Thrust chamber sizes and locations are shown in Figure 12. As shown in the left-hand sketch, three independent attitude control systems are provided. Each consists of eight 25-lb. thrust units arranged to fire in parallel pairs for yaw and pitch control or, differentially, for roll. Two of the systems are packaged in the cylindrical re-entry control system module at the forward end of the cabin section. Each of these systems has its own propellant and pressurization tankage, valves, and lines. These re-entry control systems, referred to as the RCS, are utilized only during the retrograde and re-entry portions of the mission. They are made redundant since they are considered essential to crew safety.

The third ring of attitude control thrusters is used during the orbital portion of the mission and is located at the rear of the adapter module. The eight maneuvering thrusters are arranged as shown in the right-hand sketch. Four 100-lb. units are directed through the center of gravity to provide for lateral and vertical impulses. A pair of aft-facing 100-lb. thrusters at the base of the equipment adapter section provides forward impulse. A pair of 85-lb. units facing forward and canted slightly outboard, mounted on either side of the adapter close to the re-entry module attachment station, provides reverse thrust.

The adapter-mounted orbit attitude and maneuver propulsion systems, generally referred to as the OAMS, share a common propellant supply. The OAMS thruster arrangement permits attitude and maneuver control in the event of the loss of any single thruster. Complete redundancy is not provided because mission safety is not directly involved, and because of the high weight penalties required to make a truly redundant system. As indicated in the typical section view in Figure 12, the thrust chambers are ablatively cooled with ceramic inserts at the throat section. Separate valves are provided for fuel and oxidizer.

Propellants for both the RCS and OAMS systems are nitrogen tetroxide (N_2O_4) oxidizer and monomethyl hydrazine ($N_2H_3CH_3$) fuel, and are pressure fed to the thrust chambers from bladder type tanks. Propellant capacity is 35 lbs. for each of the two RCS systems. This is sufficient to accomplish retrograde and re-entry with either system. Maximum propellant capacity of the OAMS tanks is approximately 700 lbs. This is sufficient to provide attitude control throughout a rendezvous mission plus a 700 foot/second maneuvering velocity increment. For the non-rendezvous, long duration mission, smaller OAMS tanks are used to save weight and allow extra oxygen and fuel cell reactants to be carried.

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The RCS and OAMS thrust chambers and propellant systems are being supplied to McDonnell by the Rocketdyne Division of North American Aviation.

Guidance and Control Electronics. A detailed description of the Gemini guidance and control system and its operation is beyond the scope of this report. A series of reports could be, and in fact has been, written on this phase of the spacecraft design. Therefore, only a brief description of the system and its components, along with a very cursory review of its functions, will be attempted.

Figure 14 presents a very simplified block diagram of the guidance and control system. Pilot inputs are made through either the attitude control or the maneuver control handle. These are processed through the attitude control and maneuver electronics (ACME) which contains the logic circuitry needed to select the proper thruster valves. Resulting spacecraft dynamics are sensed by rate gyros, by the horizon scanners and inertial measuring unit of the inertial guidance system and, in the case of motion with respect to the target, by the rendezvous radar. Outputs from the sensing units are fed either directly or through the computer to the several displays to command pilot action. Depending upon the particular control mode selected, outputs from the sensors may also be fed directly into the ACME to provide automatic or mixed control modes. In the case of re-entry attitude control, a direct mode exists in which control signals may be fed directly from the control handle into the thruster chamber solenoid valves.

Responsibility for the overall concept, definition and final integration of the guidance and control system into the spacecraft rests with McDonnell. Component suppliers include Minneapolis-Honeywell, Minneapolis, for the ACME and rate gyro systems; Minneapolis-Honeywell, St. Petersburg, for the inertial measuring unit; International Business Machine Corporation for the computer and ΔV indicator; Advanced Technology Laboratories for the horizon sensors; Westinghouse for the radar and range rate display; Lear-Siegler for the attitude and rate indicator; and Rocketdyne for the thruster system. IBM also has responsibility for integration of the overall inertial guidance system and for analytical studies in areas of mission planning associated with computer programming.

Although not shown on the diagram, a number of redundancies exist in the system. For example, crew selectable backup units are provided for the horizon scanner, rate gyros, and attitude control electronics. As previously discussed, redundant re-entry attitude control thrusters are available and redundant control signals are provided to them from the control handle.

The pilot has available three different manual attitude control modes: rate command, single pulse, and direct; and two automatic modes: orbital and re-entry. The rate command mode provides a spacecraft angular rate which is proportional to the control handle deflection. This mode is the primary attitude control mode used during maneuvers. With the control handle centered in this mode,

angular rates about all three axes are damped to less than $.1^\circ$ per second. Since $.1^\circ$ per second is equal to the angular rate of the large hand on a clock, this mode is equivalent - for short time periods, at least - to an attitude hold mode. The single pulse mode is one in which a single minimum duration thruster pulse results each time the control handle is deflected from neutral. It is used for precise attitude control; for example, when preparing to align the inertial platform, or for making minor adjustments to angular rates during extended orbital flight. The direct or fly-by-wire mode, as previously noted, is essentially a backup method of operating the thrusters and results in a constant angular acceleration any time the control handle is deflected.

The automatic orbital attitude control mode is a coarse slaving of the spacecraft attitude to the vertical as detected by the horizon scanner. It holds roll and pitch attitude to within approximately 5° during extended periods of orbital flight without the need for operation of the complete inertial guidance and associated electronics systems. Since no yaw reference is available in this mode, yaw attitude is manually controlled by the pilot using the single pulse mode for control and visual observation of the ground as a reference. The automatic re-entry attitude control mode provides rate damping in pitch and yaw about the aerodynamic trim point of the spacecraft, and roll attitude control in response to error signals from the inertial guidance system.

Only one mode of translation thruster control is provided. This is a manual direct control mode in which thrusters are simply turned on or off by motion of the maneuver handle.

The heart of the guidance and control system is the inertial guidance system. As noted earlier in this report, it is basically required to perform the mission objectives of rendezvous and controlled re-entry. However, as shown in Figure 15, its versatility has been exploited to perform a number of other important mission functions. A most significant one is backup for the basic radio guidance used for the Titan II launch vehicle. Although not indicated on the block diagram, error signals from the inertial guidance system may be fed to the launch vehicle autopilot to control its engine gimbal actuators. Another function to be programmed in the computer is a launch abort navigation mode which will enable touchdown from launch abort to be made at pre-selected points. Orbital navigation which is essentially keeping track of present position in orbit may also be accomplished. This program is carried a step further to allow calculation of the retrograde time required to touch down at any pre-selected point within the maneuver capability of the spacecraft in the event of an abort from orbit. Rendezvous maneuver commands are generated with the aid of the rendezvous tracking and ranging radar. The re-entry control program generates the error signals necessary to accomplish roll attitude control during re-entry.

Docking System. One of the major objectives

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of the Gemini spacecraft program is to actually dock with the target vehicle. The docking and latching procedure is shown in Figure 16. A target docking adapter (TDA) supplied by McDonnell bolts on to the upper equipment bay of the Agena target vehicle. This adapter contains a radar transponder which operates in conjunction with the spacecraft rendezvous radar, flashing lights for visual target acquisition, and a docking cone. The latter is a funnel-shaped assembly supported by shock absorbers which damp out impact loads and prevent rebound. A V-shaped slot in the cone mates with an indexing bar on the spacecraft to align the two vehicles. This permits three latches in the cone to engage fittings in the nose of the spacecraft. The cone is then retracted and locked tight to rigidize the connection between the Gemini and the Agena. The process is reversed to separate the two vehicles. The latching fittings on the spacecraft may be blown free by pyrotechnic charges as a backup means of separation.

The Gemini crew can command the Agena attitude and propulsion systems as well as the docking mechanism by an RF link either before or after docking. Just prior to final docking and while attached to the Agena, the status of the target vehicle is ascertained from displays mounted above the docking ring (not shown in the sketch).

Landing System. The paraglider is the only major Gemini system being provided as government-furnished equipment to McDonnell. It is being developed under a separate contract to NASA-MSC by North American's Space and Missile Systems Division. Detailed design insofar as installation in the spacecraft is concerned is being closely coordinated with that of the spacecraft. Certain portions of the system such as the gas supply for inflation are being provided by McDonnell.

Figure 17 presents a brief summary of the paraglider deployment and performance characteristics. The paraglider in conjunction with the re-entry trajectory control system permits touchdown at the pre-selected site. It is flown much like a two-control airplane using the same hand controller as for orbit attitude control.

The landing gear, as previously described, is a tricycle skid type. The nose gear is extended at paraglider deployment and the main gear by pilot action. The paraglider is jettisoned immediately after touchdown to avoid possible interference during the run-out.

In addition to the paraglider, an 84-foot ring sail parachute recovery system is being developed by Northrop-Ventura under contract to McDonnell. This parachute will be utilized on the unmanned launches, and possibly several of the early manned flights, pending final qualification of the paraglider. Touchdown will be in water when the parachute is used.

Escape Provisions. Probably the most noticeable difference between Mercury and Gemini when viewed externally is the absence of the escape tower

on the latter. In the early part of this paper, the addition of the ejection seat was listed as a design feature added to back up the paraglider. Once having accepted this requirement, extension of the seat's capabilities to enable crew escape on the pad and during the early phases of launch was natural. This was accomplished by incorporating a rocket-type catapult in the seat. This gives sufficient altitude and velocity to deploy the parachute and land the man at a minimum of 600 feet from the base of the launch vehicle. Application of the ejection to off-the-pad operation is really made possible by the fact that with the Titan II launch vehicle, there is no violent detonation associated with deflagration of the propellants. Therefore, the primary consideration is to achieve sufficient clearance from the fire which might result from a launch vehicle malfunction, rather than blast effects.

During the early phases of the launch, ejection conditions are essentially the same as from an aircraft. In fact, the maximum dynamic pressure of approximately 750 lbs/sq. foot encountered during a Gemini launch is only about one-half of the value which can be achieved with current fighter type aircraft. However, as the launch vehicle accelerates, temperature rather than pressure becomes the limiting condition for ejection. A maximum altitude of 70,000 feet has therefore been established for operation of the Gemini ejection seat. As shown in Figure 18, there are two additional modes of escape during the launch phase. The first of these, which extends from the ejection seat altitude of 70,000 feet to an altitude of 522,000 feet, utilizes the spacecraft retrorockets to separate the spacecraft from the launch vehicle. To achieve this capability, the ratio of maximum retrograde thrust to mass for the Gemini spacecraft is increased to almost double that of Mercury. In the abort mode, the 4 rockets are fired in salvo to give a total of 10,000 lbs. of thrust. Above an altitude of 522,000 feet, which corresponds to the point where the velocity is approximately 20,000 feet/second, separation is accomplished in the same manner as a normal injection in orbit. In this third mode, the retrorockets are retained for possible use in the normal retrograde mode to enable more flexibility in selection of a touchdown site.

In conclusion, it should be noted that space has not permitted a description of several important Gemini systems, including communications, retrorockets, and tracking systems. These are all important, and all incorporate interesting features, but in concept have not changed as drastically from Mercury as those covered in this report. However, future reports will most certainly deal in more detail with all Gemini systems.

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TABLE 1

GEMINI SPACECRAFT

MISSION OBJECTIVES	14 DAYS IN ORBIT	CONTROLLED LAND LANDING	RENDEZVOUS AND DOCK	SIMPLIFIED COUNTDOWN	MAN'S PERFORMANCE
<p>New Design Features Resulting</p>	<ul style="list-style-type: none"> • Long Mean Life • Fuel Cells • Cryogenics • Space Radiator • PCM Telemetry • Equipment Adapter 	<ul style="list-style-type: none"> • Offset Center of Gravity • Inertial Measuring Unit • Computer • Digital Command System • Paraglider • Landing Gear • Ejection Seats 	<ul style="list-style-type: none"> • Orbital Attitude • Radar • Inertial Measuring Unit • Computer • Digital Command System • Docking Mechanism • Equipment Adapter 	<ul style="list-style-type: none"> • Modular Concept • Accessible Equipment • Built-in Test Points • Automatic Checkout • Integrated AGE 	<ul style="list-style-type: none"> • On-Board Decisions • Manual Control • Partially Removable Space Suit • Egress to Space

OTHER FEATURES GENERALLY CORRESPOND TO MERCURY.

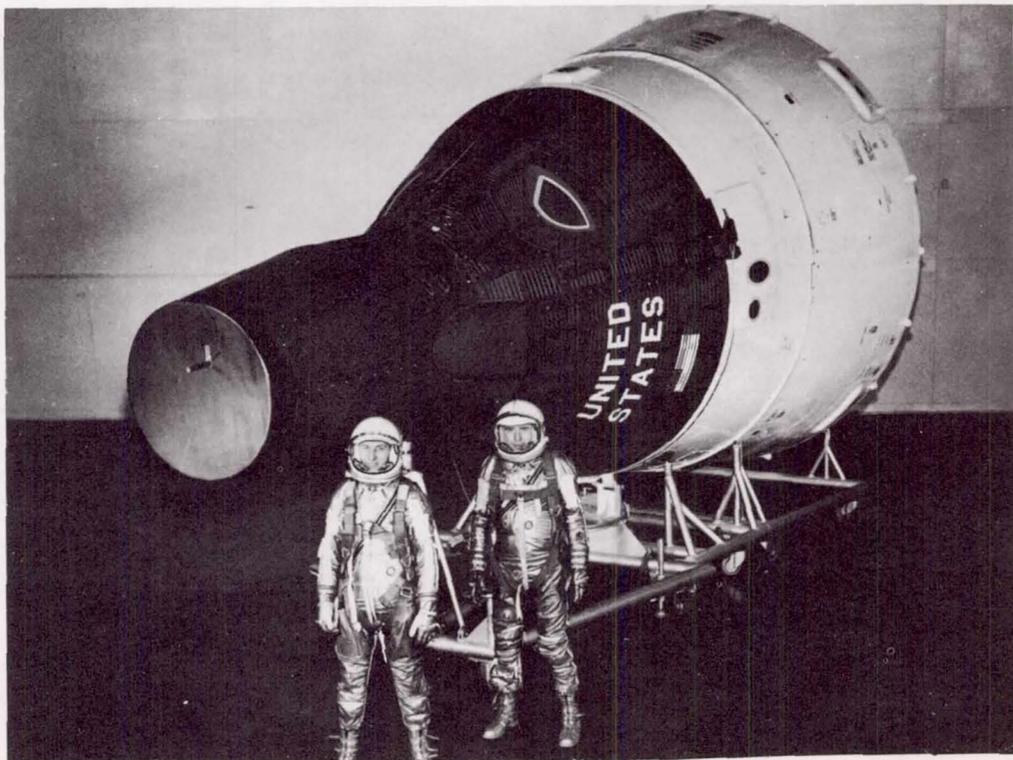


Figure 1. - Mock-up of Gemini spacecraft.

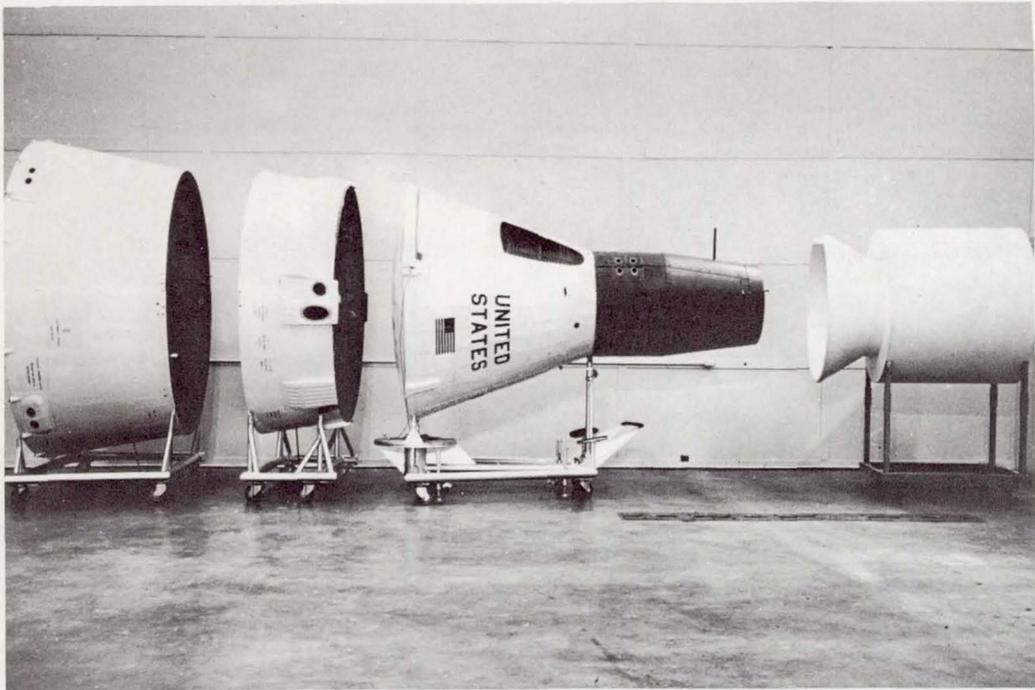


Figure 2. - Division points between major structural assembly points of Gemini spacecraft.

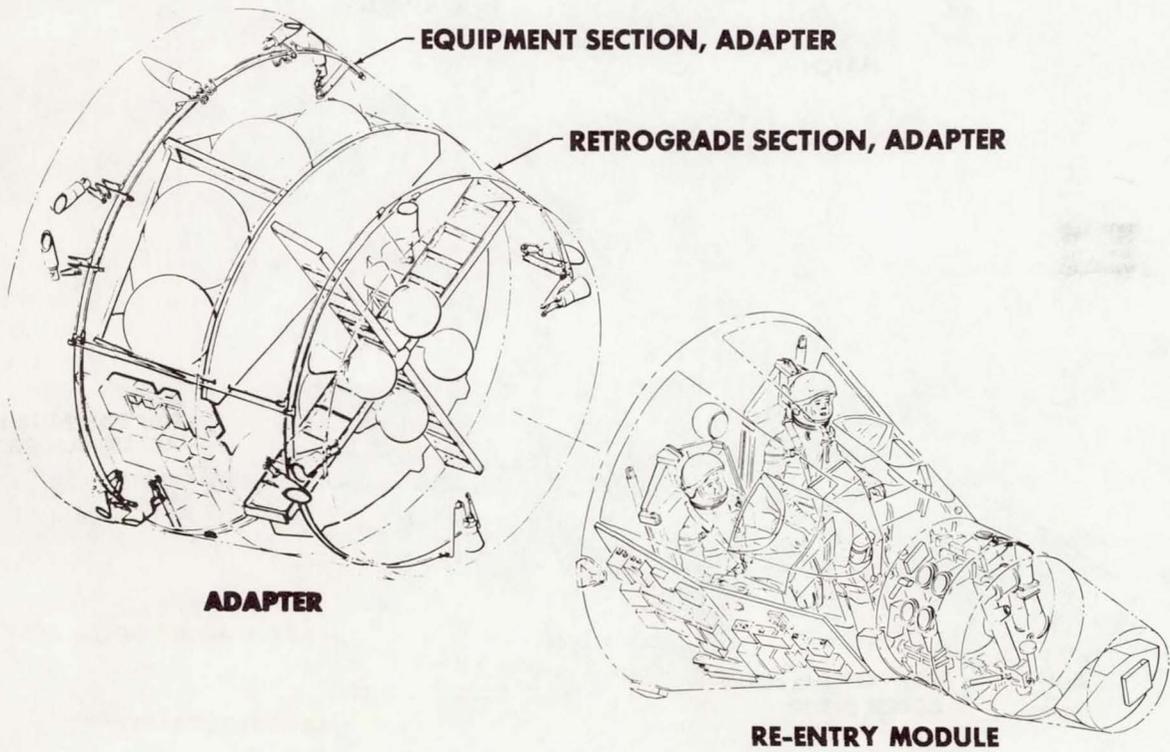


Figure 3. - Gemini adapter and re-entry module.

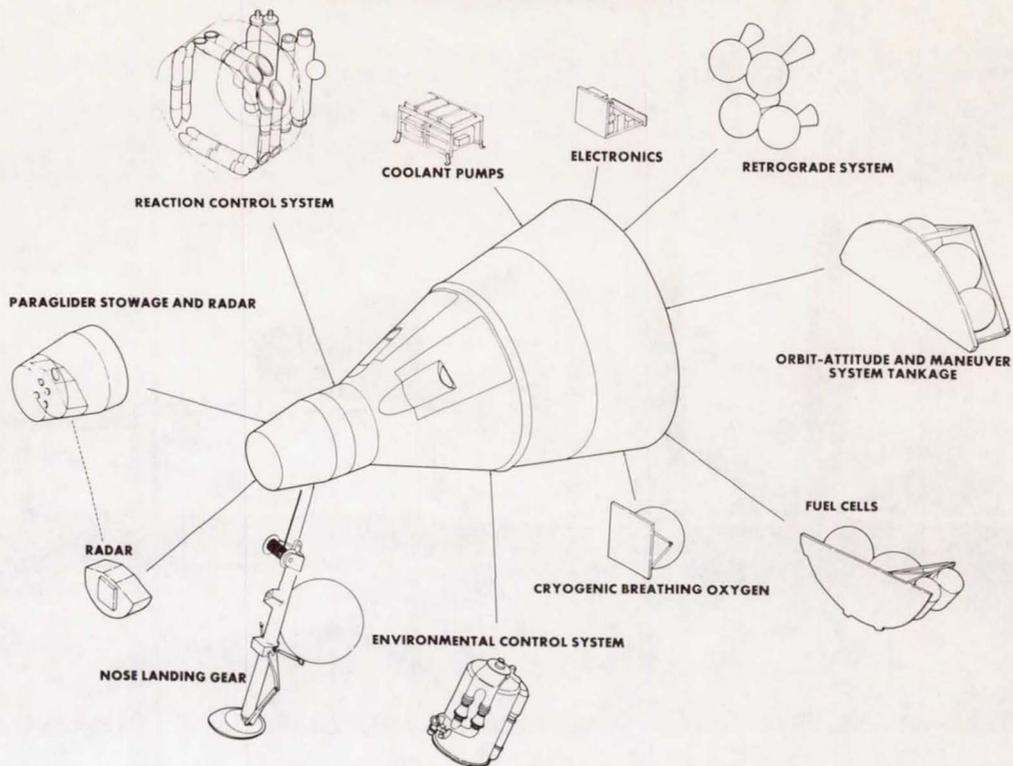


Figure 4. - Subassembly concept.

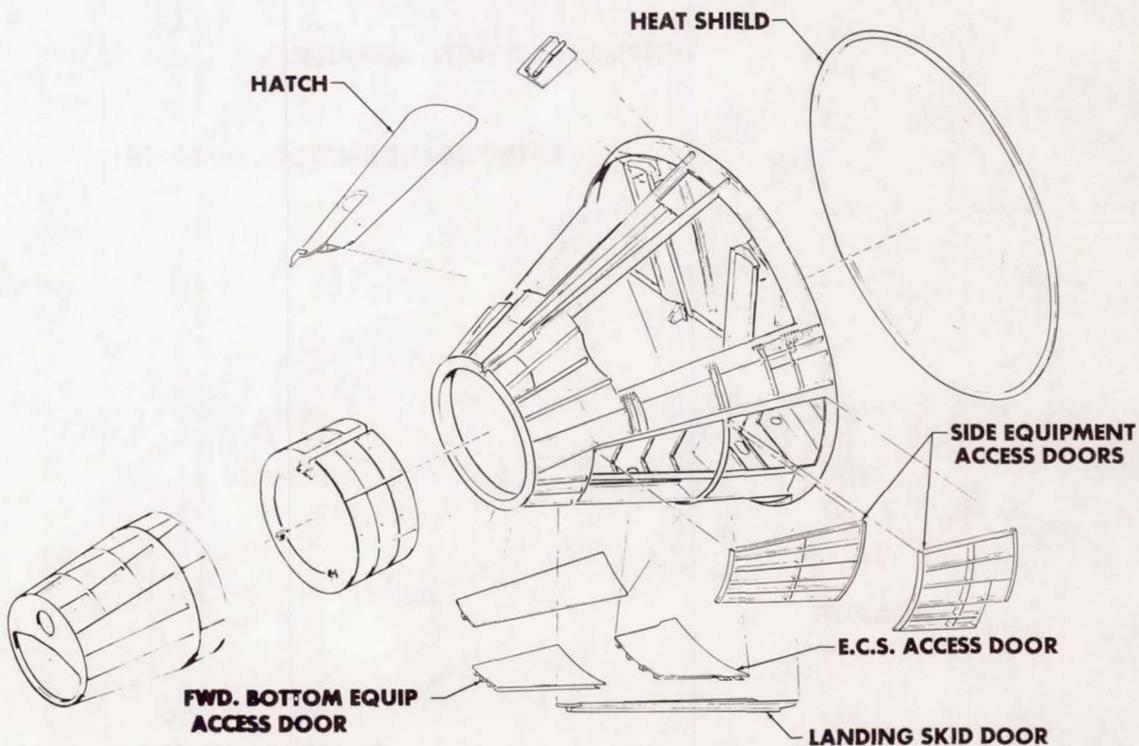


Figure 5. - Re-entry module structure.

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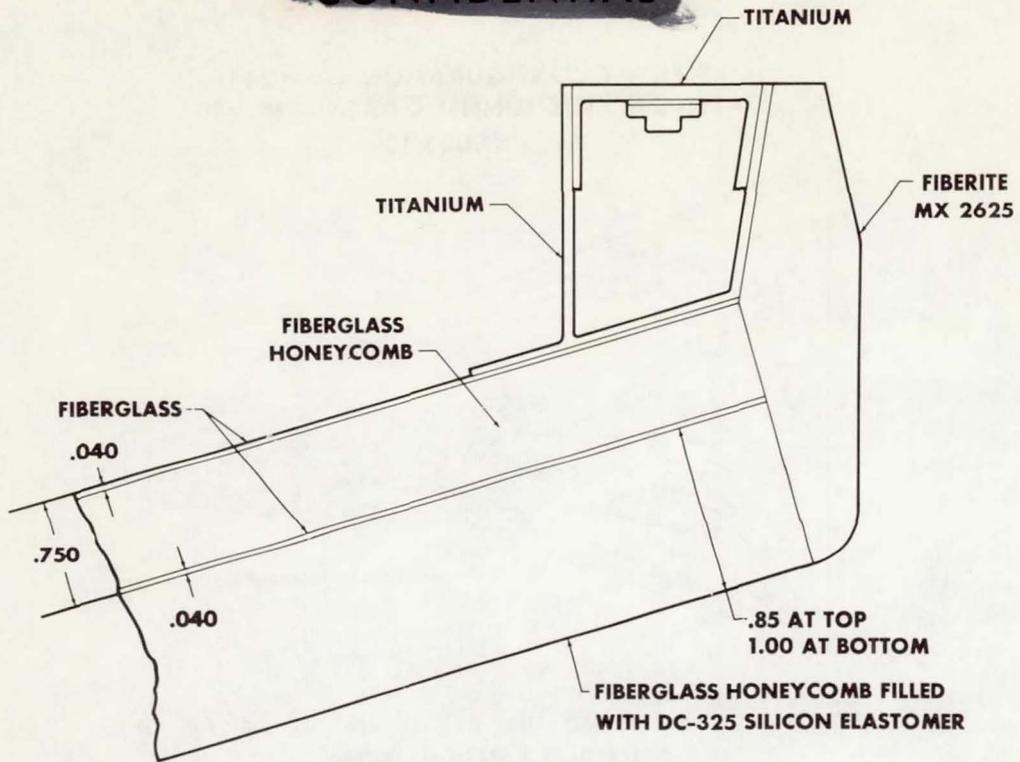


Figure 6. - Ablation shield.

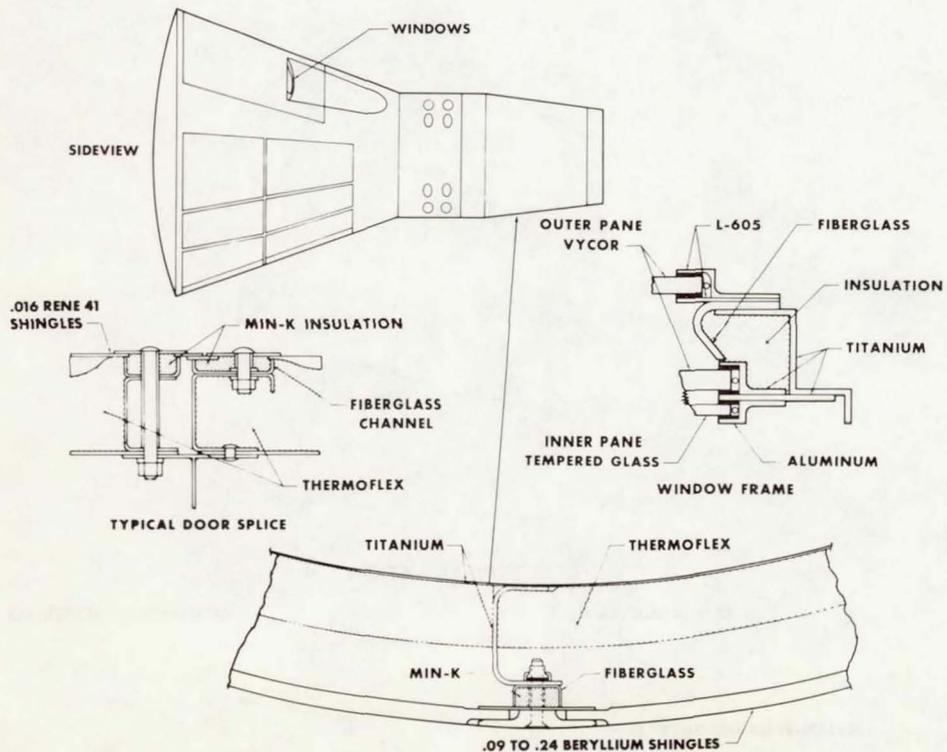


Figure 7. - Afterbody heat protection.

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RE-ENTRY CONFIGURATION $\sim \alpha = 20^\circ$
BASED ON AEDC TUNNEL C DATA $\sim M_\infty = 10$
 $Re_{x/D} = 1.08 \times 10^6$

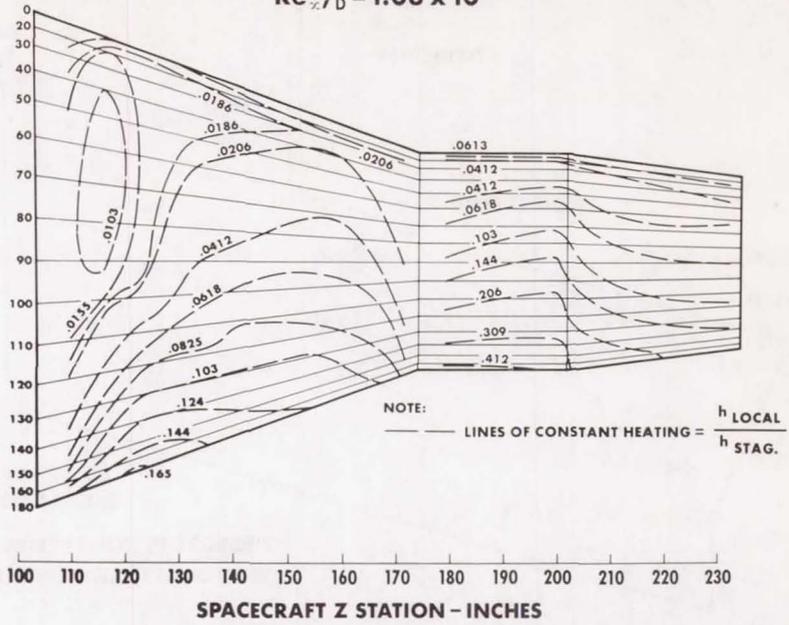


Figure 8. - Gemini afterbody heating distribution.

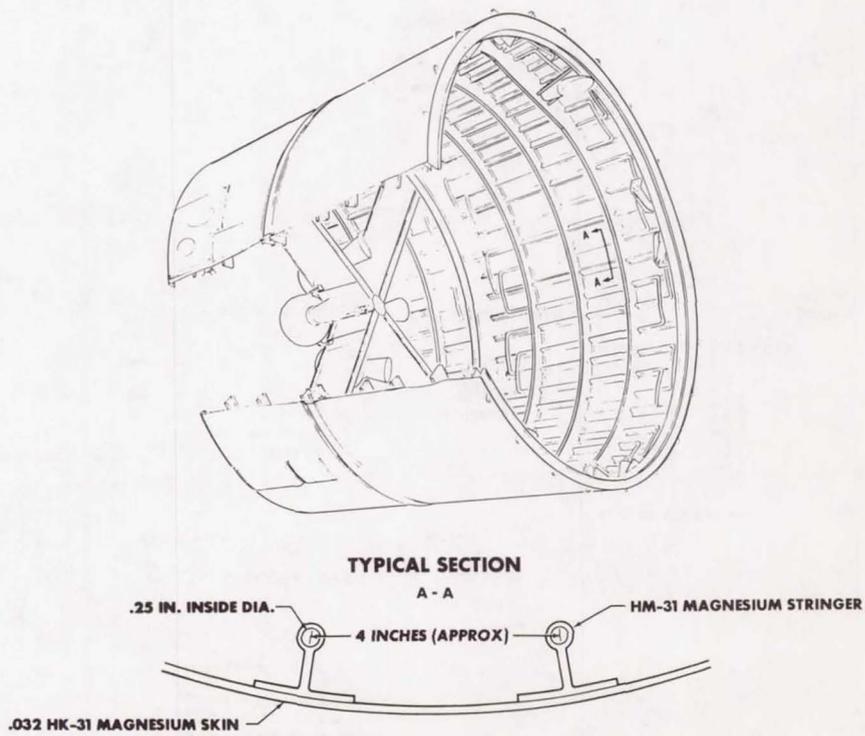


Figure 9. - Adapter structure.

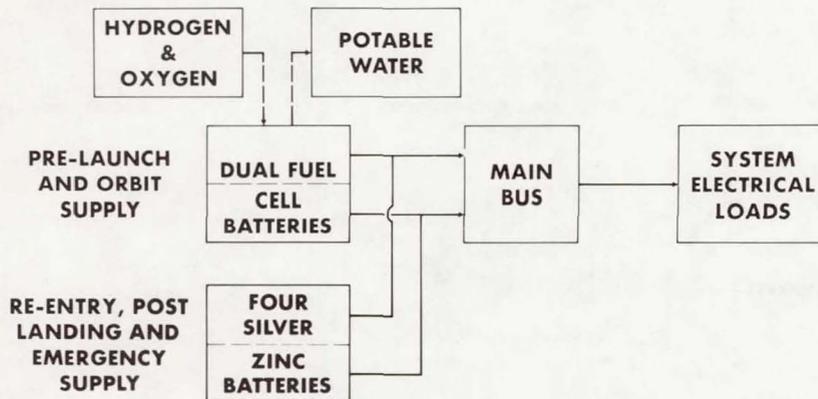
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BASIC SYSTEM CONCEPTS SIMILAR TO MERCURY:

- CABIN AND SUIT ENVIRONMENT -PURE OXYGEN
- CABIN PRESSURE-5.1 PSIA
- SUIT PRESSURE-3.5 PSIA EMERGENCY
3 IN. H₂O BELOW CABIN NORMAL
- ORBIT OXYGEN-104 LBS. SUPERCRITICAL CRYOGENIC
- RE-ENTRY AND SECONDARY OXYGEN-DUAL GASEOUS SUPPLY, 7 LBS. EACH
- SUITS ARE PARALLELED IN SINGLE CLOSED LOOP CIRCULATING SYSTEM
- WATER REMOVAL BY WICK ABSORPTION OF WATER AT THE HEAT EXCHANGER.
- CO₂ REMOVAL BY LITHIUM HYDROXIDE BED

Figure 10. - Environmental control system.

PRIMARY SYSTEM



ISOLATED SYSTEM

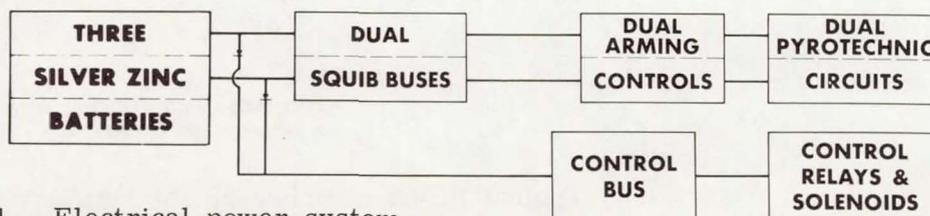
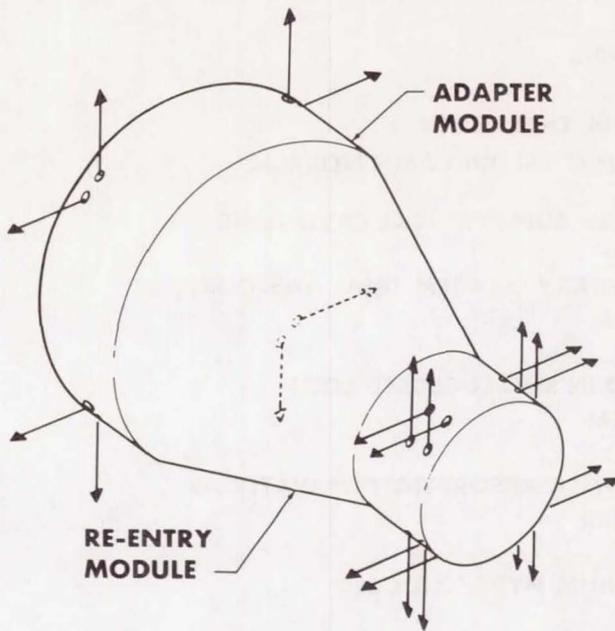


Figure 11. - Electrical power system.

**ATTITUDE CONTROL
25 LBS. THRUST PER UNIT**



**MANEUVER CONTROL
100 LBS. THRUST PER UNIT
(EXCEPT 85 LBS. FOR FORWARD FIRING UNITS)**

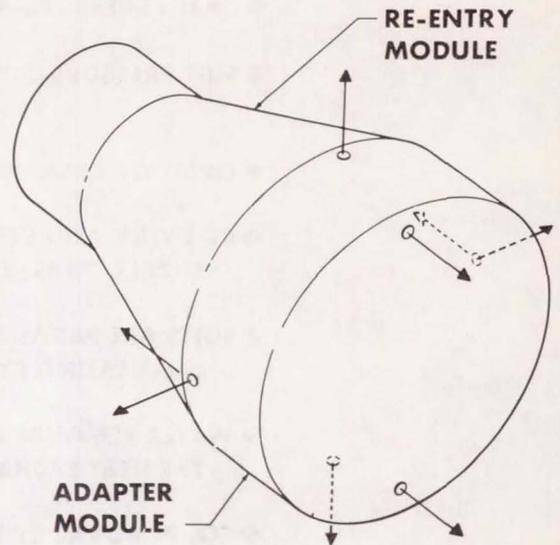
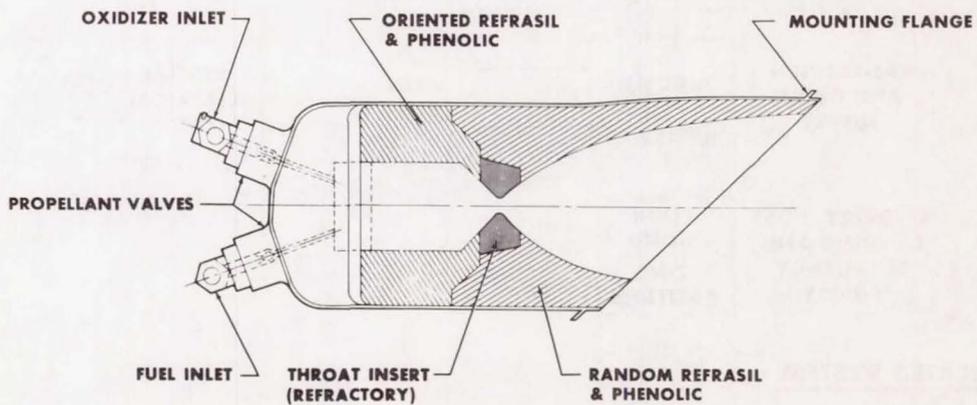


Figure 12. - Thrust chamber arrangement.

CONSTRUCTION OF "GRES" UNLESS OTHERWISE NOTED



NOTE:
VALVE SECTION ROTATED 90° WITH RESPECT
TO CHAMBER FOR CLARITY.

Figure 13. - Typical thrust chamber 25 lb. OAMS

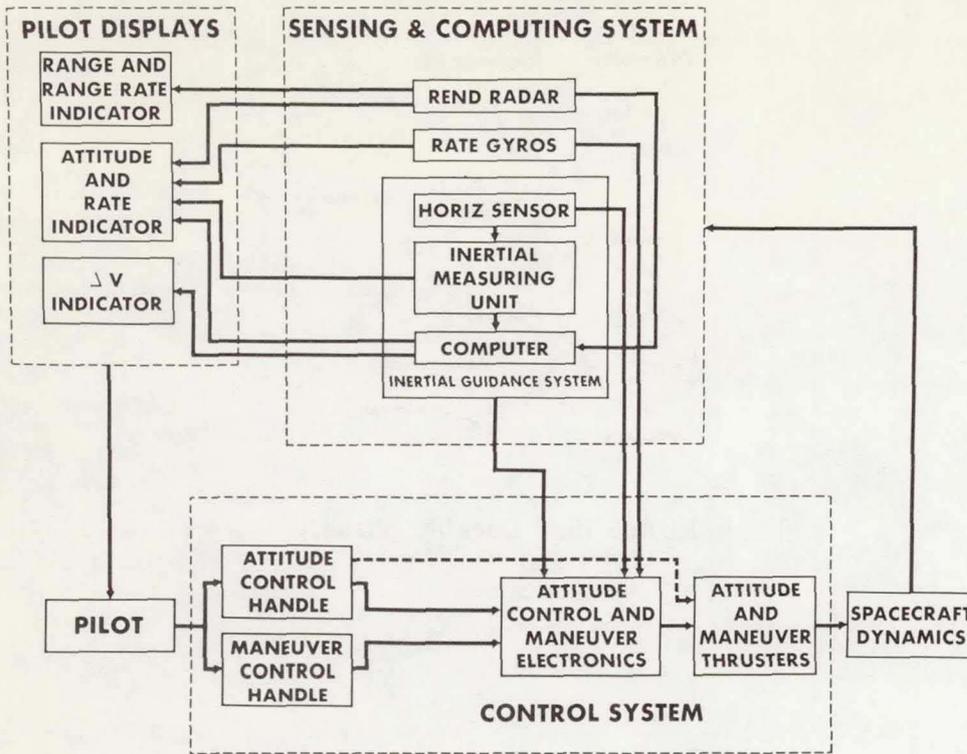


Figure 14. - Gemini guidance and control system.

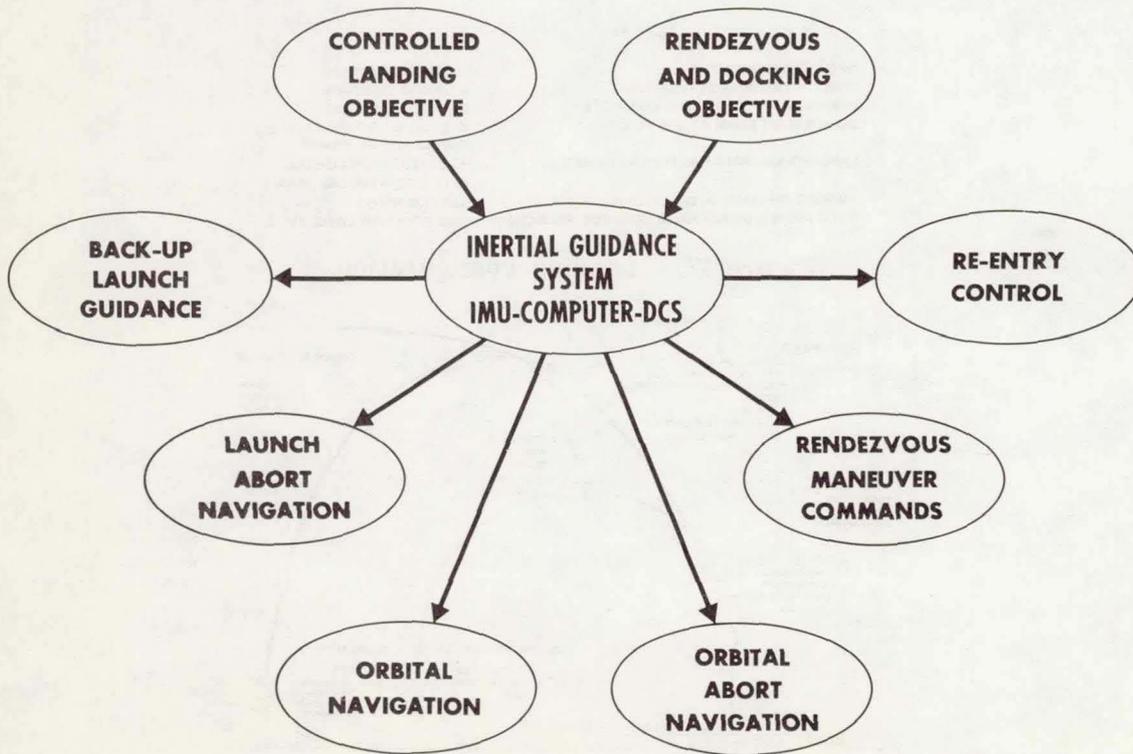


Figure 15. - Inertial guidance system mission application.

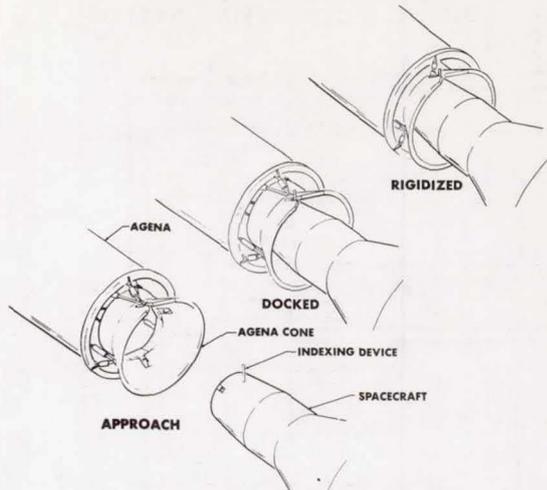
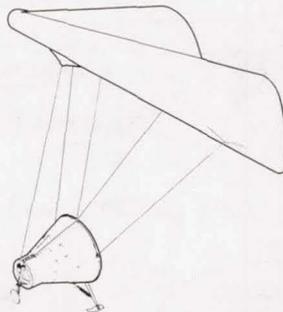


Figure 16. - Docking phase.



DROGUE DEPLOYMENT	60,000 FT. ALTITUDE
START PARAGLIDER DEPLOYMENT	50,000 FT. ALTITUDE
PARAGLIDER DEPLOYMENT COMPLETE	40,000 FT. ALTITUDE
GLIDE RANGE FROM 42,000 FT. L/D=3.1	19.2 NA.MI.-DOWN RANGE
	18.2 NA.MI.-UP RANGE
LANDING VELOCITIES AT TOUCHDOWN	95 FT./SEC. HORIZONTAL
	10 FT./SEC. VERTICAL (MAX.)
RUNOUT DISTANCE AFTER LANDING	330 FT. APPROX.

PILOT FLOWN USING HAND CONTROLLER, ACTUATES GAS OPERATED CABLE REELS.

Figure 17. - Landing configuration.

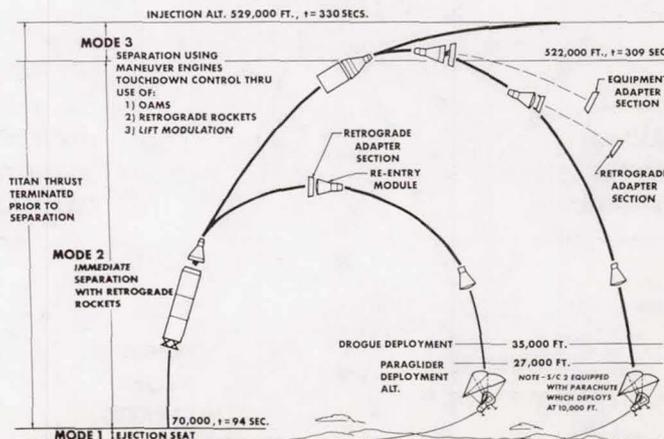


Figure 18. - Launch escape modes.

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The X-20 (Dyna-Soar) Progress Report

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Background

1. X-20 (Dyna-Soar) background encompasses an extensive time period from 1957 to date (see Figure 1). Active research and development has been accomplished during dual phase I competition between Boeing and Martin between mid-1958 and mid-1959, and during the current Dyna-Soar program, which was placed under contract in May 1960.

2. It is to be noted that the active R&D has been accompanied by considerable planning and study efforts. These studies have examined numerous alternate plans for conducting the program, as well as a large number of possible alternate vehicle configurations. Relationship of the X-20A program with other national space programs and with the Air Force Space Plan has been extensively examined in various studies.

3. The initiation of the program in November 1957, was preceded by approximately 4 years of study of methods of extending system performance into the high hypersonic speed flight regime by exploiting large rocket boosters which were under development for the ballistic missile program. It was found that as speed and altitude performance increased, that military potential became of interest. A large number of technical problems were identified and found to be of such a magnitude that a research program was required for their solution. After careful study within the Air Force and NASA, it was concluded that the various interrelated problems could best be solved by a research or "conceptual test vehicle" which would be capable of extending the flight capabilities of the X-15 into the high hypersonic flight regime up to orbital speeds.

4. A Development Directive issued in November 1957, was followed by a competition involving 9 major aircraft companies. From this competition a selection was made of The Boeing and Martin companies to further pursue the relative merits of each company's proposal. During the Phase I competition, both contractors evolved configurations of a wing-body type having very similar characteristics and capabilities. The AF/NASA evaluation concluded that the Boeing glider design and the Martin booster design should be selected for further development.

5. During this period, because of extensive NACA interests in a hypersonic flight research aircraft, a joint Memorandum of Understanding was prepared to make the program a joint AF/NASA program.

6. A three-step program was devised. Step I utilized the Titan I ICBM booster to boost the glider from Cape Canaveral down the Atlantic Missile Range to velocities of approximately 18,000 ft/sec. While not as high as desired, this speed did permit initial investigation of the high

hypersonic heating regime which occurs between 18,000 and 22,000 ft/sec.

7. The second step of the three-step program was planned to utilize the same basic glider in conjunction with a larger, but undefined booster to achieve the orbital velocities necessary for complete re-entry tests. Studies were authorized to examine all possible candidates for this step of the program and to examine possible military equipment tests which could be carried on during the orbital phase of the flights.

8. The third step envisioned future use of the technology developed by the first two steps to develop a weapon system.

9. Increased glider weight and safety considerations resulted in a change to the Titan II booster in January 1961. This change in boosters provided a suborbital capability up to 22,000 ft/sec.

10. The MMSP (Manned Military Space Program) study (November 1961) concluded that the best alternative to the current Dyna-Soar program would be to adapt the glider and the Titan III booster together to achieve orbital flight. A ten shot program limited to single orbits was proposed in a development plan dated 16 November 1961, and submitted in conjunction with a White Paper which outlined Air Force objectives in space, and the essentiality of filling the potential critical gap which then existed in the development of controllable maneuvering re-entry vehicles with man integrated into the system. This program was approved in December 1961, and resulted in the initiation of the current orbital Dyna-Soar program.

11. During 1962, two multi-orbit flights were added within the 10 flight program by direction of Hq. USAF, and a change was later made to utilize the five segment Titan III booster as a result of a change of the standard booster from four to five solid segments.

Objectives

The objectives of the X-20A Program are as stated in Figure 2. The X-20A is a R&D program of a military test system to explore and demonstrate maneuverable re-entry of a piloted orbital space vehicle which will effect a controlled landing in a conventional manner at a selected landing site. The program will gather research data in the hypersonic flight regime, will test vehicle equipments, will investigate man-machine capabilities and represents a fundamental building block for the attainment of future military piloted space capabilities.

X-20 Flight Corridor

1. Figure 3 illustrates the wide range of altitude, velocity, and flight path control over which the X-20 has the capability of gathering research data.

2. The X-20 possesses the capability of dynamically flying at any point below the recovery ceiling, but above the structural limit. Controlled equilibrium flight is possible between max. C_L and the structural limit line. The initial flight shall be in the middle of the corridor for which the thermal margins are maximum, with later flights investigating the limit lines.

Research Regime

1. The widely different re-entry durations and heat flux rates (Figure 4) for the semi-ballistic capsule and the X-20 vehicle illustrate the difference in the re-entry heating problem for the two classes of vehicles. The large heat flux rates associated with capsule re-entry dictates ablative shields which work well when the re-entry duration is of the order of 10 minutes or less. The smaller heat flux rates of the X-20 vehicle actually result in a greater total heat flux because of the longer duration. However, this heat is radiated away into the atmosphere by the outer skin and only a very small percentage (2 to 5%) is absorbed into the structure.

2. The technology associated with high heat short duration re-entry is based on past ballistic missile programs and is well defined. However, little of this technology is applicable to lifting re-entry vehicles. The X-20 will provide the aerothermodynamic technology associated with slender re-entry vehicles capable of extensive maneuverability at hypersonic speeds.

3. Present day aircraft are exploring only a small region of the potential atmospheric flight regime. While the X-15 has greatly extended the investigation at the lowest end of this corridor, the greater portion remains unexplored. Arc facilities are presently available that duplicate the gas enthalpy and density corresponding to altitudes of about 200,000 feet and flight velocities of about 10,000 ft/sec. Partial simulation of some of the flight parameters is possible in conventional hypersonic wind tunnels and shock tubes. Complete simulation of the gas conditions in the entire corridor is possible in the near future only by actual flight. The X-20 is a program that will provide the vital data required to develop the necessary technology for hypersonic flight.

Re-entry Research

1. The X-20 configuration provides many features which will contribute to a number of technical areas (see Figure 5). One of its unique features is the radiation cooled metal structure which can evaluate the effects of the dissociated, chemically reacting gas flows on

heat transfer properties, materials, and oxidation resistant coatings. The ability to fly in a real gas, high enthalpy flow regime for extended time periods will add vital new data-technological anchor points - unobtainable from ground facilities.

2. The effectiveness of blended reaction and aerodynamic controls to control the vehicle over a wide range of angles of attack (0 to 50°), densities and Reynolds Numbers will provide extensive performance and stability data. The extent of laminar flow over the vehicle surface will provide data on transitional flows and boundary layer stability. Refractory heat shields and the ceramic nose cap on the X-20 are components which could have application to future radiation cooled systems. The refractory shields are easily replaceable permitting tests of alternate designs. The flight program will also provide a large amount of test data in the areas of flutter, aeroelasticity, acoustics and vibration.

3. The X-20A program will greatly expand our technology in the area of piloted flight operation (Figure 6) from the relatively short X-15 flights to global re-entry operations. The development of sophisticated re-entry management and thermal margin displays and adaptive control augmentation will enable the pilot to exercise full command of the guidance and control functions and obtain significant research on display effectiveness and pilot control capabilities. Valuable handling qualities criteria will be obtained throughout the hypersonic corridor and during approach and landing operation. From this technology, it will be possible to verify ground based flight simulation techniques and develop improved simulation programs.

4. Re-entry flight operations research will be provided by particular investigations in the following areas:

- Abort Techniques
- Energy management techniques
- Corridor exploration
- Re-entry communications through ionized flow
- Transition from reaction to aerodynamic controls

Design Criteria Impact

The X-20 flight research program will provide design criteria (Figure 7) which will be needed for the design of efficient future systems. Since these criteria are not now available, the X-20A has been conservatively designed. Turbulent flow has been used to determine heat transfer rates and an allowance of 20% has been added to account for roughness, small waves, and joints in the skin surface. Equilibrium flow has been assumed in the leading edge region which results in the highest heat transfer. Heat transfer on the wing surface may be reduced as much as 50% if extensive laminar flow is obtained in flight. Reductions in

leading edge heat transfer up to 50% may be realized if the dissociated flow is prevented from recombining at the wall by the use of a "non-catalytic" coating. If the effects of roughness prove to be less detrimental than expected, less blunt leading edges might be used which could increase the lift/drag ratio by 25% with a corresponding lateral range increase of 50%, as well as a payload increase of up to 6000 pounds.

Re-entry Maneuverability

1. Development of capabilities for re-entry maneuverability represents a basic need of the nation and one of the prime objectives of the X-20. The ballistic re-entry concept has now been demonstrated and has further emphasized the need of distance and direction control capabilities during re-entry. The Gemini project will provide a minimal improvement in these parameters. The X-20A project represents the prime national effort to provide a system with a high degree of re-entry maneuverability.

2. The payoffs of re-entry maneuverability are many. The principle payoff is the wide choice of landing sites available during re-entry from orbit, during emergencies, or in the event unforeseen circumstances require a change in plans during the re-entry and approach phases of the flight. Another key advantage is the elimination of extensive time in orbit, waiting for an opportunity to land at a selected site. The advantages of re-entry maneuverability are discussed in the following paragraphs.

Re-entry Maneuverability (Distance and Direction Control)

1. An illustration of the use of distance and direction control during re-entry is shown in Figure 8. After re-entering the atmosphere, a maneuverable re-entry vehicle such as the X-20A is capable of employing aerodynamic lift to vary its landing point. Normally, a landing to a pre-selected site as shown in the center of the ground landing area "footprint" would be planned with flight at a nominal glider re-entry attitude (angle of attack) and L/D. By flying at relatively low glider angles of attack, it is possible with the X-20 to extend range by approximately 3,000 nautical miles over the nominal re-entry path. By flying at a high angle of attack, it is possible to shorten the landing distance by approximately 3000 nautical miles, thus providing considerable flexibility for landing at an alternate site if necessary. It is also possible to bank the glider and perform a gradual turn in order to land at sites as much as 2,000 nautical miles displacement from one side of the orbital track.

2. In comparison, a ballistic re-entry vehicle is constrained to a landing essentially along its orbital track, controlled in range by the timing of the retro rocket firing.

X-20 Maneuver Flexibility

1. Choice of landing areas available as a result of the X-20 maneuver flexibility is shown in Figure 9 for a typical orbital flight, with the ground track limited to that of a single orbit for clarity. During the orbital flight, the pilot has the option of landing at any site within the broad band indicated on the chart, whereas a ballistic device could land only along the orbital track shown within this band.

2. Typical landing footprints are shown to illustrate the size of the landing area available to the pilot after a deorbit has been accomplished. Such a footprint is always potentially available to the pilot, with its center some 8000 miles ahead of his actual position, and may be visualized as moving along the orbital track ahead of the vehicle and becoming available after deorbit. The considerable flexibility such a capability provides should be of considerable importance to operational missions which cannot always be completely preplanned, as well as facilitating the accomplishment of preplanned test missions.

Test Vehicle Equipment and Explore Man's Function in Space and Re-entry

1. One of the objectives of the X-20A program is to test the vehicle's equipment and to explore the role of the pilot during orbit.

2. Initially, the more important portions of the flight testing effort will necessarily concentrate in the boost and re-entry areas until confidence and equipment reliability are fully established. Hence, the initial flights are being planned as single orbit flights. Even so, these flights provide a significant 43 minutes in orbit in which to accomplish additional testing of both man and machine. This testing extends to all of the vehicle subsystems as well.

3. Later, multi-orbit flights will serve to extend this testing time when a shift of emphasis to broader system testing becomes appropriate.

4. With all elements adequately instrumented for research and performance testing, the X-20A then provides the means for meeting its test objectives.

Mission

Now that the history and the basic program objectives have been covered, a discussion of our present program is in order. First, the:

Air Launch Program

The purpose of the air launch program is to demonstrate low supersonic, transonic and subsonic flight and landing capabilities, operation of subsystems, evaluate the integrated glider subsystems in flight prior to ground launch, and to conduct pilot training. One glider is scheduled to accomplish 20 air launches. The test program is planned

to fully explore the low speed portion of the flight corridor (70,000 feet altitude and up to speeds of approximately Mach 1.4). The glider is air launched at an altitude of approximately 50,000 feet and at a speed of approximately Mach 0.8. The acceleration rocket will be used on four power-air-launches to obtain low supersonic performance.

Ground Launch Program

1. The first phase of the ground launch program will be a two shot unmanned configuration utilizing developmental boosters. The next phase of the program consists of manned shots of both single and multi-orbit configurations. The nature of these flights is depicted in Figure 10.

2. Prime mission of the single orbit flights is exploration of the re-entry flight regime and demonstration of controlled maneuvering re-entry. These flights are launched from Cape Canaveral and directed along the Atlantic Missile Range, but tilted over to a flatter boost trajectory than is common for ballistic launches, so as to avoid violating the aerodynamic flight recovery ceiling. Boost burnout occurs approximately 1,000 miles down range where the vehicle is injected into an elliptical orbit with an apogee over South Africa (altitude approximately 100 nautical miles) and a perigee within the atmosphere (altitude approximately 60 nautical miles) northwest of Australia. At this point, advantage is taken of the X-20's aerodynamic controllability to prevent re-emergence and thus initiate re-entry. There follows a 7,000 nautical mile hypersonic re-entry approach through the Pacific Missile Range to Edwards AFB in California, where a horizontal landing is effected on the dry lake bed. Nominal re-entry time is 50 minutes. All critical action regions of the hypersonic boost and re-entry flight are covered with SHF range instrumentation and data collection facilities.

The Multi-Orbit Flights

These are very similar to the single orbit flights in the launch and re-entry areas, except that the launch azimuth is reduced to allow for precession of the ground track due to earth rotation during the orbital time period. The Titan III transtage is retained as part of the orbital vehicle to provide propulsion in orbit. Upon reaching the apogee, the transtage rocket motors are fired briefly to circularize the orbit. Thereafter, orbital flight proceeds for three orbits to a point over the Indian Ocean where the glider orientation is reversed and the transtage again fired briefly to effect deorbit. The glider orientation is turned for re-entry and thereafter, re-entry is executed as for the single orbit flights.

Configuration

1. The X-20 (Figure 11) consists of a 12,250 pound glider, of which 1000 pounds is payload, and a

5,750 pound transition section. The glider lower surface area is 345 square feet. The maximum length is 35.3 feet, the maximum height is 8.9 and the maximum width is 20.8 feet. The re-entry and landing weights of the glider are 12,000 and 11,700 pounds, respectively. The transition section is 15 feet in length and has a maximum diameter of 10 feet. It is divided into a 4.7 foot emergency propulsion section and 10.3 foot mating and multi-orbit equipment section.

2. The glider is shown mounted on the Air Force's Standard Space Launch Vehicle (Titan III). This booster will not be discussed here, but will be the subject of a separate paper in another section of this symposium.

3. Figure 12 shows the three compartments within the glider which are cooled. The pilot's compartment and the equipment compartment are both pressurized and cooled and the rear or secondary power compartment is provided with heat protection by means of a water wall.

4. The equipment compartment is designed to provide 75 cubic feet of available space and is shaped to easily accommodate a wide variety of payloads. It is designed for 1000 pounds payload and is currently utilized to house the test instrumentation subsystem and portions of the communications subsystem. It is provided with a 100% nitrogen atmosphere pressurized to 10 PSIA, and thus is well suited for the test of prototype electronic equipment which has not necessarily been made explosion-proof.

5. The secondary power compartment houses the hydrogen tank, oxygen tanks, auxiliary power units, and other equipment required to generate and distribute power. Hydrogen is stored supercritically in order to assure expulsion under weightless conditions, and is utilized as a heat sink as well as for fuel for the APU's.

Technical Developments

1. This first portion of the presentation was to acquaint the unfamiliar with the basic Dyna-Soar program. Now we will turn our attention to some of the technical areas of interest to discuss in more detail.

2. One of the first important decisions that was made in the Dyna-Soar program was to choose a hot primary structure approach instead of an active cooled aluminum sub-structure. These two concepts were evaluated in the June 1959 evaluation between The Boeing and Martin Companies. Although the cooled approach had many desirable characteristics including much better volumetric efficiency, there was considerable doubt at the time as to the feasibility of developing a heat shield system for the cooled structure which could effectively restrict heat shorts through attachments and hot boundary layer air leakage to the cooled structure. The feasibility and reliability of employing extensive coolant tubing throughout the glider was also considered a serious problem. The feasibility of the hot structures, however, had been demonstrated by Boeing during Phase I and the inherent reliability of a passive cooling system were important factors in the decision.

3. The state-of-the-art has advanced considerably in both areas since 1959, and follow-on applications of the Dyna-Soar technology may have either a hot or cool sub-structure depending on the overall system requirements.

X-20 Structure

1. The X-20 structure is one which is subjected to a severe re-entry environment. Temperature varies between 3650°F on the nose cap to a life environment for the pilot and equipment. The vehicle is subjected to dynamic pressures up to 860 psf during boost, sonic vibrations of 147 decibels, maneuver factors between - 1g and + 4g, and sink rates up to 8 fps during landing.

2. The system consists predominantly of trusses fabricated from materials selected to sustain the thermal environment (see Figure 13). The structure is designed to operate in an environment up to 1800°F. It is capable of withstanding at least four maximum condition re-entries. The conditions of major concern to the designer are thermal gradients across the structure and maximum structural temperatures. Accommodation of maximum temperatures is primarily a matter of material selection. For Dyna-Soar, Rene' 41 (nickel-base superalloy) has been selected. This alloy exhibits the best combination of availability, workability, and strength at elevated temperature. The accommodation of thermal gradients, which are as high as 500°F across a structural section, is an arrangement and concept problem. On Dyna-Soar, the basic approach is use of truss-type construction. Trusses were chosen because of their ability to reorient to the thermally induced shape without causing excessive secondary stresses. This principle is demonstrated in

Figure 14 for a single, three-sided truss. As member AB heats to a greater temperature than the other members, and hence, elongates more than the other members, the triangle changes shape by rotating about the joints.

3. This accommodation of gradients, which are nonlinear, is also best handled by trusses since the loads are carried in discrete members separated by air spaces as opposed to shear webs which have continuous shear material between the joints. Where thermal gradients are nonlinear, high shear stresses can be created by the large differences in thermal deformation across small distances. Where the thermal gradient is linear and the structural members are isolated, corrugated shear webs function satisfactorily.

4. The Dyna-Soar glider truss arrangement is as shown in Figure 13. Structural details of the various truss areas are predicated on the loading conditions, thermal environment, space available, manufacturing capabilities, and other peculiarities in the area in question. The fuselage main beams utilize rectangular, round and square members, pinned and fixed joints, joint fittings made from forgings and bar stock, and both standard and special fasteners.

5. The exterior surface consists of Rene' 41 corrugation-stiffened panels, either uninsulated or insulated, depending on the location of the panel on the glider. Insulated panels are used in all areas where the surface temperature exceeds 2000°F and includes the entire lower surface of the glider, the outboard surface of the fin and rudder, and a small portion of the forward sides of the body aft of the nose cap. Uninsulated panels are used on the upper surface of the wing, body, and elevon, and on the inboard surface of the fin and rudder. The configuration, sizes, and materials selected for these panels resulted from design considerations that include thermal, flutter, sonic, air pressure, and shear loads, fabricability, and maintainability. The insulated panel, as shown in Figure 15 consists of a Rene' 41 corrugated panel with TZM molybdenum or D-36 columbium alloy heatshields attached with stand-off clips, and Q-felt insulation sandwiched between the two.

6. The D-36 and TZM heatshields assemblies are protected against oxidation by a disilicide coating. Individual parts are precoated prior to riveting, and the completed riveted assembly is recoated to protect the riveted area and the faying surface between the clip flange and the shield.

7. The leading edges are defined as all edges that face into the airstream. Altogether, the glider has approximately 140 running feet of leading edge construction and about 140 square feet of exposed area. Average transverse spans are on the order of 8 inches. The edge radii vary from a maximum of 7.5 inches at the nose cap to

a minimum of 2.06 inches on the inboard side of the elevon. The radii are jointed by faired and tapered sections. These sections were selected to be consistent with a maximum design short-time temperature of 2900°F and an equilibrium temperature of 2825°F. Nonmetallic leading-edge specimens have been built of graphite, ceramic, and composites. Metallic specimens have been built of forged molybdenum and sheet-metal tantalum, columbium and molybdenum. Of the metallic specimens, only the molybdenum and columbium sheet-metal have reached detail design status. The effort spent on graphite, ceramic and composite designs did not result in arrangements which were competitive with sheet-metal designs in terms of joint smoothness, suitability for sealing, and applicability to geometry. In addition, both the nonmetallic and the forged refractory specimens appear to be heavier, as shown in Figure 16, TZM molybdenum alloy sheet metal will be used for most of the leading edges, and D-36 columbium alloy for areas where temperatures do not exceed 2450°F.

Nose Cap

1. The nose cap of the Dyna-Soar glider is required to sustain very high temperatures over a much longer period than that of a ballistic re-entry vehicle. Because of this relatively long period at temperature and the desirability of maintaining aerodynamic shape, the development effort has centered around heat-sustaining materials. Two structural configurations of different material combinations are being developed, one by the Chance-Vought Corporation and the other by The Boeing Company. This dual effort has been considered necessary because this piece of hardware is so critical to the successful flight of the vehicle.

2. The Chance-Vought concept utilizes a structural shell of National Carbon RT-0029 graphite protected by a silicon carbide coating. The shell is further protected by an outer cover of zirconia tile retained by zirconia pins in such a manner that the major thermal stresses in the protective cover are relieved by mechanical motion between the zirconia tiles. This cap is illustrated in Figure 17.

3. The Boeing nose-cap effort is directed toward developing a monolithic shell of zirconia reinforced with platinum wire. The forward face of the shell is grooved to relieve the thermal stress on the surface. This surface grooving is accomplished by inserting a paper honeycomb configuration into the mold, pressing, and burning the paper out during the firing operation.

4. The mounting of the nose-cap shell to the glider structure has been a joint effort of the two companies. The mounting is so arranged that the attachment of the two nose-cap shells to the support ring differs in only minor details.

Landing Gear

The Dyna-Soar landing gear configuration is an all-skid, tricycle arrangement utilizing

yielding metal (energy strap) shock absorbers. Each of the two main and the single nose gear are composed of three major elements: a skid, a pivoting support strut, and an energy strap (see Figure 18). The main skids are wire brush types to generate a high coefficient of friction, and the nose skid is hard coated to provide a low coefficient of friction. The support struts are assembled from machined Rene' 41 forgings and are designed to pivot aft under load. This pivoting motion causes the energy straps to yield and absorb the landing impact energy.

2. All landing-gear doors are operated mechanically by the extension motion of the gear. The gear itself is extended at 275 knots by a high-pressure pneumatic system which moves the gear to an external position where aerodynamics and gravity complete the extension cycle. The major portion of this pneumatic system, as well as the gear itself, will experience a high-temperature soak in the 1600°F to 1800°F range.

3. A test program is presently being conducted at Holloman Air Force Base, Track Test Division on both the nose and main skids. Asphalt, concrete, and lakebed surfaces have been laid down in the sled track trough so that 5000-foot slide-outs can be made to verify the coefficient of friction, wear, and bump capability on each type surface. A special rocket sled permits the glider to start the slide-outs at the maximum glider landing velocity of 220 knots and to coast to a full stop in 5000 feet.

Integrated Power and Cooling

1. The operation of Glider Subsystems results in a 34HP Design Requirement for Secondary Power Generation. This total can be broken down into the primary electrical load, such as guidance, communications, flight controls, TIS, cockpit displays and lights that account for 6.9 KVA, and secondary electrical loads associated with environmental control equipments and cryogenics supply requiring 3.8 KVA. The remainder can be attributed to the 8.5 GPM, 3000 psi hydraulic load. Considering the duty cycle of the subsystems, the total energy demand could vary from approximately 12 to 80 horsepower-hours.

2. Figure 19 shows the secondary power generation spectrum derived from initial and projected program requirements superimposed over load regimes within which particular energy conversion units operate most effectively. Note that the chemical dynamic APU is shown as the most suitable prime mover for the X-20A application. A cryogenic bipropellant, hydrogen and oxygen, was selected on the basis of results comparing many propellant combinations. The two most promising schemes are shown in Figure 20. Here hydrazine weight requirements are approximately 2 1/4 times that of the hydrogen-oxygen unit. On this basis, the hydrogen-oxygen bipropellant combination was selected. The operation of electrical and hydraulic equipment, combined with the effects of aerodynamic heating

results in a total heat load of approximately 200,000 BTU's. Two approaches were taken to dissipate this energy:

a. Equipment cooling would be accomplished by the environmental control system within the framework of the 3 compartments.

b. The major portion of aerodynamically generated heat passing through the outer surface would be removed by a system mounted to the outer face of the compartment walls.

3. The selection of this propellant combination resulted in the use of hydrogen as heat sink for equipment cooling, since a comparison with water (see Figure 21) indicates a considerable weight saving and a wide temperature range to accommodate the cooling of equipments having different operating temperatures. The effect of adding ammonia to water results in a wider temperature range at low altitude.

4. When the implications of Figures 20 and 21 are resolved in terms of hardware and subsystem requirements, the impact of specific concepts can be evaluated. Two of the most promising approaches were selected for comparison: an integrated hydrogen-oxygen system utilizing hydrogen-oxygen for power generation and hydrogen for cooling, and hydrazine power generation units combined with a water-ammonia cooling system.

5. The weight advantages of an integrated hydrogen-oxygen system are shown in Figure 22. Although a comparison of re-entry weights shows only a small savings for the cryogenic systems, the growth capability for multi-orbit missions is significant.

6. As a result of this study, the integrated cryogenic system was selected and a hydrogen-oxygen reaction control system incorporated by including propellant for attitude control in tankage common to both systems. This additional feature was short-lived since analog flight simulator studies indicated hydrogen requirements for attitude control that exceeded the capability of the hydrogen storage system and tank pressure controls.

7. A schematic of the integrated system is shown in Figure 23. Hydrogen, transported directly from the permanent vacuum insulated storage vessel is utilized in the primary heat exchanger to absorb heat from the pressurized compartments and a number of equipments. A secondary loop, employing an aqueous solution of ethylene glycol and water as the working fluid, transports heat from the compartment atmosphere, hydraulic oil, AFU gearbox and controls, and the alternator to the primary coolant. After passing through the primary heat exchanger, hydrogen is combined with oxygen in the combustion chamber of the AFU to drive the hydraulic pump and alternator through a 3-stage re-entry turbine. Both cryogens are stored above the critical pressure by supplying heat to the fluids to maintain a constant

expulsion pressure and are stirred to prevent stratification. When the hydrogen requirement for cooling exceed that for power generation, the excess is exhausted overboard and, if the reverse is the case, the additional hydrogen is supplied to the prime mover via the heat exchanger by-pass line.

8. Several problems encountered in the development of the integrated system are mentioned in Table 1. Satisfactory design approaches have been adapted to solve most, and in many cases, operation of revised development hardware has been demonstrated.

Water Wall

1. Thermal protection for the X-20 during re-entry flight is provided by a radiation-cooled outer surface employing coated refractory metals or René 41. Since this method is not totally effective in preventing the influx of aerodynamically generated heat to the vehicle interior, additional protection must be included to absorb this energy to minimize the effect on the internal environmental control system.

2. Two possible choices are available: insulating the compartments with a sufficient quantity of material to prevent heat from reaching the interior, or combining insulation with a cooling system. From a weight standpoint, Figure 24 shows that when re-entry times and average surface temperatures are considered, the concept of insulation and cooling results in the lightest weight.

3. After considering many possible insulations, a light-weight fibrous quartz material, Q-felt, was selected as one of the most thermally effective material for application to the X-20.

4. The selection of a cooling system considered both active and passive types. The passive system was selected because it offered more inherent reliability, was of simple construction, and was readily adapted to a hot structural concept that has few heat shorts to the cooled compartments. Also, the weight of the passive system was less.

5. A schematic of the water-wall system is shown in Figure 25. The insulation is covered with a 2 mil metal foil, that acts as a retention sheet. This outer surface is supported by perforated discs to distribute the load into the cover and to provide outlets for outgassing of air from the insulation during boost.

6. The cooling system dissipates the heat transferred from the hot outer surfaces by utilizing the latent heat of vaporization of an expendable coolant. It is an open ended type consisting of an assembly of polyurethane foam sections contained by aluminized mylar laminated faces. A gel, composed primarily of water, is retained within the cellular foam structure from the time of system fabrication until evaporated

during flight of the X-20 through the earth's atmosphere.

7. Since the coolant is not circulated, successful operation depends upon the ability of the system to contain a sufficient supply at desired locations. The coolant supply will be installed during fabrication of individual panels and remains in tact until the time of use.

8. Problems encountered in the development of this system included difficulties in meeting life requirements and developing "field" filling procedures. As a result, it was decided to factory fill the panels and replace them after each flight.

Flight Control

1. Now I would like to turn our attention to the flight control subsystem. The X-20 flight control system utilizes the self-adaptive control principle as the primary technique for stability and control of the glider and the glider plus transition configurations. Early self-adaptive flight control work was accomplished by the Flight Control Laboratory at ASD, Wright-Patterson Air Force Base, Ohio. This work was followed by the application of this development in the X-15 flight control subsystem. The X-15 self-adaptive flight control program is being monitored for application of this experience to the X-20 flight control development.

2. Figure No. 26 illustrates the flight control subsystem as planned for the X-20 vehicle. In the manual mode of operation, the flight control subsystem electronics utilizes signals derived from the pilot's sidestick controls and rudder pedals. These controls are provided with dual position transducers to provide electrical signals for the flight control subsystem electronics. Electrical signals are sent to the servo valves of the aerodynamic and thrust vector controls for activation of these portions of the system. The thrust vector controls are used only in the event of an abort. Electrical signals are also sent to the reaction control system for activation of the reaction control solenoid valves. Dual and triple redundancy is employed throughout the entire flight control subsystem. Switching logic is employed with monitors for fail safe operation in event of a malfunction of the dual redundant electronics. These monitors provide automatic switching to switch out any malfunctioning channel of operation. Automatic operation is provided by signals derived from the primary guidance system. These signals command the correct pitch attitude or angle of attack, bank angle and zero sideslip.

3. A variety of control, stabilization and gain techniques are used in the flight control system. The automatic mode utilizes the self-adaptive gain control principle. Two manual modes are provided. The manual augmented mode utilizes the self-adaptive gain control principle. Additionally, provision is made for pilot selection of an appropriate gain. The manual direct mode provides for pilot selectable gain adjustment of

the controlling element gain, i.e., aerodynamic control and thrust vector controls. The manual direct mode provides through threshold switches the direct electrical control of the reaction control solenoid valves.

4. Flight control subsystem electronics development, analysis and design is essentially complete. The functional requirements and performance requirements have been established. Production flight control subsystem electronics mechanization diagrams have been released. The first production prototype unit has been fabricated and delivered to The Boeing Company for installation in the guidance and control development model. This equipment is presently being installed in a mock-up wherein all interfacing electronic equipment is also installed. Tests during this phase will determine equipment compatibility. Qualification testing of the production flight control electronics is planned to start approximately September 1963.

5. It is apparent to most of you that the X-20 flight control subsystem is a very sophisticated development. Now let us turn our attention to the problems it must solve and why it must be complex. In Figure 27, the stability and control problems are shown as a function of the mission. During the boost phase of the mission, the X-20 stability and control problem is primarily that of the potential abort configuration. During the boost phase, the aerodynamic controls are locked by hydraulic means to fixed positions most favorable to the worst abort conditions. The fact that the center of gravity is behind the aerodynamic center of pressure for the abort configuration imposes exacting requirements in the flight control subsystem design. The self-adaptive flight control system must therefore have suitable initial condition gains and be capable of adapting to the optimum gain rather quickly. The static instability is sufficiently great that stability augmentation must be relied upon. It is questionable whether the pilot could provide the necessary damping in the event of stability augmentation failure in one or more axes. The orbital phase of operation provides problems in the area of maintaining the desired attitude accuracy in the automatic mode in view of fuel utilization restrictions. Consequently, trade-offs are being made involving the attitude accuracy in the automatic mode. Present indications are that fuel utilization will be satisfactory in the manual modes of operation. During re-entry both reaction and aerodynamic controls are utilized for stability and control. The use of reaction controls is discontinued when the aerodynamic pressure increases to a point where the aerodynamic controls provide the majority of control effectiveness. During this phase of operation very low load factor limits are observed in order to preclude exceeding the glider temperature limits. Considerable work has been done in this area to define acceptable handling qualities requirements at these low dynamic pressures. The hypersonic glide regime provides stability and control problems in terms of providing satisfactory roll

control and high angles of attack. The fundamental nature of the problem is that aerodynamic surfaces produce moments about the body axis where it is required that a moment be produced about the roll stability axis. This problem is further complicated by the fact that the elevons produce relatively strong yawing moments. Several solutions have been found to this problem including the cross feed of roll commands into the rudder surfaces. As shown in Figure 27, the close proximity of structural frequencies, self-adaptive limit cycle frequencies, the aerodynamic short period and handling quality requirement frequencies have required careful attention to detail. The aero-servo-elastic coupling problem has resulted in the design of structural coupling filters in the flight control subsystem electronics to provide a very high attenuation of any structural feedback signals to the gyros. Additionally, careful attention has been given to the design of the self-adaptive limit cycle circuitry to preclude the possibility of structural mode oscillations reducing the self-adaptive gain unnecessarily. Attention has also been given to gust and pilot input frequencies in order to preclude undesirable changes of self-adaptive gain due to these inputs. The basic fundamentals of the self-adaptive technique utilized in the X-20 flight control system are reasonably simple. The concept that is employed involves use of a high gain control loop preceded by a model or filter designed to provide the characteristics of the desired handling qualities. The assumption being that if the loop gain is sufficiently high, the outer loop performance will conform to that defined by the model. Gain is maintained by the self-adaptive gain computer. This device utilizes signals obtained from the moment producing control element, for example, the elevator in the pitch axis. The gain computer maintains the necessary gain to keep the pitch rate innerloop on the verge of an unstable oscillation. This is accomplished by virtue of measurement of the elevator deflection. The deflection signals are passed through logic filters, through a rectifier to obtain the absolute value of motion of the surface, then through appropriate limiters and shaping circuits, and finally to the variable gain circuitry. The logic filters are designed for frequencies of approximately four tenths of a cycle per second for the up gain logic and four cycles per second for the down gain logic. Operationally, this will mean that any oscillatory energy of the elevator in the vicinity of four tenths of a cycle per second will result in increasing the gain of the flight control system. Similarly, elevator activity in the vicinity of four cycles per second will result in a decrease of the flight control system gain. Nominally, a very small amplitude oscillation of the control surface will exist during flight with a frequency of approximately one to two cycles per second.

Guidance

1. The Dyna-Soar program, at its inception presented the first requirement for a full navigation and guidance capability from launch

thru re-entry and thence to landing of a manned space vehicle. The configuration requirements were established about a primary system that would provide the greatest reliability of performance at a minimum cost thru employment of proven system elements to the greatest extent feasible. A reliable simple backup capability was to be provided to enable safe re-entry in the event of failure of the primary system.

2. Initially, a guidance configuration was established during the boost portion of flight by providing guidance and control from the glider inertial guidance subsystem. Backup was to be provided by an available Radio Guidance System in the event of an IGS failure. Upon reorientation of the program to the Titan III Space Launch Vehicle, the boost guidance configuration was revised to control this portion of the trajectory from the available booster Inertial Guidance System. Currently a booster guidance backup capability has not been established. However, simulation investigations have indicated the feasibility of the pilot to control the booster, with the aid of proper instrument displays, thru the flight control system to the point of injection within acceptable limits. Studies are currently underway to determine manner and cost associated with mechanization of such a capability.

3. During orbit and re-entry, navigation and guidance capability will be provided by the glider's Inertial Guidance System. The elements of this system are shown in Figure 29. This system provided by Minneapolis-Honeywell consists of three major elements: the inertial platform, which is a further refinement of a platform initially developed for the NASA (Centaur Program) a combined general purpose (g.p.) and digital differential analyzer computer popularly known as Verden digital computer employed on the GAM-77 missile; for the X-20 application, its g.p. computation capacity will be increased about four fold; and a coupler electronics unit which houses the various circuit elements of the system. As displayed in Figure 30, the IGS provides an attitude reference for the automatic flight control system as well as the necessary steering commands to automatically control the glider on its path. In addition, the flight instruments are also provided their sensing inputs from the IGS to facilitate pilot manual control of the glider. The key instrument receiving these inputs (Figure 31) is an instrument known as an Energy Management Display, which thru overlays calibrated for speed and landing destination controlled from the IGS, provides the pilot with a display showing bank angle and angle of attack relationship with his footprint capability of attaining the desired landing area. The guidance then accomplished during re-entry is maintaining the desired angle of attack and bank angle so that the vehicle's kinetic and potential energy is dissipated in such a manner that the structural and thermal limits are not exceeded and the vehicle arrives at the desired high key point for landing with the proper energy for a landing.

4. The landing phase of flight will commence about 100 miles from Edwards with approximately a 4000 ft/sec velocity and an altitude of 130,000 ft. A visual approach, let down and landing will follow.

5. The glider will employ an Emergency Re-entry Subsystem as a backup to the Inertial Guidance Subsystem. This system will consist of an all attitude reference which will operate the pilots attitude indicator. In the event of an IGS failure, this reference will enable the pilot to maintain a safe attitude during the critical portion of re-entry.

Communications and Tracking

1. A reliable communications net is essential for the early flights to control and gather data from the first exploratory flights. At that time, there will be urgent needs for flight safety, design verification and/or failure analysis data coverage. This coverage entails overflying a chain of interconnected surface communications, tracking, and data collection range stations positioned along the (Atlantic and Pacific) orbital and re-entry track (see Figure 32). Many of these stations already exist in the Atlantic and Pacific Missile Ranges and the NASA Mercury net. A major problem presented itself in utilizing these range stations and existing equipment. Experience with the preceding ballistic missile and orbiting satellite programs had demonstrated that a vehicle re-entering the atmosphere at hypersonic speeds becomes enveloped with a thermally ionized plasma-sheath configured to the flow field around the vehicle, as illustrated in Figure 33. This plasma sheath effectively acts as an electrical conductor, thus forming a highly reflective and absorptive media about the vehicle, which serves to obstruct and black-out conventional tracking radar and radio communications to and from the vehicle. A black-out occurs in the region of the re-entry hypersonic flight regime where Dyna-Soar is required to carry out its prime flight-research mission. To solve this problem, advantage was taken of concurrent research on the interaction of electromagnetic radiations and plasma fields and those findings extended. This research had demonstrated a distinct frequency sensitive behavior for the plasma sheath. In fact, it indicated the existence of a window in the frequency spectrum above the expected plasma resonant frequencies and below the onset of absorption by water vapor, oxygen and other constituents of the atmosphere (see Figure 34). For Dyna-Soar lifting re-entry flight conditions, a choice of communications frequencies in the SHF band in the vicinity of 10 KMC to 15 KMC was indicated.

2. Other approaches, such as seeding or cooling of the plasma adjacent to the affected antennas and using special propagation modes established by magnetic fields, appeared possible. Another possibility was the use of a thin sharp spike antenna which would not produce a dense

shock wave and associated plasma in the vicinity of the radiating elements. These latter approaches, while attractive, were still in early stages of development and have not been adequately proven for flights similar to those planned for Dyna-Soar. The bulk of the available research data suggested that greatest confidence would result from pursuing the frequency-choice route, which was done.

3. The configuration adopted is shown on Figure 35. The figure illustrates the configuration adopted for both the airborne and ground (prime) communications subsystems to be used in the launch and re-entry areas. The SHF ground-to-air link frequency selected was in the region of 10.4 KMC. The air-to-ground link frequency selected was at 13.5 KMC to take best advantage of available microwave equipment components. Not specifically identified in Figure 35, but included in the system are a pair of similar UHF voice communications links and a C-band transponder to be compatible with the range station equipments existing along the established missile and orbiting-satellite ranges in the non-re-entry regions.

4. New SHF equipment for both glider and surface station adaptation is being developed and procured. The surface station adaptation equipment is self-tracking in both azimuth and elevation. Inclusion of a tone-ranging circuit also provides a measurement of slant range; thus providing simultaneously for both the needed radio communications and vehicle position tracking in the otherwise blacked-out re-entry region of the mission. In addition, a higher-powered (5 watt peak) UHF rescue beacon/transceiver is being provided to yield greater homing range capability for pilot rescue.

Test Instrumentation Subsystem

The Test Instrumentation Subsystem of the X-20A program encompasses all areas of airborne data collection, signal conditioning, multiplexing, translating and recording of data in the glider. Also included is the necessary ground based equipments for demultiplexing, detranslating recording/reproducing, formatting and data calibration up to the point of providing calibrated data tapes to the various data users for the required analysis. The X-20A Program Office is responsible for the overall management of the test instrumentation area of the program. However since the X-20 is a joint effort between the USAF and NASA, a team of instrumentation specialists was established to provide the Program Office technical support and recommendations in the area of test instrumentation. This team is composed of members of the USAF and NASA and is chaired by a NASA member.

Design Considerations

1. The basic design considerations for the TIS subsystem consisted of the number and type of sensors to be employed and bandwidth impairment

of transmission range. A list of measurements was established that included approximately 1000 parameters. Flight safety and failure analysis type data received top priority with design validation and basic research data following a close second. The majority of the parameters to be measured are quasi-static or have a very slow rate-of-change, thus lending themselves to narrow-band digital time division multiplex. However, not all the measurements fall into this category. Required are a number of continuous time-history parameters best cared for with analog (frequency division) multiplex, at the expense of transmission range, i.e., there are 3 parameters with frequencies from 50 cycles per second to 10,000 cycles per second (acoustics data). Eleven parameters with frequencies from dc to 2000 cycles per second (vibration), and 12 parameters of dc to 1000 cycles per second (flutter). Thus, it is seen that a combined digital/analog system was needed to care for both classes of data. A further limitation was imposed on the instrumentation subsystem, that of weight. In the early design phase of the program, a payload allocation was made based on X-15 experience. This allocation was 1000 lbs. In the research version of the X-20, the 1000 lbs. is allocated to the test instrumentation subsystem. Approximately half of this weight allocation is used for wiring, tubing, racks and environmental control. A majority of the parameters to be measured are located in a very high temperature environment requiring special type wire and insulation. Also, tubing is used from pressure ports on the X-20 surfaces to an environmentally controlled compartment where the pressure sensors are installed. This is necessitated by the present state-of-the-art pressure sensors.

2. Instrumentation configurations depicting the locations of the various sensors have been established, (see Figure 36). The primary change in the instrumentation configuration from flight is the type and location of external surface sensors. One configuration emphasizes external surface pressures while another configuration emphasizes external surface temperature measurements. In all configurations the internal subsystems measurements remain the same. This configuration change approach is used to obtain the numerous research measurements required to meet program objectives within the number of flights and weight limitations imposed on the test instrumentation subsystem.

3. On-board recording of the data is required on the X-20 so the validation and research data can be obtained throughout the flight regime of the glider. Telemetry is being used in areas of the flight regime where engineering analysis indicates the glider will be subjected to the maximum environmental hazard, such as high temperatures, aerodynamic loads, potential flutter etc. These areas present the higher probability of structural failure and are instrumented to obtain data for failure analysis in the event that the mission is not successful.

Telemetry Equipment Considered

1. A considerable number of different types and/or combinations of telemetry equipment were considered for use on the X-20A program. A basic philosophy established early in the program was that the test instrumentation subsystem design was to use "off-the-shelf" type techniques. We did not want to run a research program while we were still testing the basic means of obtaining data. Basically, we have held to this philosophy in the design of the system. However, there are some cases where slight modifications had to be made to off-the-shelf techniques to make them suitable for our requirements. As an example, the use of a video-recorder on-board the glider. To provide the bandwidth and channel capability, design effort was required to achieve tighter phase delay compensation and the reduction of time-base instabilities induced by flutter, wow and tape skew. This design effort is underway and tests on engineering models indicate the system will operate satisfactorily.

2. FM/FM and PDM/FM/FM telemetry subsystems were considered. Due to the large number of measurements, the bandwidth of high frequency response parameters require excessive transmitter power and exceeded the weight limitations allowed and was removed from further consideration.

3. From time to time throughout the existence of the program, an all PCM/FM telemetry subsystem seemed attractive. In the early phases of the program, the PCM/FM system was considered to be beyond the state-of-the-art due to the high bit rate required. Also, there was reluctance on the part of some data users to accept the low-rate sampled data as sufficient for analysis purposes. At this early point in the program, a decision was made to incorporate the present system, a hybrid PCM/FM/FM telemetry equipment, into the X-20A. The hybrid PCM/FM/FM system uses a frequency translation technique. The high frequency analogue parameters are fed into standard telemetry voltage controlled oscillators. The outputs of the oscillators are then grouped according to frequency and translated to a higher frequency.

4. There are 42 of these high frequency response channels that are grouped and translated to six different frequency bands. These six frequency bands and the PCM (144,000 bits per sec) are then mixed in three combinations.

5. All data measured on the glider is combined into the largest bandwidth for on-board recording only (see Figure 37). Two abbreviated combinations are separated out for sequential telemetering to the surface data collection stations in the terminal and mid-course regions of the mission. In the terminal areas, the acoustic noise measurements are omitted and the vibration data analyzed on-board into simpler power spectral density data for transmission to the ground along with the remainder of the complex and called the wide band case (see Figure 38). In

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the mid-course regions (beyond the two terminal flutter inducing regions) all the flutter data is omitted and a single slant-range measuring a signal channel substituted to form the third, or narrow-band combination. This conserves bandwidth and extends data transmission to a maximum in the regions where range is a prime consideration.

Summary and Conclusions

1. The major program milestones are shown in Figure 39. Ninety percent drawing release is scheduled for September 1963. The first air launch is scheduled for January 1965; the initial unmanned ground launch in November 1965; the first manned ground launch in May 1966; and the final flight in September 1967.

2. Significant progress has been made on the program to date. The development effort is essentially completed. Production drawings are being released to the manufacturing shops and the qualification test program has begun. The problems to come should not be in the development or state-of-the-art area, but rather in the hardware and integration of the various system elements. Though we have not yet reached the flight test part of the program, a significant step forward has been made in the specific areas which have been covered as well as in innumerable other technological fields. It is our view that the lifting re-entry technology, which is being developed by the X-20, is filling an important gap in this country's overall research and development effort which will in turn provide a sound technological base for the design and development of future systems in the National Space Program.

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PROBLEM AREA	CONCLUSION	ACTION
INTEGRATION OF H ₂ -O ₂ REACTION CONTROL	INTEGRATION NOT PRACTICAL	INDEPENDENT H ₂ -O ₂ SYSTEM BEING PROCURED
POSITIVE EXPULSION OF CRYOGENS	SOLUTION NOT WITHIN X-20 TIME PERIOD	SUPERCRITICAL STORAGE SELECTED (EXCEPT FOR N ₂) WITH HEAT ADDITION
STRATIFICATION OF CRYOGEN IN STORAGE TANK	POSITIVE APPROACH TO BE TAKEN	FORCE CIRCULATION ADOPTED USING CENTRIFICAL BLOWERS
HEAT LEAK TO CRYOGENIC SYSTEMS	USE EFFECTIVE INSULATION	PERMANENT VACUUM JACKETED TANK AND LINES SELECTED AND DEVELOPMENT COMPLETED
GLYCOL-WATER FREEZING IN PRIMARY HEAT EXCHANGER	DEVELOP A DESIGN TO ELIMINATE THE POSSIBILITY OF FREEZING	RECIRCULATION OF WARMED H ₂ TO HEAT EXCHANGE INLET SELECTED AND DEMONSTRATED
APU HIGH SPEED ASSEMBLY FAILURES	CHANGE NATURAL FREQ. OF TURBINE BLADES	REVISED DEVELOPMENT PLAN SUCCESSFUL TO DATE
ZERO G LUBRICATION OF APU GEAR BOX	SELECT POSITIVE SOLUTION	FORCED FEED LUBRICATION SELECTED & DEVELOPMENT COMPLETE
DEVELOPMENT OF 4 LIGHT WEIGHT H ₂ TANK	THIN OUTER SHELL NOT PRACTICAL FOR THIS APPLICATION	STRUCTURAL OUTER SHELL IN DESIGN

Table I. Major Problem Areas

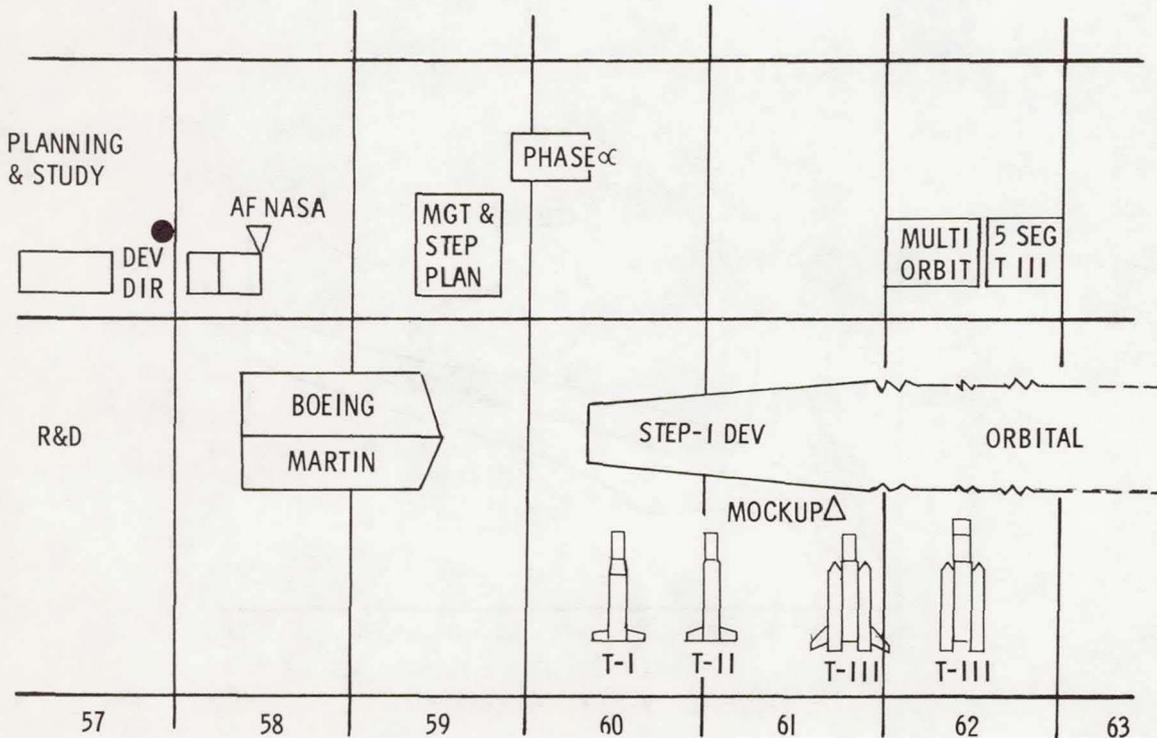


Figure I. Dyna-Soar Background

- PROVIDE PILOTED, MANEUVERABLE GLIDERS AND ASSOCIATED SUPPORT EQUIPMENT FOR THE CONDUCT OF FLIGHT TESTING IN THE HYPERSONIC AND ORBITAL FLIGHT REGIME TO INCLUDE:

GATHERING OF RESEARCH DATA TO SOLVE DESIGN PROBLEMS OF CONTROLLED, LIFTING RE-ENTRY FROM ORBITAL FLIGHT

DEMONSTRATE PILOTED, MANEUVERING RE-ENTRY AND EFFECT A CONVENTIONAL LANDING AT A PRESELECTED LANDING SITE

THE TESTING OF VEHICLE EQUIPMENTS AND EXPLORATION OF MILITARY MAN'S FUNCTIONS IN SPACE

FOLLOWING SUCCESSFUL ORBITAL DEMONSTRATION, TO PROVIDE THE CAPABILITY FOR QUICK EXPLOITATION OF TECHNOLOGICAL ADVANCES THROUGH FUTURE TESTS

- EXPLORE THE POTENTIAL OF MAN TO ACCOMPLISH MILITARY FUNCTIONS IN SPACE

Figure 2. Program Objectives

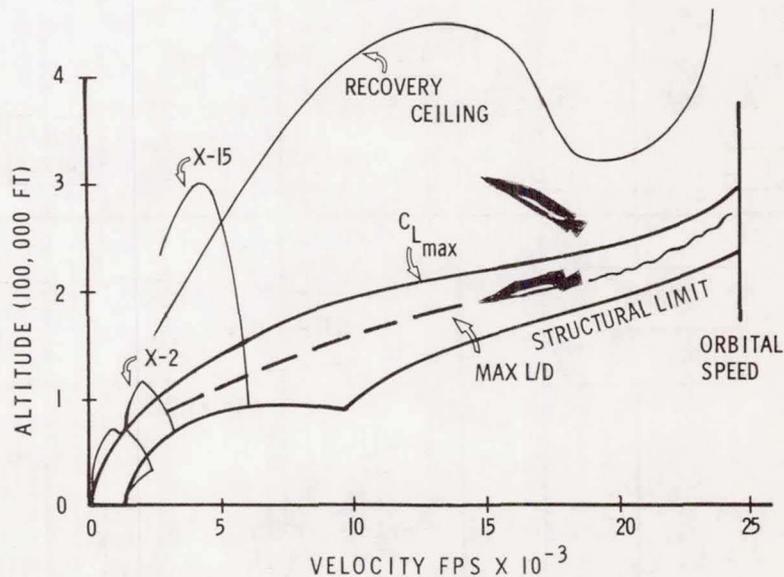


Figure 3. X-20 Flight Corridor

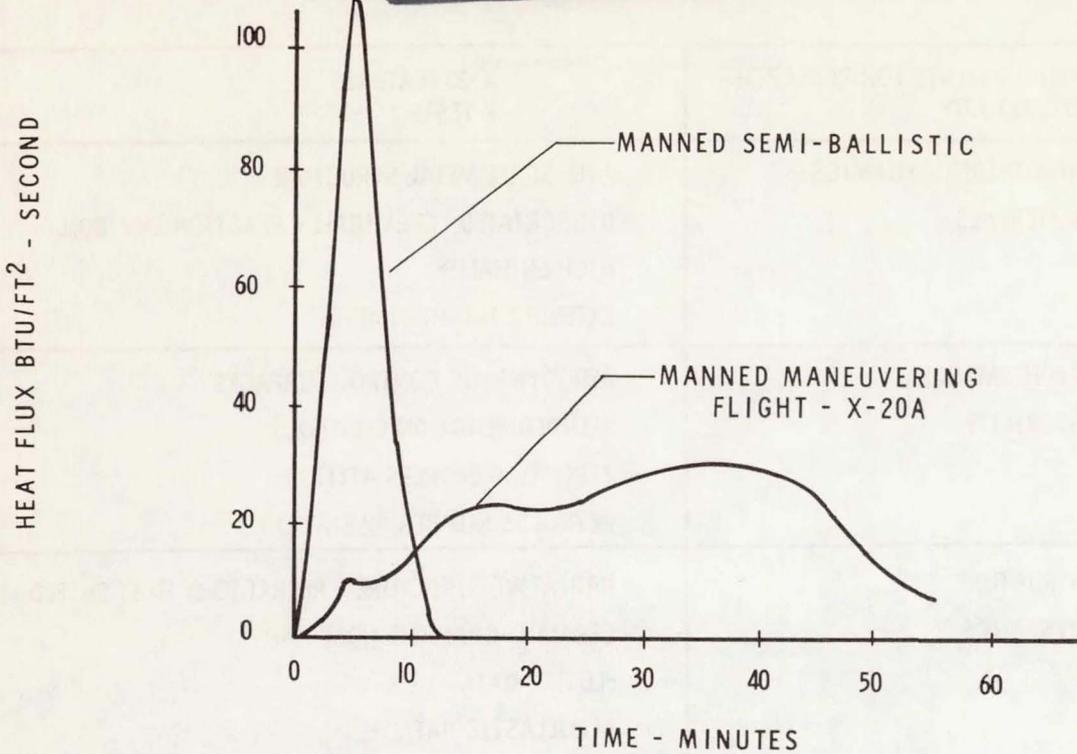


Figure 4. Research Regime

REQUIREMENTS FOR RESEARCH TECHNOLOGY	X-20 FEATURES & TESTS
PILOT CONTROL PILOT DISPLAYS FLIGHT SIMULATION HANDLING QUALITIES	PILOT INTEGRATED IN CONTROL-GUIDANCE LOOP ADAPTIVE CONTROL AUGMENTATION ENERGY MANAGEMENT AND DISPLAYS HYPERSONIC FLIGHT ENVIRONMENT RE-ENTRY APPROACH & LANDING TECHNIQUES
FLIGHT OPERATIONS	ABORT TECHNIQUES GUIDANCE EVALUATION DURING RE-ENTRY CORRIDOR EXPLORATION COMMUNICATIONS THROUGH ION SHEATH INSTRUMENTATION RESEARCH

Figure 5. Re-Entry Research

REQUIREMENTS FOR RESEARCH TECHNOLOGY	X-20 FEATURES & TESTS
AEROTHERMODYNAMICS MATERIALS	FULL SCALE METAL STRUCTURE DISSOCIATED, CHEMICALLY REACTION ENVIRON. HIGH ENTHALPY EXTENDED FLIGHT TIME
PERFORMANCE STABILITY	AERODYNAMIC CONTROL SURFACES BLENDED REACTION CONTROLS ZERO TO 50 DEGREES ATTITUDE REYNOLDS NUMBER VARIATION
STRUCTURE DYNAMICS	RADIATIVE STRUCTURE - REFRACTORY HEAT SHIELD CERAMIC-GRAPHITE NOSE CAP FLUTTER DATA AEROELASTIC DATA ACOUSTICS AND VIBRATION DATA

Figure 6. Re-Entry Research

RESEARCH AREA	X-20 DESIGN REQUIREMENTS	FUTURE POSSIBILITY
HEAT TRANSFER	TURBULENT FLOW ROUGHNESS MARGIN - 20% EQUILIBRIUM FLOW	LAMINAR FLOW — 50% REDUCTION IN HEAT TRANS. SMOOTH SURFACES — UP TO 6000# INCREASE IN RE-ENTRY PAYLOAD NON-CATALYTIC WALL — 50% REDUCTION IN LEADING EDGE HEAT TRANSFER
PERFORMANCE	TURBULENT FLOW "BLUNT" LEADING EDGES	LAMINAR FLOW "THIN" LEADING EDGES — 25% INCREASE IN L/D 50% INCREASE IN LATERAL RANGE

Figure 7. Design Criteria Impact

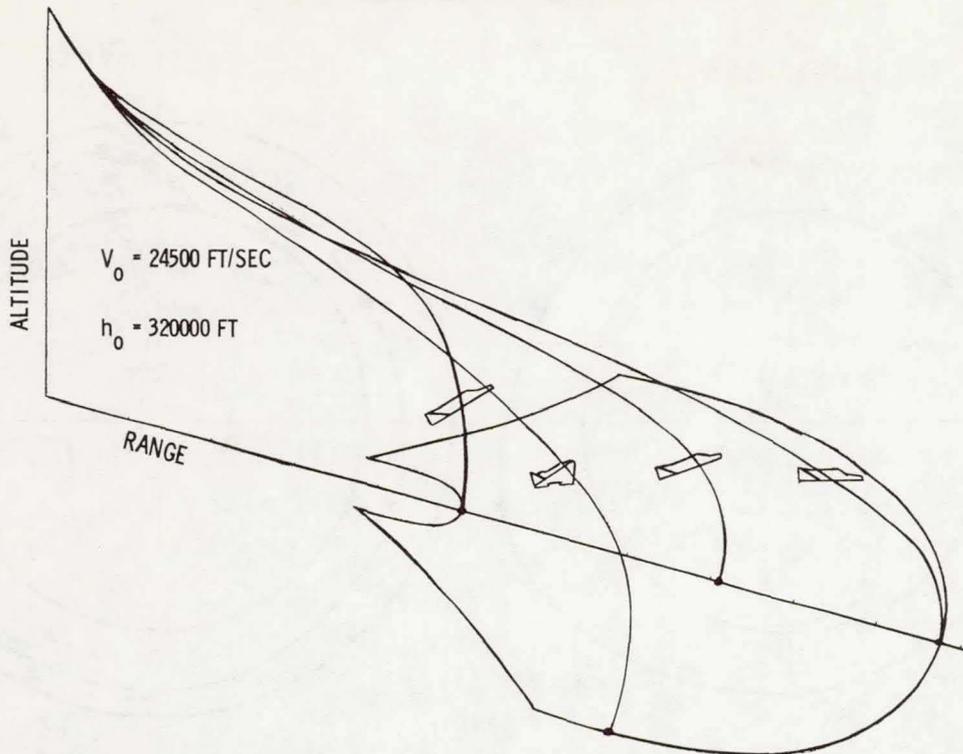


Figure 8. Distance and Direction Control Re-Entry Maneuverability

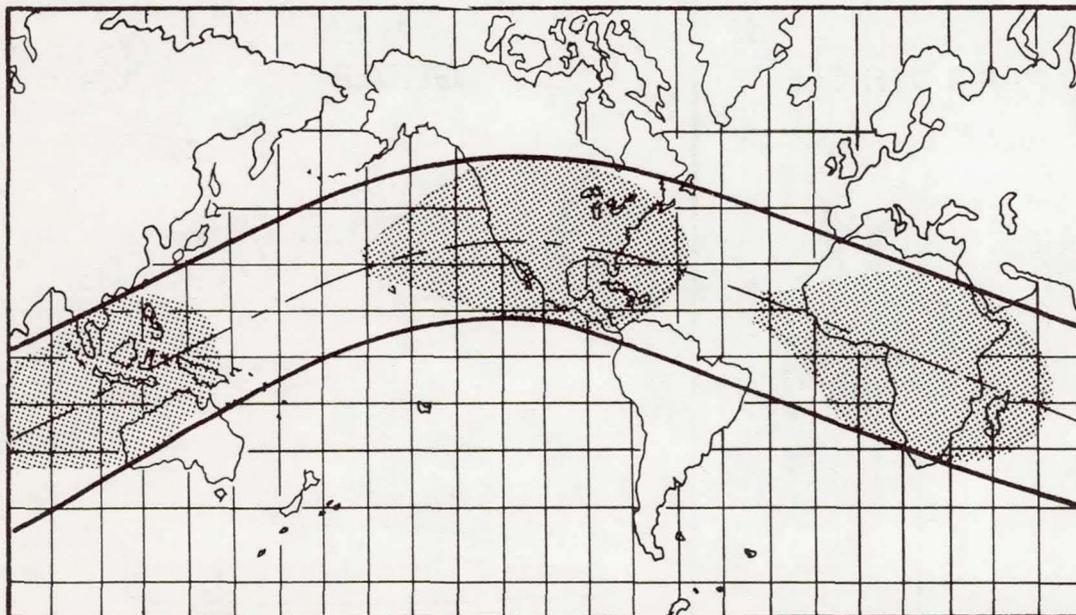


Figure 9. X-20 Maneuver Flexibility
110° Launch From CCMTC

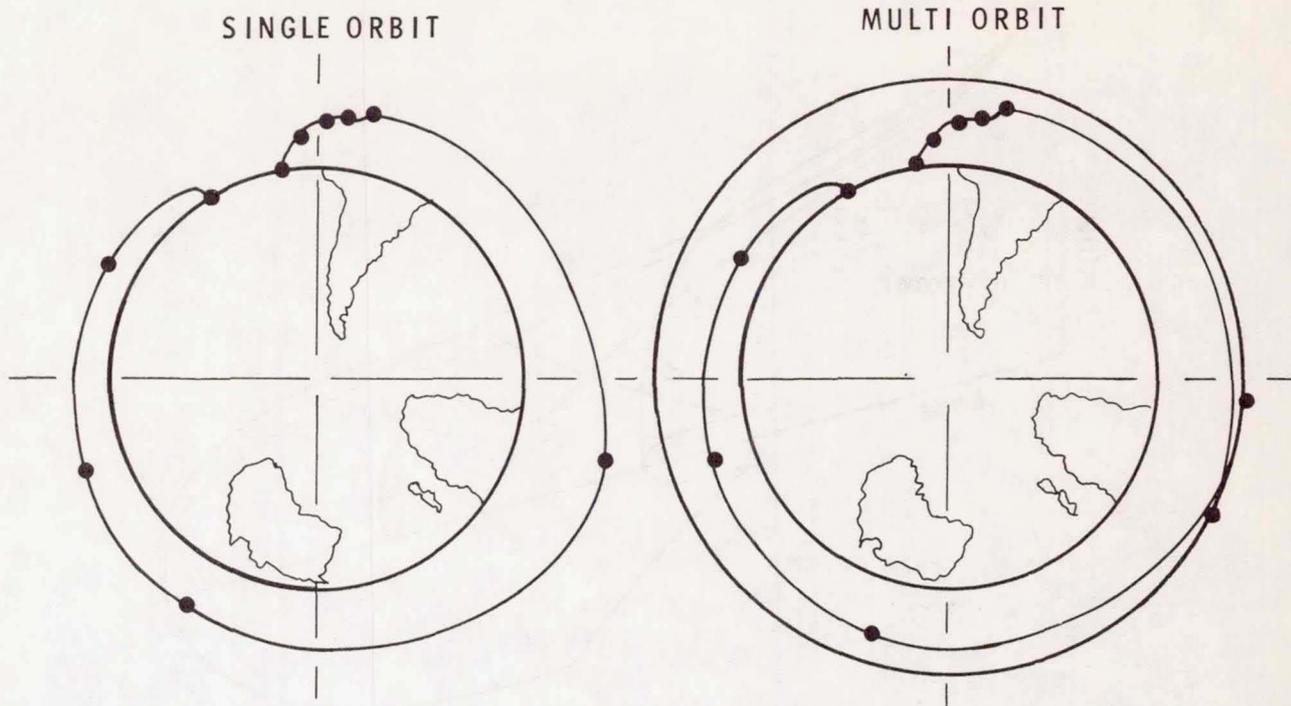
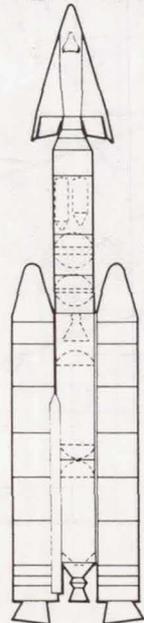


Figure 10. Mission Profiles

AIR VEHICLE



GLIDER

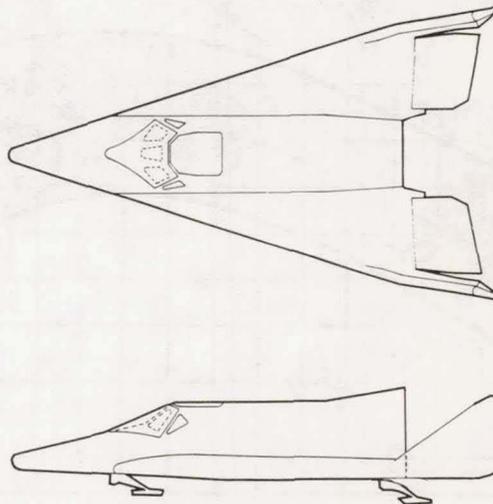


Figure 11. X-20 Configuration

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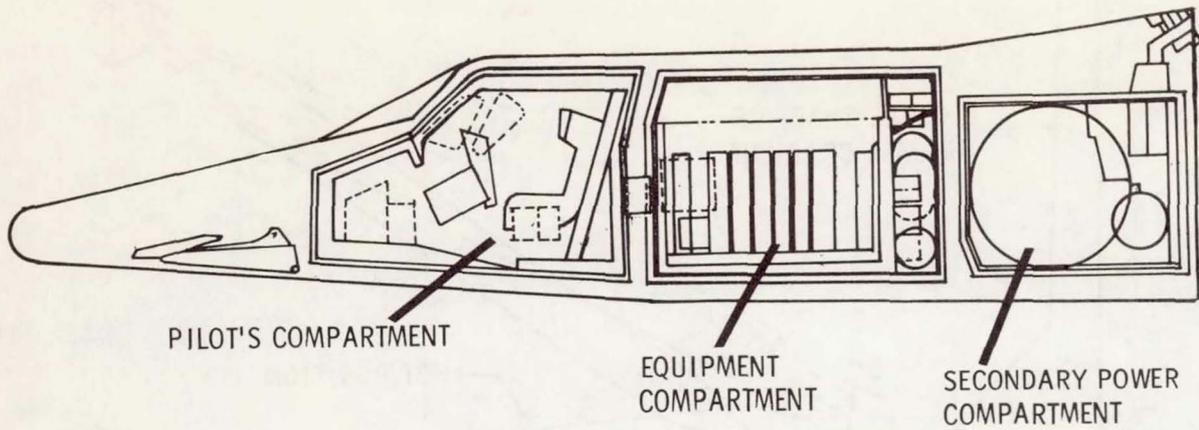


Figure 12. Inboard Profile

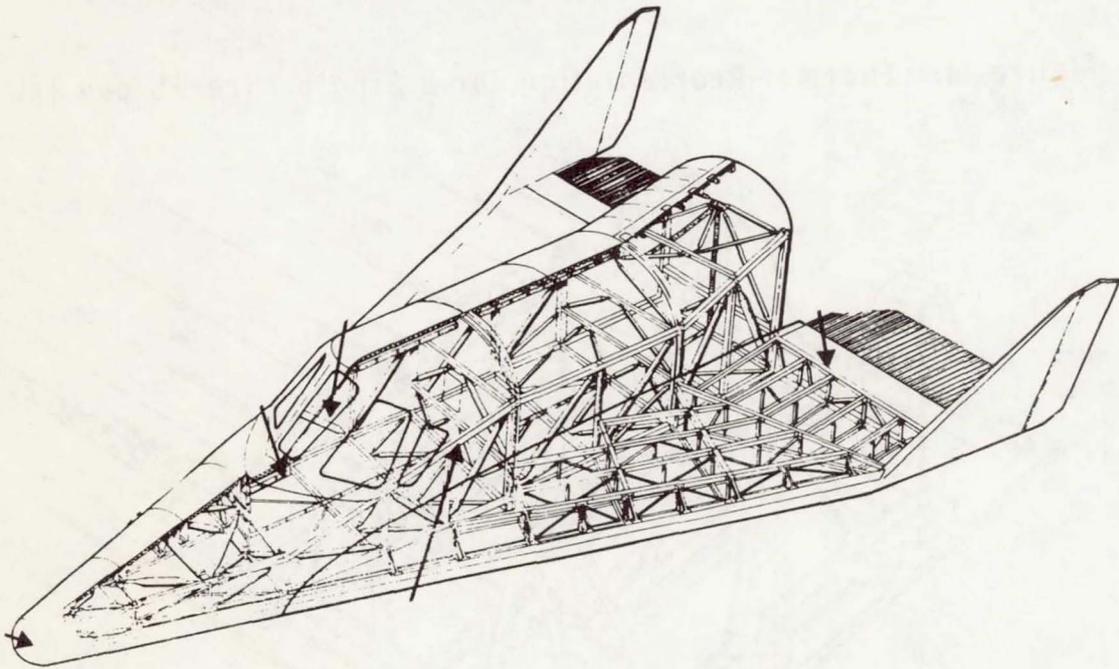


Figure 13. X-20 Truss Structure

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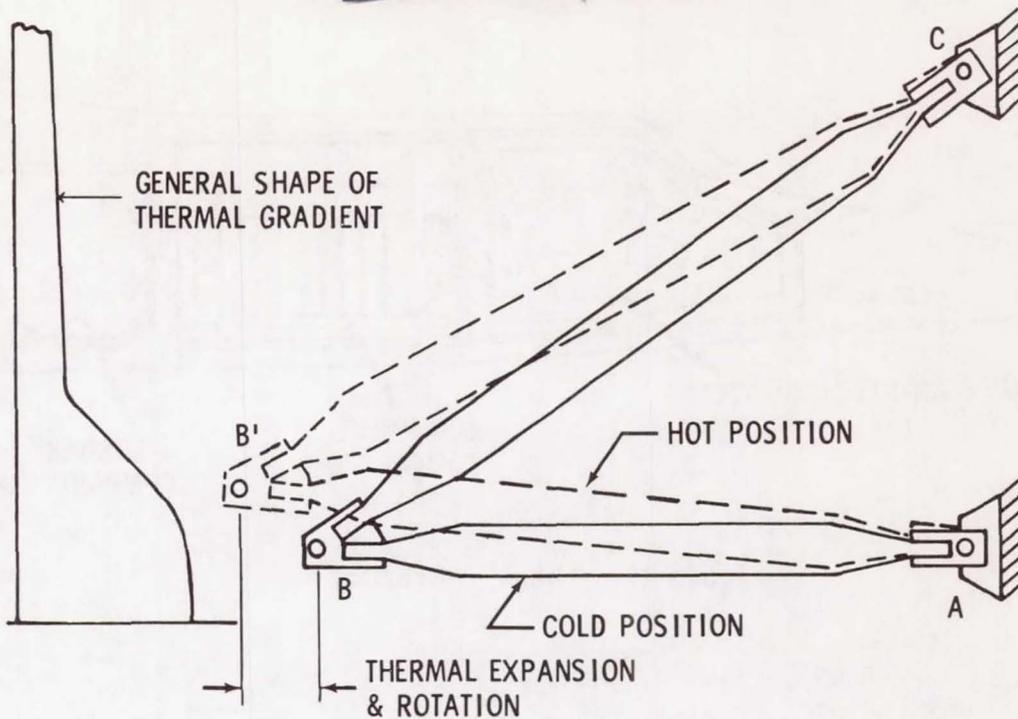


Figure 14. Thermal Reorientation for a Single Three-Sided Truss

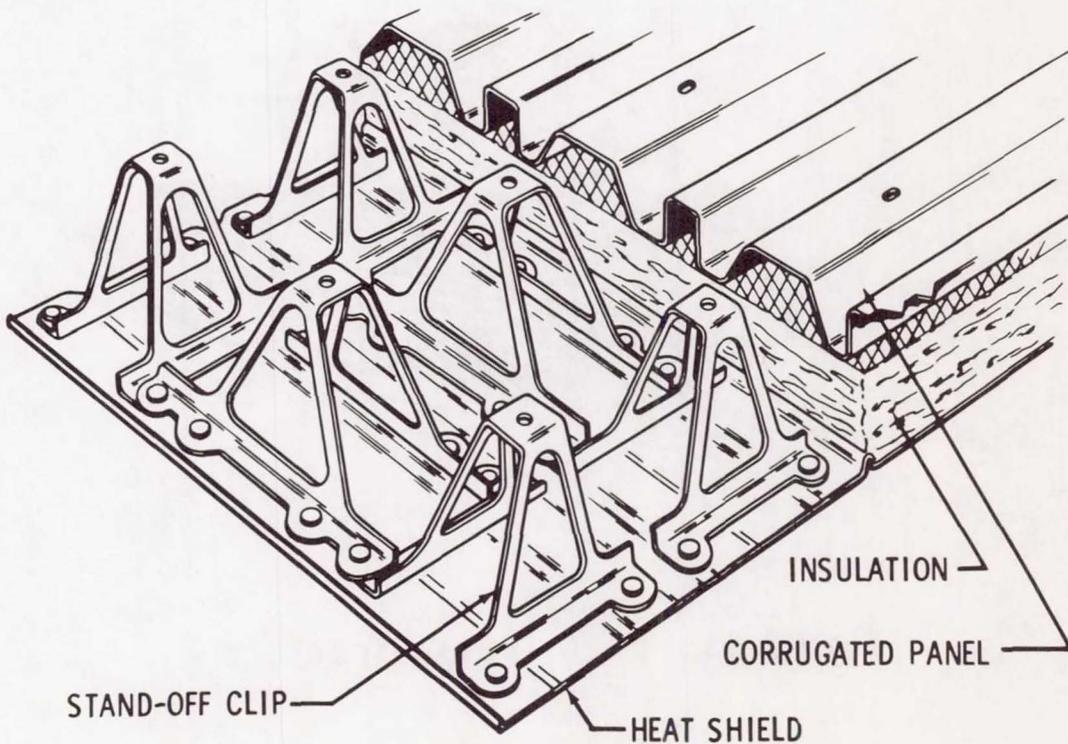


Figure 15. Typical Insulated Panel

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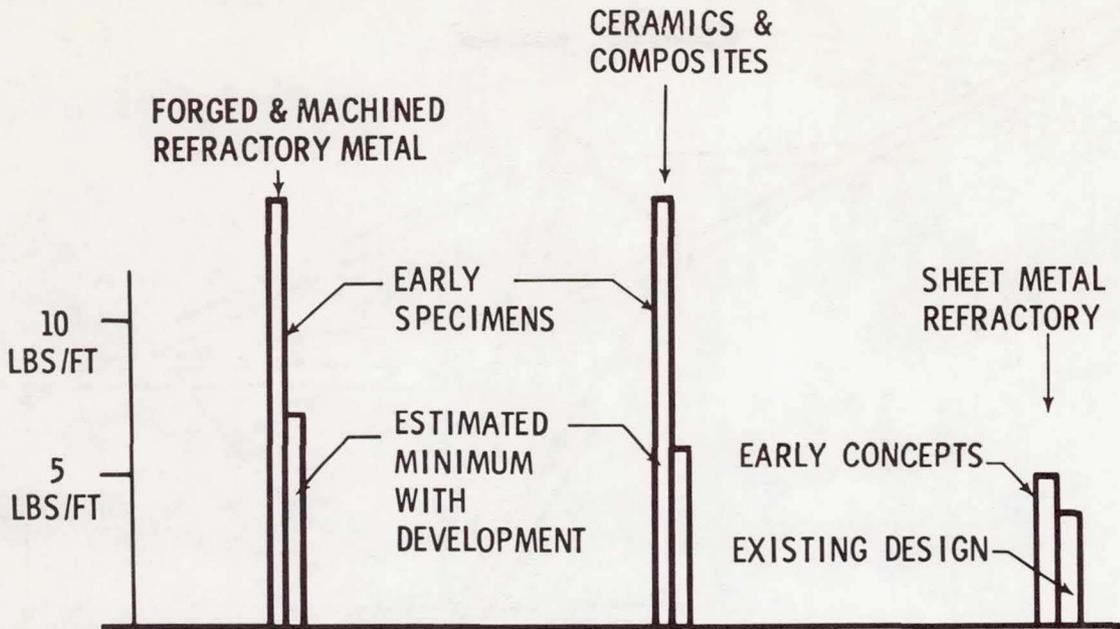


Figure 16. Comparative Weights of Leading Edge Specimens

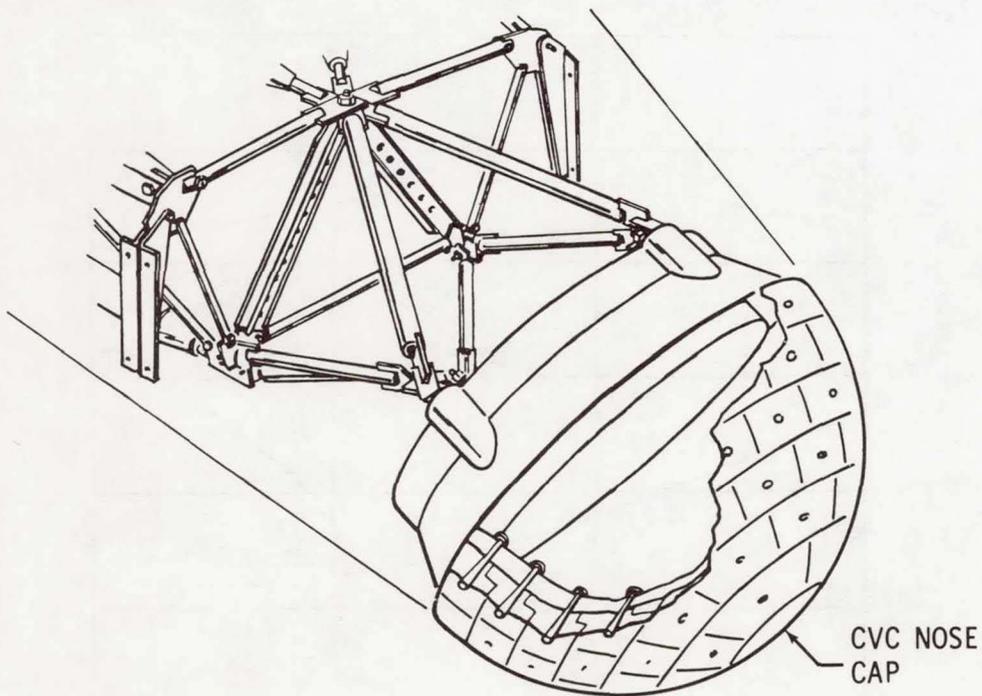


Figure 17. Nose Cap and Support Structure

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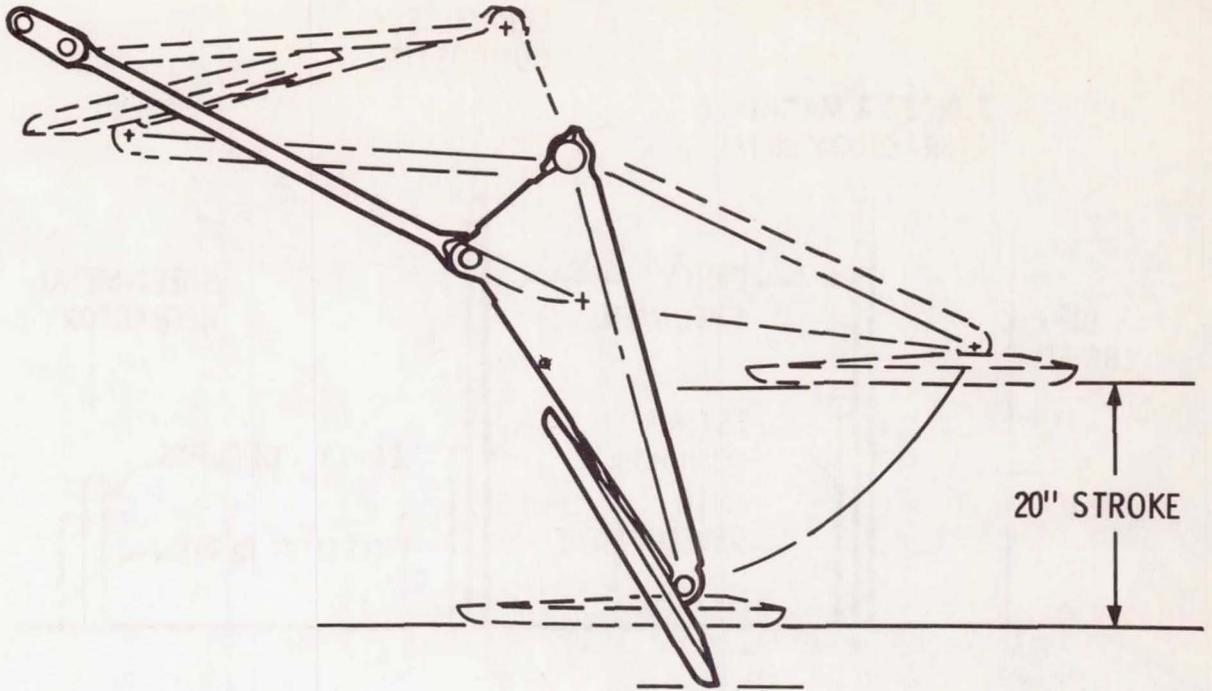


Figure 18. Nose Gear Arrangement

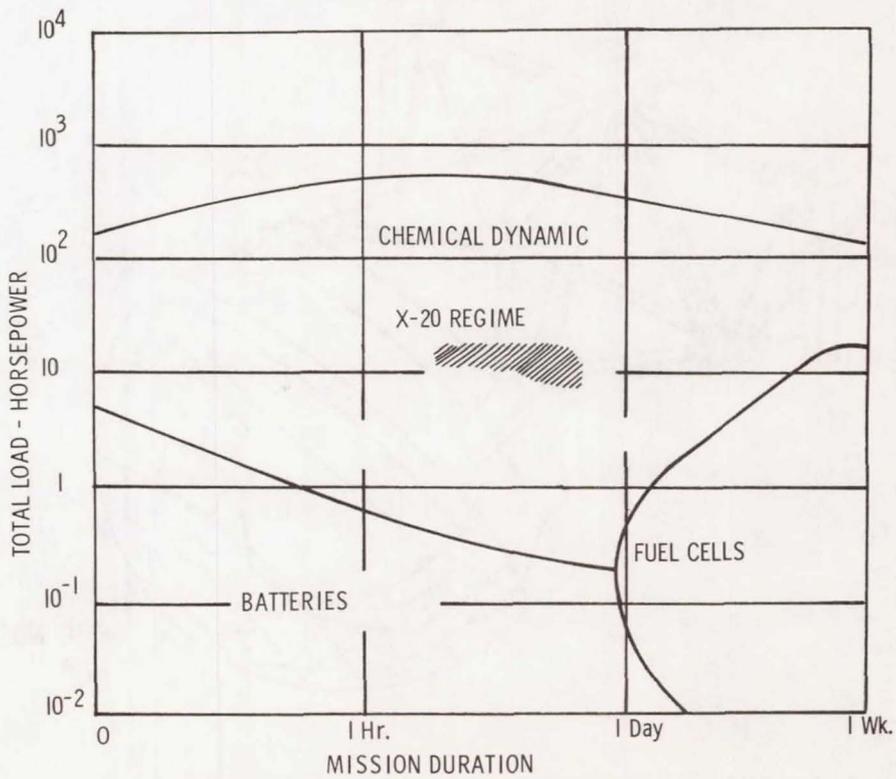


Figure 19. Source of Secondary Power Generation

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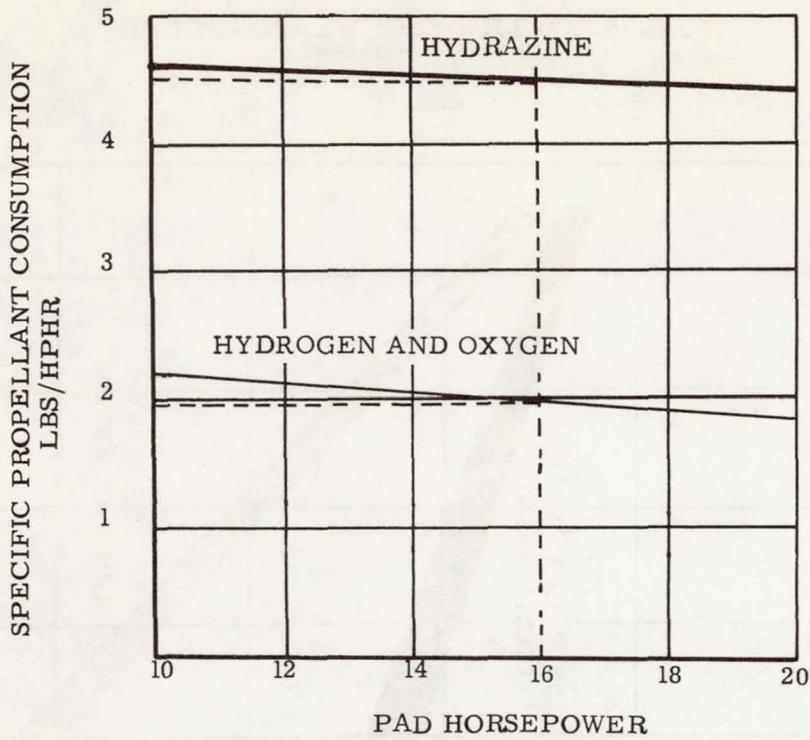


Figure 20. APU SPC Comparison

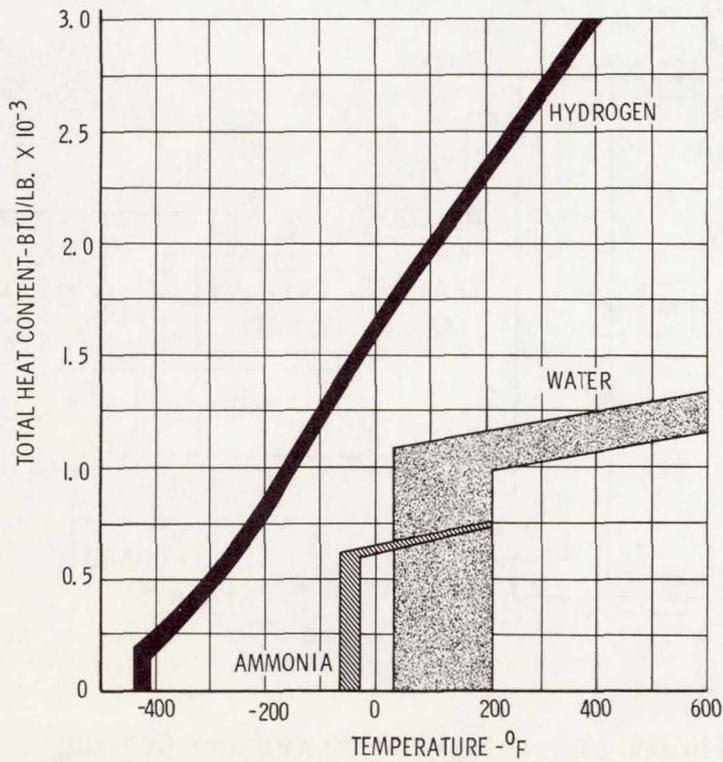


Figure 21. Potential Heat Capacity of Three Coolants

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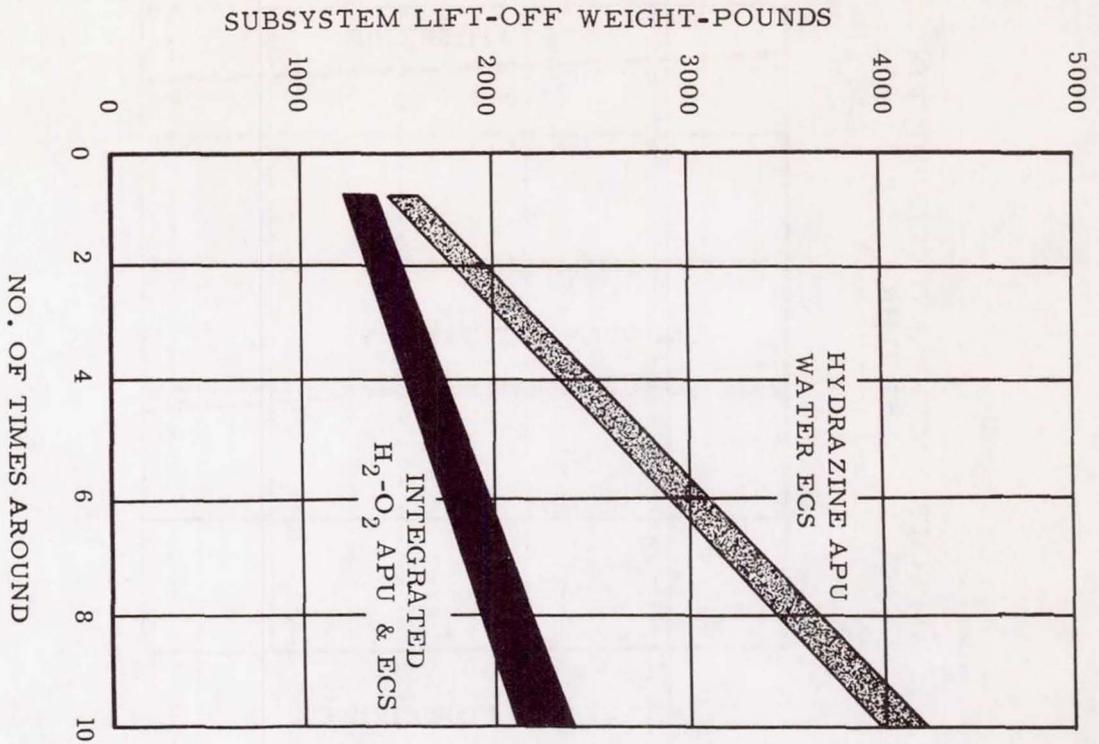


Figure 22. Weight Comparison

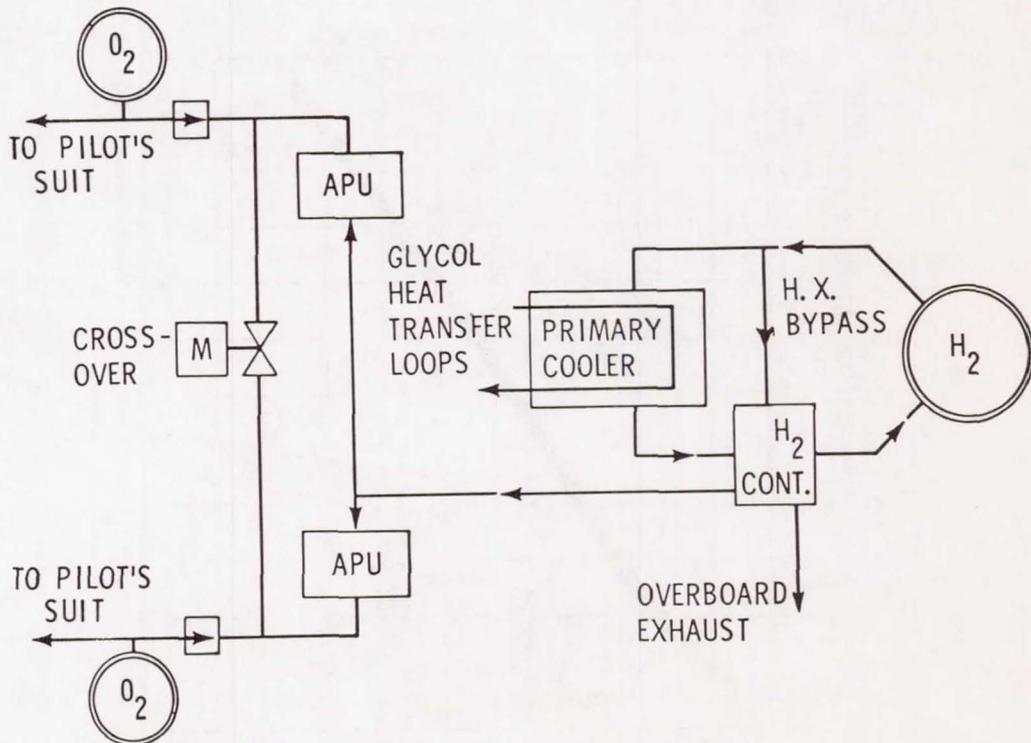


Figure 23. Integrated Power and Cooling

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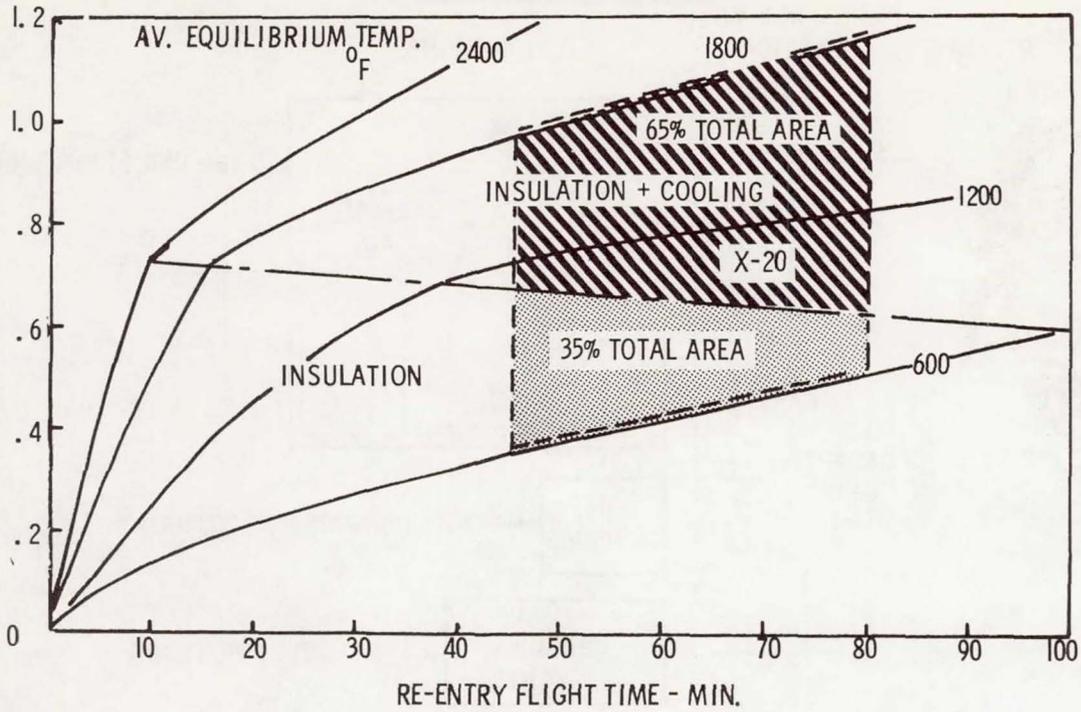


Figure 24. Minimum Weight Thermal Protection

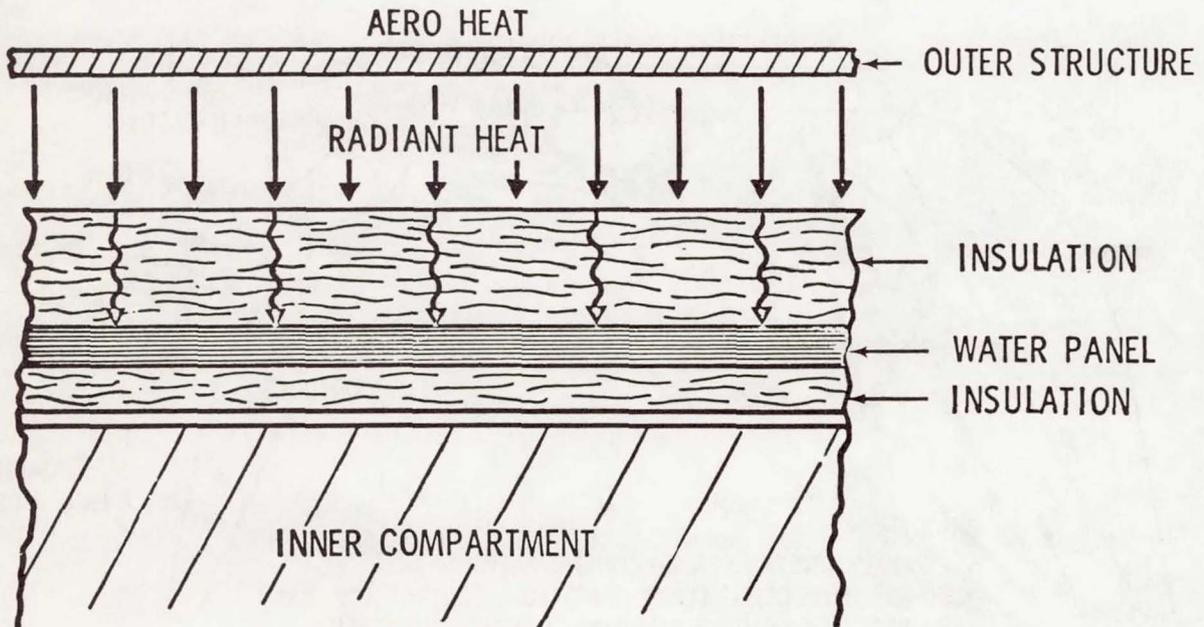


Figure 25. Water-Wall System (Schematic)

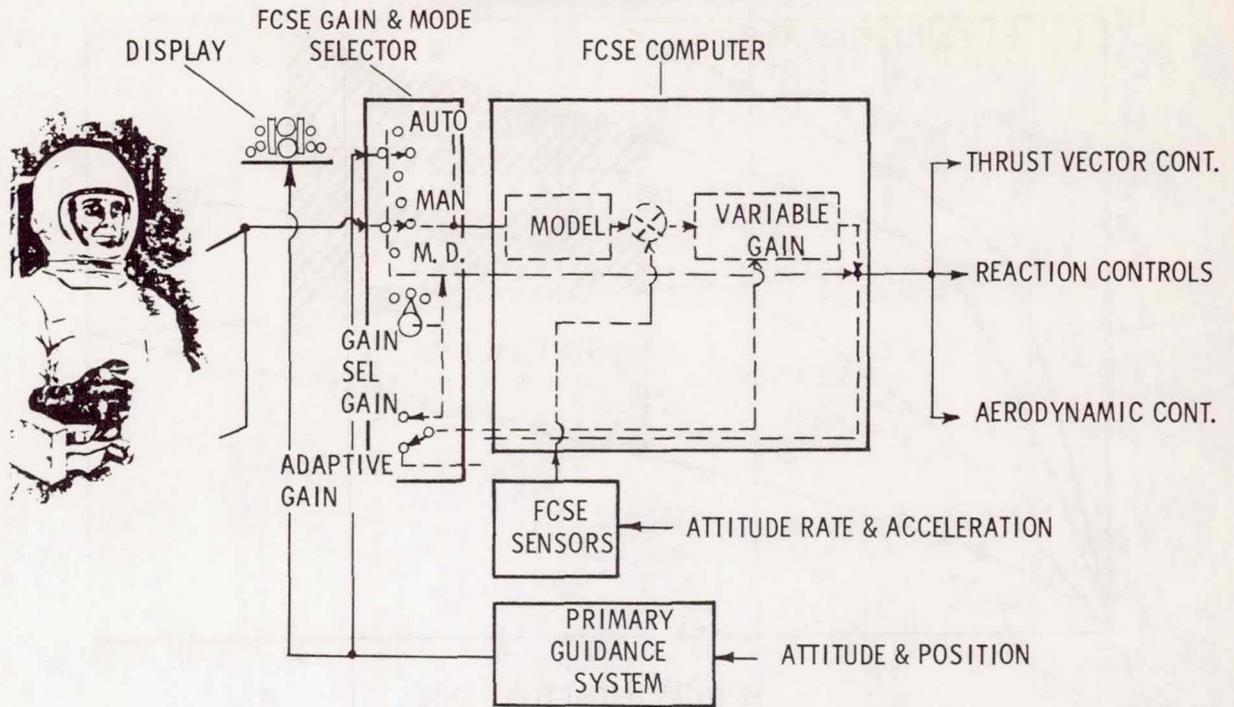


Figure 26. Flight Control Subsystem Electronics (Sensors, Computer & Mode Selectors)

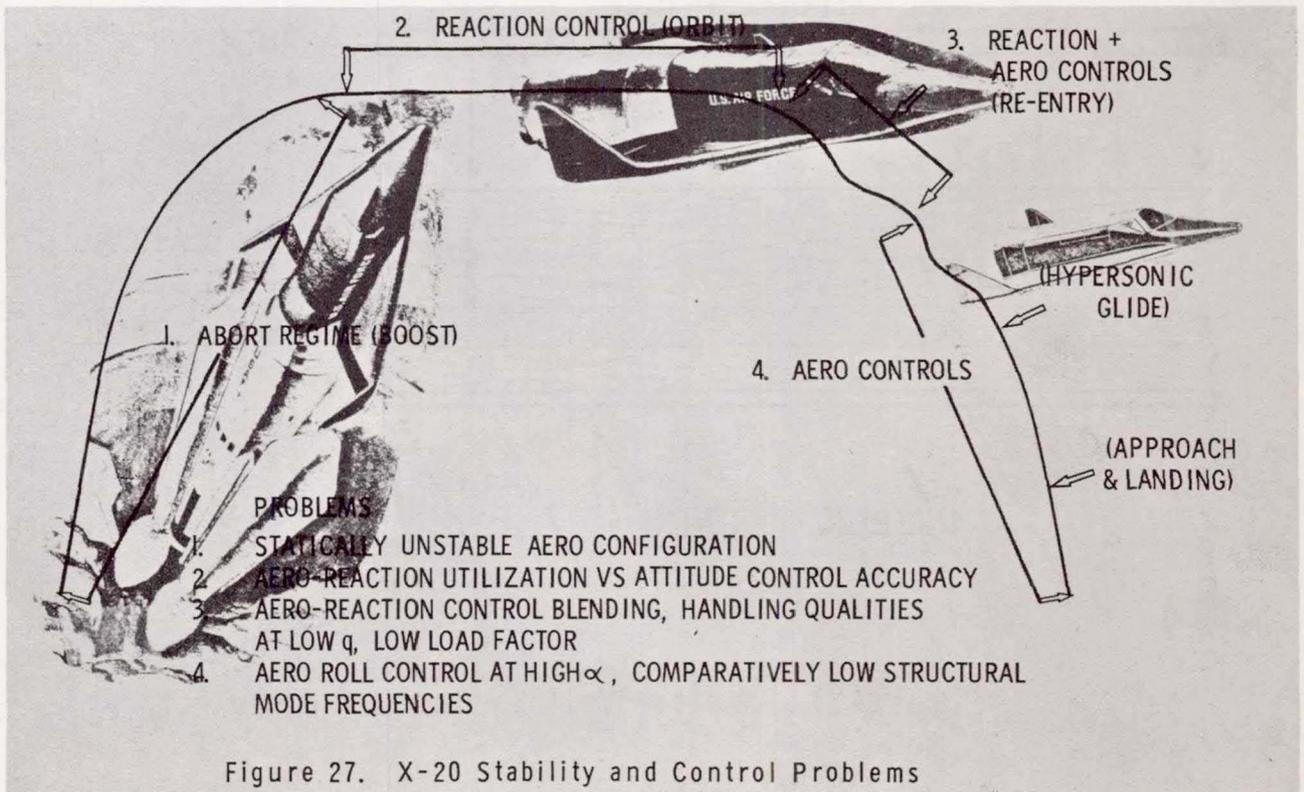


Figure 27. X-20 Stability and Control Problems

CLOSE PROXIMITY OF STRUCTURAL & CONTROL FREQUENCIES RESULTS IN EXCITATION OF STRUCTURE BY CONTROL SYSTEM

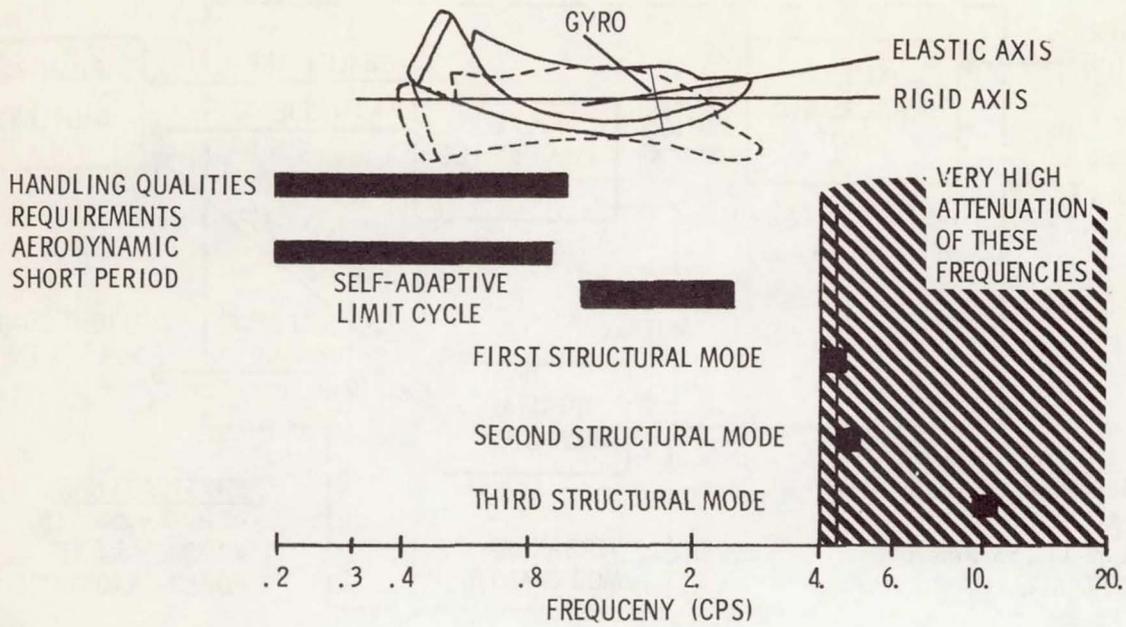


Figure 28. Problem Aero-Servo-Elastic Coupling

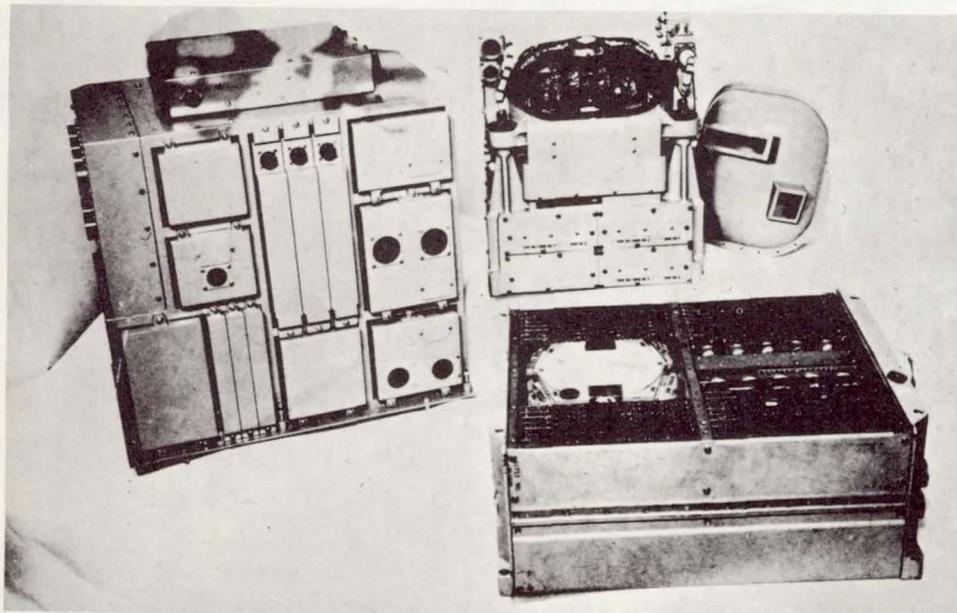


Figure 29. X-20 Inertial Guidance Subsystem

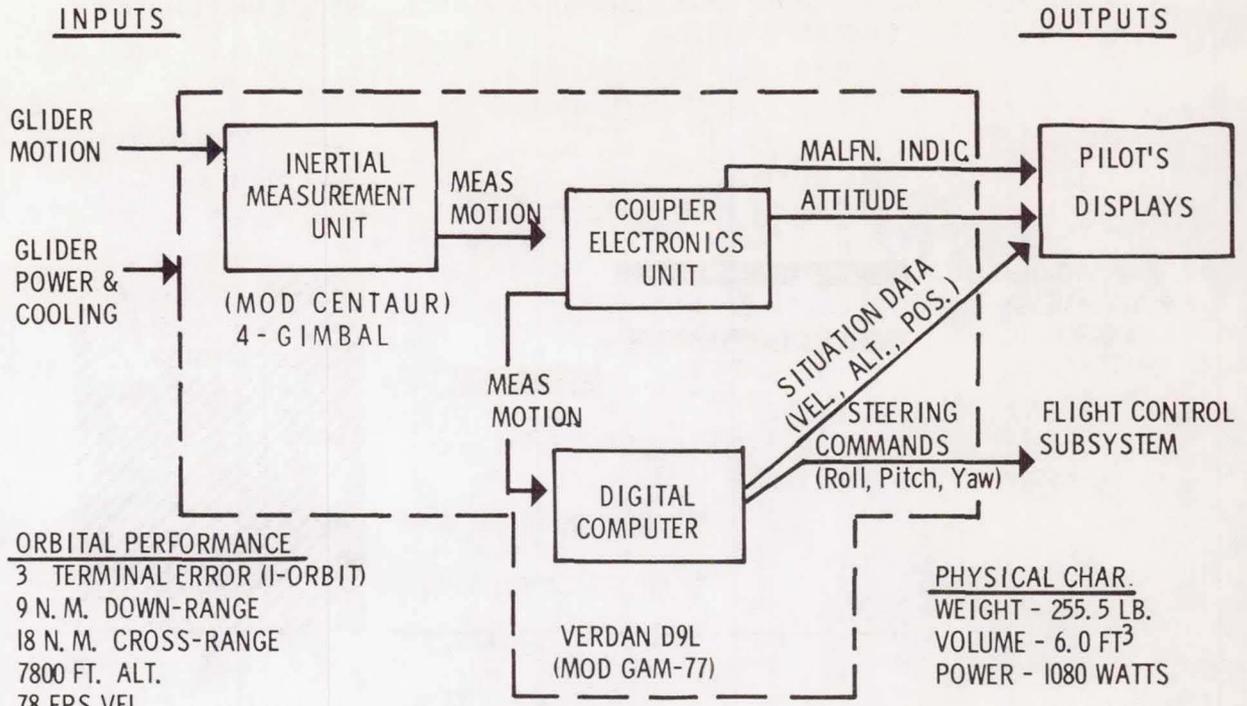


Figure 30. X-20A Inertial Guidance Subsystem

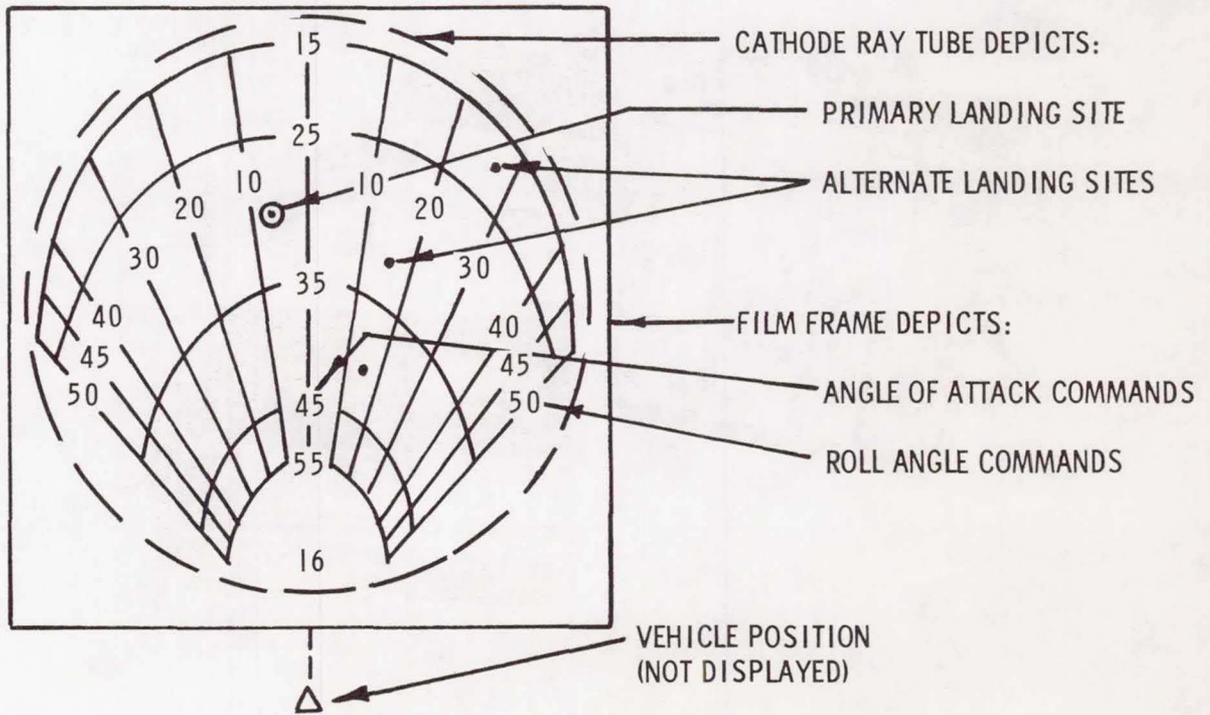


Figure 31. Typical Energy Management Display

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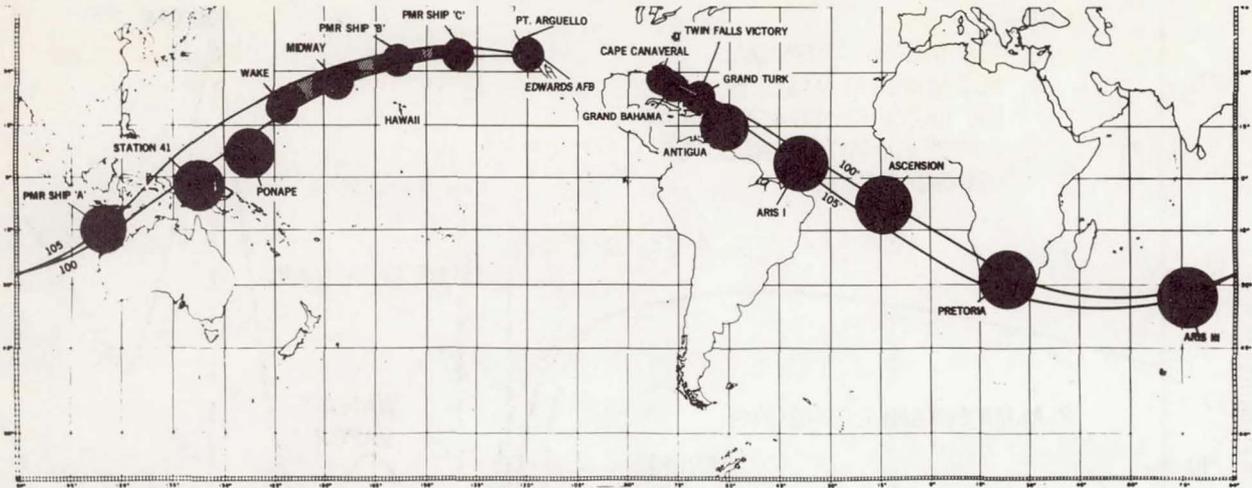
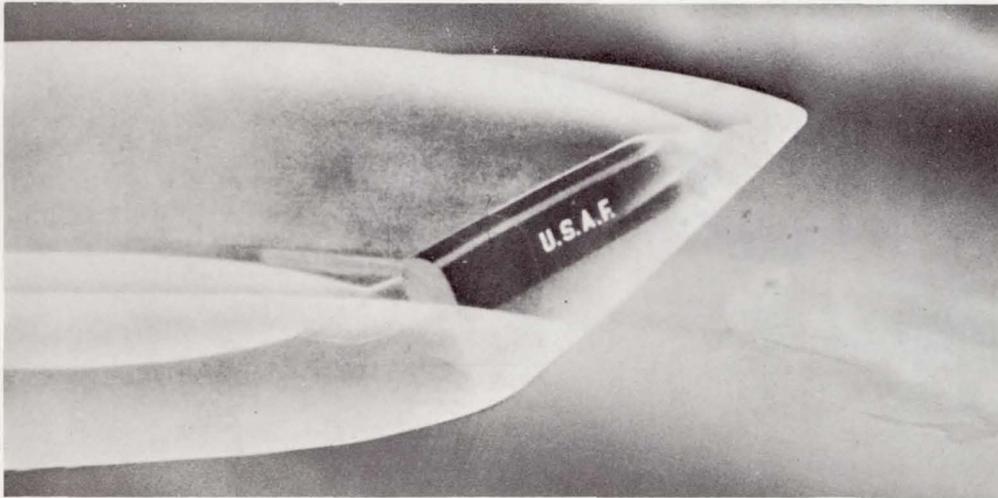


Figure 32. Single Orbit



PLASMA SHEATH

ELECTRO-OPTICAL (ABSORPTION, REFLECTION, & REFRACTION)

ANTENNA VOLTAGE BREAKDOWN

PLASMA-INDUCED NOISE

Figure 33. Hypersonic Re-Entry Communications

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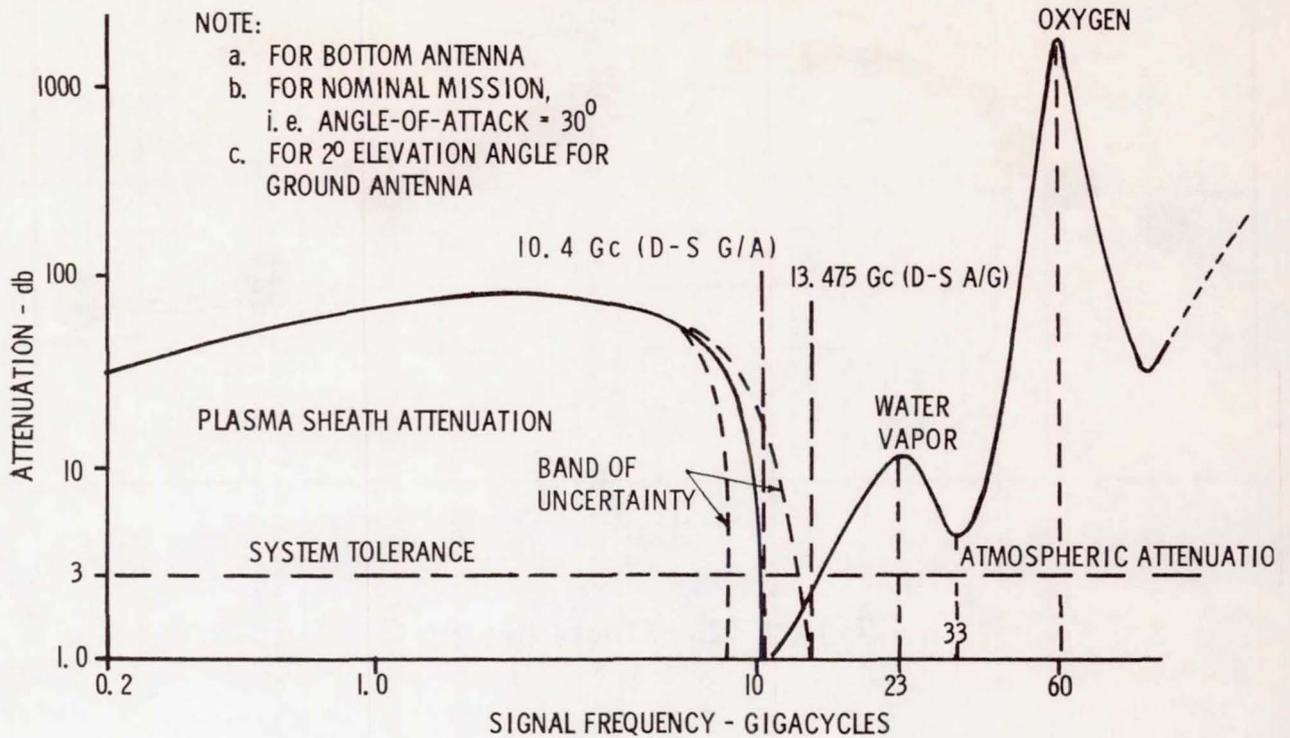


Figure 34. Attenuation Vs Frequency for X-20 (Dyna-Soar) Re-Entry

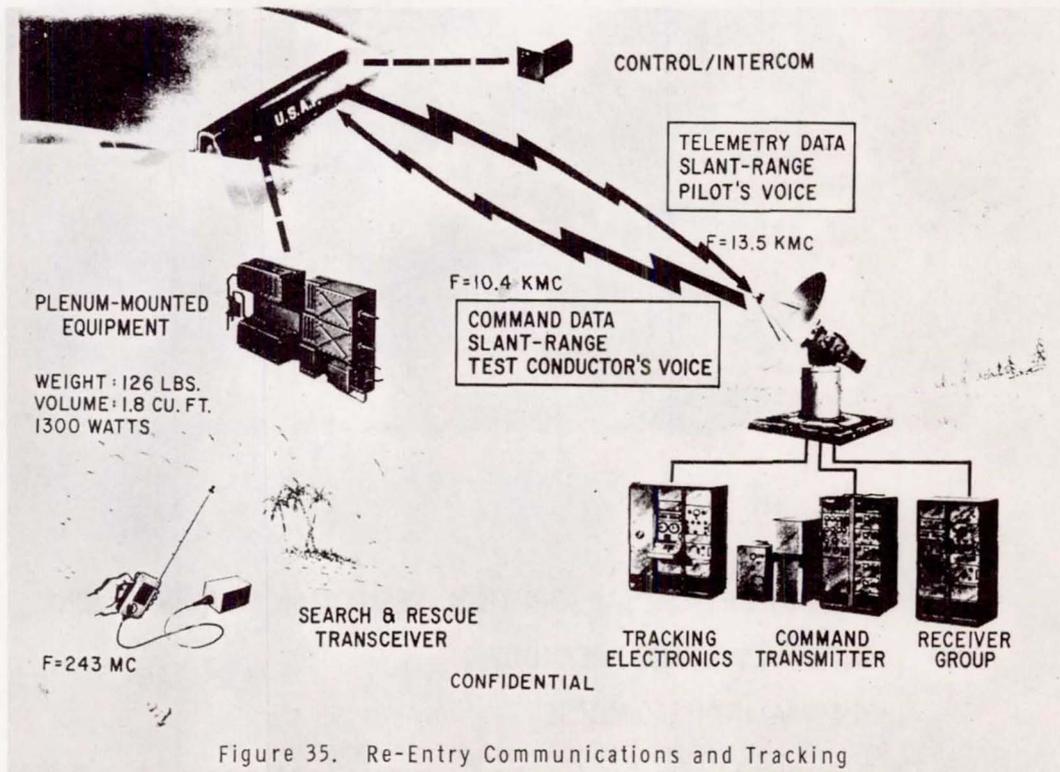


Figure 35. Re-Entry Communications and Tracking

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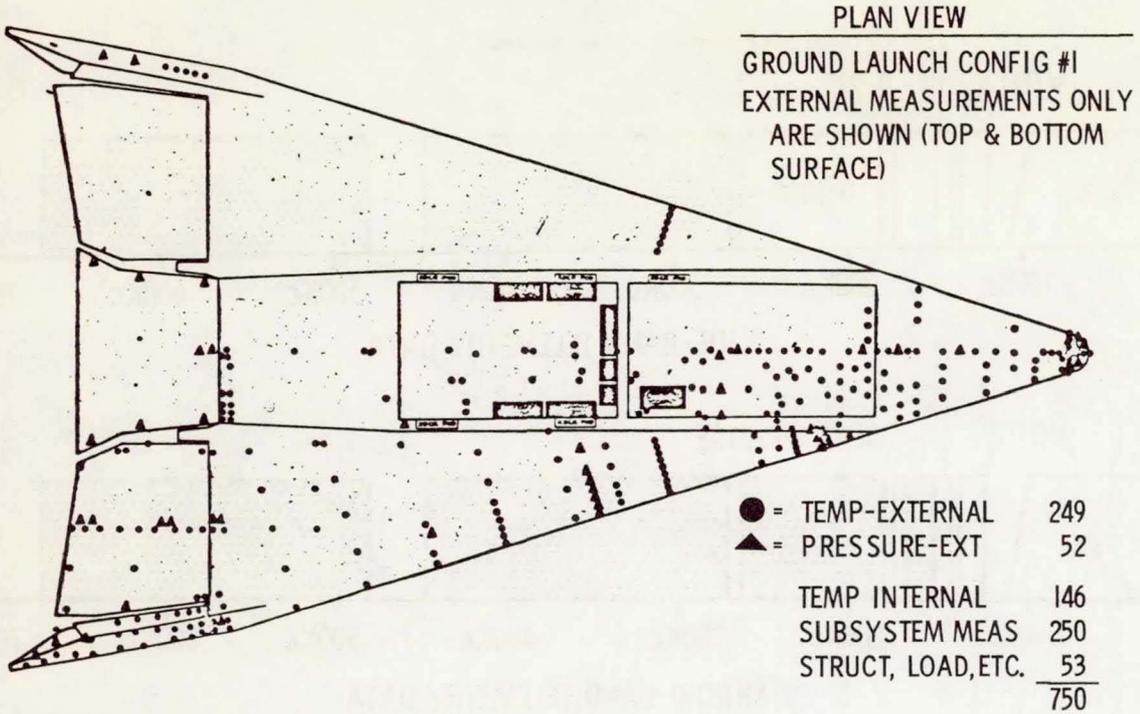


Figure 36. TIS-Equipment Locations

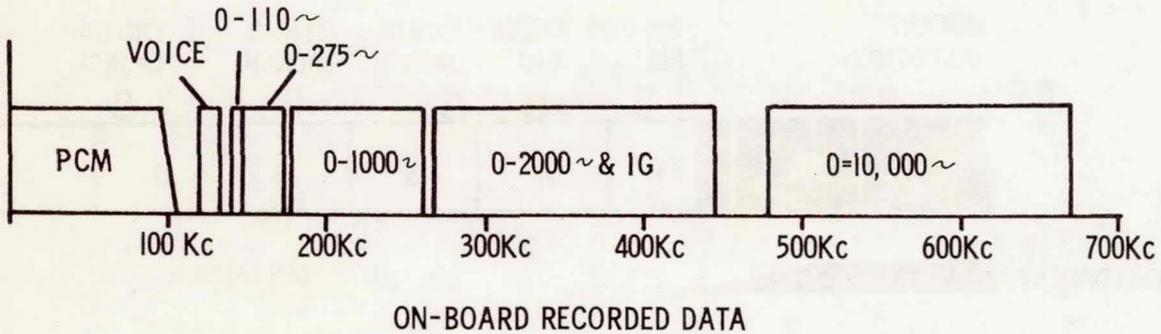


Figure 37. Data Baseband Spectrum

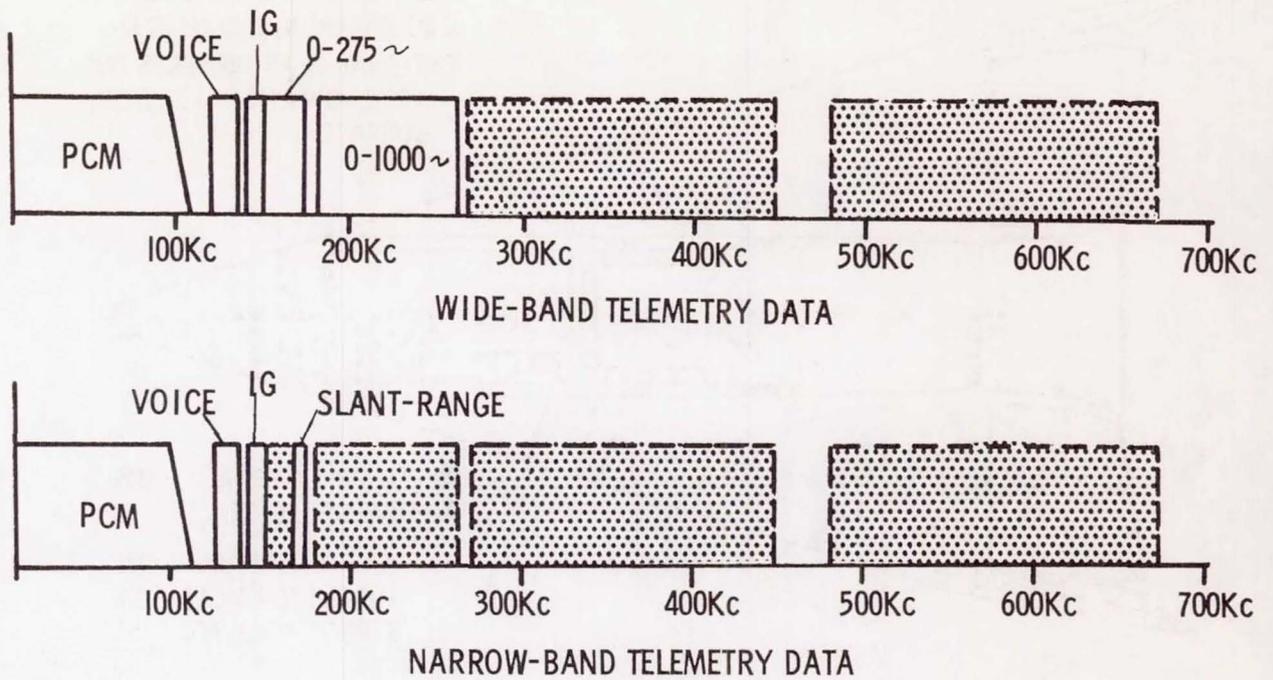


Figure 38. Data Baseband Spectrum

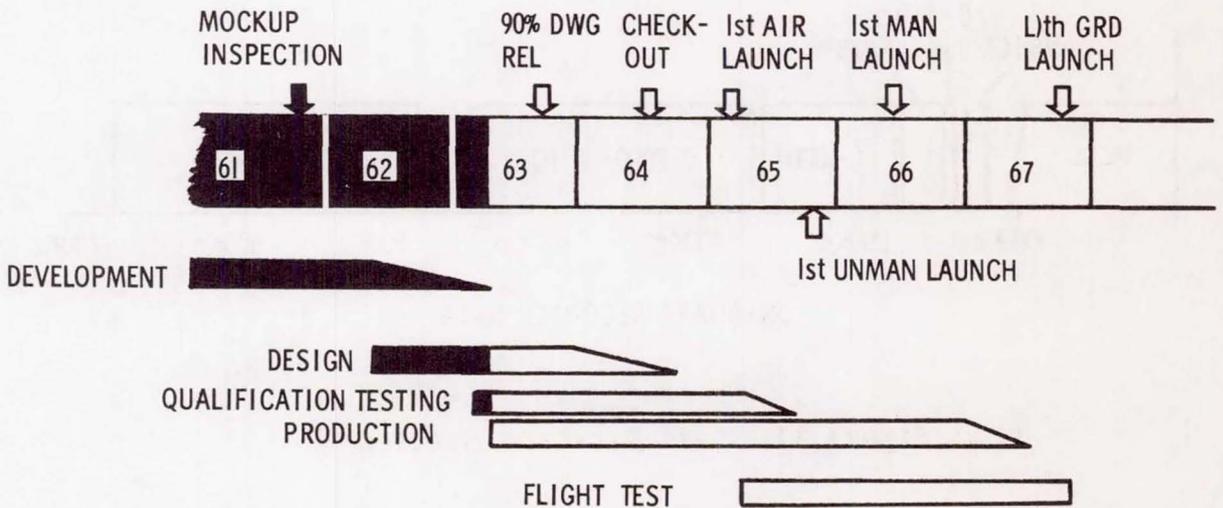


Figure 39. X-20A Schedule And Status

APOLLO DESIGN FEATURES

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Space and Information Systems Division, North American Aviation, Inc.

Abstract

This paper presents some Apollo design features that were dictated by special problems associated with a manned lunar landing and return mission. Design features primarily attributed to booster limitations, crew safety, and natural mission requirements are discussed. Emphasis is placed on those features considered unique. Examples of specific topics considered are the general designs of the command module, heat shield, environmental control system, service module propulsion system, and Earth landing system.

Introduction

Although unmanned space probes have penetrated into deep space and, in particular, have transmitted information back to Earth regarding our neighboring planet Venus, man's personal venture into space has thus far been confined to Earth-orbital flights. The success of the Mercury program has been phenomenal. Project Gemini is an extension of the Mercury program with a greater number of Earth-orbits, two men in the capsule, and Earth-orbital-rendezvous missions. Projects Mercury and Gemini are logical steps in man's systematic attempts to conquer space, and as such, they are fundamental to future manned space flights extending beyond the gravitation of the Earth.

The next big step after Gemini in the United States manned space program is Project Apollo. Unlike Earth-orbital missions, the Apollo mission to land American astronauts on the Moon and return them to Earth necessitates escaping the Earth to reach the Moon and then escaping the Moon to return to Earth. This jump from manned Earth-orbital missions to manned lunar-landing missions demands propulsion capability far in excess of that ever before required. In addition, mission durations longer than ten days must be anticipated. During this time, the spacecraft and its crew must survive the environment of outer space. The Apollo mission thus imposes severe demands on booster capabilities and introduces many technological and environmental problems that are peculiar to a manned lunar-landing and Earth-return mission.

This paper presents some of the design features dictated by the special requirements of the Apollo mission. In particular, design features primarily attributed to booster limitations, crew safety, and natural mission environment are discussed. Emphasis is placed on those features that are considered unique.

Design Features Arising From Limitations in Available Boosters

The relative sizes of the various launch vehicles that are either in use or considered for use in the United States manned space program are shown in Figure 1. Of these vehicles, only Saturn V or NOVA has the performance capability to fulfill the Apollo objectives. For a direct lunar-landing mission, the NOVA vehicle would be the most desirable from the standpoint of performance, but because of the longer development time and higher cost of the NOVA, NASA selected Saturn V as the Apollo launch vehicle.

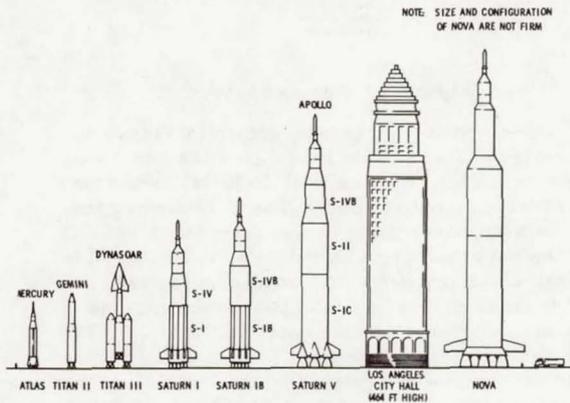


Figure 1. Launch Vehicles

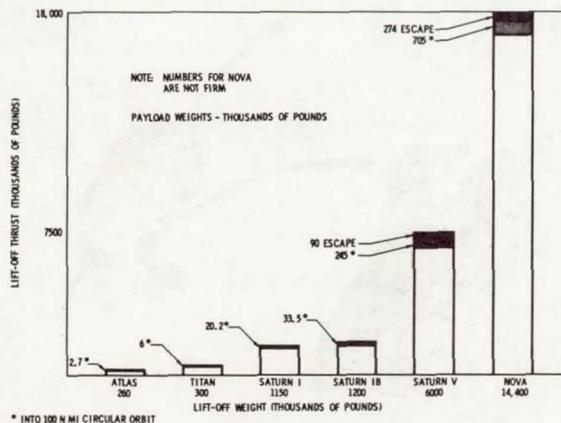


Figure 2. Launch Vehicles—Weight/Payload/Thrust

The jump from Atlas/Titan to Saturn V is a big one. As shown in Figure 2, Saturn V has an Earth-orbital payload capability approximately 90 times that of the Atlas and 40 times that of the Titan. Although Saturn V is capable of injecting about 90,000 pounds to the Moon, mission require-

ments of this weight impose severe design restrictions, not only on the spacecraft and associated components, but also on the over-all configuration of the Apollo spacecraft. Because every extra pound that is landed on the Moon and subsequently returned to Earth increases the gross take-off weight by nearly 500 pounds, weight control is a very critical problem. Therefore, some of the design features of the Apollo spacecraft can be attributed primarily to limitations in the performance capabilities of the available boosters.

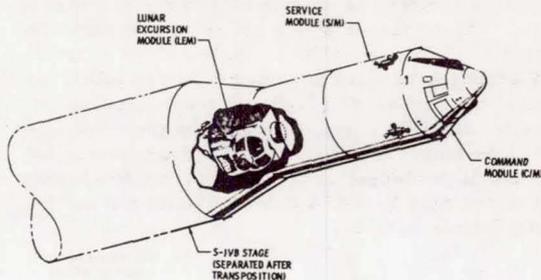


Figure 3. Apollo Spacecraft

The Apollo spacecraft, shown in Figure 3, consists of three basic modules—the command module (C/M), service module (S/M), and lunar excursion module (LEM). The C/M houses the three astronauts going to and from the Moon. It is the only module to be returned to Earth. The S/M, which provides the propulsion for the midcourse corrections and the return trip, is jettisoned prior to Earth entry of the C/M. The LEM houses two astronauts for the lunar-landing and return-to-orbit phases of the mission. The landing gear portion of the LEM is left on the lunar surface, and the remainder is left in lunar orbit after transfer of the astronauts back into the C/M.

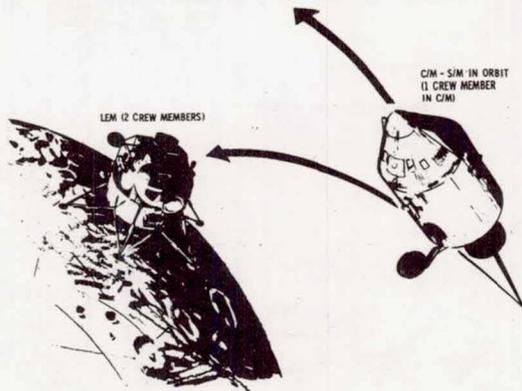


Figure 4. Apollo Approach

Two separate vehicles, each capable of sustaining human lives, are needed to accomplish the lunar-landing mission. These two vehicles are the C/M and the LEM, and their simultaneous existence reflects the decision of the United States to go to the Moon via the lunar-orbital-rendezvous mode. This method was chosen partly because of the limitations in booster capabilities. Figure 4

shows the Apollo approach employing this technique. The LEM is descending to land on the Moon, while the C/M and S/M remain in lunar orbit.

It is possible to eliminate the LEM by going to the Moon via the Earth-orbital-rendezvous (EOR) mode, but two Saturn V launch vehicles and a large spacecraft lunar-landing propulsion unit would be required. In addition, there is the operational problem of having to make two consecutive launches successfully within a specified period of time: one would place a tanker or a booster loaded with cryogenic fuel into an Earth orbit, and the other would place the spacecraft into the proper position for rendezvous with the vehicle in orbit.

From the decision to use the lunar-orbital-rendezvous method, the following design feature was established: the LEM is to be initially transported behind the S/M, and then shortly after translunar injection the LEM is to be transposed and mated with the C/M. This transposition phase of the flight is necessary in order to expose the S/M engine for use in midcourse guidance corrections. Abort requirements make it impracticable to launch with the C/M and LEM mated. A promising scheme for making the transposition and docking is illustrated in Figure 5. The action begins by igniting the four S/M reaction-control-system engines and then blowing off the adapter. Separated from the launch vehicle, the C/M-S/M unit free-flies around to mate with the LEM, which is stabilized by the empty S-IVB stage and its stabilization system. After the mating of the C/M-S/M unit with the LEM, the S-IVB stage is jettisoned, and the Apollo spacecraft proceeds to coast toward the Moon.

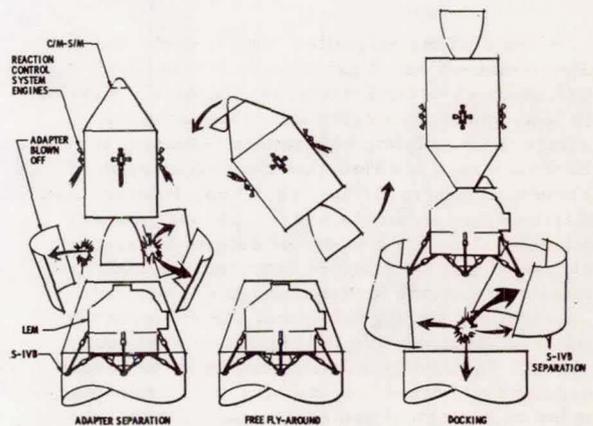


Figure 5. Free Fly-Around Transposition and Docking

It has been stated that the S/M is jettisoned prior to Earth entry of the C/M. Unlike the Mercury and the Gemini vehicles, which require retrothrusting to deorbit for the Earth entry, the C/M, moving with an inertial velocity of approximately 36,000 ft/sec, enters the Earth's atmos-

phere directly. Partly because of weight limitations, a retropackage is not used to reduce this high velocity. The result is that the C/M must be capable of dissipating the energy (virtually all kinetic) associated with Earth entry in such a manner that the integrity of the spacecraft remains intact and its human occupants unharmed. In addition, the C/M must be capable of correcting guidance errors in order to reach a given landing site. The present C/M, in fulfilling these requirements, presents the following design features.

The C/M is essentially a body of revolution and, with the center of gravity (c.g.) along its longitudinal axis, will develop no aerodynamic lift (Figure 6). By offsetting the c.g., however, the C/M trims at an angle of attack approximating -33 degrees. In this trimmed attitude, the axial force is resolved to yield a lift-to-drag ratio of 1:2. It should be noticed that on this vehicle positive lift is generated at negative angle of attack. The actual c.g. offset is achieved by locating the heavy equipment on one side of the longitudinal axis. This requirement critically restricts the space available for the installation of various components.

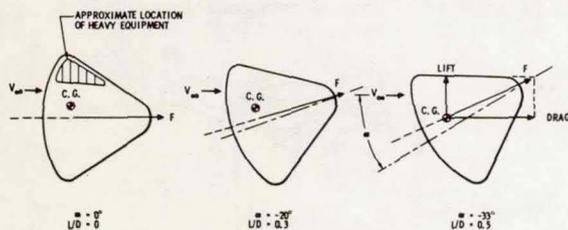


Figure 6. Command Module Aerodynamics

The C/M can be flown by rotating the vehicle about the instantaneous velocity vector. This maneuver, however, forces the lift vector out of a given plane of action so that any effort to maneuver in the vertical plane automatically produces horizontal displacements. Figure 7 shows the C/M with its lift vector fully up, partly tilted to the right (with resulting vertical and horizontal components), and fully down. The four roll reaction-control engines shown in Figure 8 are used to rotate the C/M about the stability axis. Each reaction jet can deliver 100 pounds of thrust. Note that there are 12 reaction-control engines on the C/M. Since only six engines are needed to control roll, pitch, or yaw, the 12 engines represent a completely redundant reaction control system. With a lift-drag ratio of 1:2, the C/M can enter the Earth's atmosphere and maneuver to the landing site from as far out as 5000 nautical miles or as close in as 1400 nautical miles. Figure 9 illustrates the Earth entry range limits.

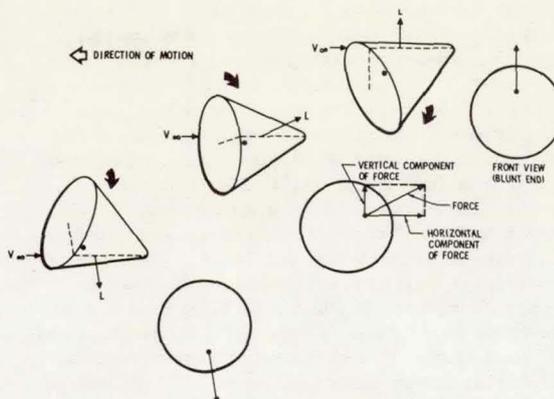


Figure 7. Lift Vector Control

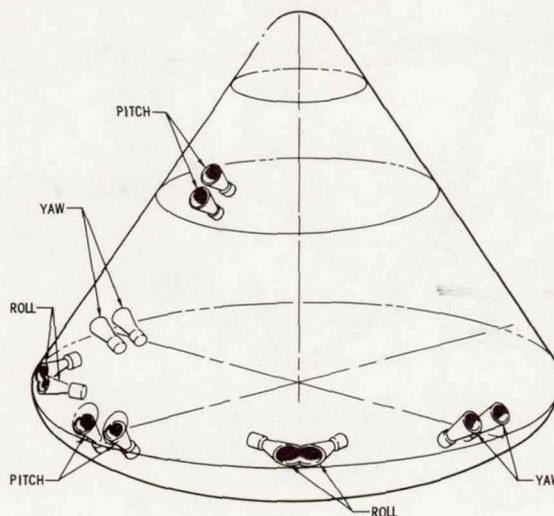


Figure 8. Command Module Reaction-Control-System Engines

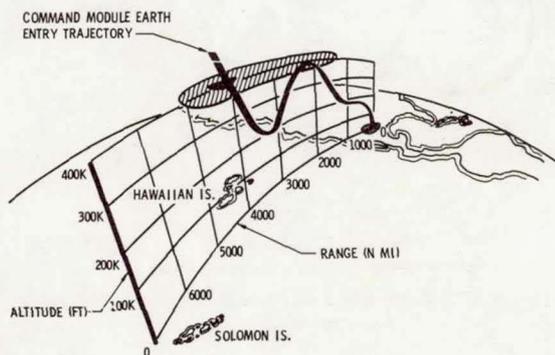


Figure 9. Entry Range Limits

During Earth entry, depending upon the particular trajectory flown—e.g., high deceleration with short flight time or low deceleration with long flight time—the total heat load on the

C/M can vary between approximately 5 to 8 million Btu's. These heat loads are many times larger than those occurring during entry from an Earth orbit. The heat shield being developed for use in the C/M to dissipate the entry heat loads incorporates a fiberglass honeycomb matrix that is bonded to the outer body substructure and then filled with ablative material. This type of construction yields a well-integrated heat shield that can withstand thermal stresses associated with temperatures as low as -260 F. Because of the stringent weight restrictions in the Apollo spacecraft injected payload, the C/M heat shield is tailored in thickness (Figure 10) to the imposed local heat load. The surface temperature of the C/M during Earth entry can reach 5000 F, but the ablator bond line will not exceed 600 F.

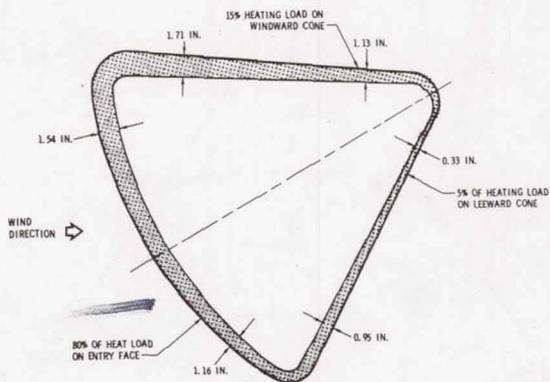


Figure 10. Apollo Command Module Local Heating Load and Heat Shield Thickness

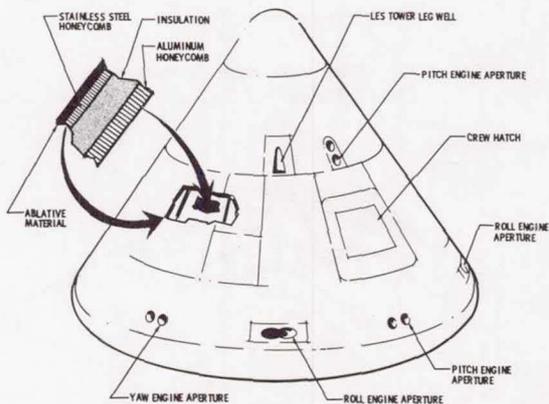


Figure 11. Command Module Exterior Structure

Figure 11 shows a cutaway view of the complete C/M, exposing a cross sectional view of the heat shield and the basic C/M structure. A design feature of this structure is its light-weight, double-shell construction. The outer shell is made of brazed stainless steel honeycomb, and the inner shell (Figure 12) is made of bonded aluminum honeycomb. This inner substructure constitutes

the pressure vessel and is maintained at a pressure of 5 psi in a 100-percent oxygen environment for altitudes above 20,000 feet. The two shells are separated by floating fiberglass stringers, and the space between is filled with Q-felt insulation material. Although this type of construction is partially influenced by weight limitations, it is primarily developed from heat transfer considerations. This construction also serves as an effective barrier for meteoroids, trapping any meteoroid that might penetrate the outer layer of the honeycomb structure.

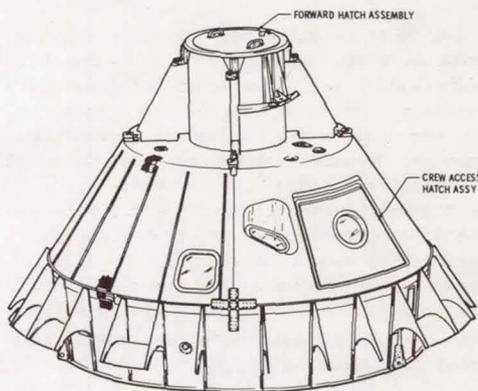


Figure 12. Command Module Inner Structure

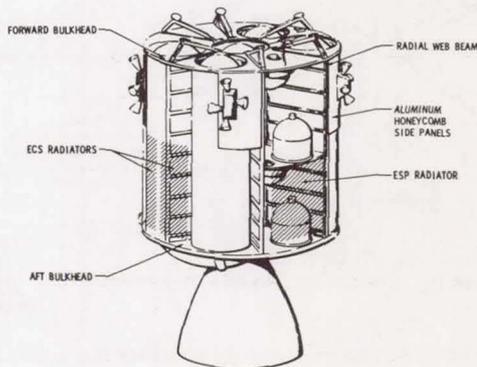


Figure 13. Service Module Structure

The S/M structure (Figure 13) also reflects a light-weight, simple type of construction. The basic structure consists of six equally spaced radial beams that divide the cylindrical S/M into six bays. These bays are used to house various items, such as the fuel and oxidizers for the S/M engine and the fuel cells. Aluminum honeycomb side panels and aft and forward bulkheads are bolted onto the solid aluminum beams to form the outer shell of the S/M. Four radiators, bonded directly to the side panels, are integral parts of the S/M outer structure. Two of these radiators are for dissipating heat from the environmental control system (ECS), and two are for dissipating heat from the electrical power system (EPS). The fuel cells, S/M engine, ECS, and EPS are discussed in the following sections.

Design Features Arising From Requirements
for Safety of Astronauts

The United States' philosophy of maximum concern for the safety of the astronauts dictates some operational design features that may or may not be manifested in specific pieces of hardware. Particular examples of nonhardware safety considerations are the circumlunar "free" return trajectory, LEM and C/M-S/M equal-period orbits, and over-all mission abort flexibility.

The circumlunar free return trajectory permits a return to Earth with a minimum change in the velocity vector if an abort is necessary after translunar injection. This means that in the event of a failure of the service propulsion engine, the reaction-control-system engines can be used to correct guidance errors to place the spacecraft into the proper circumlunar trajectory for the free return to Earth. The use of this type of trajectory, together with an Earth-to-Moon transit time of about 70 hours, makes it necessary to land on the Moon in retrograde motion with respect to the natural rotation of the Moon about its axis. Inasmuch as a point on the surface at the equator is moving with a tangential velocity of about 15 ft/sec, the LEM would have to land and take off against this velocity. This is a loss in velocity-change capability of 30 ft/sec. It is a direct consequence of flying such a circumlunar free return trajectory. The free return feature, however, is desirable from a crew safety and morale point of view.

The LEM and C/M-S/M equal-period orbit is a part of the over-all abort flexibility. Its use provides for a possible pickup of an inactive LEM by the C/M-S/M. For example, assume that the spacecraft is orbiting the Moon at 80 nautical miles altitude and that the LEM is ready to deorbit for the lunar landing. (See Figure 14.) A velocity increment of approximately 460 ft/sec toward the center of the Moon is imparted to the LEM. This action injects the LEM into a transfer ellipse that takes it to an altitude of 50,000 feet at perilune, with an orbital period equal to the circular orbital period of the C/M-S/M in its parking orbit. This equal-period orbit provides the LEM with an automatic (without propulsion) rendezvous point with the C/M-S/M in the event of an abort, as well as permits the C/M-S/M to follow the LEM optically down to perilune in a normal mission. For an abort situation, about two hours after the LEM deorbit maneuver, the two vehicles will meet again. The C/M-S/M has chase capability, and if at this time, a 460 ft/sec velocity increment toward the center of the Moon is imparted to the C/M-S/M, it will be placed in the same orbit with the LEM. The C/M-S/M can now actively rendezvous with a disabled LEM.

The over-all mission abort flexibility feature permits the astronauts to abort anytime up to the actual lunar landing. Figure 15 indicates points

along the Apollo Earth-to-Moon trajectory where it is possible to abort the mission.

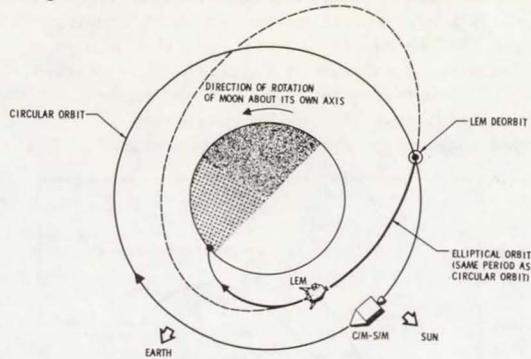


Figure 14. Equal Period Orbits

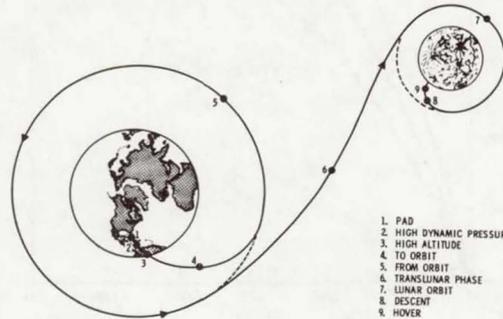


Figure 15. Abort Opportunities

Of the crew safety design features that do manifest themselves in particular pieces of hardware, the most obvious one is the launch escape system (LES). Although the Mercury also utilizes a launch escape rocket, the larger size and more stringent abort requirements for the Apollo make this LES unique. Paraglider and ejection seats are used in the Gemini, but they are considered too heavy for incorporation into the Apollo program. The Apollo LES is designed for abort on the launch pad, during high dynamic pressure, or at high altitude.

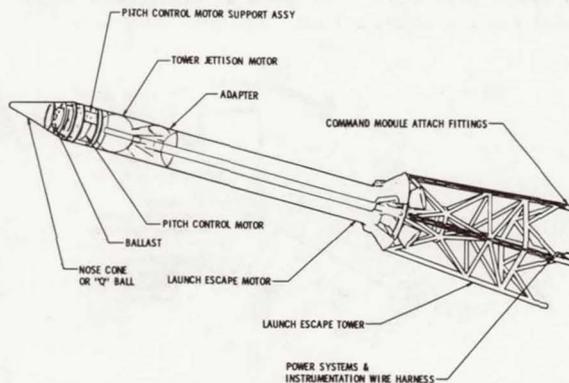


Figure 16. Launch Escape System

Figure 16 shows the basic construction of the LES. Titanium is used for the tower because of

its light weight and high structural strength. The thrust of the launch escape motor is about 150,000 pounds. A pitch control motor having an impulse of nearly 1700 lb-sec is used to pitch the LES over for pad abort. As shown in Figure 17, the system is capable of carrying the C/M to a minimum altitude of 4000 feet at 3000 feet downrange. The minimum safe range at touchdown is about 2000 feet.

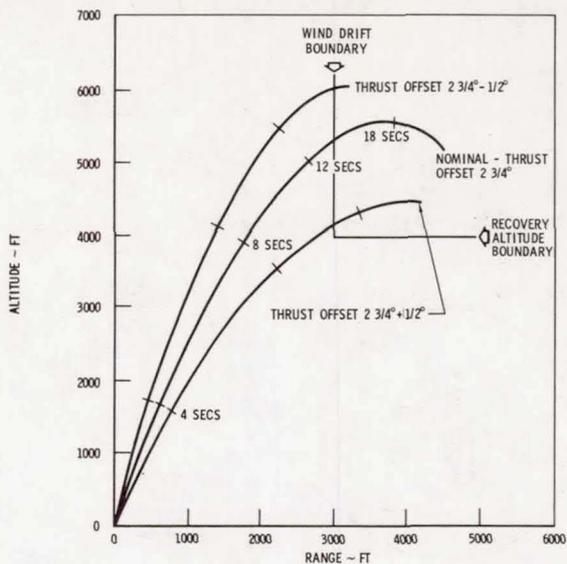


Figure 17. Pad Abort Trajectories

In a normal launch, the LES is jettisoned shortly after ignition of the second stage. Unlike the Mercury, which uses a Marman band for the launch tower separation, the Apollo uses explosive bolts. (These bolts are unique in that there are provisions for loading or unloading the explosive charges.) During LES jettison, there is a possibility that the jet plume might damage the windows of the C/M. Partly because of this reason, but mainly because of the adverse effects from aerodynamic heating during atmospheric exit and entry, the windows have covers. Figure 18 illustrates the C/M window configuration.

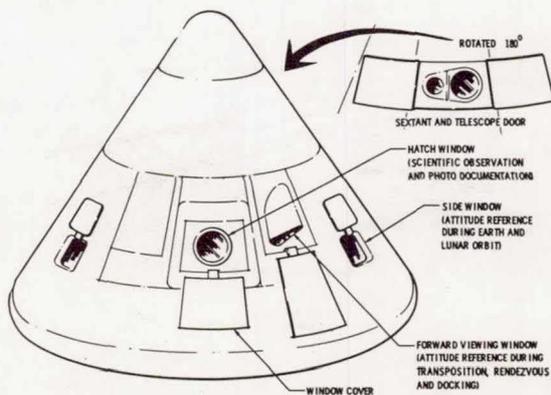


Figure 18. Command Module Window Configuration

For a launch from Cape Canaveral, a high-altitude abort (about 180,000 feet) would force the C/M to land in the ocean. Although San Antonio, Texas, and Woomera, Australia, are being considered for the primary landing sites, the possibility of a water landing requires that the C/M be designed for landing on either land or water. By way of comparison, the Gemini also has a dual landing capability. The Mercury, however, has a water landing capability only. Because of the offset c.g., the C/M has two stable orientations in water. These orientations are shown in Figure 19. As designed, position 1 is the more stable of the two because of the geometry of the C/M and the c.g. location with respect to the water. If the c.g. were low enough or sufficiently offset, the C/M would float in only one orientation.

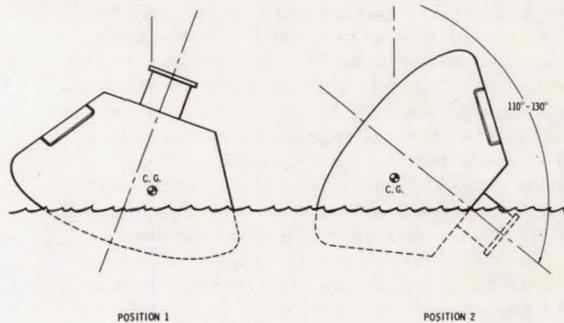


Figure 19. Command Module Flotation Positions

During a high-altitude abort, tumbling may cause the C/M to come in apex forward. In order to eliminate this apex-forward trim point, which is not acceptable from a crew safety point of view, two strakes are installed on the C/M. Although the final size and shape of the strakes are not firm, their approximate geometry and location are shown in Figure 20.

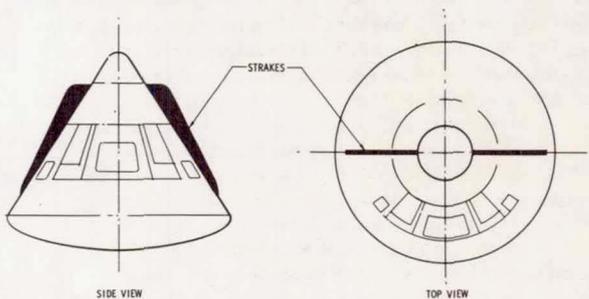


Figure 20. Command Module Strakes

A critical phase of the Apollo mission is the Earth landing of the C/M, whether the landing is being made in connection with an abort or a return from a lunar mission. Whatever the case may be, the Earth landing system must reduce the landing speed of the C/M to assure the safety of the astronauts. Unlike the Mercury, which uses a single

main parachute, or the Gemini, which uses a paraglider for the Earth landing, the C/M deploys three main parachutes, any two of which will land the C/M without exceeding emergency limits. The three-chute system was chosen because of its light weight and high reliability.

Figure 21 illustrates the operational sequence of chute deployment. The normal rate of descent of the C/M with all three parachutes deployed will be approximately 24 ft/sec; the emergency descent rate with two parachutes opened will be nearly 30 ft/sec. A couch impact attenuation system is used to reduce the landing impact. As illustrated in Figure 22, the system consists of hollow struts filled with crushable honeycomb that is arranged to fold like a telescope upon landing.

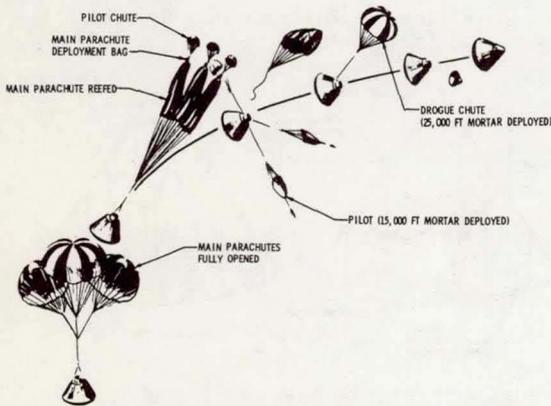


Figure 21. Earth Landing System

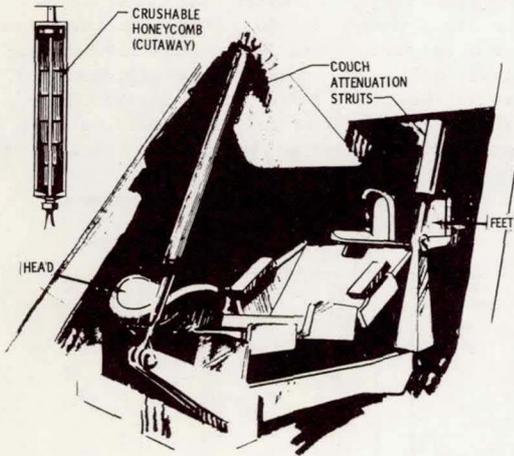


Figure 22. Couch Impact Attenuation System

From the standpoint of mission success (probability of success = 0.900) as well as crew safety (probability of safety = 0.999), a high over-all system reliability is mandatory. One way of assuring high reliability is to incorporate component or system redundancies where practicable. An example is the S/M propulsion engine

shown in Figure 23. This is a single swiveled-nozzle engine that must be operable at any time throughout the entire flight. Multi-engine configurations were considered for the S/M, but, based on factors of weight and reliability, it was decided to use a single engine. As shown in Figure 24, the service propulsion propellant system uses a series feed. In order to achieve a high engine reliability, double series and parallel regulator and check valve systems are deployed in the fuel system. This redundancy technique safeguards against possible fail open or fail close situations.

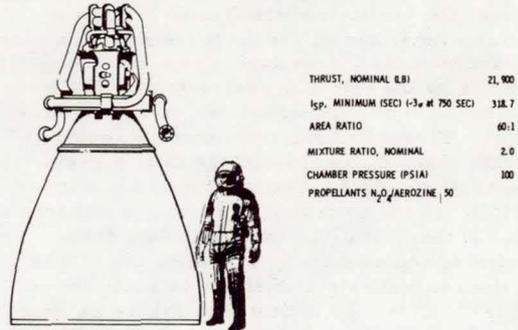


Figure 23. Service Propulsion Engine Configuration

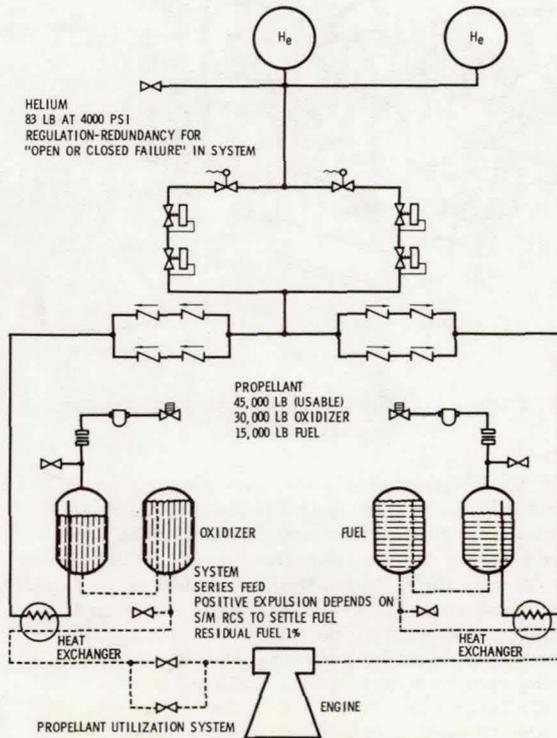


Figure 24. Service Propulsion System Schematic

three batteries. These units constitute the electrical power sources (Figure 29). While it is clearly desirable to have all three fuel cells operating, any two of these cells will satisfy the mission requirements. The three batteries located in the C/M are for use during Earth entry, but they can be used at anytime in the event of an emergency.

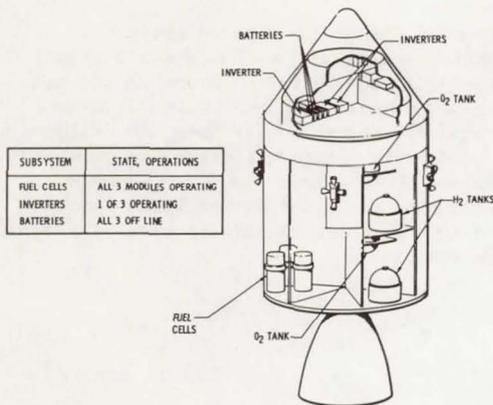


Figure 29. Electrical Power System

The Apollo spacecraft, traveling to and from the Moon, is placed in a radiation environment that can produce surface temperature variations from 250 to -290 F, depending upon the orientation of the spacecraft to the sun. Lengthy exposure to these temperatures can be avoided by properly controlling the orientation of the vehicle. This method of solution, however, is not desirable, and the spacecraft is consequently being designed to withstand temperature extremes for various orientations of the vehicle with respect to the sun.

In addition to the requirements for a habitable spacecraft, there also exists a requirement for suitable communication with the Earth, which is essential to the well-being of the astronauts as well as to mission success. The various antenna equipment located in the C/M and S/M are illustrated in Figure 30. For distances greater than 40,000 miles from the Earth, the 2-kmc high-gain antenna is used in transmitting signals to the Deep Space Instrumentation Facilities (DSIF) located at Goldstone, California; Woomera, Australia; and Johannesburg, Africa. The vhf omniantenna is used with the Ground Operational Support System (GOSS) for near-Earth communication. The frequencies will be the same as those now used on the present GOSS complex for Mercury. A design feature of the communication system is that voice communication between the spacecraft and the Earth is available almost continuously. Blind spots will occur during certain phases of Earth operations and when the spacecraft is traversing the back side of the Moon.

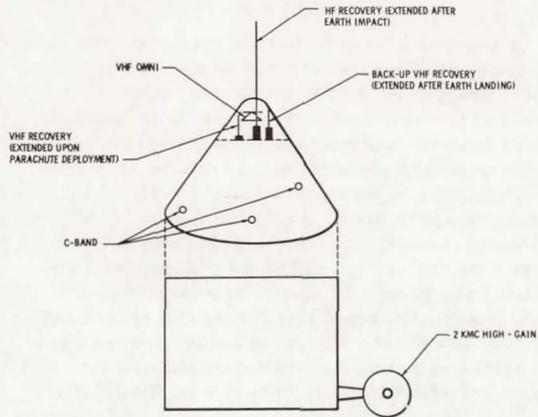


Figure 30. Antenna Equipment

A final design feature to be presented in this paper is the personal communication assembly (Figure 31). The assembly consists of a bump hat, a microphone with amplifier, and earphones. It is worn by the astronauts when they are not in their spacesuits. Identical microphones and earphones are incorporated in the helmet of the spacesuit. These components are compatible with hardware or wireless communication equipment. Communication within the cabin is achieved through the intercommunication system, using a hardwire plug-in. Radio frequencies are used for voice communication exterior to the spacecraft. This personal communication system is especially required during the actual exploration of the Moon. It is mandatory that the astronauts, one of whom will be walking on the lunar surface, be in voice contact with one another.

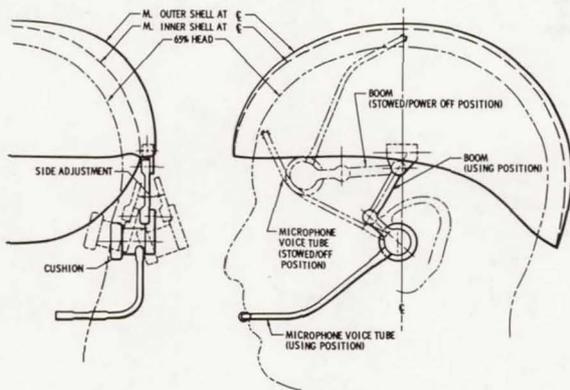


Figure 31. Communication Assembly—Personal

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Concluding Remarks

A number of Apollo design features have been discussed to illustrate the broad spectrum of the Apollo spacecraft design problems. Not all the technical problems have been covered. Each design feature, before final incorporation, must endure stringent experimental tests to verify its acceptability. There will be flight tests of the launch escape system, using the Little Joe II booster to investigate aborts at high dynamic pressures and at high altitudes. There will be Saturn I and Saturn IB Earth-orbital missions for flight qualification tests of the Apollo spacecrafts. Aircraft drop tests are being made to investigate the performance of the Earth landing system, and drop tests of boilerplate versions of the C/M are

being made to assess landing impact loads. The flotation and stability of the C/M have been explored by dropping and towing boilerplate versions of the C/M in water.

Some of the design features presented undoubtedly will be modified as a result of new experimental data and information. In addition, some new problems will arise that will dictate other design features. Therefore, the design must be flexible enough to incorporate changes as needed. At this time, there is no known technical reason why the United States cannot successfully complete the Apollo mission within the present decade.

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GEMINI LAUNCH VEHICLE

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Introduction

This paper presents a broad technical description of the changes made to the Titan II ICBM to enable it to perform the Gemini mission. In effect these changes created an essentially new product, the Gemini Launch Vehicle.

The data presented in this paper has been collected from numerous program documents.

Program Objective

The purpose of this program is to develop launch vehicles which will place the Gemini Spacecraft in trajectories designed to meet the following operational objectives:

- (1) Perform a 14-day earth orbital flight.
- (2) Demonstrate that the spacecraft can rendezvous and dock with a target vehicle in orbit.
- (3) Develop simplified spacecraft and launch vehicle countdown techniques in order to optimize the rendezvous mission.
- (4) Develop a fully reliable man-rated launch vehicle system.

Mission and Performance

Mission

The objective of the basic launch vehicle is to inject the spacecraft into orbit at an altitude of 87 nautical miles with sufficient overspeed to maintain a perigee of 87 nautical miles and an apogee of 161 nautical miles.

The general trajectory mechanization for the Gemini Launch Vehicle is similar to that used on the basic Titan ICBM, except for inclusion of a variable launch azimuth capability which has been added to meet the conditions imposed by the rendezvous missions.

Sequentially, the Gemini launch is characterized by an engine start signal, followed by a 1.08-second span in which engine thrust is built up to 77%. At that point, the Thrust Chamber Pressure Switch (TCPS) activates a two-second timer and, at the end of that period, the launch bolts are blown and liftoff begins. Then follows a vertical rise of approximately 20 seconds. During the vertical rise, the roll program is inserted to obtain the desired launch azimuth. The first of three open loop pitch commands is initiated approximately 20 seconds after liftoff in order to approach a zero lift trajectory during the Stage I flight regime. Figures 1 and 2 show the results of this type of trajectory on a few of the basic nominal design parameters. As in Titan II, a fire-in-the-hole technique is used to separate the first and second stages.

Sustainer flight is guided by a closed loop Radio Guidance System (RGS) which employs an explicit guidance law similar to that used during the Mercury-Atlas program. Figures 1 and 2 show the characteristics of this portion of the trajectory. Injection conditions are supplied by a velocity cutoff signal which is activated through the guidance system at the required attitude and altitude.

Performance

The performance capability of the Gemini Launch Vehicle is shown as a function of altitude and velocity in Fig. 3. For the mission objectives just described, the vehicle is capable of launching a payload weight greater than the combined weight of the Gemini Spacecraft with the adapter.

Fundamentally, the injection altitude chosen for the launch vehicle is governed by the design premise that minimum modifications will be made to the basic Titan II structure. Such parameters as aerodynamic heating, first-stage dynamic pressure, staging dynamic pressure and minimum elevation angle required for guidance were considered in determining this injection altitude (Fig. 4). A concession was made to the flight loads criteria in that the wind environment used for the Gemini Launch Vehicle is reduced in comparison to that normally used on the SM68B vehicles. Explicitly, Avidyne winds are used in this design application as representative of the environment experienced at the Atlantic Missile Range. Dynamic pressure in the first-stage regime is in excess of that used in SM68B vehicle design. Aerodynamic heating limits, which are derived from SM68B performance, and the minimum angle required for guidance provide the constraints which limit the injection altitude to approximately 87 nautical miles.

Description of Changes From Titan II

As has been mentioned, the Gemini Launch Vehicle is a version of the Titan II. The differences between the two vehicles can be categorized into three classes:

- (1) Changes needed to physically adapt the launch vehicle for the spacecraft.
- (2) Changes required to accomplish the mission of accurately injecting a spacecraft into an 87-nautical mile orbit with enough overspeed to achieve a 161-nautical mile apogee.
- (3) Changes or additions made because men are part of the payload.

In Class 1, the diameter of the top of the vehicle has been increased to 10 feet. No other

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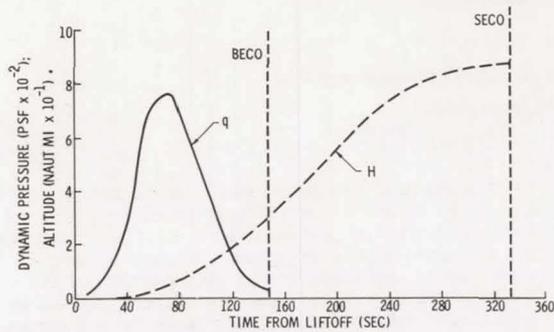


FIG. 1. ESTIMATED TRAJECTORY CHARACTERISTICS

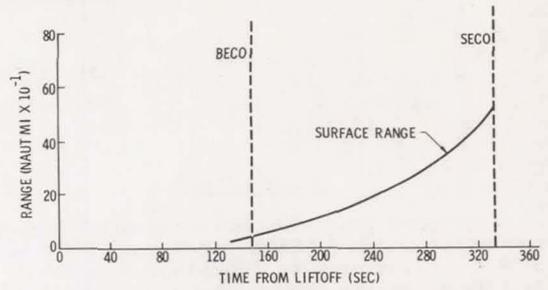


FIG. 2. ESTIMATED TRAJECTORY CHARACTERISTICS

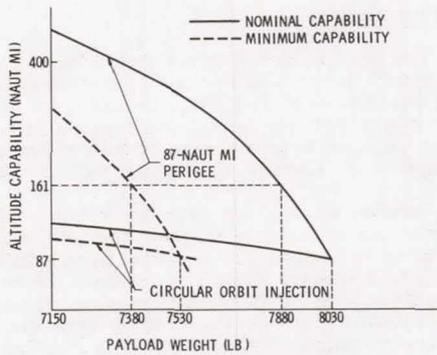


FIG. 3. PERFORMANCE CAPABILITY

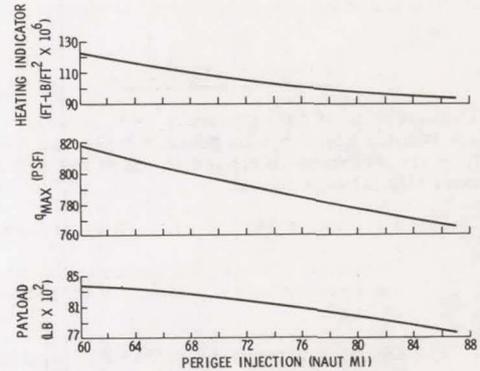


FIG. 4. INJECTION ALTITUDE PARAMETERS

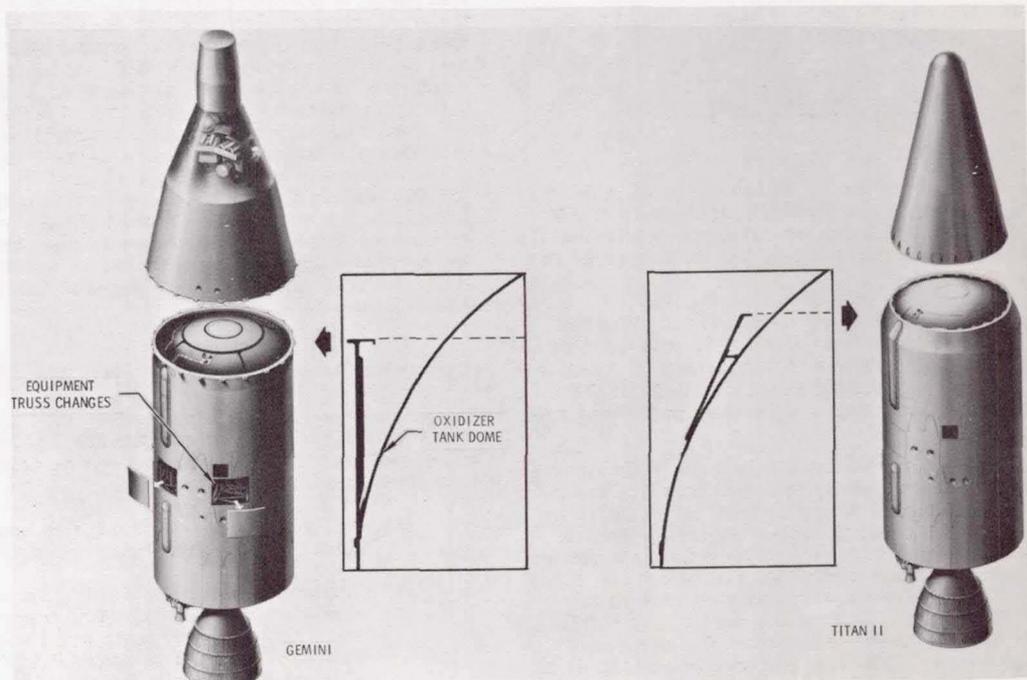


FIG. 5. CLASS I CHANGES

basic changes are required (Fig. 5) because the weight of the spacecraft is less than the maximum warhead weight carried by Titan II, and the trajectory flown will not impose loads which exceed those for which the ICBM was designed.

While some refinements were required, the environment and criteria used for the structural design of the Gemini Launch Vehicle are essentially those of Titan II. Figure 6 shows four major trajectory parameters which directly affect

the vehicle structural design. Dynamic pressure (q) and axial acceleration are essential to loads calculations, while structural heating is dependent upon the altitude-velocity relationship. The flight path shown in Fig. 6 is one of the numerous trajectories studied in defining the Gemini Launch Vehicle performance requirements. This trajectory is based upon nominal conditions for a 7400-pound payload injected at an orbital altitude of 87 nautical miles at perigee.

All load and structural heating calculations were obtained by using the atmospheric properties given by the 1959 ARDC model atmosphere (NASA Technical Note D595). Figure 7 presents the ground and flight wind profiles used in the loads calculations; as shown, both ground and flight winds represent 1% risk values. The ground wind profile, which is used for prelaunch and launch loads development, is based upon climatic data for Patrick Air Force Base as interpreted by Geophysical Research Directorate, Hamson Field, Bedford, Massachusetts. The first two-thirds of the wind profile is applied as a steady wind condition, while the final one-third is applied as a gust. The flight winds used are those developed by Avidyne for the winter months at Cape Canaveral. A 1-cosine, 20-fps, true gust is added to the Avidyne profiles at any given altitude. In the example shown, the predominant wind is from the west.

Figure 8 shows the net effect for the critical air load condition. The Gemini Spacecraft-Launch Vehicle configuration creates a different air load distribution at the forward end, and this different distribution causes higher internal structural

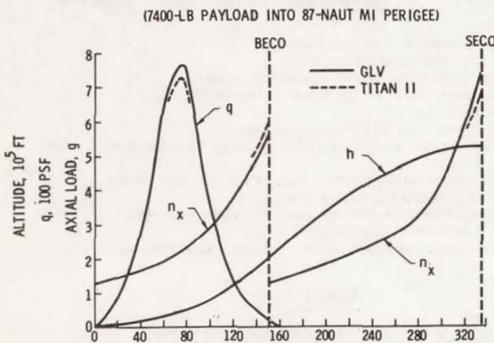


FIG. 6. STRUCTURAL DESIGN TRAJECTORY PARAMETERS

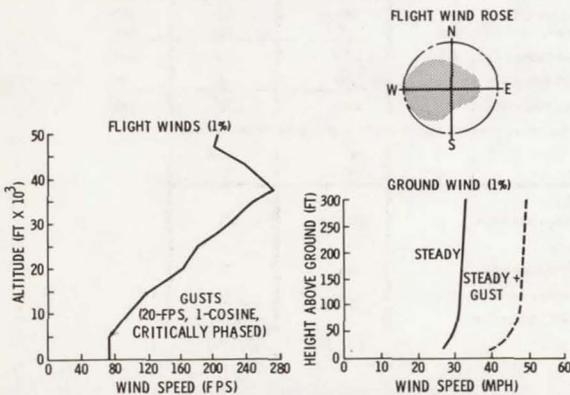
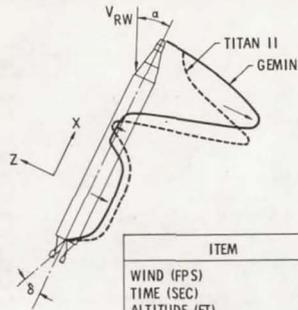


FIG. 7. DESIGN WIND ENVIRONMENT

stresses. These differences are offset by using a lower engine gimbal angle, 3.5 degrees instead of 5 degrees (Fig. 9). The substitution is justified because the control requirements for the most dispersed cases are less than 3 degrees.



ITEM	GEMINI	TITAN II
WIND (FPS)	AVIDYNE (259)	SISSEWINE (300)
TIME (SEC)	69	63.5
ALTITUDE (FT)	36,000	31,000
MACH NO.	1.44	1.27
DYNAMIC PRESSURE (PSF)	685	705
WEIGHT (LB)	216,000	224,000
ANGLE OF ATTACK (α) (DEG)	-10.2	-11.5
ENGINE GIMBAL ANGLE (δ) (DEG)	-3.5	-5.0
AXIAL LOAD FACTOR (γ_x) (g)	2.03	1.88
LATERAL LOAD FACTOR (γ_z) (g)	-0.31	0.365

FIG. 8. MAXIMUM AIRLOAD CONDITION

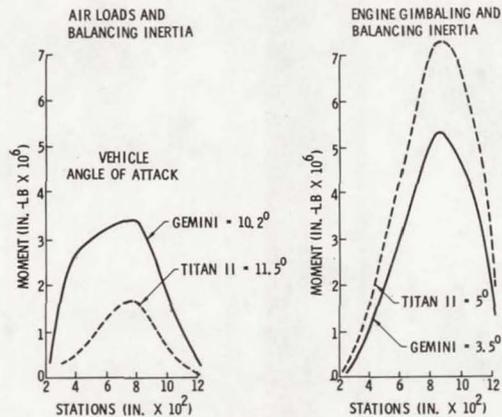


FIG. 9. VEHICLE BENDING MOMENT

Class 2 modifications (Fig. 10) deal with those changes needed to increase the payload capability for the required orbit. The following steps were taken to meet these new requirements.

- (1) Delete the Titan II Inertial Guidance System. The Gemini Launch Vehicle system uses a Three-Axis Reference System during the first stage flight and a Radio Guidance System during the second stage. Since the GE Mod III-F is used as a tracking and impact predictor for Titan II, a complete Radio Guidance System (GE Mod III-G) was developed by simply adding a decoder.
- (2) Use MISTRAM only on the Gemini Launch Vehicle. Titan II uses both MISTRAM and Azusa tracking equipment.
- (3) Remove the Titan II retro and vernier rockets.
- (4) Change the instrumentation system from a 0- to 40-millivolt system, to a 0- to 5-volt system.

TABLE 1

COMPARISON OF THREE STAGE II CONFIGURATIONS FOR THE GEMINI MISSION

Vehicle Part	Titan II N-11 (lb)	GLV No. 5 with RGS & MI STRAM (lb)	GLV with IGS & MI STRAM (lb)
Body	2,262 ⁽¹⁾	2,262	2,262
Separation and destruct	66	82	82
Propulsion	1,328	1,332 ⁽²⁾	1,332 ⁽²⁾
Power generation	100	104	104
Static inverter	0	68	0
Orientation controls	338 ⁽³⁾	138	138
Mod 3-F	32	33	0
Decoder	0	14	0
TARS	0	72	0
Autopilot No. 1	38	38	38
Autopilot No. 2	38	38	38
Adapter	0	17	0
IGS	282	0	282
MI STRAM	30	30	30
Azusa	29	0	0
Command receivers	50	50	50
Strobe light	72	0	0
Wire and bracketry	528	254	440
Environmental control	24 ⁽⁴⁾	14	14
Instrumentation and telemetry	1,074	684	753 ⁽⁵⁾
MDS	96 ⁽⁹⁾	96	96
Unaccountable variation	-58	0	0
Translation system	120 ⁽⁶⁾	0	0
Total Weight Empty	6,449	5,326	5,659
Residual Propellant	445	400 ⁽⁷⁾	400
Burnout weight	6,894	5,726	6,059
Disposable propellants	58,896	60,083 ⁽⁸⁾	60,083
Engine bleed	13	11	11
Solid propellants	145	0	0
Starter grain	3	3	3
Gross Weight	65,951	65,823	66,156

NOTES:

- (1) Normalized to remove N-11 warhead adapter.
- (2) Revised Gemini engine specification weight.
- (3) Stated with vernier system weight included (200 pounds).
- (4) Reflects ducting in equipment compartment for air conditioning while the vehicle is on pad.
- (5) Includes AC-Spark Plug (IGS) telemetry packages.
- (6) Used to rotate the burned out Stage 2 out of the flight path of the payload after separation.
- (7) Based on propellant loading statement issued 20 February 1963. These values are nominal and include mean outage.
- (8) Based on cold propellant loading statement issued 20 February 1963.
- (9) Included to normalize comparison basis.
- (10) All weights include malfunction detection and redundancy provisions.

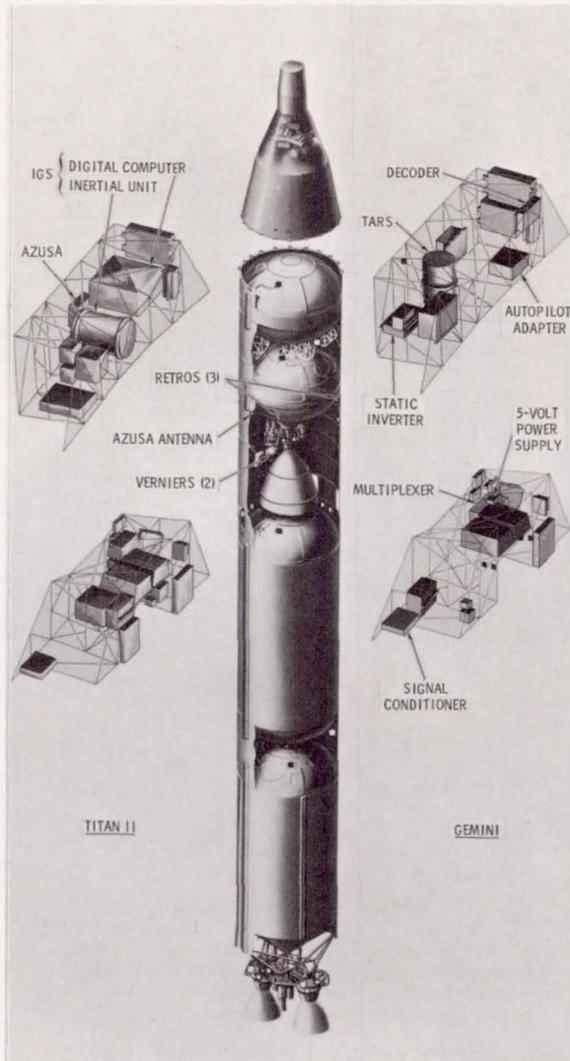


FIG. 10. CLASS 2 CHANGES

Figures 11 and 12 show the modifications made to the guidance and instrumentation trusses in order to adopt the Titan for the Gemini mission.

Table 1 shows three Stage II configurations which have the necessary equipment to perform

TABLE 2

INCREASED PROPELLANT AND PAYLOAD

Items	Stage I (lb)	Stage II (lb)	Total (lb)
Cold propellant	2090	900	2990
Tank volume considerations	1260	200	1460
Total Loaded	3350	1100	4450
Nonusables, transients and bias	650	60	710
Total Steady-State	4000	1160	5160

The increase in payload capability which results can be stated as follows:

Stage	W (lb)	Payload (lb)
Stage I		
△ Propellant weight	4000	133
Stage II		
△ Propellant weight	1100	73
△ Empty weight		1168
△ Total Payload Gain		1374

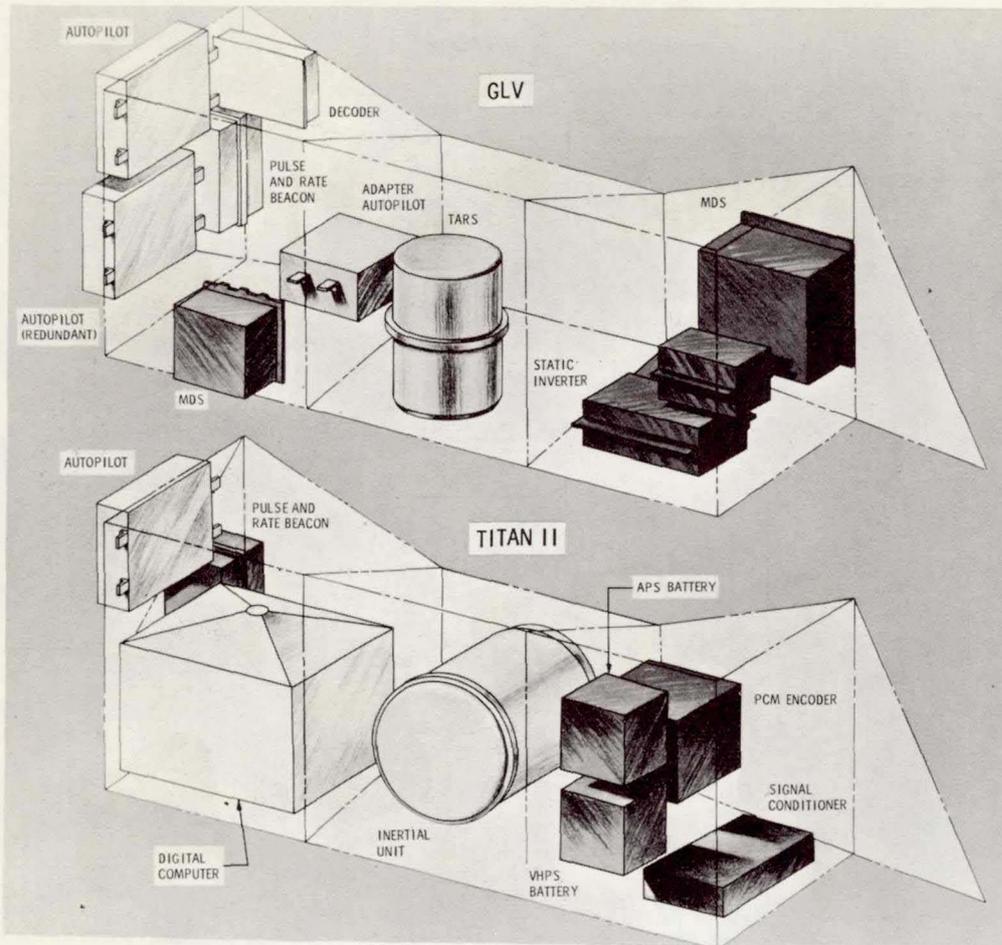


FIG. 11. EQUIPMENT TRUSS NO. 1 (GUIDANCE)

a Gemini mission. The tabulation indicates that a payload increase of 1168 pounds was realized because of the differences between the Titan II research and development ship No. 11, which served as the base for the Gemini Launch Vehicle, and the Gemini configuration finally chosen. In addition, it is shown that there is a payload differential of 264 pounds between a stripped Titan II with inertial guidance and the final Gemini Launch Vehicle configuration.

Table 2 shows the increased payload and propellant that the Gemini Launch Vehicle is capable of handling. There are four reasons why the Gemini Launch Vehicle can carry this additional propellant: (1) calibrated tanks with nominal rather than minimum values are used; (2) The area between the prevalves and thrust chamber valve can be used for propellant storage; (3) a more accurate loading system is provided; and (4) lower propellant temperatures are maintained. Table 2 shows how the additional 5160 pounds of propellant which can be loaded on the Gemini is distributed.

The preceding tabulation explains the payload gains realized to date; it does not include additional gains that could be effected through:

- (1) Reducing ullage requirement and loading more propellant.

- (2) Using selective injectors to bring about I_{sp} gains.
- (3) Using chambers selected to optimize burning mixture ratios.
- (4) Devising additional means of reducing weight.

The changes in instrumentation hardware, some of which resulted in the weight savings just discussed, are summarized in Table 3 and are schematically indicated in Fig. 13. The summary of all the Class 2 changes is shown in Fig. 10.

Class 3 modifications (Fig. 15) deal with those changes which have been introduced to ensure the safety of the two astronauts who will be aboard the spacecraft. The Man-Rating and Pilot Safety Program which was developed to do the job involves many considerations. These are summarized in Fig. 14.

Gemini changes related to hardware are considered under the category of system design. Specifically, the major considerations made in this category can be delineated as:

- (1) Addition of a Malfunction Detection System (MDS).

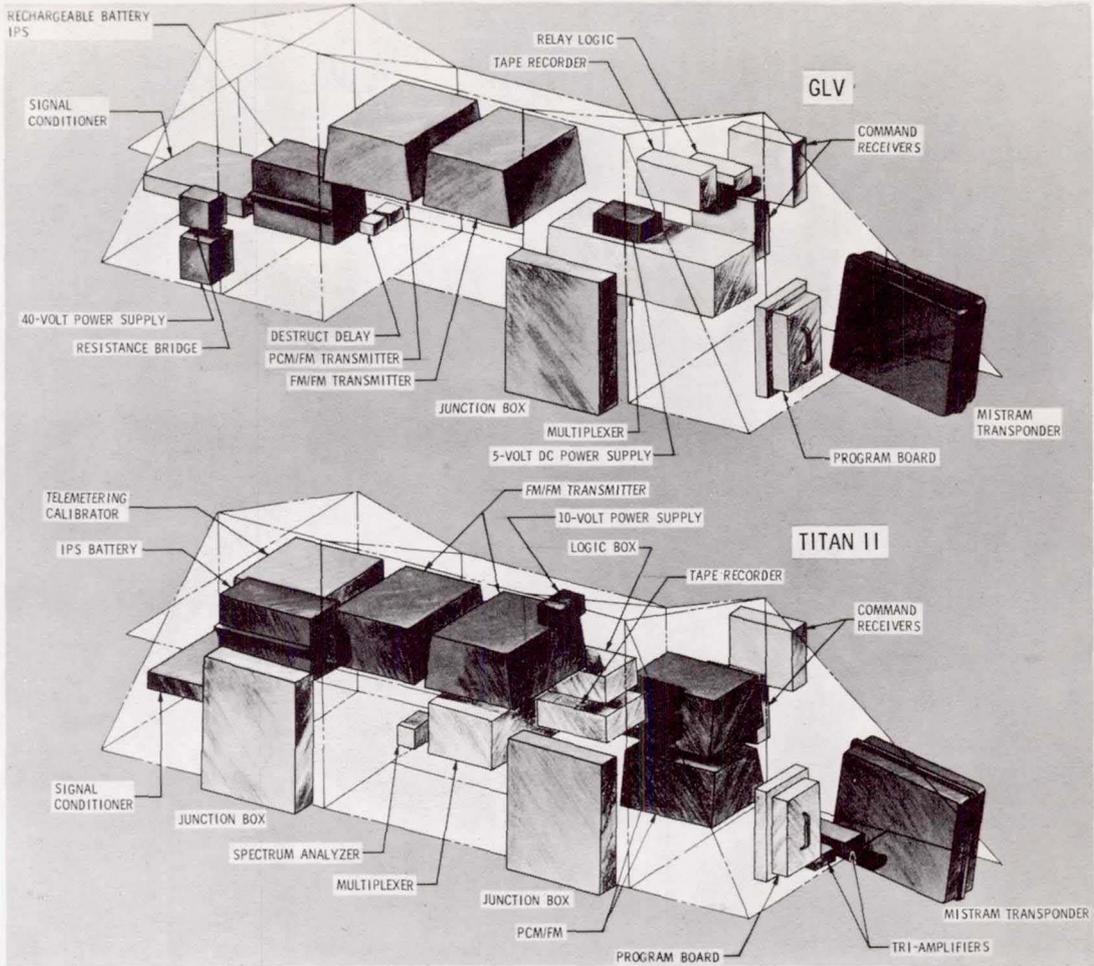


FIG. 12. EQUIPMENT TRUSS NO. 2 (INSTRUMENTATION)

TABLE 3

INSTRUMENTATION SYSTEM

Components	Same as Titan II	Extent of Changes: Remarks
PCM/FM transmitter	Yes	None: Denver supplied
FM/FM telemetry	Essentially (98%)	Five FM low level oscillators changed to high level: Denver supplied
PCM multiplexer	Essentially (90%)	Channel capacity, format, and sampling rates are same as Titan II. Changed input section for Gemini Launch Vehicle to high level, 5 volts. Weight saving on Gemini Launch Vehicle is 25 pounds
Power divider, 5 port	Yes	None
Antenna telemetry, 4 required	Yes	None
Diplexer	No	Repackaged to cover 2 RF links
Program board	Yes	None
Signal conditioner types of modules		
(1) 400-cps phase demodulator	No	Same as Titan II, except that the Gemini Launch Vehicle has a TARS package, while Titan II doesn't
(2) 800-cps phase demodulator	No	Titan I 400-cps phase demodulator has been modified for 800 cps
(3) 115-vac discriminator	No	Same as Titan II; required for Gemini Launch Vehicle because of 400-cps static inverter and TARS
(4) 26-vac discriminator	No	Titan II modified to give 5-volt instead of 40-mv output
(5) 400-cps frequency deviation	No	Same as Titan II; required for Gemini Launch Vehicle because of 400-cps static inverter and TARS
(6) DC amplifier	No	Same as Titan II; high level for current monitoring of IPS and APS
Transducers		
(1) Temperature sensor system	No	Same as Titan II; has a 5-volt output without signal conditioning, replaces thermocouples used on Titan II
(2) Static accelerometer	No	Unit has high level output: similar units on Titan II are low level
(3) Pressure transducer	No	Unit has high level output: Titan II uses low level; sensing element is solid state bridge
Airborne tape recorder	Yes	None
Connectors	Yes	None
Wire	No	Titan II uses twisted pair shielded for each measurement, while Gemini Launch Vehicle uses single conductor shielded. Weight saving 142 pounds

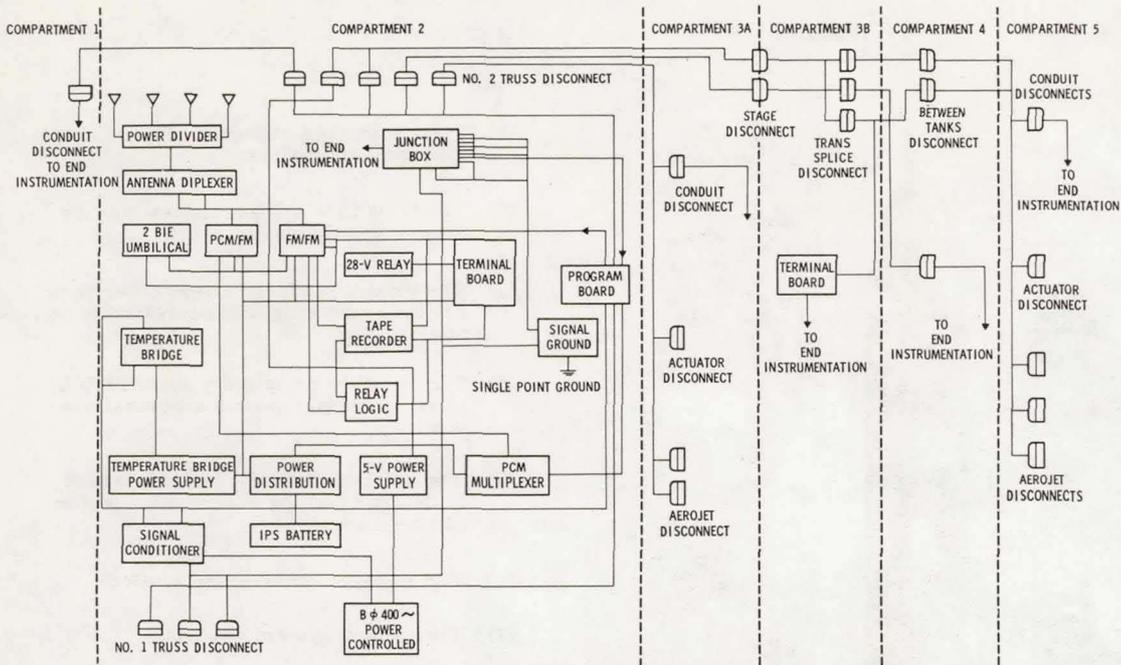


FIG. 13. INSTRUMENTATION SYSTEM

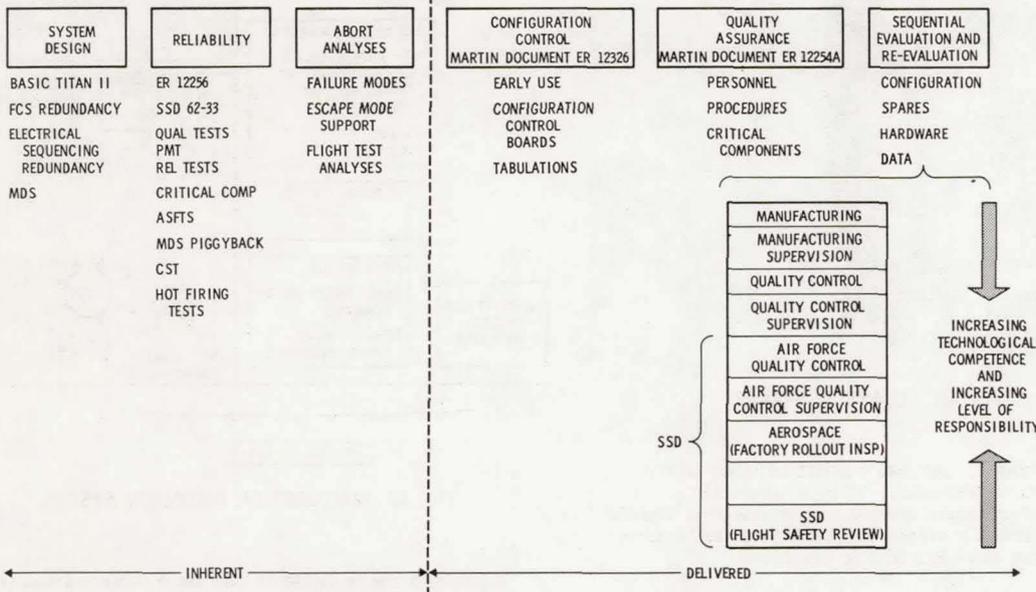


FIG. 14. MAN-RATING AND PILOT SAFETY

- (2) Addition of those features required to produce flight control system redundancy.
- (3) Addition of time delays in the flight termination system.
- (4) Addition of redundancy provisions in the electrical circuits of the flight sequencing system.

Malfunction Detection System (MDS)

Effective implementation of a Man-Rating and Pilot Safety Program, like the one shown in Fig. 14, will ensure a launch vehicle which will perform more reliably. Even though the goal is perfection, realistically, there is always some possibility of hardware failures. In order to minimize losses due to this possibility to the lowest attainable level, a highly sensitive Malfunction

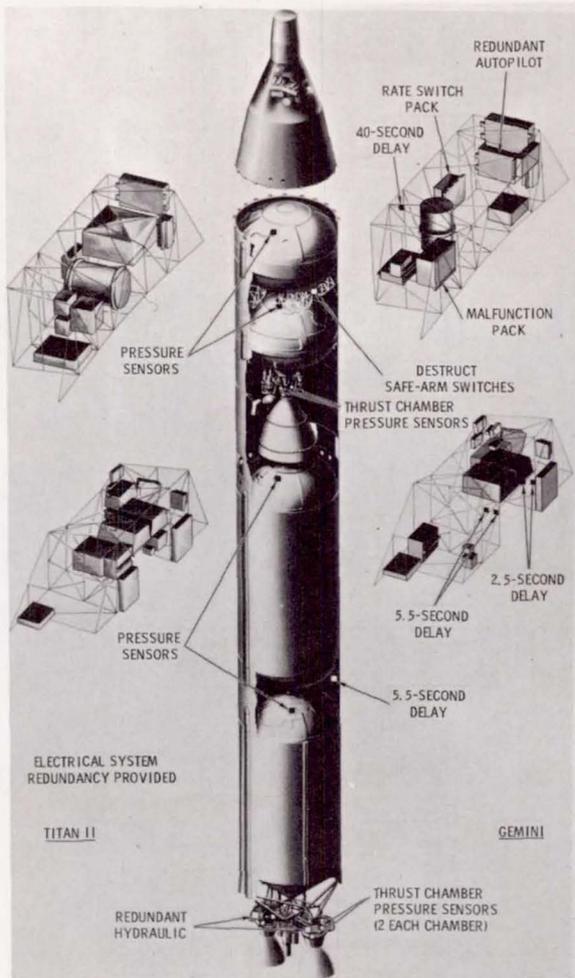


FIG. 15. CLASS 3 CHANGES

Detection System has been incorporated in the Gemini Launch Vehicle. This system (Fig. 16) provides information on those parameters which most significantly affect the safety of the astronauts and the success of the mission.

The fundamental question which must be answered in developing a Malfunction Detection System is, "How will the sensed information be used?" Stated simply, the question can be reduced to determining the degree of automatic action which should result; that is, should the sensed information cause automatic ejection or should the information be displayed to the pilots who would then decide what to do. Before a valid decision can be made, the following factors must be considered.

- (1) Time histories of launch vehicle action following anomalies.
- (2) The time in which anomalies may be sensed and displayed.
- (3) The extent to which "cues" other than hardware sensing will be available and useful.
- (4) The relative complexity and reliability of an automatic versus a manual system.
- (5) The astronaut's role: the role which is desired and the contribution which can be made.
- (6) The mission requirements effect.
- (7) The escape system concept.

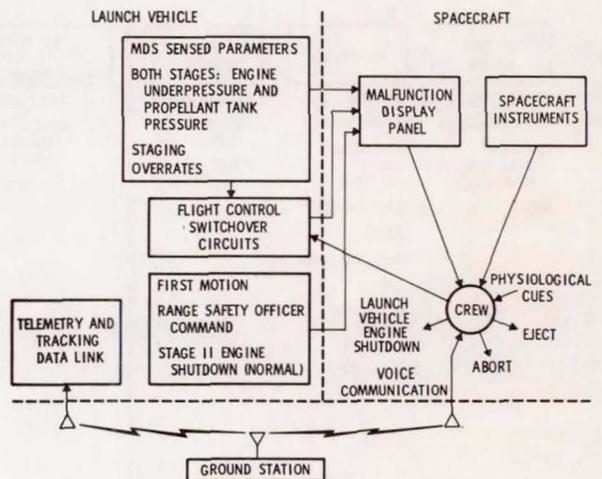


FIG. 16. MALFUNCTION DETECTION SYSTEM

Although these factors can be evaluated independently, many of them are necessarily inter-related. For example, in the case of the Gemini Launch Vehicle, Items 4, 5, 6 and 7 were inter-meshed and basic decisions in these areas indicated a need for a manual rather than an automatic abort system. However, this meant that Items 1, 2 and 3 had to be evaluated in order to determine whether a safe manual system could be developed. Once it was proven that such a system could be provided, the Gemini Malfunction

Detection System was implemented to provide information to the astronauts who must ultimately decide what action is to be taken.

Project Gemini's design philosophy is summarized effectively in a February 1963 article in "Astronautics and Aerospace Engineering" by Chamberlain and Meyer. An analysis of a few quotes from this article enables one to understand the need for a manual abort system.

The Atlas is so instrumented that it will automatically abort the Mercury Spacecraft if any one of a number of malfunctions is sensed in the launch vehicle. The automatic abort modes in Mercury are very complicated and have caused the loss of complete spacecraft in the early development unmanned flights. In each instance, had a man been on board, he could have manually salvaged the situation.

In Gemini, a launch vehicle malfunction activates lights and gages on the instrument panel and the astronauts exercise judgment as to the seriousness of the situation and the best procedure to follow during any special circumstances. With this sort of system, more than one cue can be used to verify an abort situation. Simulations reveal that in many cases, much reliance is placed on the audio-kinesthetic cues for this purpose. These cues are not only very reliable, but instill confidence in the pilots in the validity of the systems when they are checked by this means.

A further quote from this article shows that one of six primary objectives of the program is:

To perfect methods for returning and landing the spacecraft on a small preselected landsite. This objective involves re-entry control and a paraglider for spacecraft recovery. The ejection seats not only provide a substitute for a reserve parachute, but also provide an escape mode both early in flight and on landing.

This latter quote is offered to indicate some of the background that led to the choice of ejection seats as one of the escape modes. Their use and speed of reaction is one of the factors that was considered in deciding whether a manual abort system was feasible.

The factors just evaluated cover Items 4, 5, 6 and 7 of the characteristics which had to be considered in evaluating the desirability of a manual versus an automatic abort system. Logically, the next step in such an evaluation was to examine all possible malfunctions in order to determine the more critical malfunction times.

The first step in such an analysis was to determine the frequency of failures by systems. Primarily, this information was gathered by re-viewing Atlas, Titan I and Titan II histories. During these analyses, the following information was particularly sought:

- (1) Probability of occurrence
- (2) Mode of failure.
- (3) Time until critical limits are exceeded.

From these studies, a summary of what might be expected on the Gemini Launch Vehicle was prepared; the summary indicated the probabilities of malfunction by systems (Fig. 17). Each system was then considered independently, and the consequences of a failure at different times during the flight on better than 1000 analog simulations of this kind were made for the Gemini Launch Vehicle Program. Typical results of these studies are shown in Figs. 18, 19, 20 and 21. From these data, the time required to reach a critical limit

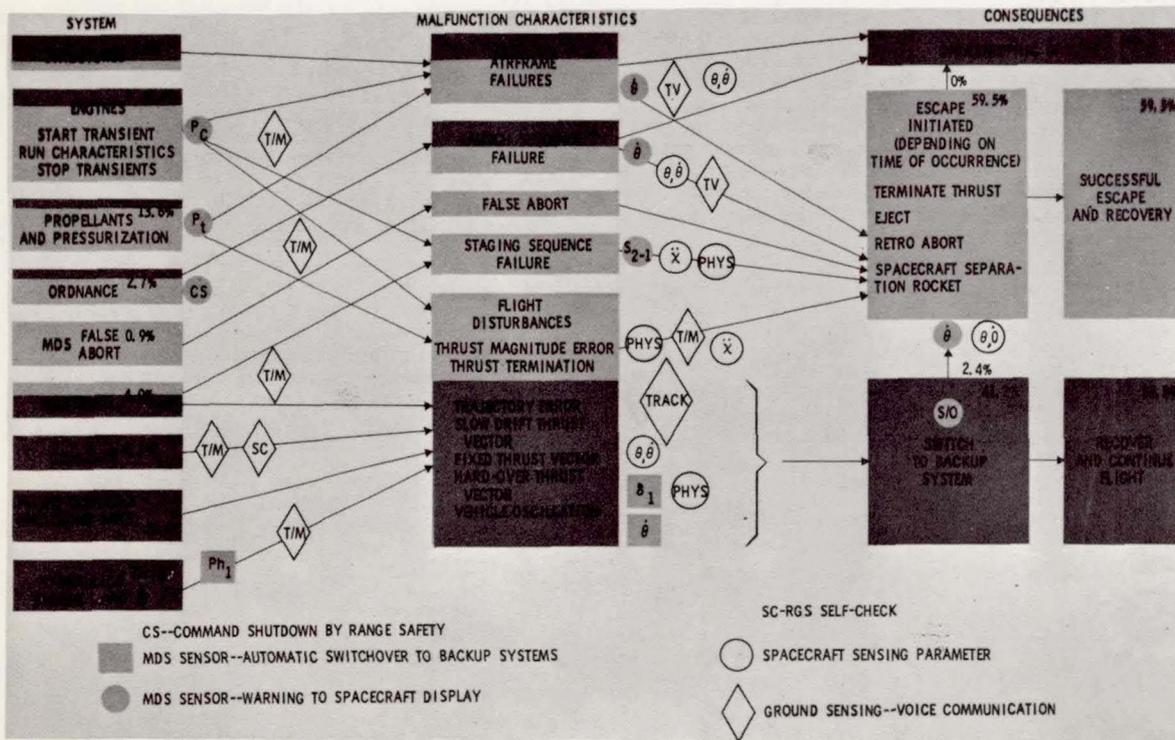


FIG. 17. ORGANIZATION OF MALFUNCTION PROBLEM

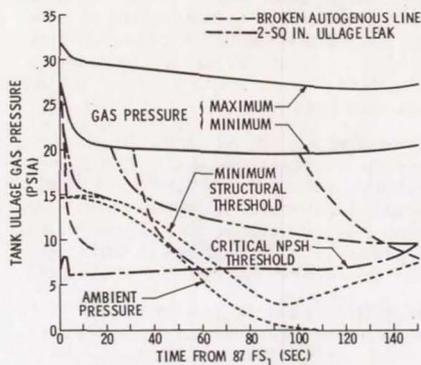


FIG. 18. STAGE I FUEL TANK MDS REQUIREMENTS

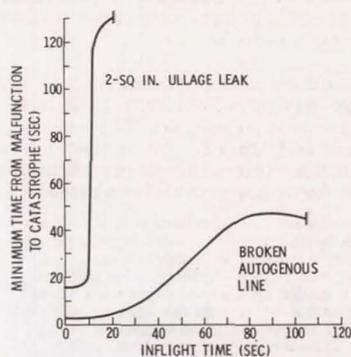


FIG. 19. STAGE I FUEL TANK TIME TO CATASTROPHE

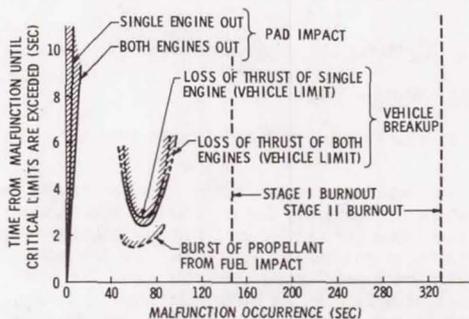


FIG. 20. INADVERTENT THRUST TERMINATION

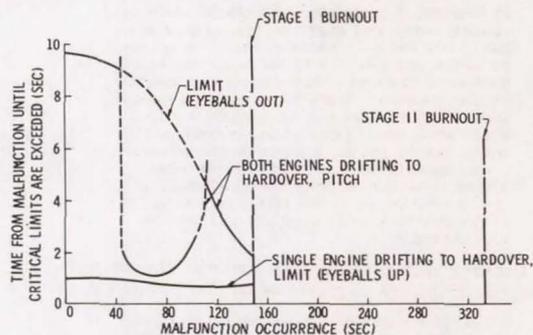


FIG. 21. VIOLENT CONTROL MALFUNCTION

was determined. For example, Fig. 20 shows that if an engine failure occurs at approximately 70 seconds, the vehicle would break up in approximately three seconds. With a manual abort system, the sensing, indication, reaction and escape actions would all have to occur within three seconds. The results of these analyses indicated that it is possible to react to all failures in a timely manner, with the exception of engine hard-over cases which will be discussed under Flight Control System Redundancy. From these analyses, it was determined that the following parameters must be monitored while the Gemini Launch Vehicle is in flight:

- (1) Four tank pressures (structural limit or minimum NPSH).
- (2) Engine chamber pressure switches set at 68% of rated thrust for Stage I and 65% for Stage II; this is equivalent to 550 psia \pm 30 psi for both stages.
- (3) Vehicle attitude rates.

	Stage I (deg/sec)	Stage II (deg/sec)
Pitch	+3.5, -4	10
Yaw	\pm 3.5	10
Roll	20	20

- (4) Staging signal: the light goes on at staging signal (87 FS₂, 91 FS₁) and

goes off at separation approximately 87 FS₂ + 0.6 second.

The tank pressure sensors provide analog signals to the spacecraft indicators. Redundant sensors, which are connected in independent, parallel circuits individually routed to the spacecraft, are supplied for each tank. All other sensors are bi-level. They are also redundant for each parameter, but, in this case, they are connected in series. Consequently, the contact of both sensors in the redundant pair must be closed before a signal is initiated (Fig. 22).

In addition to the parameters measured in flight, sensors have been added in those lines which contain the propellant tank pressurants. These sensors measure whether gas for the tank pressurization is being generated to a value which will be high enough to pressurize the tanks. The values sensed are:

Values	Stage I	Stage II
Fuel	50 \pm 4 psi	None
Oxygen	385 \pm 25 psia	None

If the sensed values are not high enough, an engine kill is initiated prior to liftoff.

In addition to the flight considerations, there are ground abort conditions which also had to be evaluated. These conditions are shown in Fig. 23.

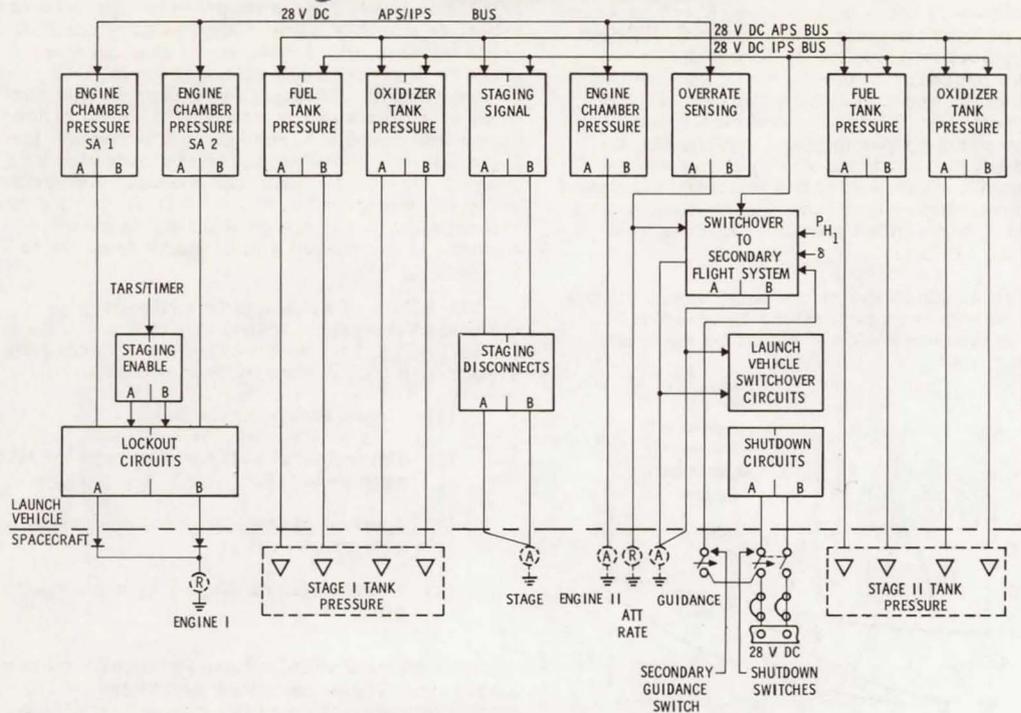


FIG. 22. DETAILED MALFUNCTION DETECTION SYSTEM

The figure shows that the Gemini recovery area is being cleared and leveled for recovery of the two Gemini pilots in the event of a pad abort. The legs of this triangular-shaped area are each 1000 feet long and the angle between them is 54 degrees. All elevated obstacles are being removed; even pad illumination lights will be installed flush in the ground. The highlighted area (dashed line) will be deluged with water in the case of booster explosion. In present Gemini capsule design, the pilot's seats are angled 9 degrees above horizontal and 12 de-

grees apart. The ejection motor on each seat will develop 2500 pounds of thrust and burn for 1 second; pilot should be clear of capsule 0.4 second after motor ignition. Barostats will activate seat-mounted chutes 3 seconds later when the pilots are about 300 feet above the ground. Pilots will have a maximum 5.5 seconds in which to initiate escape procedures after notification from Range Safety Officer of his intention to destroy a malfunctioning booster.

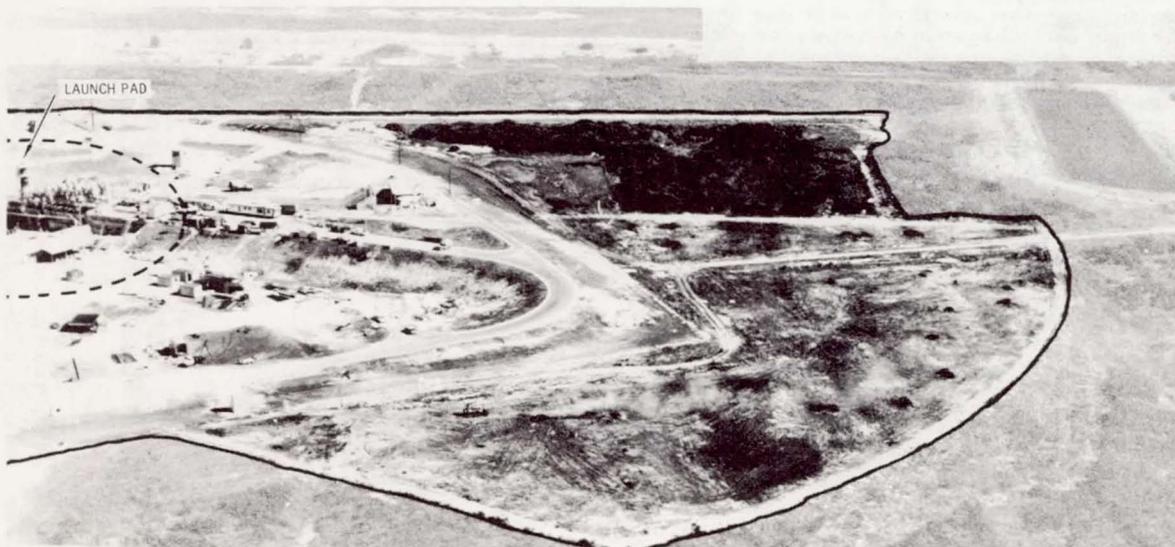


FIG. 23. COMPLEX 19 RECOVERY AREA

One switch will eject both seats. Ejection seats will be the primary escape mode up to 70,000 feet. After that, pilots will escape by firing the spacecraft's solid propellant retrorockets, each developing 2500 pounds, and separating the capsule from the launch vehicle. Pilots would then fly their capsule back to earth by paraglider. NASA, Martin and McDonnell are studying ways of pilot escape from the launch stand before the erector is dropped, preparatory to engine ignition. These include a cherrypicker, high-speed elevator, cork-screw type slide and lifelines.

The times at which the remaining escape modes (use of spacecraft retrorockets or longitudinal spacecraft maneuver rockets) would be used are shown in Fig. 24.

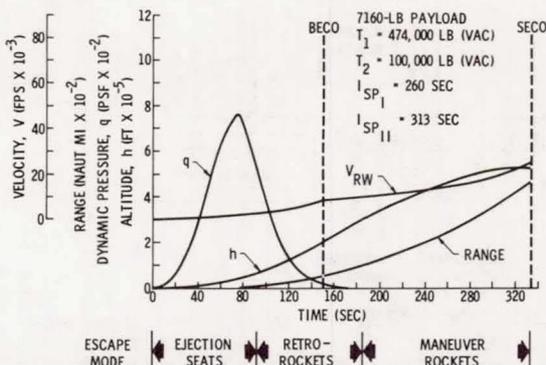


FIG. 24. ESCAPE MODE FLIGHT PARAMETERS

Flight Control System Redundancy

As previously indicated, analyses were made for a number of postulated malfunctions to determine how much time would elapse from the instant when a malfunction was sensed until critical limits were exceeded. These times were then examined to define whether there was sufficient time for pilot warning and reaction. The engine hard-over condition, that is a failure in the flight control system or hydraulics which causes or allows one or two engines of Stage I to drift hard-over, was examined carefully. Figure 21 shows the time histories accumulated during these analyses. As seen, it takes approximately 1.25 seconds to reach vehicle destruction if both engines drift to hard-over in pitch and one second or less to reach a physiological limit should a single engine drift hard-over and cause a yaw-roll buildup.

In order to determine whether there would be enough time for astronaut reaction for this and other cases, NASA decided to conduct a series of experiments. These were conducted at Chance Vought in a simulator where the malfunctions were simulated and response time measured. In all cases, except those for engines hard-over, there was sufficient time for positive astronaut reaction. In no case was the time for engine hard-over met.

These experiments showed that a manual abort system was desirable, possible and practical, except in the case of engine hard-over. The question then remained as to whether an

automatic abort should be provided for this condition or whether some compensatory method could be devised. A number of studies were made to determine the effect of various degrees of redundancy. These studies showed that the most effective system was one in which redundancy was provided from guidance through the flight control systems and to the hydraulics of Stage I (Fig. 25). With this system, the probability of an engine hard-over failure is reduced appreciably, while the probability of mission success is increased significantly from 90 to 93.6% (Fig. 26).

The effect of sensing and switchover to maintain the vehicle within structural limits is shown in Fig. 27. Switchover to the secondary system can be effected by four methods:

- (1) Command from the pilot.
- (2) Detection of vehicle overrate by MDS rate sensors.
- (3) Loss of Stage I primary hydraulic system pressure.
- (4) Positioning of Stage I hydraulic actuator.

Pilot command is initiated manually by the astronaut. These decisions are based on the pilot's interpretation of the spacecraft display,

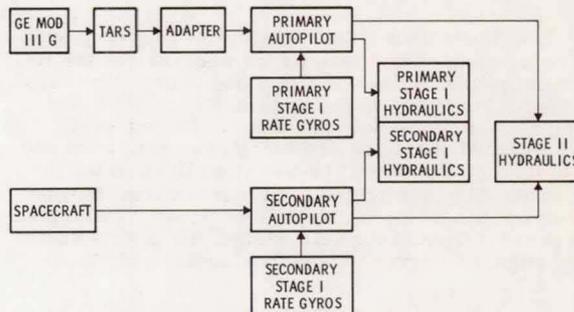


FIG. 25. REDUNDANT GUIDANCE AND FLIGHT CONTROLS SYSTEMS

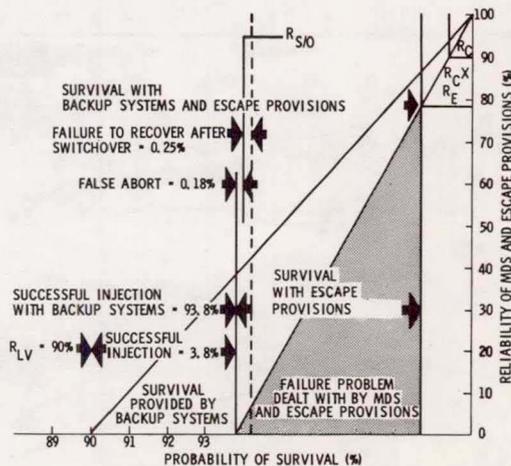


FIG. 26. SURVIVAL CONSIDERATIONS

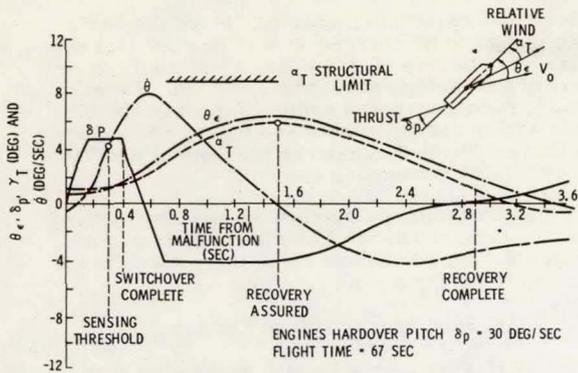


FIG. 27. SIMULATION OF CONTROL MALFUNCTION EVENT

plus information which he receives from the ground station. The MDS overrate sensors will automatically initiate a signal when the vehicle's motion exceeds a predetermined safe limit. In addition, the hydraulic pressure switch automatically initiates switchover when the pressure on the primary side is reduced to a preset value.

Each of these methods produces a signal which simultaneously energizes the hydraulic switchover valve solenoids in the Stage I hydraulic system, and a relay which switches the Stage II hydraulic system input signals from the primary to the secondary autopilot.

Flight Termination System

Except for the following differences, the

Gemini Launch Vehicle flight termination and destruct system (Fig. 28) is the same as that used on Titan II (N-1).

- (1) Crew safety switches have been added between the airborne 28-v d-c power supply and the destruct switches.
- (2) The 28-v d-c power is isolated from the destruct switches until after flight termination system shutdown command has been initiated.
- (3) Time delay relays have been added to prevent the flight termination system from giving a destruct command until 5.5 seconds have elapsed after the shutdown command has been initiated.
- (4) Time delay relays (5.5 seconds) have been added to the Stage I automatic destruct system; consequently, the system reacts only if there is an inadvertent separation of Stage I from Stage II during the boost phase.
- (5) Stage I is shut down and destroyed if it inadvertently separates from Stage II during boost phase.
- (6) The Stage I inadvertent separation destruct system is made safe at approximately 10 seconds prior to normal separation by independent signals transmitted from both the Three-Axis Reference System and 140-second timers.

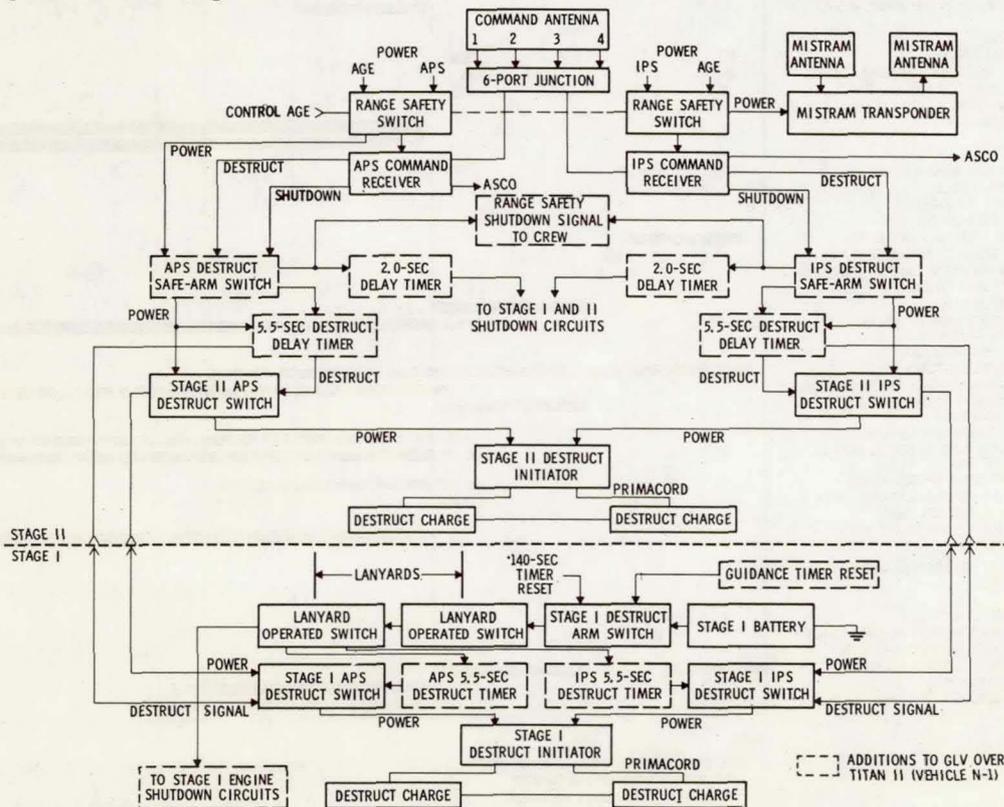


FIG. 28. RANGE SAFETY SYSTEM

Stated simply, these changes, which have been made to protect the men aboard, provide information with respect to Range Safety Officer action and adequate time for independent astronaut action. A summary showing the specific escape mode against the time of flight during which the mode would be employed is shown in Fig. 18. As further evidence of the planning which has been done to provide maximum crew safety, Fig. 29 shows a summary view of tracking, flight termination and destruct systems actions which occur prior to and after launch.

Figure 30 shows the flight termination sequence times during the various modes of escape. Vehicle destruct is accomplished by another independent action and a signal from the Range Safety Officer following destruct enable.

Gemini Electrical Sequencing

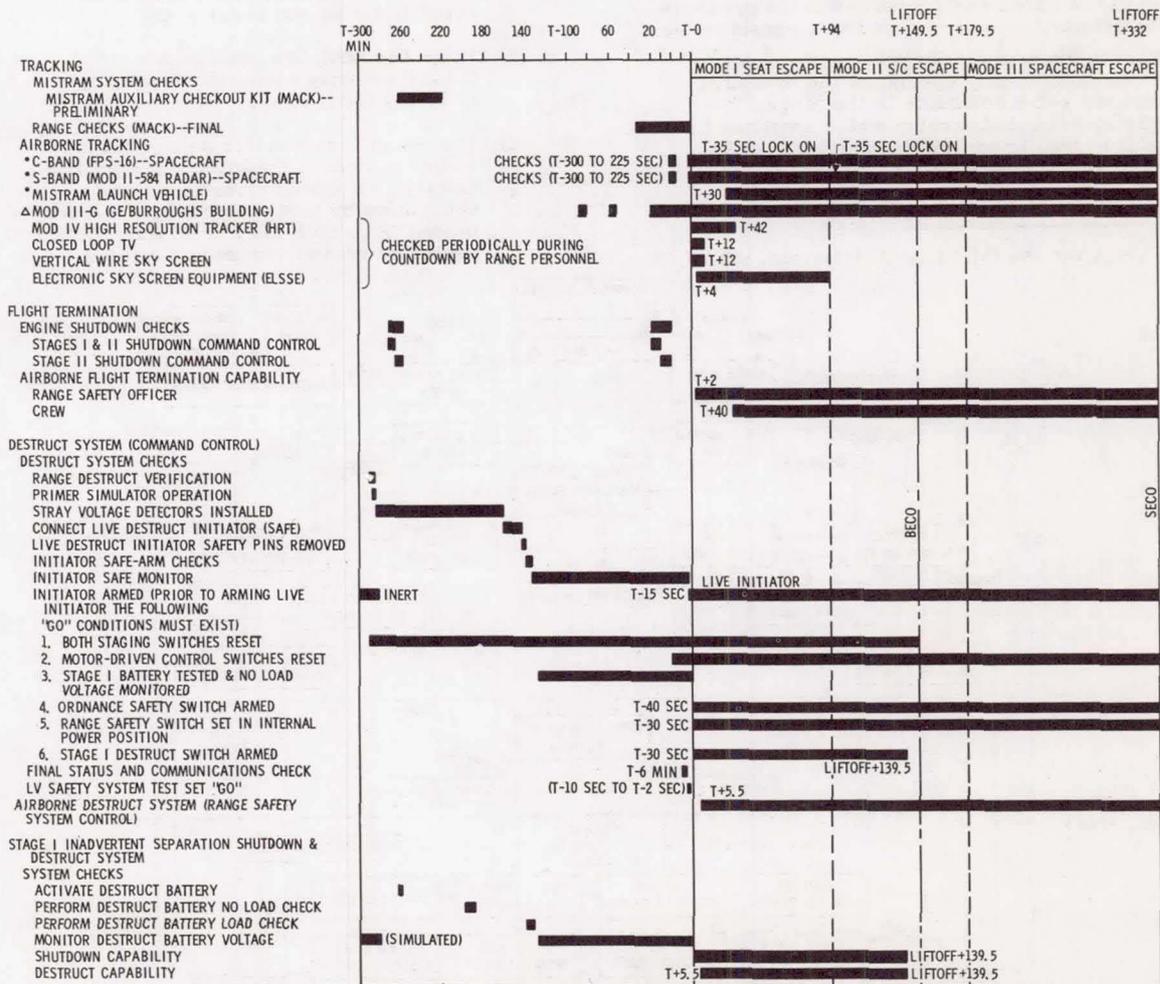
The addition of the Malfunction Detection System and the modifications made to the guidance system brought about a number of changes in the

electrical sequencing circuits. Since the basic design had to be changed, it was decided that the maximum degree of redundancy, within the context of the change, should be provided. Essentially, redundancy was achieved through the circuit wiring design without adding any new components. Table 5 compares the Gemini and Titan II electrical sequencing systems.

The controlling electrical sequencing system for the Gemini Launch Vehicle consists of the motor driven switch and relay logic which is required to perform such functions as:

- (1) Shut down the Stage I engine.
- (2) Fire Stages I and II separation nuts.
- (3) Start Stage II engine.
- (4) Command autopilot gain changes.

The system is shown in detail in Fig. 31.



NOTE:
 * PROCESSED THROUGH DATA SELECTOR BEFORE ENTERING 7090 IMPACT PREDICTOR COMPUTER (AUTOMATIC DATA SELECT PROGRAM) Δ GE BURROUGHS DATA TO BE TIED IN WITH ADSP

FIG. 29. RANGE SAFETY SYSTEM SEQUENCE

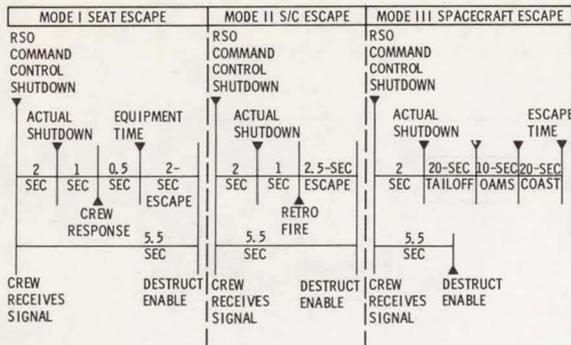


FIG. 30. FLIGHT TERMINATION SEQUENCE

Titan II Electrical Sequencing

While the Gemini and Titan sequencing systems are similar, Gemini has four additional provisions:

- (1) The system is redundant.
- (2) There is a Stage I fuel shutdown sensor.
- (3) There are 40- or 140-second time delay relays. In Titan these arming functions are performed by the Digital Control Unit.
- (4) There are two staging switches.

The APS staging switch performs the same function in both the Titan and Gemini Launch Vehicle. However, the Gemini can also call on

a backup IGS switch to perform the APS functions. The degree of redundancy which has been added is summarized in Fig. 31.

The sequencing system, which is fully redundant, is set into operation when the launch vehicle actually lifts off from the pad. The following operations occur simultaneously during lift-off:

- (1) The 40-second time delay relays (Nos. 1 and 2) start timing.
- (2) The Three-Axis Reference System starts timing.
- (3) The 140-second time delay relay starts timing.
- (4) The spacecraft receives a liftoff signal.

After 40 seconds has elapsed, the 40-second time delay relays are timed out, and the astronaut then has the capability to command a launch vehicle shutdown by operating the appropriate shutdown switches. After 140 seconds has elapsed, the stage separation circuitry is armed by both the Three-Axis Reference System and the 140-second time delay relay.

Normally, at approximately 150 seconds, the oxidizer will be depleted and a low stage I engine chamber pressure will result. The Thrust Chamber Pressure Switches will sense this condition, supply a ground to the staging circuitry, and staging will occur. If the fuel is depleted before the oxidizer, the Stage I fuel shutdown sensors will supply a ground and initiate staging.

TABLE 4
FLIGHT SEQUENCING FUNCTIONS

Function	Gemini Launch Vehicle Implementation	Titan II Implementation
Program Initiate	Redundant pad disconnect at liftoff. Redundant Program Initiate relays Nos. 1 and 2. Relay No. 1 applies 400-cps power to Three-Axis Reference System and starts 40-second relay No. 1. Relay No. 2 starts 140-second time delay relay and 40-second time delay relay No. 2.	Signal from Master Operations Console at T-3.7 seconds starts Digital Control Unit.
Spacecraft enable for launch vehicle engine shutdown	After 40 seconds has elapsed, the crew can shut down the launch vehicle (redundant relays).	N/A
Stage I fuel and oxidizer shutdown sensing	Thrust Chamber Pressure Switch sensors and fuel shutdown sensors sense depletion of oxidizer or fuel.	Thrust Chamber Pressure Switch only.
Staging arming	Redundant staging control relays Nos. 1 and 2 are armed by the Three-Axis Reference System 139.5 seconds after liftoff, and the 140-second time delay relay arms these relays 140 seconds after liftoff.	One staging control relay is armed by the Digital Control Unit 140 seconds after liftoff.
Staging	APS staging switch (1) Stage I engine shutdown. (2) Stage II engine start. (3) Autopilot gain changes at staging. (4) Fire separation nuts on the Stage II side. IPS staging switch (1) Stage I engine shutdown. (2) Stage II engine start. (3) Autopilot gain changes. (4) Fire separation nuts on Stage I side.	APS staging switch (1) Stage I engine shutdown. (2) Stage II engine start. (3) Autopilot gain change. (4) Fire separation nuts Stages I and II
Stage II low level shutdown	Fuel and oxidizer depletion is sensed by Stage II shutdown sensors. These units are armed by the Stage I low level shutdown control relay, and the relay in turn, is armed by the Three-Axis Reference System at 322.56 seconds after liftoff.	N/A
Stage II guidance shutdown	Shutdown is accomplished by Radio or Inertial Guidance System. Switchover is accomplished by Relay No. 2. The output is fed to redundant Stage II shutdown, relays Nos. 1 and 2.	Shutdown is accomplished by the Inertial Guidance System through one guidance shutdown relay.

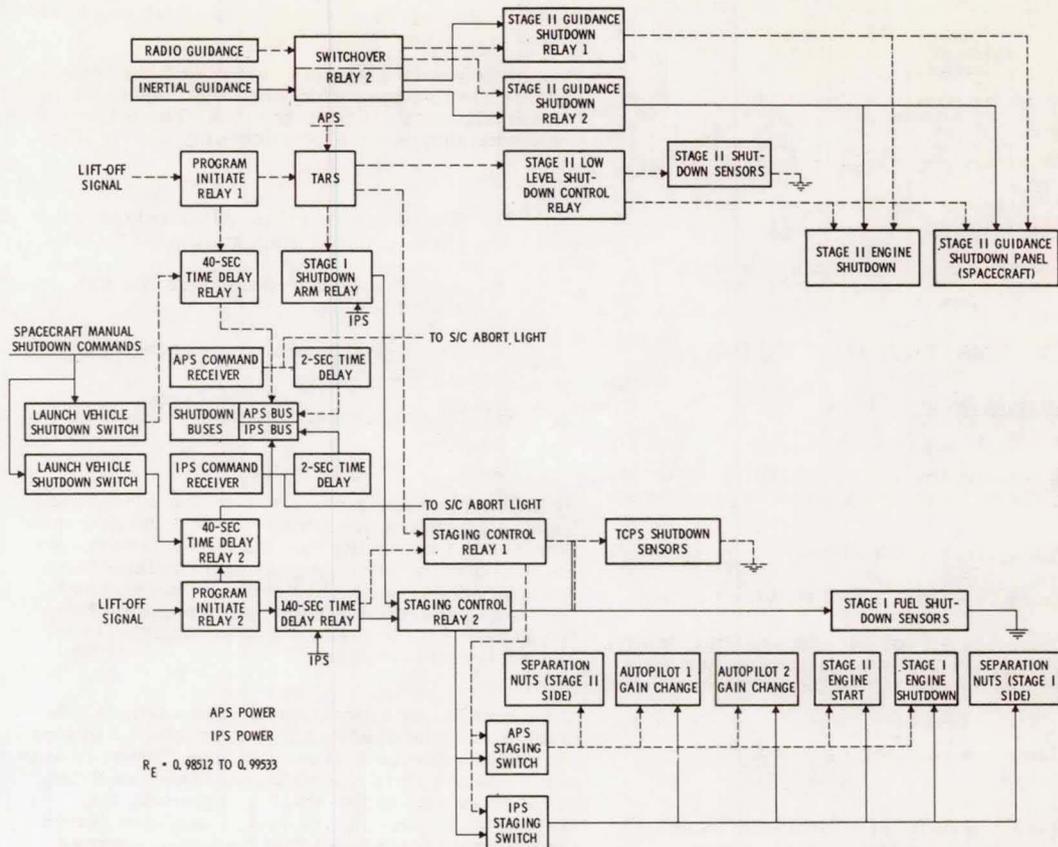


FIG. 31. ELECTRICAL SEQUENCING SYSTEM

TABLE 5
FLIGHT SEQUENCE DIFFERENCES: TITAN II AND GLV

Function	Time from Liftoff (sec)		Source of Function	
	GLV	T-II	GLV	T-II
98FS1 Stage I ignition	-3.3	-3.3	MOC	MOC
Thrust chamber switch closure	-2.2	-2.2	MOC	MOC
Fire nuts	-0.2	-0.2	MOC	MOC
Program initiate	0	0	Program initiate Relays Nos. 1 & 2	Digital control
Roll program start	+4.0	+5.0	TARS	DCU
Roll program end	+20.48	+10.0	TARS	DCU
Pitch program start Step No. 1	+23.04	+12.0	TARS	DCU
Spacecraft shutdown lockout	+40.0	N/A	40 seconds to relays Nos. 1 & 2	N/A
Pitch program, complete Step No. 1, start Step No. 2	+89.6	N/A	TARS	N/A
Flight control gain change	+104.96	+105	TARS	DCU
Start telemetry FM/FM recorder	+139.52	140	TARS	DCU
Arm staging initiate sensors	+139.52	140	TARS	DCU
Staging	+149	+150	140 seconds to relay	N/A
(1) 87FS2 Stage I shutdown			TCPS, shutdown sensors	TCPS, APS staging switch
(2) 91FS1 Stage II ignition			APS & IPS staging switch	
(3) Flight control staging gain change				
(4) Remove power to Stage I gyros				
Pitch program complete Step 2	+156.16	N/A	TARS	N/A
Radio guidance initiate	+156.16	N/A	TARS	N/A
Arm Stage II low level sensors	+322.56	N/A	TARS	N/A
91FS2 Stage II shutdown	+332	326	RG5 or IGS	IGS

NOTE:
MOC Master Operations Console
DCU Digital Control Unit
TARS Three-Axis Reference System

TCPS Thrust Chamber Pressure Switch
APS Accessory Power Supply
N/A Not applicable

Stage II shutdown is normally accomplished by the Radio Guidance System command; however, it may also be accomplished by:

- (1) IGS.
- (2) Astronaut.
- (3) APS and IPS command control receivers.
- (4) Stage II propellant shutdown sensors.

Relay No. 2 switches shutdown capability from the Radio Guidance to Inertial Guidance System.

Aerospace Ground Equipment

The selection of Aerospace Ground Equipment (AGE) for the Gemini program was influenced by two major considerations: first, that the launch vehicle is a modified Titan II; second, that Launch Complex 19 will be available for this program.

A comparison of equipment selected shows that, of the 208 AGE control points, 143 involve Titan equipment used "as is," while 33 involve Titan-modified, and 32 Gemini-peculiar control points.

The Ground Instrumentation System at the launch complex consists of a telemetry ground station, data recording equipment, signal conditioning, power monitor and control, time code distribution, control console and associated patching and cabling equipment. This system provides a flexible recording system which can be used to acquire data through umbilical or transmitted telemetry links.

Checkout and Launch Control

Essentially, the checkout philosophy calls for a decentralized approach; i.e., for each major airborne system, an equivalent piece of

equipment is provided to check the appropriate airborne system. Hence, the flight control system test set will check out the airborne flight control system, etc. The relationship of the various airborne systems and the checkout equipment is illustrated in Fig. 32.

Each checkout set can operate on its equivalent airborne system virtually independently of the other equipment. However, during the countdown phase, all operations performed by the checkout equipment must be coordinated by the launch control equipment. The checkout equipment will be predominantly manual, with automatic operation being used only during critical events or time periods. This philosophy assumes more importance than ever now that redundant flight controls and hydraulic components have been incorporated into the Gemini Launch Vehicle.

Launch control is obtained with the Master Operations Control System and other related equipment, including closed circuit television and a community time display board. The Master Operations Control System will provide time coordination during checkout of the launch vehicle and remote control of facilities such as the process water system and erector. It will also display the state of readiness of the entire complex as the various time checkpoints are reached. Lastly, through use of hold-fire and kill signals, it will provide the means of permitting or inhibiting launch at the predetermined T-O point.

Activation

Martin has been assigned the responsibility of integrating activation of Launch Complex 19 and the Gemini Launch Vehicle Support Area at AMR (Figs. 33 and 34).

Complex 19 is currently being activated, with all activities progressing as scheduled. Prima-

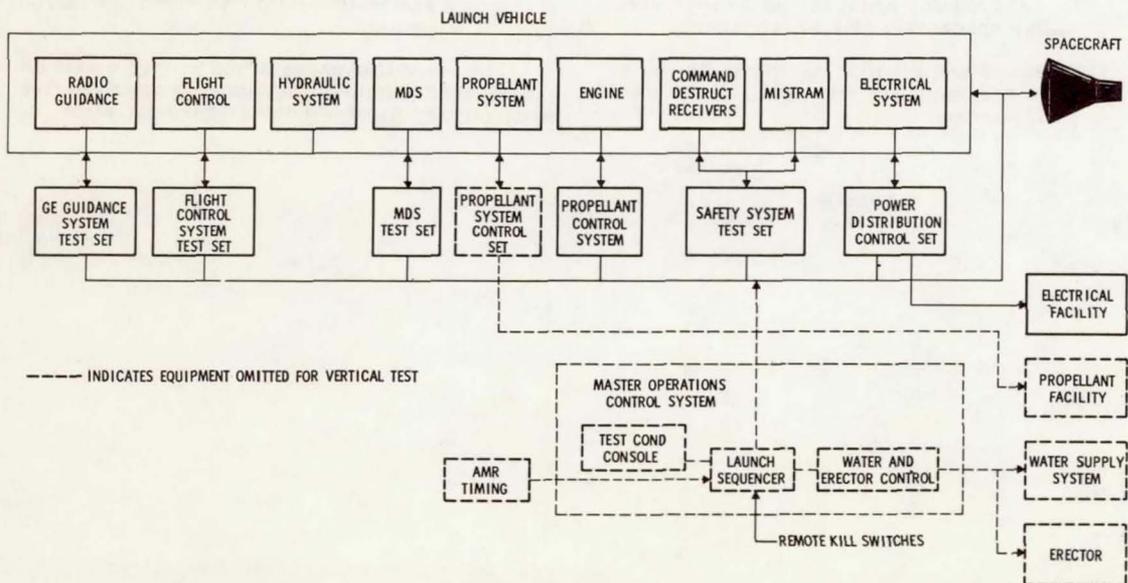


FIG. 32. AGE SYSTEMS

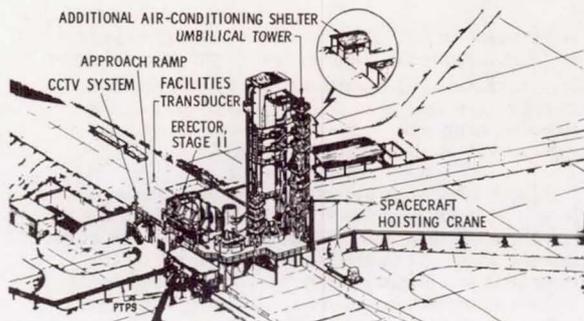


FIG. 33. AGE INSTALLATION --COMPLEX 19

rily, the activation effort on the complex consists of modifying the following existing facilities:

- (1) Blockhouse: the air-conditioning system only.
- (2) Ready building: double size to house NASA, McDonnell and Martin personnel.
- (3) Launch deck: external north end.
- (4) Complete vehicle erector: add white room, second elevator and spacecraft hoist system.
- (5) Second-stage erector: relocate work platforms.
- (6) Complete vehicle umbilical tower: extend height to accommodate two additional booms for spacecraft.
- (7) Second-stage umbilical tower: relocate existing booms.
- (8) Flume: enlarge and rearrange to permit quick runoff of expended fluids.
- (9) LOX holding area: use as storage area for spacecraft AGE service carts.
- (10) Roads and grading: modify south road to accommodate fuel and oxidizer holding areas.



FIG. 34. CAPE CANAVERAL COMPLEX LAYOUT

In addition to the facilities to be modified, the following new facilities will be added to Complex 19: a new road, located at the north end running north and south for delivery of the LH₂ to the spacecraft on the pad; an oxidizer holding area; a fuel holding area; a decontamination building and an air-conditioning facility for spacecraft servicing. No new facilities are required in the launch vehicle support area, except for a components cleaning facility which is expected to be provided by AFMTC for all contractors to use.

The design of modified and new facilities has been accomplished by Rader and Associates of Miami, Florida, in accordance with Martin's "Facilities Design Criteria," ER 12053. The construction of these facilities will be accomplished by the Army Corps of Engineers. New and modified AGE will be installed in all those facilities previously mentioned. All AGE to be installed and checked out is listed in the plan.

Martin will install all AGE on Complex 19 and in the Launch Vehicle Support Area. Each agency providing such equipment for installation will check out and maintain its own equipment throughout the program.

The activation phase of the program will be considered complete immediately after the first satisfactory flight-readiness demonstration.

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SATURN I STATUS REPORT

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SATURN I/IB Project Manager
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Gentlemen, I will present today the status of NASA's SATURN I program. In doing this, I will cover NASA's requirement for SATURN I, the scope of the job being undertaken and a brief summary of the SATURN development history. I will also give a brief review of the vehicle configuration, the schedule and development status, our flight test objectives and accomplishments, and will close with a short film of our last test flight, vehicle SA-4.

Let us first look at the NASA requirement for SATURN (fig. 1). SATURN I will give us our first large orbital payload capability. NASA will specifically use this capability for inflight qualification of the APOLLO command and service module and provide crew training. Further, SATURN I gives us the basic first stage for the SATURN IB vehicle and pioneers hydrogen technology for SATURN IB and SATURN V.

How big a job is SATURN I? Today (fig. 2) at the Chrysler Corporation Michoud Operations at the NASA Michoud plant, we have some 3,000 persons engaged in manufacture of the S-I stage. This number will rise to 4,000 as the SATURN IB program begins to be felt. At Douglas Aircraft in Santa Monica, 2,200 people are engaged in the development and production of the S-IV stage, while an additional 500 engineers and technicians are handling the static test program at Sacramento. At Marshall, we have 2,500 civil service people engaged in the systems integration, design, booster assembly and checkout, and instrument unit assembly and checkout. Engines, the H-1 from Rocketdyne and the RL-10 at Pratt and Whitney, employ an additional 4,500 persons at these companies. These major centers of activities are supported by a large complex of subcontractors, suppliers, and vendors.

To develop SATURN I and complete the ten vehicle development launch program will cost the country some 795 millions (fig. 3). This includes the flight test of ten SATURN I vehicles, development and manufacture of thirteen S-I stages, the establishment of the Chrysler Michoud operation, the development and flight testing of the guidance system, and the development and manufacture of eight instrument units, and the establishment of two launch complexes at the AMR. Further, significant steps are being taken in vehicle launch automation which give early development progress toward the SATURN V vertical assembly and launch concept.

Historically (fig. 4), SATURN I started as an ARPA project in 1958, the objective being to static test a multi-engine booster of 1.5 million pounds of thrust. ARPA next initiated a series of studies on upper stage configurations and mission requirements. In May of 1959, a modified Titan first stage was selected. This lasted some six months, and in December of 1959, the Silverstein Committee

recommended a lox-hydrogen stage for higher payload and long-range goals. This stage, a four engine S-IV, was intended as a third stage of the C-2 vehicle but was developed first due to the availability of the RL-10 A-3 engine. In April of 1961, we modified the vehicle design by adding two engines to the four engine S-IV stage, eliminating the third stage, improving the first stage and today we have this SATURN I vehicle.

The SATURN I has two stages. (See fig. 5.) The first stage, the S-I, has eight H-1 engines, uses lox-kerosene for propellants, is 80 feet long, and carries 850,000 pounds of propellants. The second stage, the S-IV, has six RL-10 A-3 engines, uses lox-liquid hydrogen for propellants, is 41 feet long and carries 100,000 pounds of propellants.

In a standard flight, the S-I stage is ignited and held down for 3.5 seconds to assure satisfactory H-1 engine operation. Prior to initiation of the tilt program, the vehicle is rolled into its flight azimuth from a fixed launch azimuth. Ten seconds after lift-off, we begin a gravity tilt program achieving a 66 degree path angle at 146 seconds, the burn-out of the first stage. After staging, the S-IV stage burns some 470 seconds, injecting the payload into orbit at some 1,400 miles from the launch point.

NASA has 16 SATURN I flight vehicles scheduled. (See fig. 6.) Ten of these vehicles are considered as launch vehicle development flights. The remainder are considered operational flights and will carry a manned APOLLO mission. Our flight test program began in October 1961 and we have had four successful flights of the Block I, or single live stage, configuration. Our next flight, a two-stage vehicle, is scheduled for launch in August of this year. If all goes well, this flight will put some 17,000 pounds payload in orbit. We have five additional two-stage flights scheduled for the period December 1963 through December 1964, prior to our first manned flight on vehicle 111 in March 1965.

The H-1 engine used in the S-I stage has an extensive test history. (See fig. 7.) We have accumulated approximately 29,000 seconds of firing time on production H-1 engines.

On the S-I stage, the cluster of H-1 engines, we have accumulated approximately 3,000 seconds of static test time (fig. 8).

The RL-10 engine history has over 100 hours of hot firing time to date (fig. 9).

The S-IV stage has 3,160 seconds of static test time to date and we project some 1,400 additional seconds prior to the first flight and some 22,360 seconds prior to the first manned flight.

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I would like to cover, in somewhat more detail, flight mission plans for the next seven SATURN I vehicles. (See fig. 10.) As we see from the chart, we are talking about vehicles SA-5 through SA-111. All these will be two-stage vehicles. All will be programmed to achieve an orbit with the spacecraft. On SA-5, we will fly our guidance system, with principle components being a Bendix stable platform and an IBM guidance computer as a passenger. Our goal is to have active guidance on SA-6 and thereafter.

On SA-5, we will have a standard nose cone. SA-6 and subsequent vehicles will carry either APOLLO boilerplate or flight spacecraft modules.

As previously mentioned, we consider the vehicle R&D program to end at SA-10. Vehicle SA-111 will be identical to SA-10 but will have a major portion of the R&D instrumentation removed. The SA-6 and SA-7 are intended to secure APOLLO spacecraft launch phase environmental data, SA-8 and SA-9 will test the crew abort system, SA-10 will be a complete flight test of an unmanned APOLLO command module and service module, and SA-111 is planned for the first manned orbital flight of APOLLO. Other missions we will undertake will be a tape recorder in SA-5 and a micrometeorite detection satellite flown on vehicles SA-8 and SA-9.

This is where we stand today:

1. The S-I stage is in an advanced development state.
2. The S-IV stage has had good static and ground test results. The flight test program remains to be accomplished.
3. Guidance components passenger flights have been successful. Full system tests remain.
4. Flight and dynamic control systems tests have been successful and give no indication of potential problems.
5. Industrial, test, and launch facilities required to support the total program will be completed by the end of this year.

FIRST LARGE
ORBITAL
PAYLOAD
CAPABILITY



Figure 1. - NASA requirements for Saturn I.

LOCATION	NO. EMPLOYED
CHRYSLER MICHOU D	3,000
DAC, SANTA MONICA	2,200
DAC, SACRAMENTO	500
MSFC, HUNTSVILLE	2,500
ROCKETDYNE, CANOGA PARK AND NEOSHO, MO.	1,500
P&W, PALM BEACH AND HARTFORD	3,000

Figure 2. - Scope of activities.

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ESTIMATE TOTAL COST 795.0 MILLIONS.

WHAT DOES IT INCLUDE:

- **TEN LAUNCHES**
- **DEVELOPMENT AND MANUFACTURE OF 13 S-I STAGES**
- **DEVELOPMENT AND MANUFACTURE OF 10 S-IV STAGES**
- **ESTABLISHMENT OF CSD MICHOU D OPERATIONS**
- **DEVELOPMENT AND PROCUREMENT OF THE GUIDANCE SYSTEM**
- **DEVELOPMENT AND MANUFACTURE OF 11 INSTRUMENT UNITS**
- **SIGNIFICANT STEPS IN AUTOMATION LEADING TO SATURN V LAUNCH CONCEPT**

Figure 4. - History.

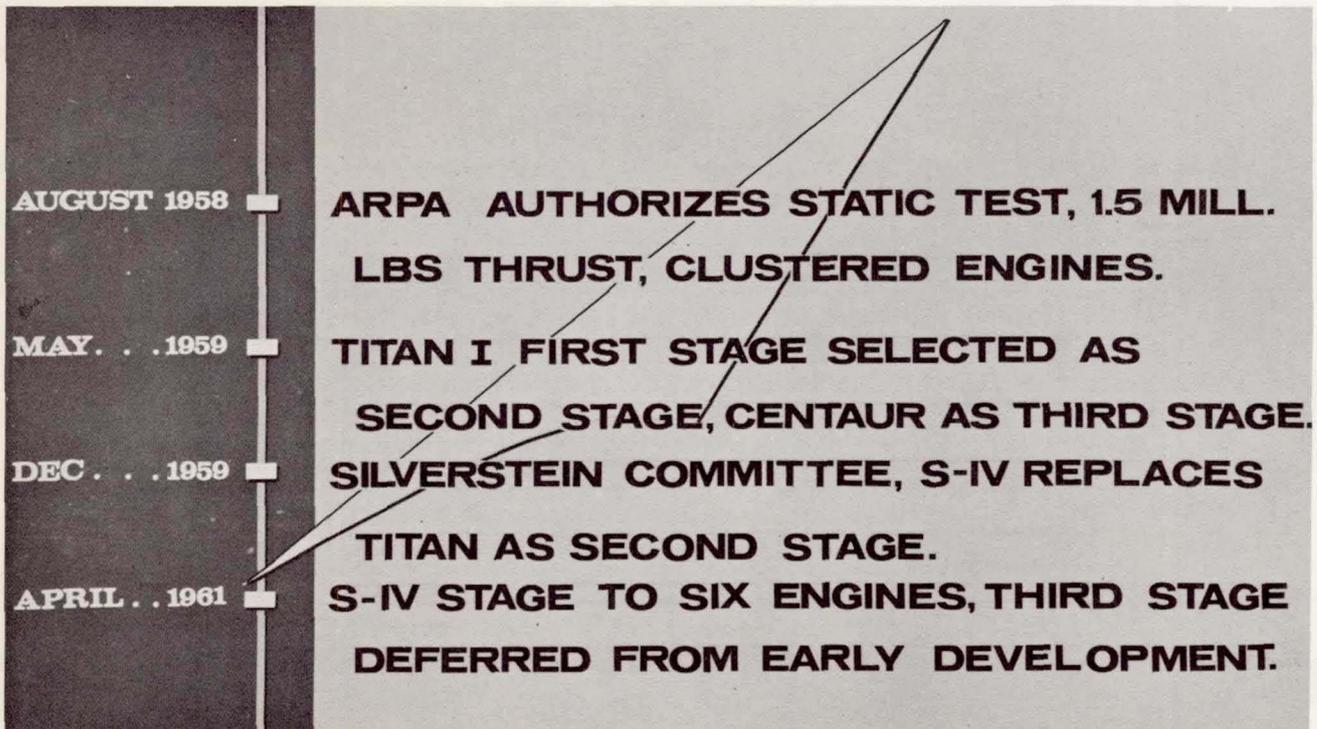
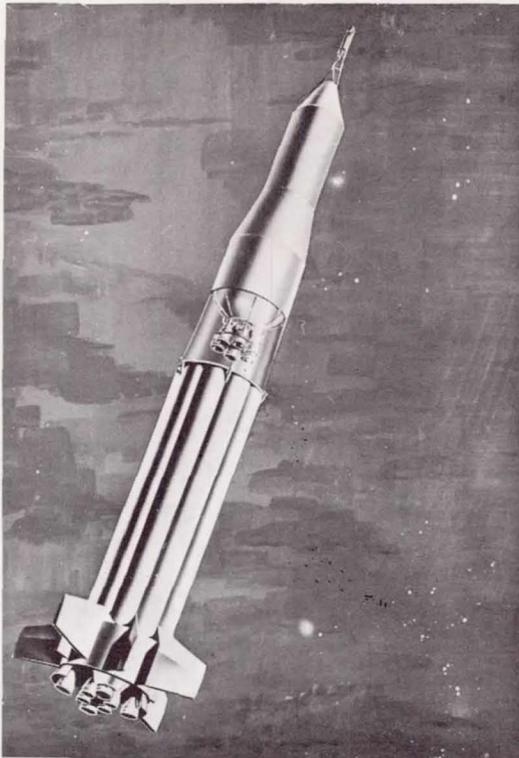


Figure 3. - Saturn I development program.

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GUIDANCE: INERTIAL
CONTROL: GIMBALLED ENGINES
PAYLOAD: 7 TONS IN 300 MILE ORBIT
OR- 10 TONS IN 140 MILE ORBIT
VEHICLE: LENGTH _____ 184 FT
WEIGHT FUELED 550 TONS
WEIGHT EMPTY 65 TONS
STAGE SIZES:
S-I _____ 257" x 80'
S-IV _____ 220" x 41'
STAGE THRUST:
S-I _____ 1,500,000 LB
8 H-1 LOX/RP-1 ENGINES
S-IV _____ 90,000 LB
6 A-3 LOX/LH₂ ENGINES

M-MS-G 14-1-63 MAR 4, 63
M-CP-D FEB 28, 63 M-CP-D 1002

Figure 5. - Saturn I (Block II) characteristics.

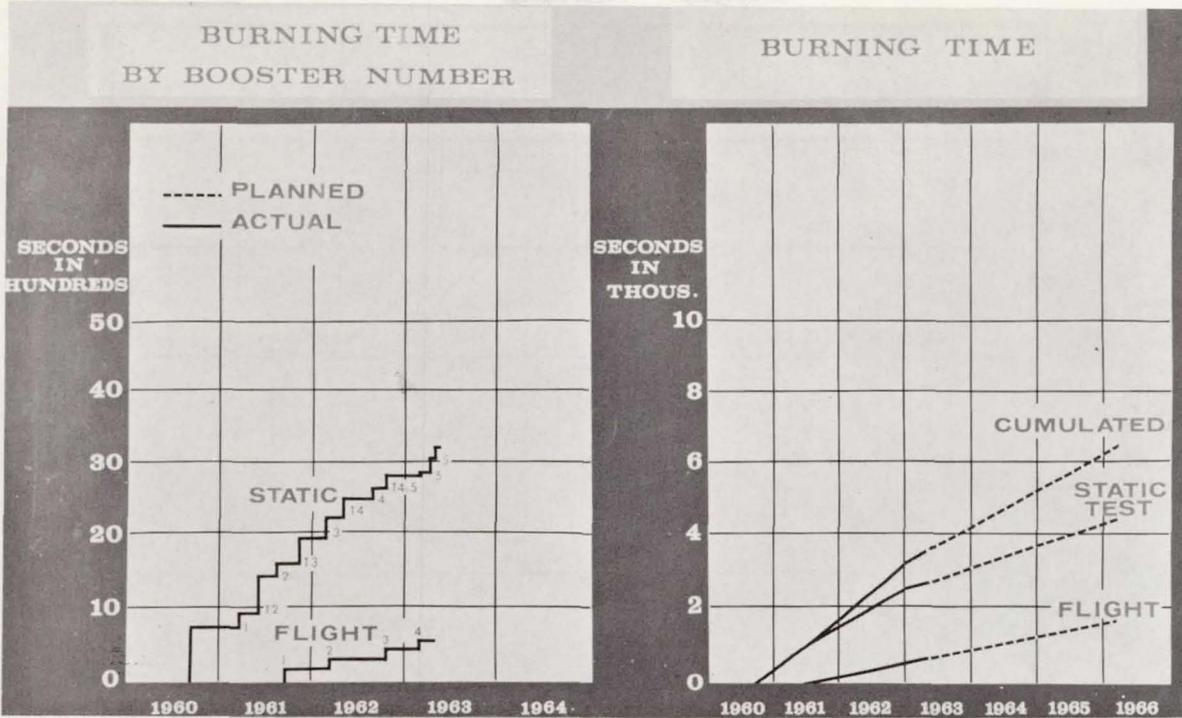
		BLOCK I						
		1961	1962	1963	1964	1965	1966	1967
Live S-I Dummy S-IV Dummy S-V Jupiter Nose Cone	SA-1	▲ OCT						
	SA-2		▲ APR					
	SA-3			▲ NOV				
	SA-4			▲ MAR				
		BLOCK II						
S-I S-IV Instrument Unit Jupiter Nose Cone (SA-5) Apollo Boiler Plate CM & SM	SA-5			◇ AUG				
	SA-6			◇ DEC				
	SA-7				◇ MAR			
	SA-9				◇ JUN			
	SA-8					◇ OCT		
	SA-10					◇ DEC		
	SA-111						▲ MAR	
	SA-112						▲ JUN	
	SA-113						▲ SEP	
	SA-114						▲ DEC	
	SA-115	SPARE						▲ MAR
	SA-116	SPARE						▲ JUN

SCHEDULE ◇ LAUNCHED ▲ OPERATIONAL ▲

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Figure 6. - Saturn I launch schedule.

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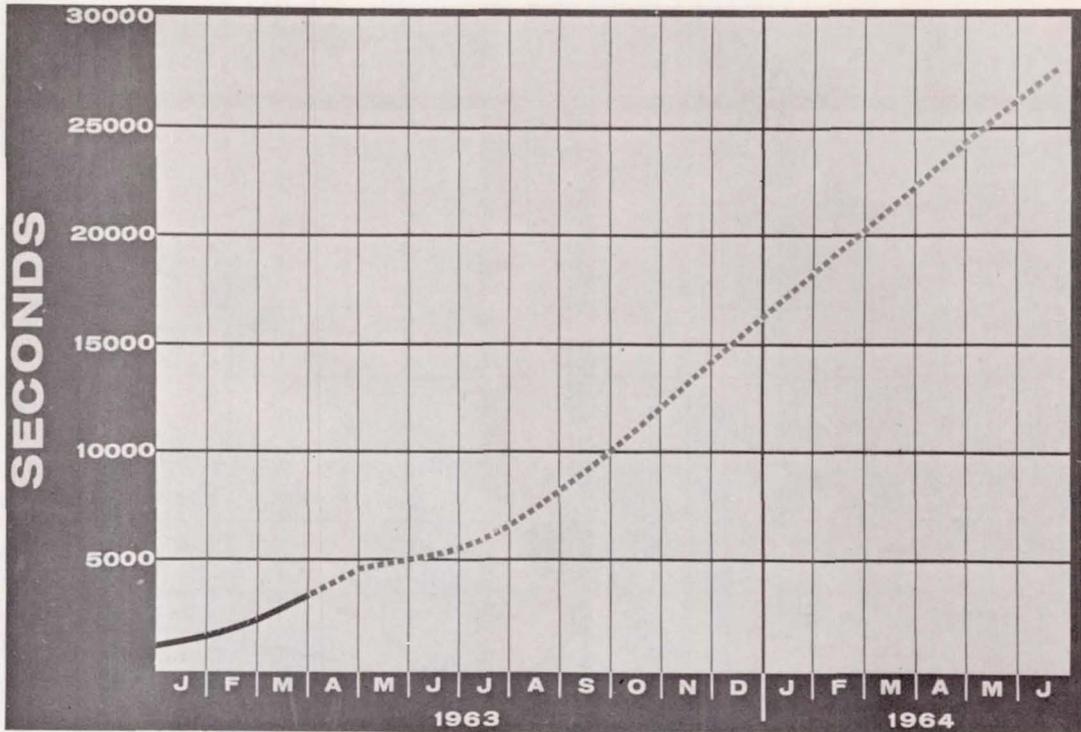


Figure 9. - S-IV stage static test times.

SATURN I VEHICLE	LAUNCH VEHICLE DEVELOPMENT	GUIDANCE		APOLLO			
		PASSENGER	ACTIVE	ENVIRONMENT	ABORT SYSTEM	SPACE CRAFT QUALITY	MANNED FLIGHT
SA - 5	▲	▲					
SA - 6	▲		▲	▲			
SA - 7	▲		▲	▲			
SA - 8	▲		▲		▲		
SA - 9	▲		▲		▲		
SA-10	▲		▲		▲	▲	
SA-111			▲		▲		▲
& SUB.			▲		▲		▲

OTHER MISSIONS

1. VOICE TRANSMISSION FROM SA-5 IN ORBIT
2. MICROMETEOROID SATELLITE, SA-8, SA-9

Figure 10. - Saturn I missions.

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Saturn V Launch Vehicle Program

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For a brief glimpse of the Saturn V Program, I will discuss the following areas: (1) the background of the Saturn V, (2) the vehicle characteristics, (3) the ground test program, (4) the flight test program, and (5) a general program status.

Background

The Saturn V launch vehicle emerged from a series of studies conducted at Marshall Space Flight Center during 1961 and consistent with the NASA overall manned lunar landing program definitions. The NASA requirement for the launch-vehicle portion of the manned lunar landing task was studied in three principal modes of operation: (1) earth orbit rendezvous, (2) lunar orbit rendezvous, and (3) direct ascent.

The selection of the Saturn V configuration was made in early 1962 on the basis of the following performance capabilities for the three modes of operation: (1) earth orbit rendezvous - 125 tons - to near earth orbit, (2) lunar orbit rendezvous - 45 tons - to the 72-hour translunar injection point, and (3) direct ascent - 20 tons soft landed on the lunar surface.

In mid-1962, NASA selected lunar orbit rendezvous as the operational mode for accomplishing the lunar landing mission. All development efforts for the Saturn V launch vehicle are directed toward supporting the LOR mode of operation.

Our present project authorization is based upon a ten-vehicle R&D flight development program; however, our planning is extended to include five follow-on operational vehicles, and our long-range plan is based upon a sustained manufacturing, testing, and launching capability of one vehicle per month. A few of the major accomplishments and milestones are listed in the following chart (Figure 1).

Vehicle Characteristics

The characteristics of the Saturn V launch vehicle are illustrated in Figure 2. Of the 6 million pounds launch weight of the vehicle, 5.56 million pounds are propellants. These weights are broken down as follows: 4.4 million pounds of liquid oxygen/JP fuel in the first stage, .93 million pounds of hydrogen/oxygen in the second stage, and .23 million pounds hydrogen/oxygen in the third stage. I have not included the propellants contained in the spacecraft.

The first stage (S-IC) is propelled by five F-1 engines, each developing a thrust of $1\frac{1}{2}$ million pounds, for a combined liftoff thrust of $7\frac{1}{2}$ million pounds. The second stage (S-II) is propelled by five J-2 engines, each developing 200,000 pounds, for a total thrust of 1-million pounds. The third stage (S-IVB) is propelled by a single J-2 engine, providing a thrust of 200,000 pounds.

An Instrument Unit rides atop the third propulsive stage and aft of the spacecraft. This unit contains the guidance and control instrumentation for the three propulsive stages. The first and second stages have a four-outer-engine-gimbal capability to provide roll, pitch, and yaw control. Auxiliary attitude control is provided to the third stage by attitude control modules.

Operating times for the stages are essentially as follows: (1) first stage, approximately 150 seconds, (2) second stage, approximately 400 seconds, and (3) the first burn of the third stage is approximately 165 seconds into a low-earth waiting orbit. After a waiting orbit of up to $4\frac{1}{2}$ hours, the second burn of the third stage is initiated; this burn time, expected to be in the order of 310 seconds, injects the payload into the 72-hour earth-moon transit.

Ground Test Program

The principal elements of the Saturn V ground test program are illustrated in Figure 3. A major emphasis is placed on an adequate ground test program. Since the expense of each flight test vehicle is quite large, the number of flight tests is kept to a minimum, consistent, of course, with a reasonable number to provide correlation between ground and flight environments.

The ground test program for component selection and qualification is underway, at this time, in many areas of piece parts and what we call "speciality" items, such as valves, bellows, seals, flanges, switches, electrical boxes, etc. These items are not only under continuous design review of a theoretical nature involving "criticality" evaluations, but are also under strenuous testing to reveal short-comings that can be corrected before the stage systems development tests get underway.

Development test capability is provided in close proximity to the design and engineering activities. For example, in first stage (S-IC) activities, MSFC, with the assistance of Boeing, is fabricating and assembling the first ground test stage and the first flight test stage. The static test and development stage will be test fired at Huntsville on the test stand which is now well along in construction. Stage structural testing will be accomplished in the laboratories of the Marshall Center.

The development test area for the second stage, S-II, is located in Santa Susana at the North American Aviation propulsion development site. Two test stands are being prepared for early battleship and all-systems testing.

For the S-IVB, the Douglas Aircraft test area in Sacramento will be utilized for development testing.

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The S-IVB stage is a common item for both the Saturn V and Saturn IB programs.

Vehicle systems development testing will be conducted at Marshall Space Flight Center, where all combinations of flight configuration will be subjected to dynamic analyses.

At the Launch Operations Center, in Merritt Island Launch Area, a functional launch vehicle system will be provided for facilities checkout. The degree of automation and the complexity of operation involved in a multi-stage vehicle of this type have prompted a very thorough operational development program for the NASA launch complex 39. The respective stages will be assembled into a functional configuration, so that, generally, the entire operational procedure can be developed prior to the receipt of the first flight vehicle. This step is taken to assure that the flight stages are not exposed to the initial activities of the Integrated-Test-Launch concept.

Flight Test Program

The flight test program (Figure 4) will start in early 1966. The first three flights are established to test, progressively, the flight stages. For the initial Saturn V flight a live first stage is to be used with inert upper stages. On the second flight, both the first and second stages are planned to be live, with the third stage inert. All three stages are to be live on the third flight. The fourth flight is backup and will provide development confidence and reliability. The fifth and sixth flights are considered to be preliminary launch vehicle qualification flights; that is, these vehicles should be capable of demonstrating full performance capability. Launch vehicles seven through ten are termed "developmental - manned qualification." This series of flights will commence in mid-1967. The operational program begins with vehicle number 511, scheduled for early 1968.

General Status

The final comments of my presentation deal with program status as of this time.

We are in the sixteenth month of the configured and approved program. The decision as to the operational mode was made nine months ago and a further refinement of the launch vehicle criteria, involving structural definitions based upon mission profile, began at that time.

The manpower presently engaged in the development effort totals in excess of 12,000 direct personnel in the major contract areas of: (1) Boeing Aircraft Company - S-IC stage, (2) Space and Information Systems Division (NAA) - S-II stage, (3) Douglas Aircraft Company - S-IVB stage, and (4) Rocketdyne (NAA) - F-1/J-2 engines.

Peak manpower estimates for the four major contract elements noted above are forecast at some 15,500 direct personnel in the 1964/1965. This increase will be in the test, operation, and manufacturing buildup since most areas of engineering are at near peak at this time.

Let me again emphasize that these figures are for the first-tier development contracting only.

At Marshall Space Flight Center, 1,000 direct civil

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service personnel are engaged in the management, systems integration and the design, manufacturing, test and quality control of the Saturn V project. This number is expected to increase to 1,800 in fifteen months as the phase-over from Saturn I to Saturn V continues.

With regard to the longest leadtime item, that is, facilities (authorization and construction), we are now at the estimated 85% point with regard to approvals, authorizations, etc. and about 60% in the construction phase. Many items are being activated and placed into operation; for example, S-IC tooling installation which is going on in the Huntsville shops at this time. Other examples are the static test facility at Huntsville and the structural test facility, also at Huntsville. In the Michoud area, facility modification has been completed in many areas, and the Vertical Assembly Building is under construction. These facilities deal solely with the first stage.

Concerning the second stage, the entire Seal Beach construction program is underway, and the first building was completed to the status of joint occupancy in January 1963. Tooling is being installed for structural fabrication. Stage development test facilities are under construction at Santa Susana, California.

With regard to the S-IVB stage, facility modification in the Douglas Aircraft Company's plant at Santa Monica is underway. Components will be fabricated in this plant. The final assembly of S-IVB will be performed by Douglas in the Huntington Beach area, a new location being developed by Douglas. Occupancy of the major buildings will be phased-in by October 1963.

The static test facility at Sacramento is presently in a site-preparation phase. Construction awards were made in March 1963.

We estimate that we have completed approximately 45% of the detail design and engineering for the Saturn V vehicle and released about 15%.

By the end of this year our schedules require the release of the major portion of all engineering.

Tooling designs are complete for all major structural elements, and tooling fabrication is approximately 75% complete. As you have already seen, some of this tooling is in operation.

With regard to structural components, Boeing has delivered, out of the Michoud Plant, two Y-Rings which involve a major machining operation. These rings are fabricated from three 120-degree segments welded together to give a 33-foot-diameter ring. The first Y-Ring has been delivered to Huntsville for final welding operation into the early structural test tanks.

In Wichita, gore segments are being fabricated for S-IC tanks. First delivery of F-1 engines for stage assembly will be accomplished by the end of this year.

The same general status exists for the two upper stages. J-2 engines will be delivered by the end of this year to start the first preparation for stage mating. These early engines are scheduled for use with heavy-wall, battleship-type tankage.

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Structural components are being fabricated at this time.

In conclusion, the Saturn V Project is proceeding at a rapid pace. Our schedules are tight but are within bounds of our capability, assuming timely and adequate funding. We have a highly competent industrial team already functioning in the development of major vehicle elements. We have a real sense of urgency toward the task we have undertaken. I am confident we can provide a launch to meet the President's stated requirement for "a manned lunar landing in this decade."

<u>1961</u>	
May	Engineering studies of Advanced Saturn Rocketdyne selected to develop upper stage engine
July	First firing of F-1 engine system
September	Michoud Plant selected for NASA use Douglas Aircraft selected to develop S-IVB stage S&ID selected to develop S-II stage
October	Test location selected - MTF
December	Boeing Aircraft Company selected to develop S-IC
<u>1962</u>	
January	<u>Saturn V configuration selected by NASA</u> First firing of J-2 engine
March	Sverdrup parcel selected to plan and design - MTF
April	DX priority established for program
May	Full thrust/full duration firing of F-1 engine
July	<u>LOR mode selected to accomplish first manned lunar landing</u>
October	Full thrust/long duration firing of J-2 engine
November	First major tooling for S-IC delivered
<u>1963</u>	
January	First increment of Seal Beach (S-II) fabrication facility readied
February	Delivery of first S-IC structural components from Michoud Plant
March	First S-IC bulkhead gore segment welded

Figure 1.- Saturn V milestone chronology.

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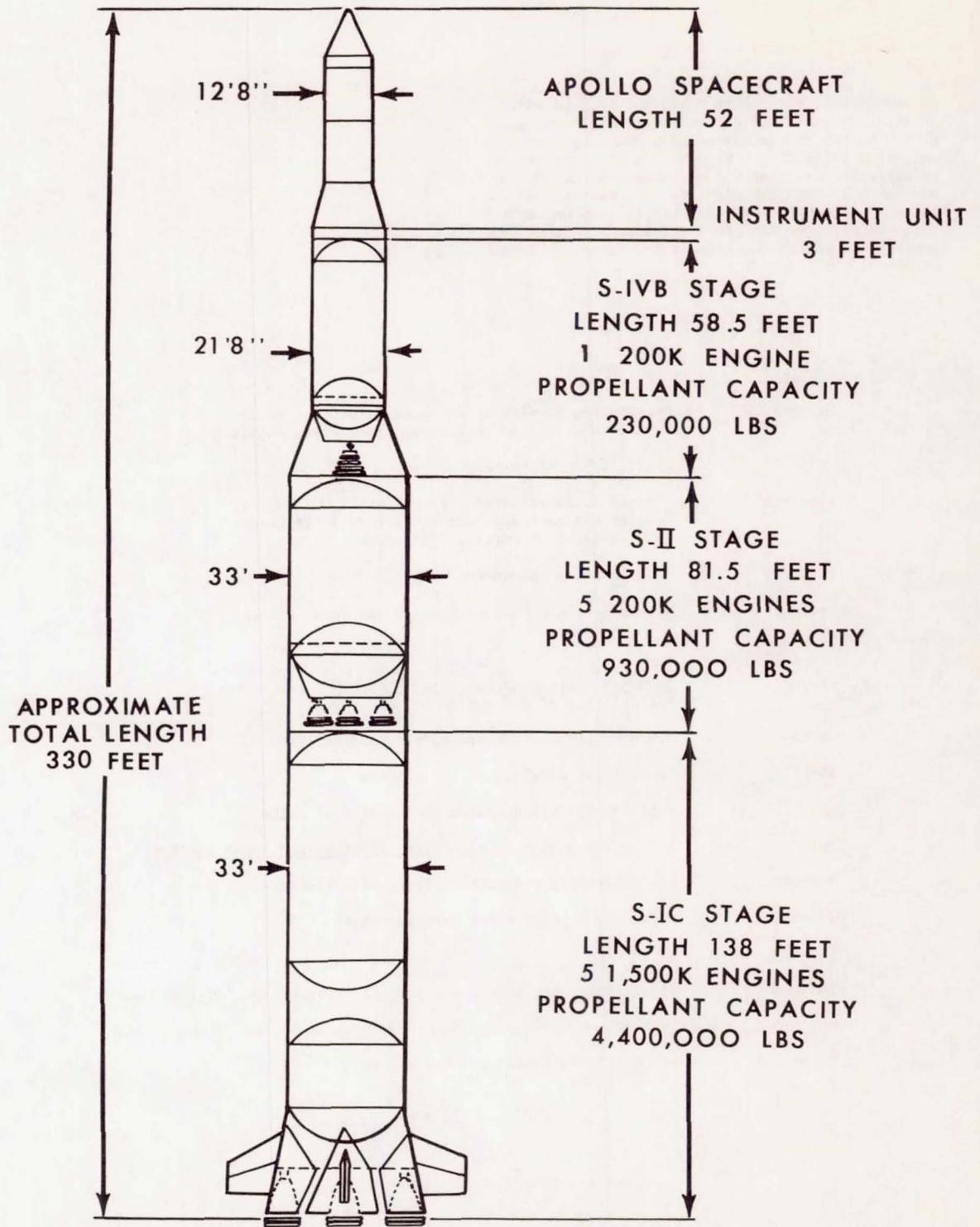


FIG 2 SATURN C-5 LAUNCH VEHICLE

Figure 2.- Saturn V launch vehicle.

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<u>Designation</u>	<u>Configuration</u>	<u>Mission</u>
SA-500-S	<u>Structural Stage</u> non-functional	Certify structural integrity of each complete stage structure under simulated critical load conditions.
SA-500-T	<u>Battleship Stage</u> functional systems	Captive testing to develop functional, operational, design, proof, performance, reliability of stage system
	<u>All System Stage</u> all flight systems	
SA-500-D	<u>Dynamic Vehicle</u> Flight Configuration, functional systems	Determine under various flight configurations the dynamic response, structural flexure, etc.
SA-500-F	<u>Facilities Vehicle</u> Flight Configuration functional systems	Complete checkout LC 39 determine functional compatibility of vehicle with instrumentation system, automatic GSE, facilities support system, etc. before arrival at first flight vehicle

Figure 3.- Saturn V ground-test program.

<u>Designation</u>	<u>Configuration</u>	<u>Mission</u>
SA-501	S-IC - Active S-II - Inert S-IVB - Inert	Structural Integrity, Flight Environment, First Stage Flight Performance
SA-502	S-IC - Active S-II - Active S-IVB - Inert	Structural Integrity, Flight Environment, Separation and Control First and Second Stage Flight Performance
SA-503 SA-504	All Stages Active All Stages Active	Structural Integrity, Flight Environment, Vehicle Performance, Separation and Control
SA-505 SA-506	All Stages Active All Stages Active	Preliminary qualification, Performance and Control accuracies Vehicle capability and reliability
SA-507 SA-508 SA-509 SA-510	All Stages Active All Stages Active All Stages Active All Stages Active	Developmental-Manned qualification
SA-511 & Subs.	All Stages Active	Operational

Figure 4.- Saturn V flight-test program.

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TRENDS IN MANNED SPACECRAFT SUBSYSTEMS

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Aero-Space Division
The Boeing Company

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

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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 (f_p) 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 f_p . 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 World-wide 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 (f_p). Below this frequency attenuation in the order of 60 db (transmission of only a millionth of the energy) is experienced. f_p is a function of the electron density and collision frequency and is defined here by the following equation:

$$f_p = 8.98 \times 10^3 \sqrt{N_e}$$

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

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 f_p 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 f_p at the next atmospheric window ≈ 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 f_p 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 ≈ one but a higher frequency and thus more ground stations per mile of coverage may be required for the L/D ≈ 0.5 vehicles. The next atmospheric window is at ≈ 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 flies 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.

$$\ddot{h} = -g_0 + \frac{v^2}{r_e} + A_L \approx 0 \text{ -----(1)}$$

$$\alpha A_L = g_0 - \frac{v^2}{r_e} \text{ -----(2)}$$

Where:

- h = Altitude
- g₀ = Gravitational constant
- r_e = 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.

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PROGRAM	TYPE		USE OF MAN	CAPABILITY SELF-CONTAINED	PERFORMANCE VS MISSION TIME	WT
	PRIMARY	BACKUP				
MERCURY	GROUND BASED	PILOT FIRE RETROCKET	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)	INERTIAL & GROUND RADAR	SECOND INERTIAL MANUAL STADIA-TRIANGULATION DSIF	EXTENSIVE • MODE SELECTION	• PRIMARY MODE	• SYSTEM DESIGNED TO COMPLETE MISSION WITH MULTIPLE FAIL 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.

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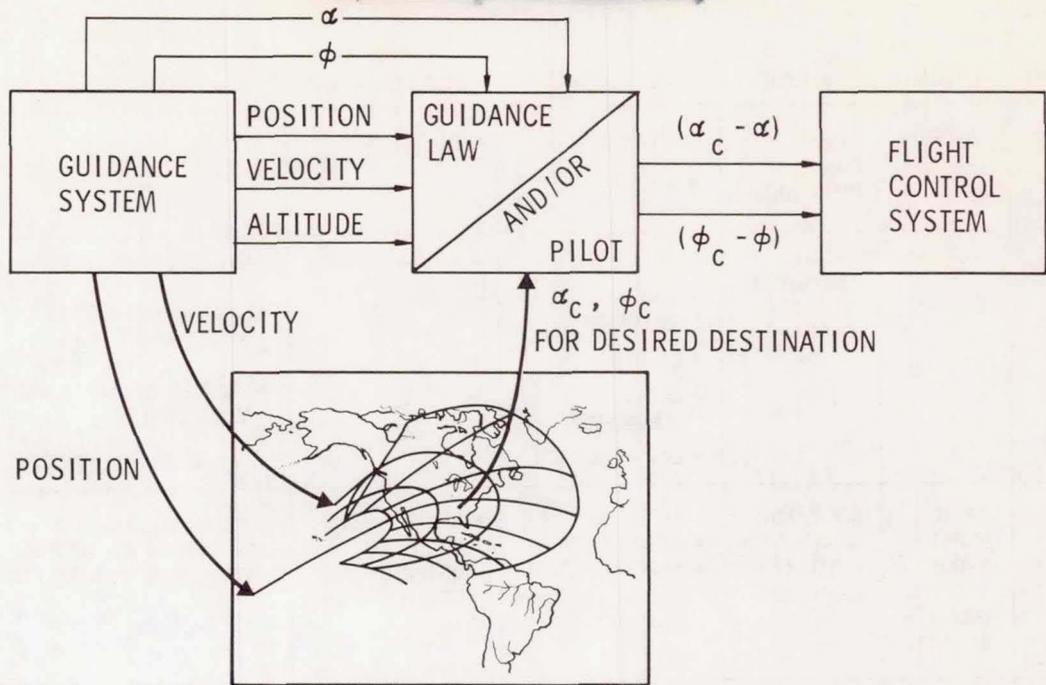


Figure 3.- Conceptual energy management system.

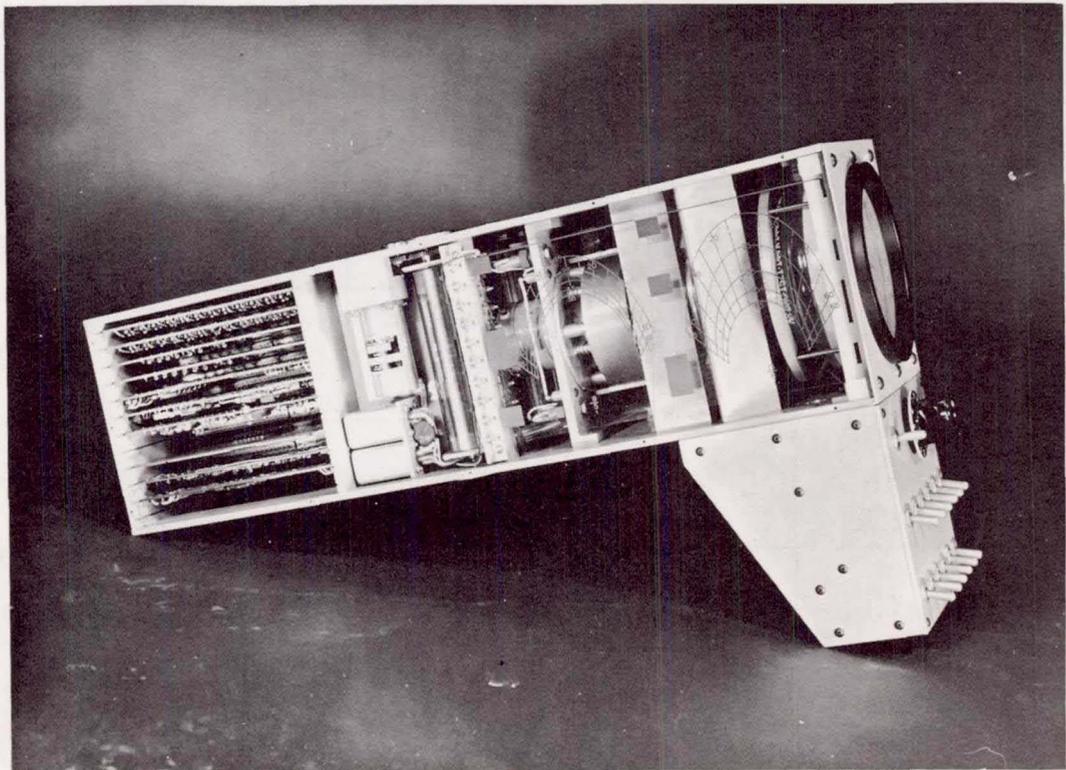


Figure 4.- Energy management display.

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

Figure 5.- Earth orbit and re-entry.

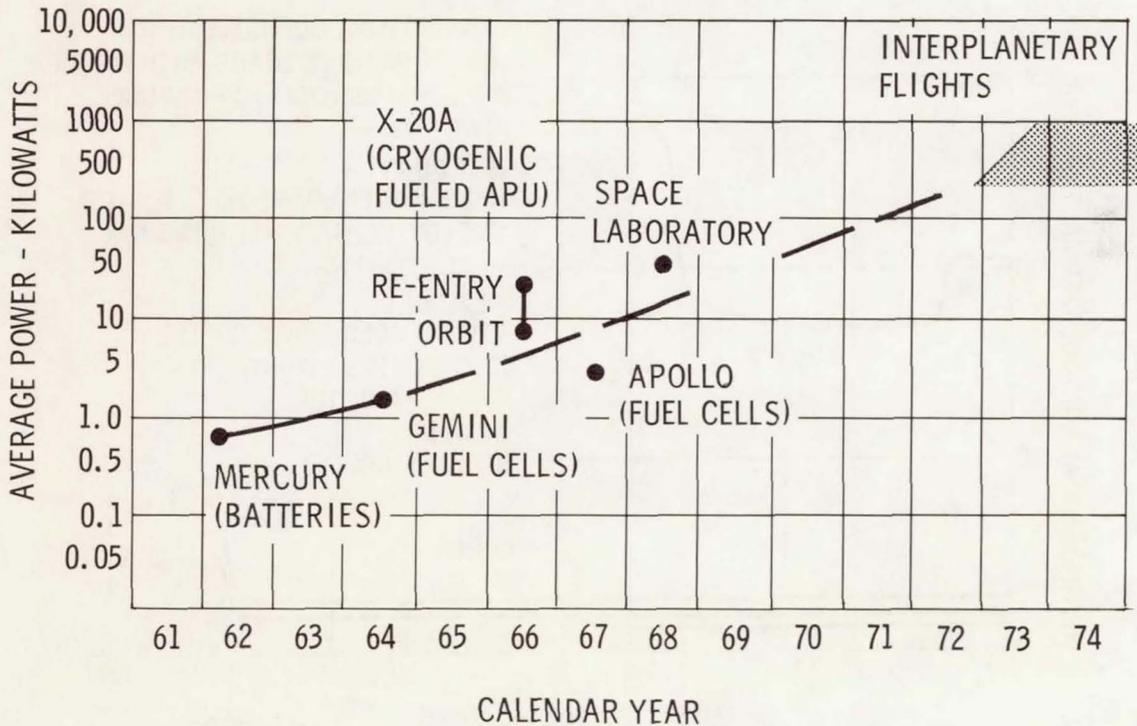


Figure 6.- Power requirements.

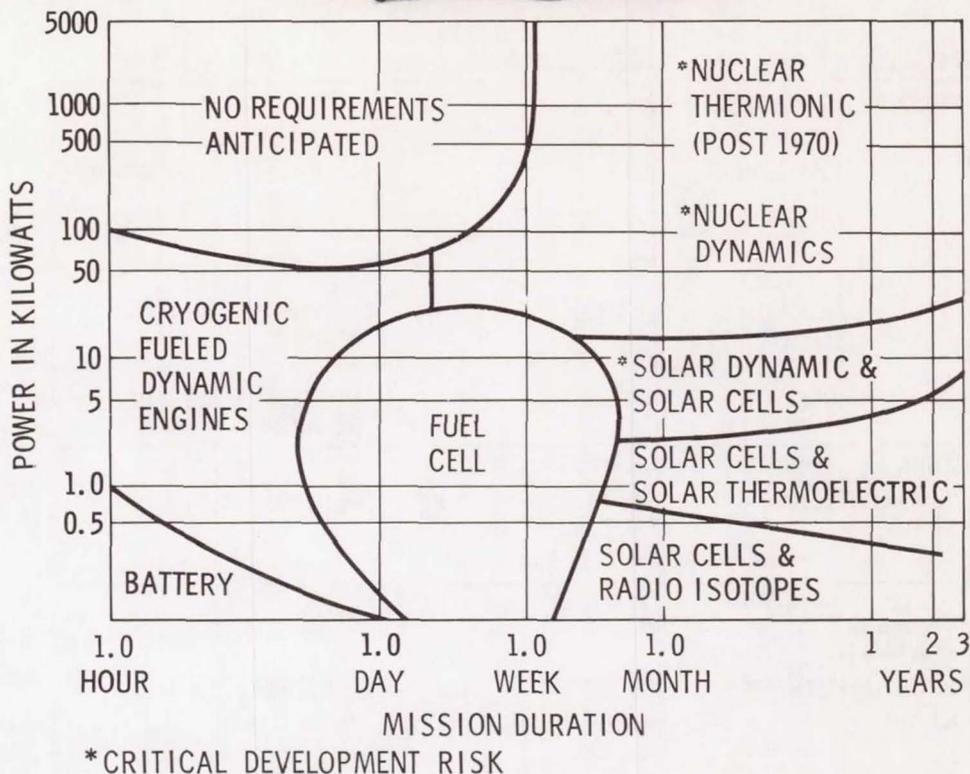


Figure 7.- Estimated optimum power systems, 1966.

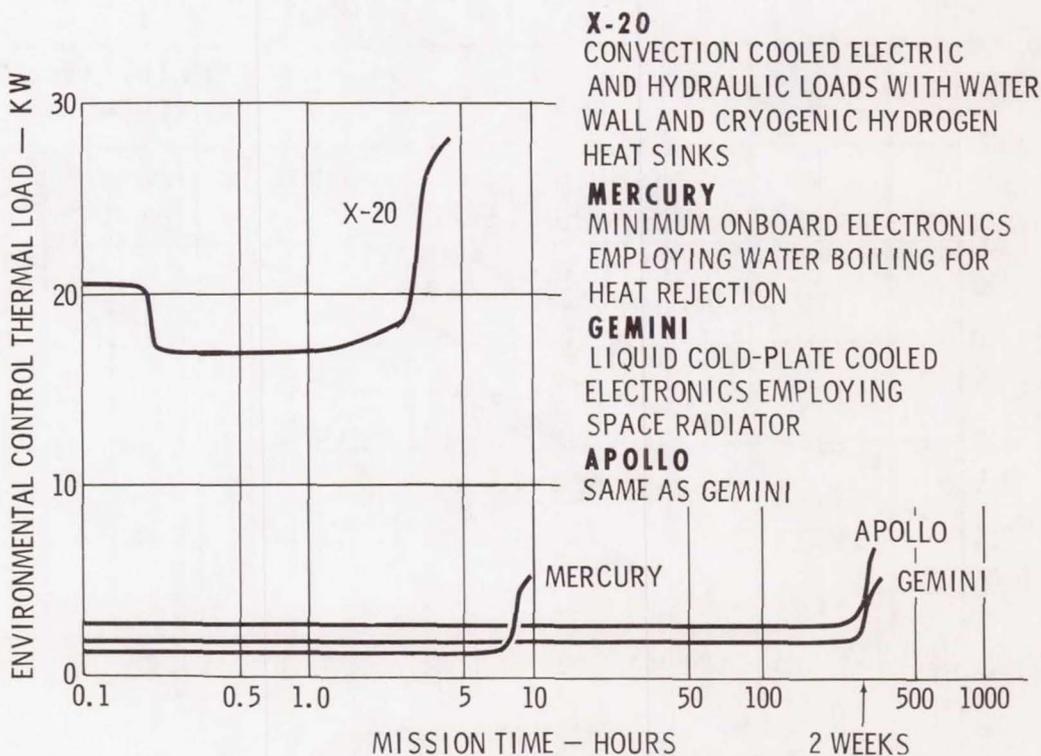


Figure 8.- Thermal control technique.

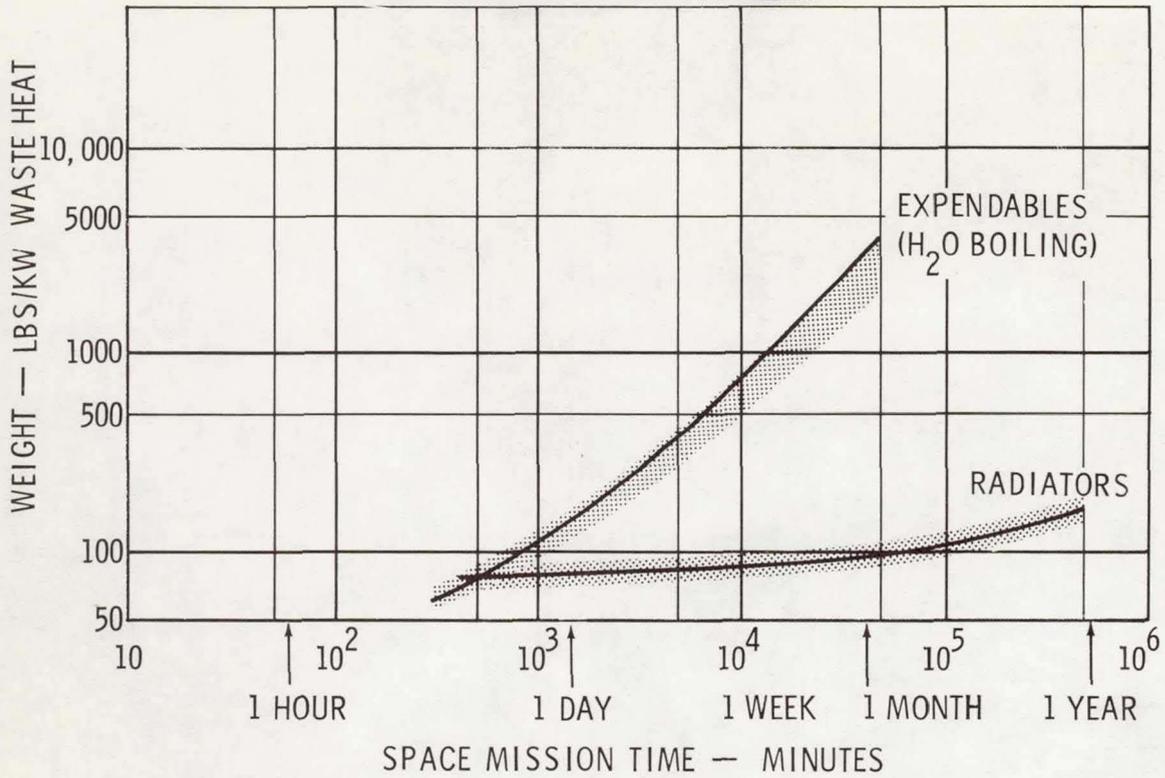


Figure 9.- Methods comparison for manned spacecraft thermal control.

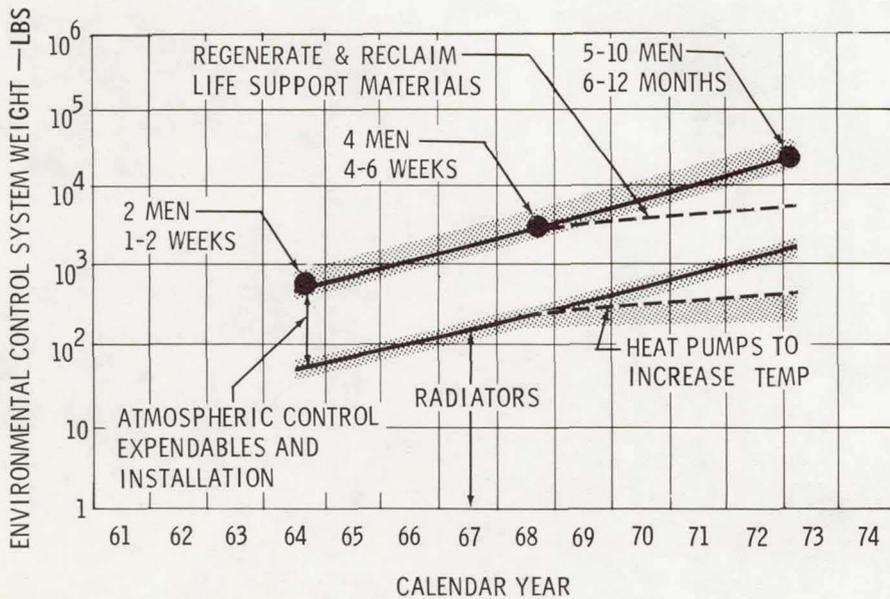


Figure 10.- Environmental control systems weight.

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

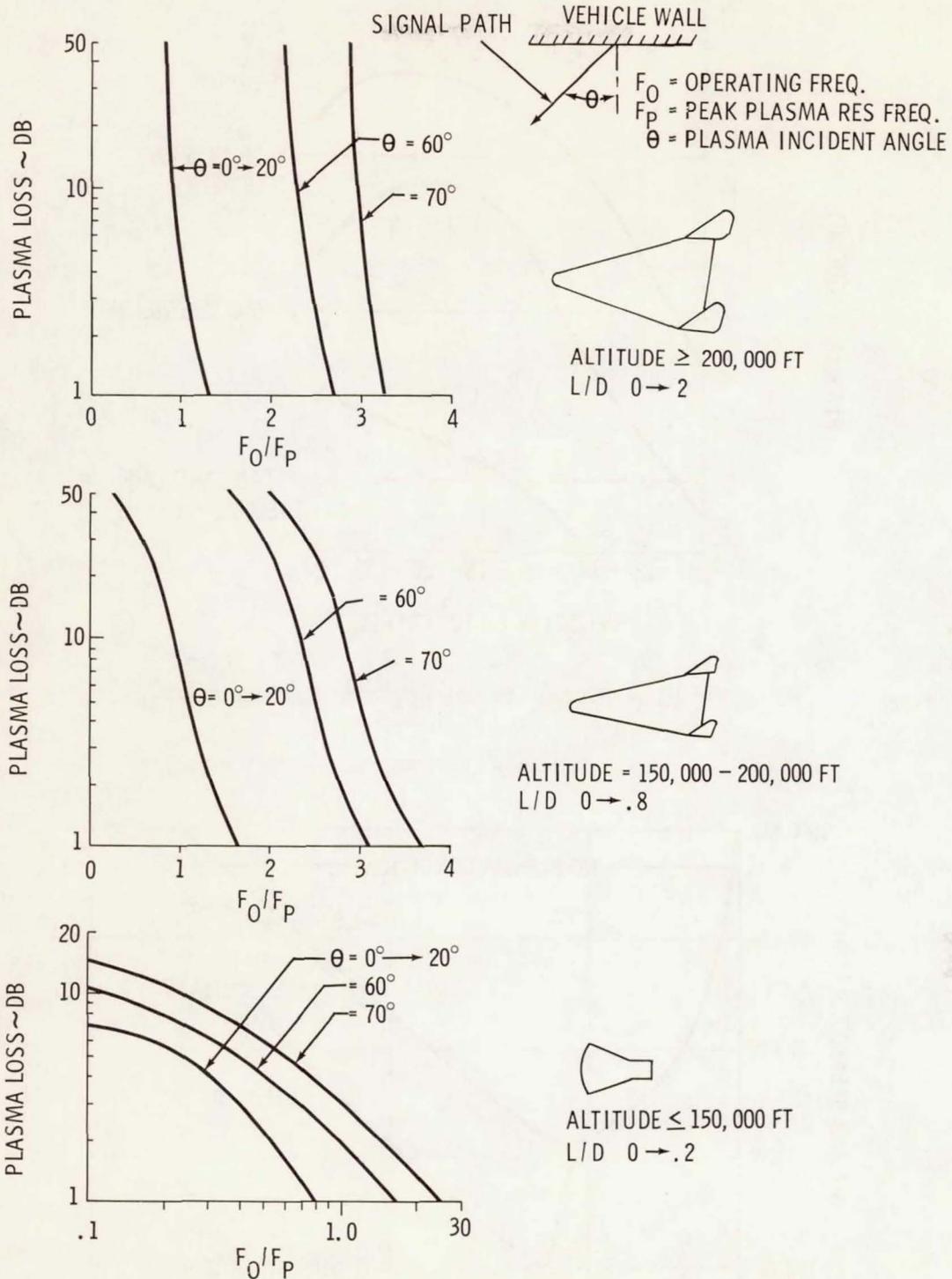


Figure 12.- Plasma attenuation versus plasma incident angle.

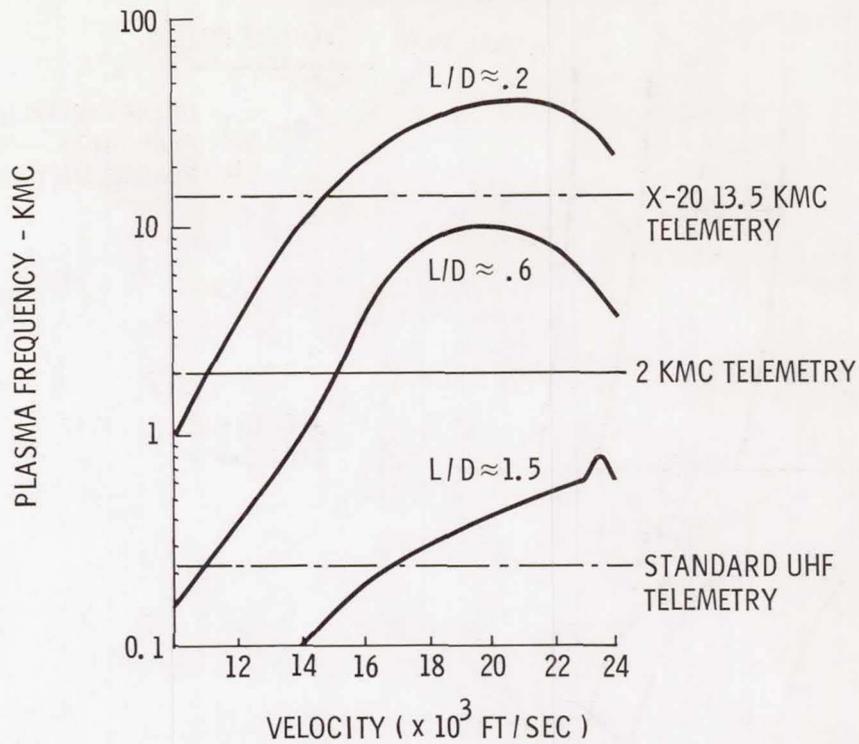


Figure 13.- Plasma frequency for L/D ranges.

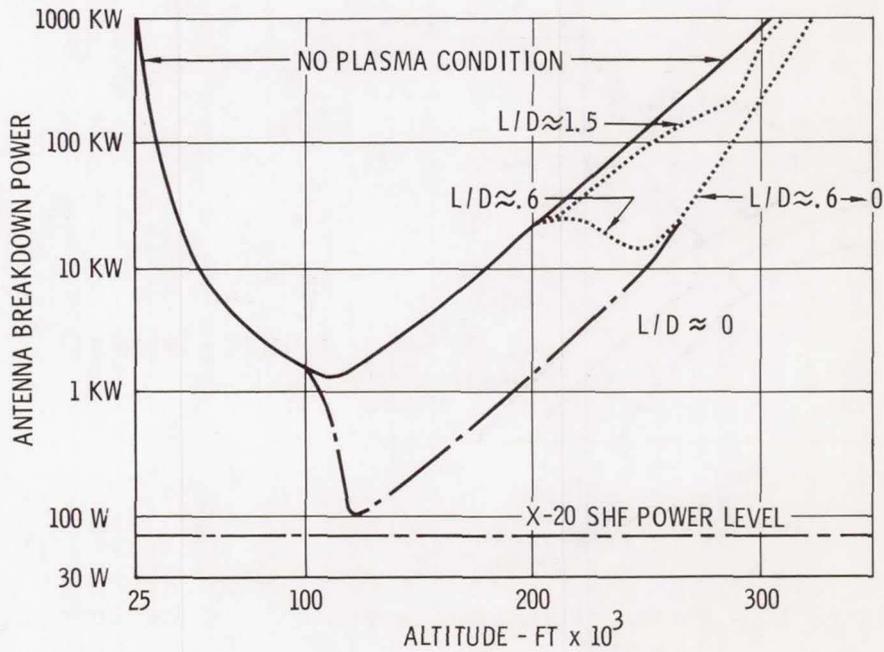


Figure 14.- X-band telemetry antenna power-handling capability.

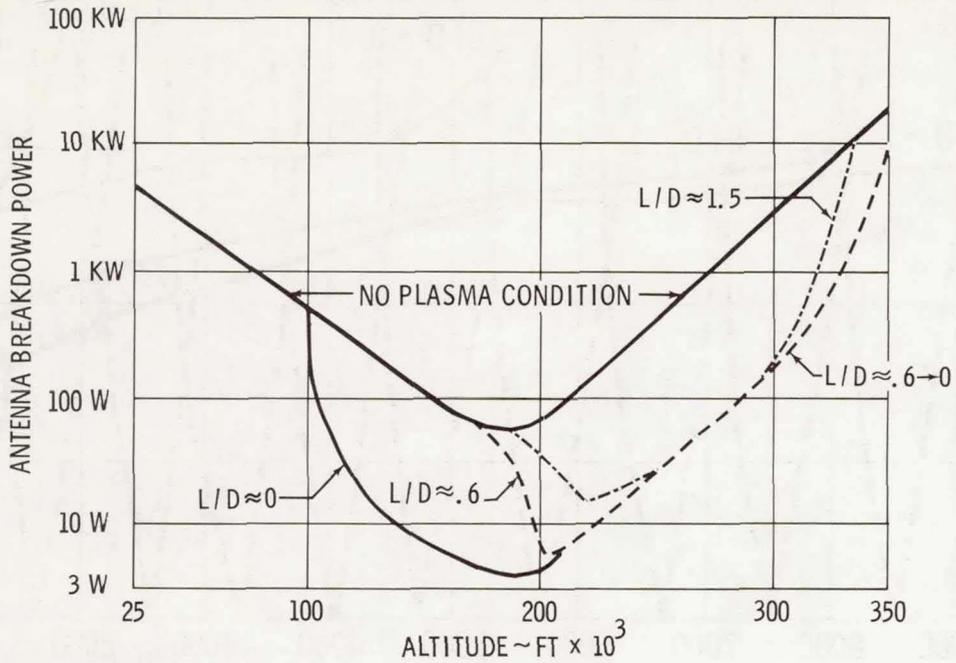


Figure 15.- UHF antenna power-handling capability.

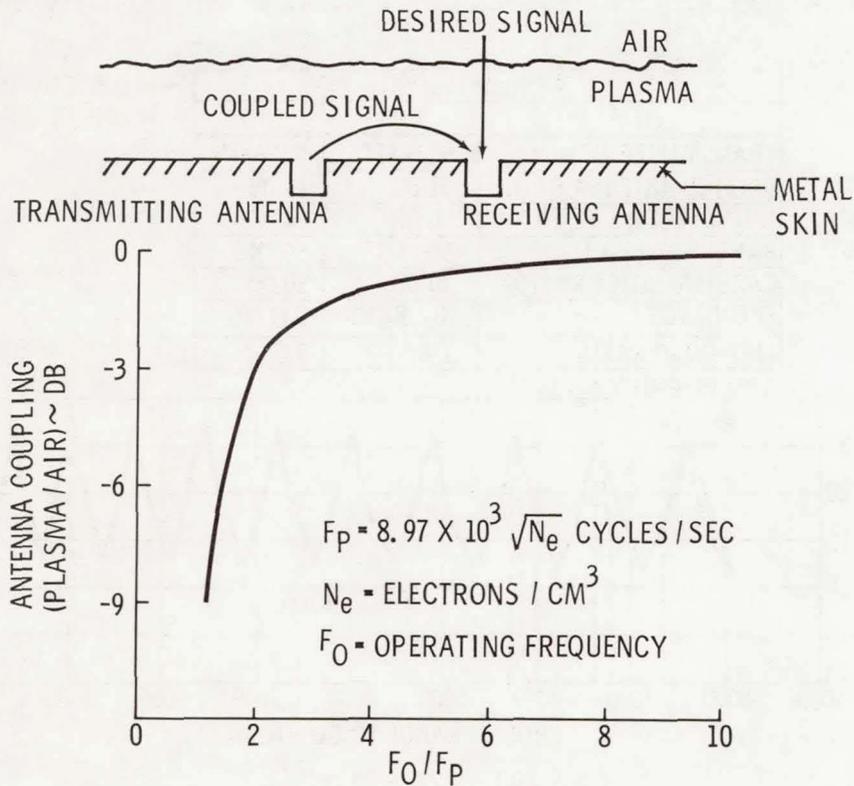


Figure 16.- Antenna coupling in plasma.

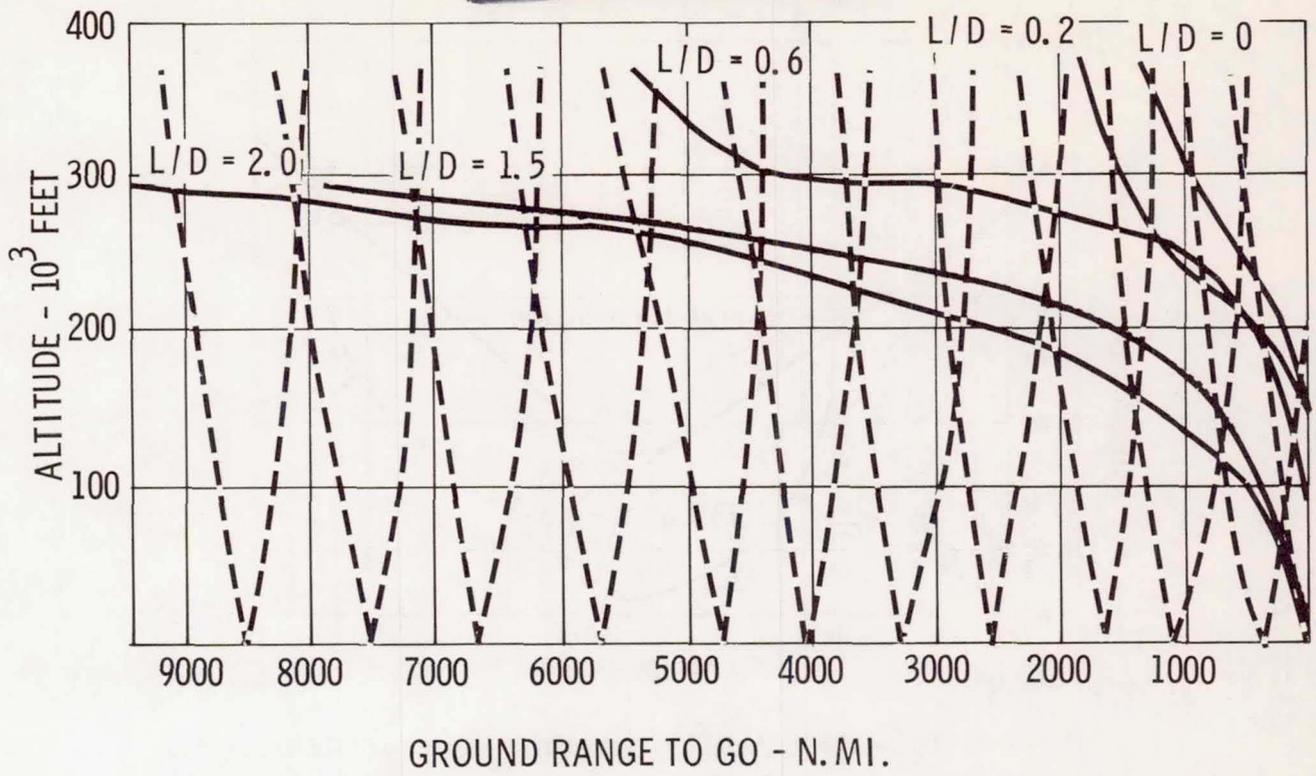
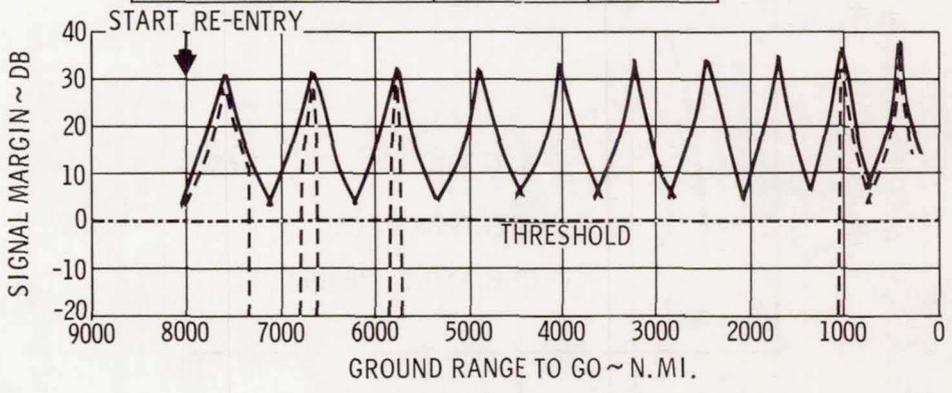


Figure 17.- Re-entry ground stations for L/D, with continuous coverage assumed.

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

— SHF COVERAGE
 - - - UHF COVERAGE

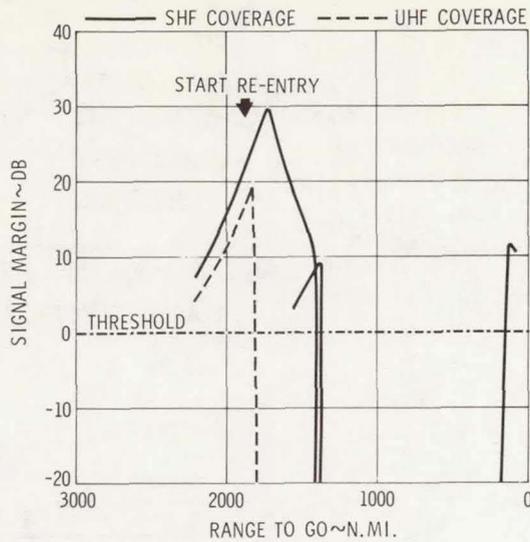


(a) L/D = 1.5

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



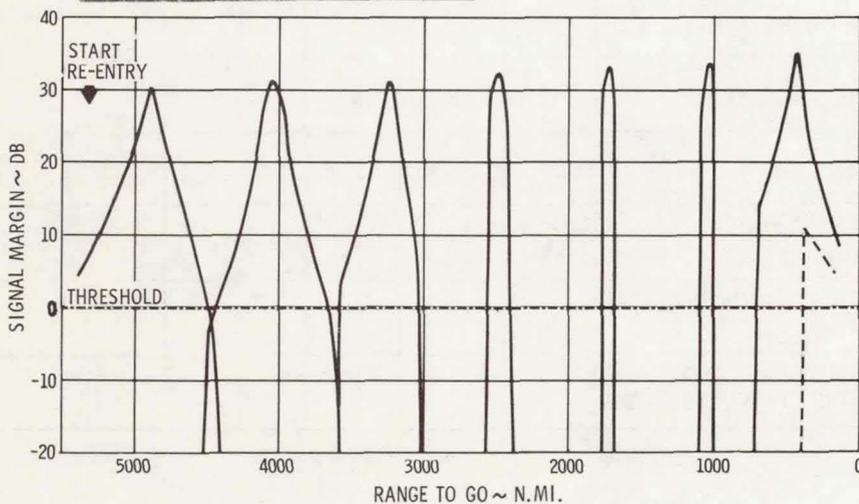
(b) $L/D = 0.2$

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

SHF COVERAGE
 UHF COVERAGE



(c) $L/D = 0.6$

Figure 18.- Concluded.

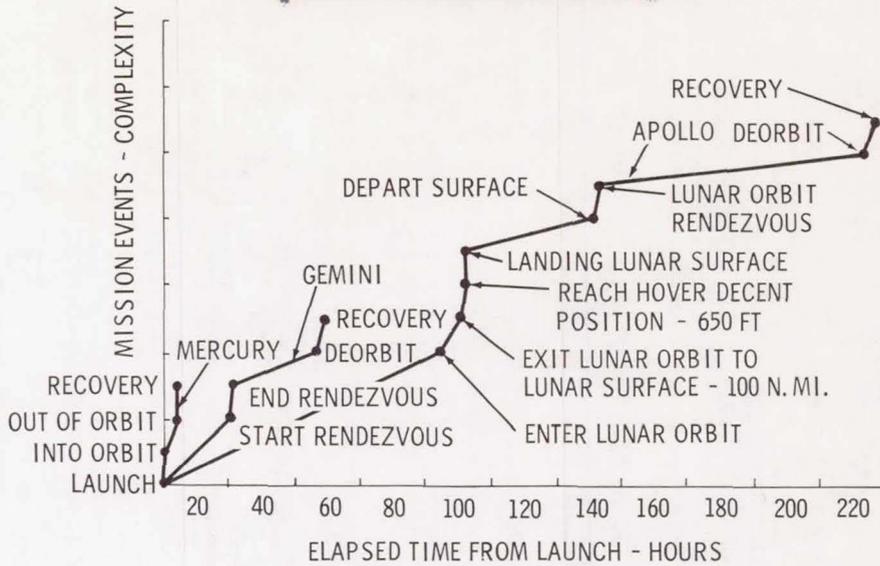


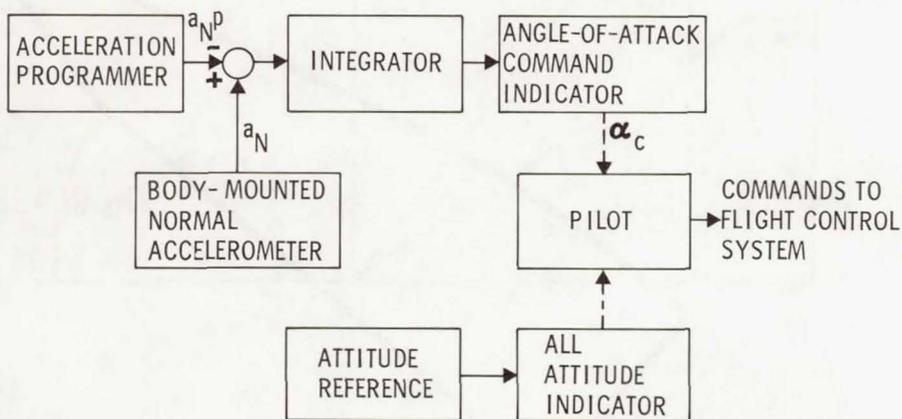
Figure 19.- Mission complexity and duration.

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.



$$\alpha_c = \left[\alpha_n + K \int (a_{np} - a_n) dt \right]$$

$$\alpha = C_{L_{MAX}}$$

$$\alpha = L/D_{MAX}$$

Figure 22.- Backup acceleration guidance subsystems.

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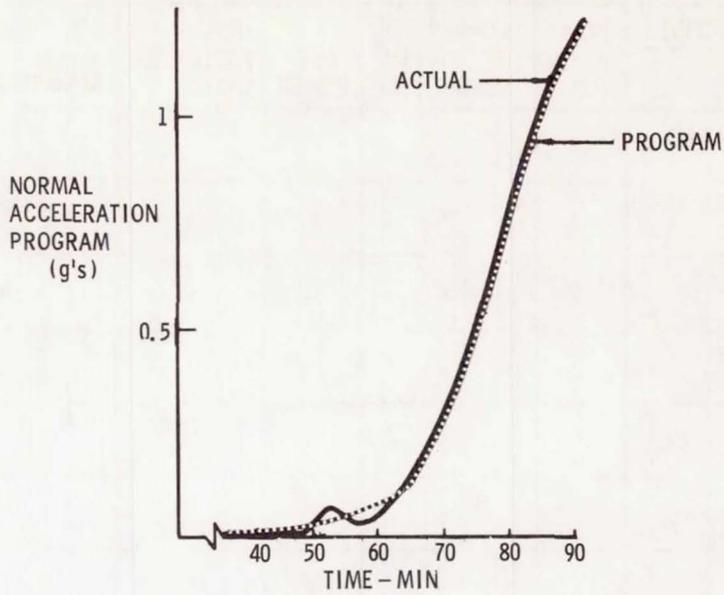
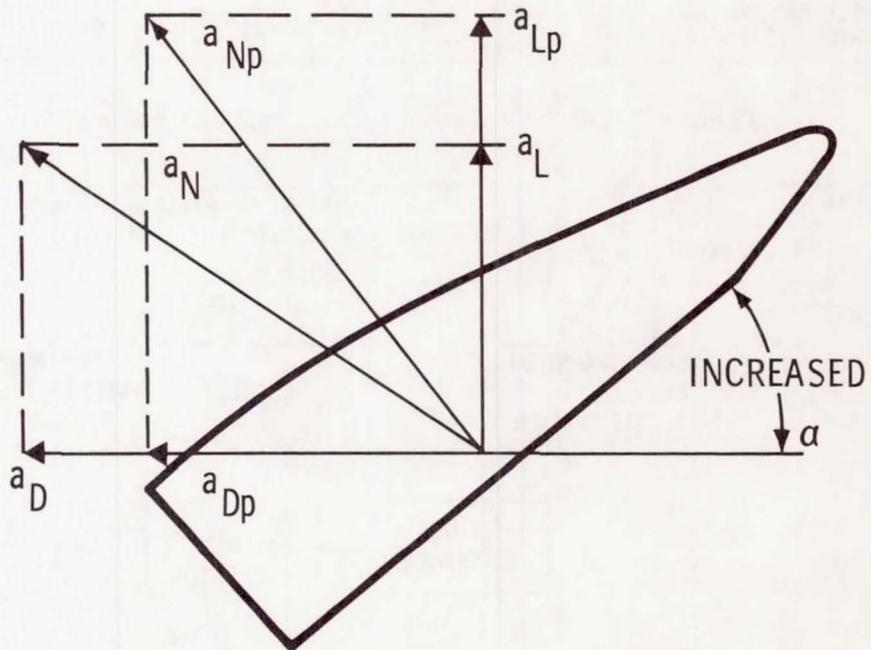


Figure 23.- Point slope acceleration program.



α IS INCREASED UNTIL $a_N = a_{Np}$ A NEW EQUILIBRIUM GLIDE IS ESTABLISHED WHERE $a_D > a_{Dp}$

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

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ACCELERATION COMPARATOR SUBSYSTEM OPERATION WITH INITIAL VELOCITY ERRORS

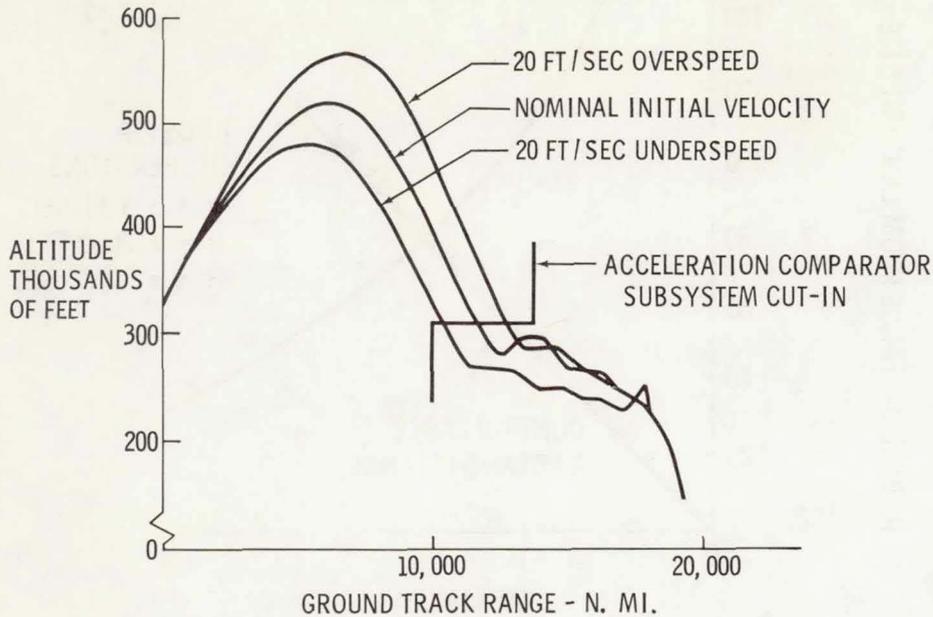


Figure 25.- Acceleration comparator subsystem guidance capability.

DOWN-RANGE		CROSS-RANGE	
SOURCE	(3σ) (N.MI.)	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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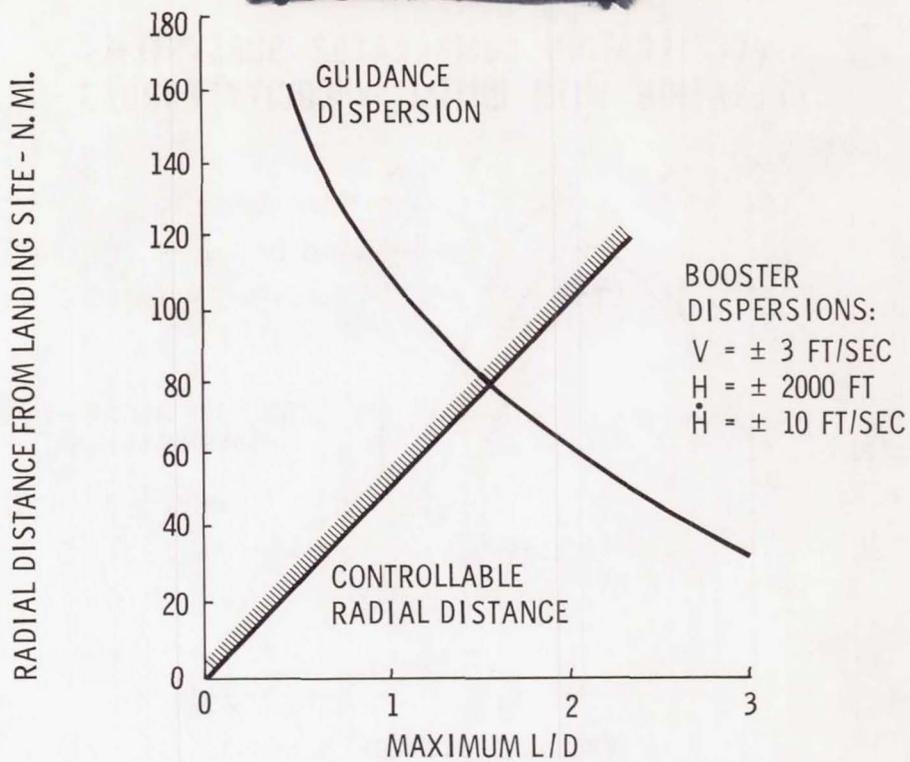


Figure 27.- 3σ performance versus vehicle lift capability.

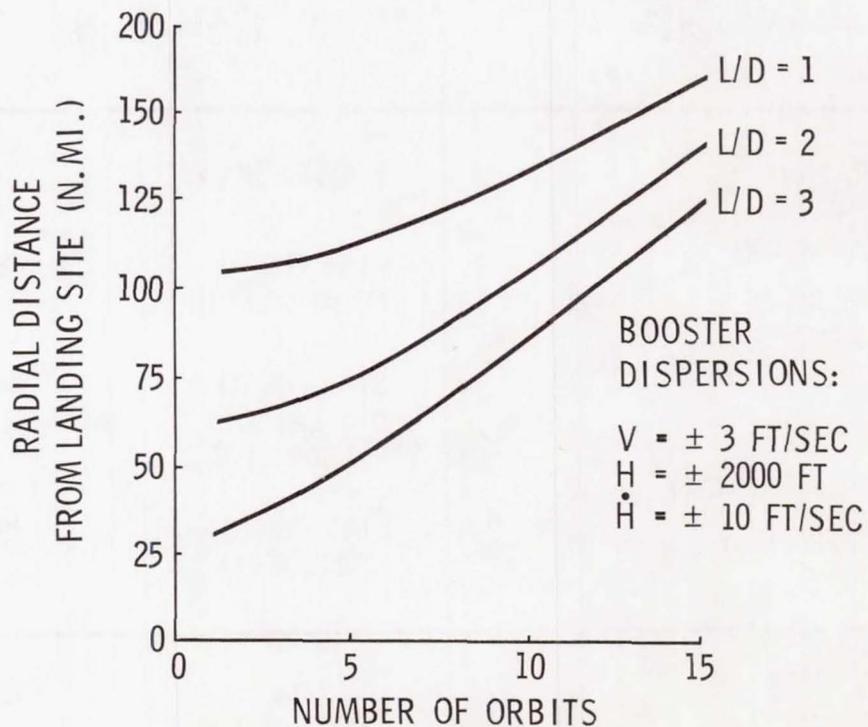


Figure 28.- 3σ dispersions at landing approach.

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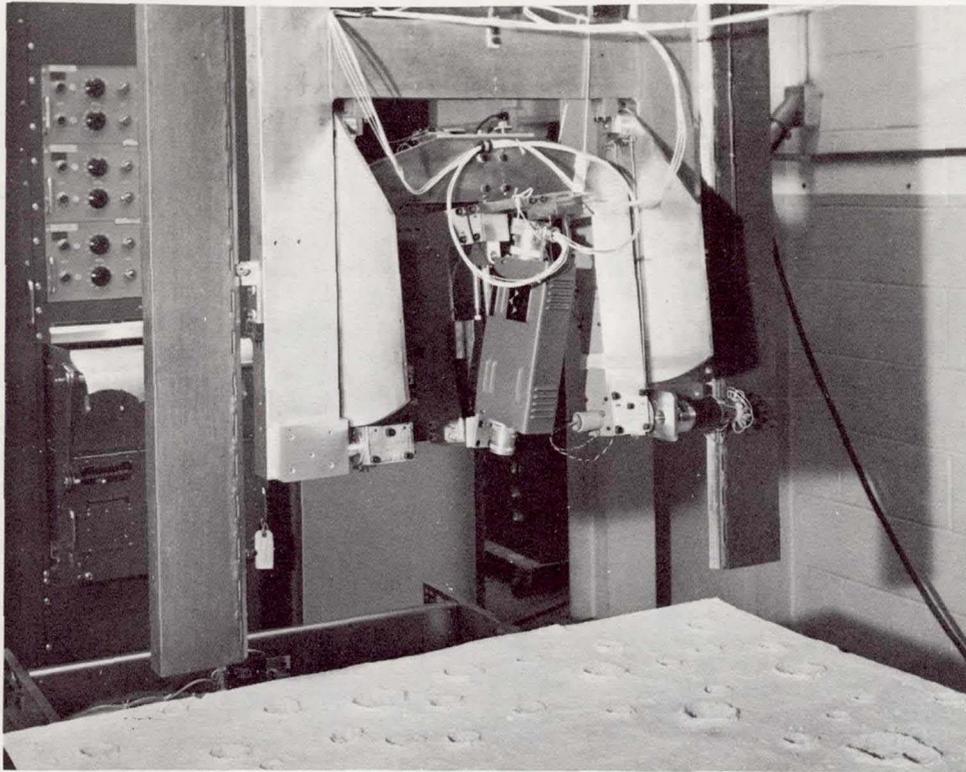


Figure 29.- Lunar-visual landing simulator.

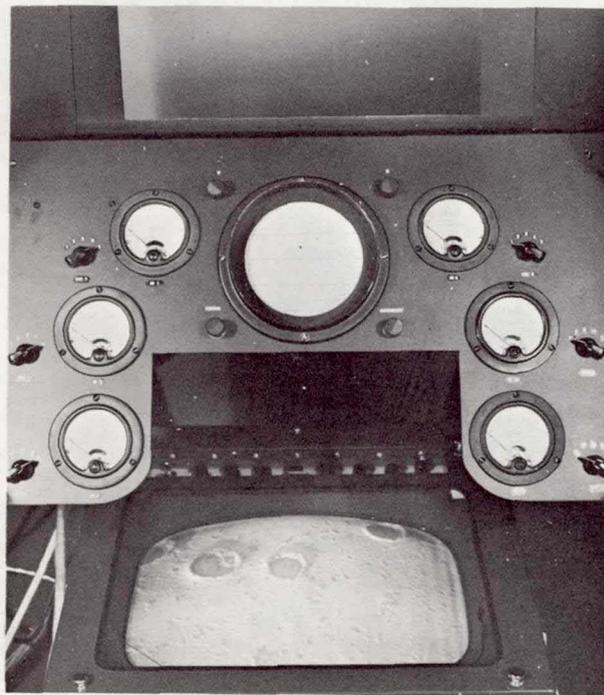
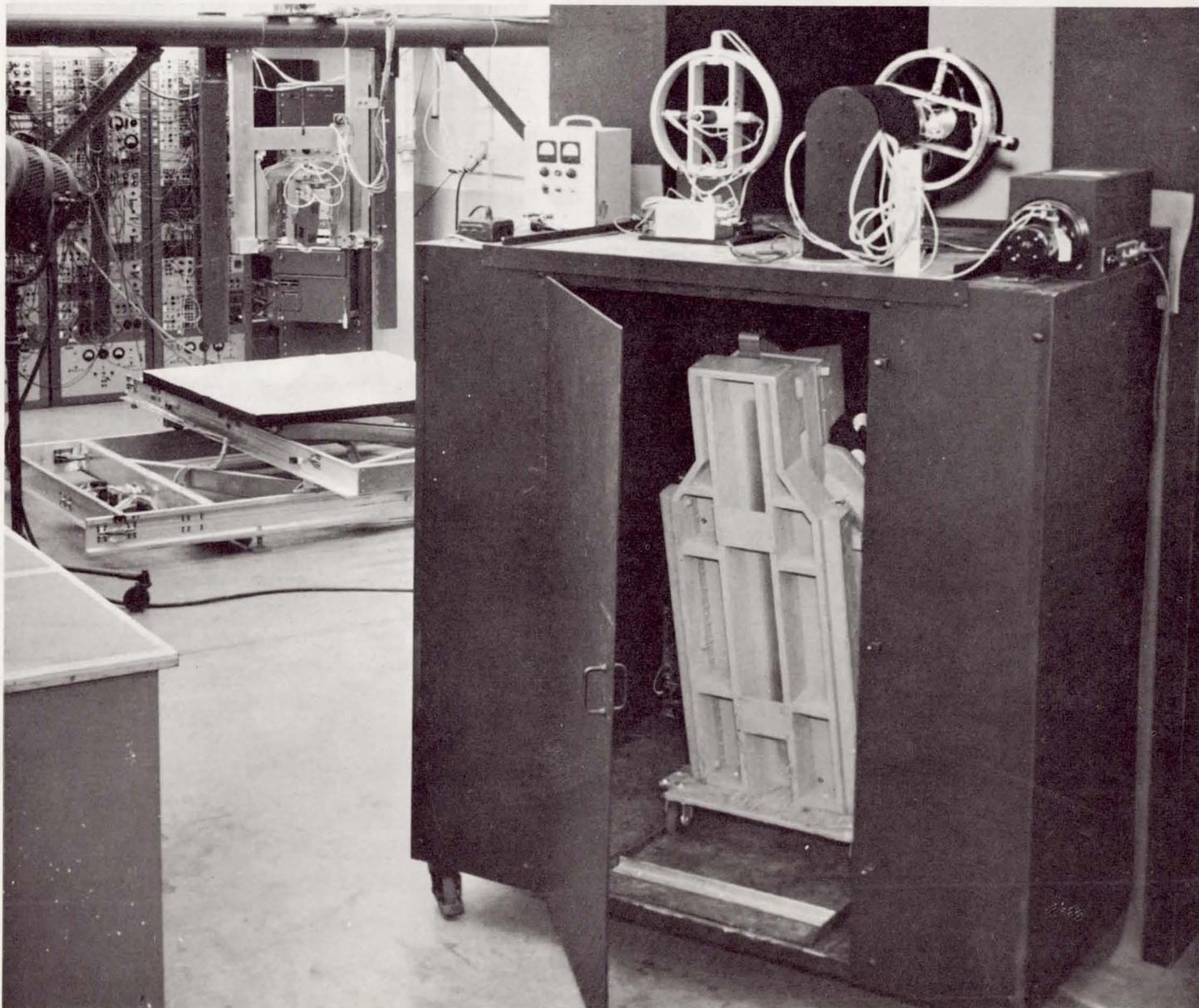


Figure 30.- Lunar-visual landing simulator.

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Figure 31.- Lunar-visual landing simulator.

APOLLO GUIDANCE AND NAVIGATION -
A PROBLEM IN MAN AND MACHINE INTEGRATION

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Abstract

The decision to send man to the moon created the need for development of accurate measurement and data processing equipment integrated into a man controlled operation. This report shows the design of the Apollo guidance and navigation equipment and the displays, controls, and operations utilized by the astronauts in performing a difficult and necessarily accurate task. The compromise between a completely automatic system and one configured for extreme dependence on the man is met with one solution having good features of both approaches. The system is described in which the navigator has complete choice and control of the system operation using his senses and judgement where they are superior, and depending upon mechanisms where man is unable or too stressed to be utilized. The details of the design of the sensors, the computer, and the displays and controls are described in enough detail to illustrate the astronaut operation of the Apollo Guidance and Navigation System.

Section 1. Introduction

When this nation's greatest identified space mission, Apollo, gets underway later this decade after years of planning, design, and experimentation, three men will be responsible to carry through an almost fantastic operation: the landing of man on the moon and his safe return.

This voyage will depend upon near perfect operation of a series of events and equipment. A failure of any of these will be a serious obstacle to mission achievement if not peril to the crew. The boost vehicle, the spacecraft, its propulsion system, ground operations, the crew life support, communications, and so on, are links in this chain. This paper is concerned, in particular, with the equipment and its operation which navigates the space vehicle and steers it through required maneuvers. This is the Guidance and Navigation system of Apollo, herein called G&N.

As part of a manned operation, it became necessary for the NASA and its contractors to determine the degree of involvement that the astronauts would have in the use of their craft. Groundrules had to be formulated as some compromise best understood by describing the extremes. . . .

Completely Automatic. Certainly the manned lunar landing objectives requested by President Kennedy in May 1961 would be met by automatic equipment delivery of an astronaut, wrapped and bundled as it were, in a life maintaining cocoon to the lunar surface; and then, abruptly carrying him back home like any inert payload. But certainly the astronauts, once aboard the vehicle, can con-

tribute mightily to attainment of objectives. The lessons of the Mercury manned space flight program emphasize this.

Completely Manual. At the other extreme could be a design wherein the men are given a rocket, a control stick, a big window, and appropriate charts and tables. This point of view was suitable for Lindberg's adventure where the most energy-efficient path from New York to Paris was only slightly better than that followed by the "Spirit of St. Louis". However, the possibility of a trip to the moon's surface and back is extremely sensitive to the velocity change attainable by rocket propulsion technology now available to push the required payload. The day of "seat of the pants" flying in outer space may not have to wait until Buck Rogers' twenty-fifth century, but today project Apollo must depend upon efficient paths determined by accurate and complex guidance and navigation equipment.

This report will describe the status of the Apollo Command Module G&N system, its relation to the astronaut, and the particular engineering compromises selected for this complex man and machine operation.

First the Apollo mission will be described briefly using Figure 1 to provide foundation for the description of the G&N equipment and operation.

In current plans, an Advanced Saturn Booster will launch the complete Apollo spacecraft and the upper stage boost rocket into a low altitude parking orbit. In this circular satellite it is envisioned that equipment will receive a final period of checkout before committing the spacecraft to escape velocity. With one or more orbits of the earth, the on-board navigation can determine accurately the actual ephemeris required for precise initial conditions for the next phase.

A second thrusting period of the booster, using the last Saturn stage, will inject the spacecraft to the necessary translunar velocity for the mission. After cutoff and staging, the Apollo is made up of the Command Module (CM), Service Module (SM), and Lunar Excursion Module (LEM). These components must first be arranged from their boost configuration to the cislunar operational configuration shown in Figure 2.

As soon as possible after translunar injection, a continuing set of navigation measurements must be made to determine the actual trajectory parameters and velocity corrections necessary. The first correction will be made a few hours after injection using the rocket in the service module. This will be followed by further navigation measurements and with one or two more velocity corrections.

The approach to the moon would now require a final correction about an hour before the larger thrust period to inject into lunar orbit.

The spacecraft assembly would orbit once or twice around the moon taking navigation measurements for an accurate ephemeris, inspecting the proposed landing area, and performing the count-down of the LEM.

The letdown of two of the men in the LEM to the lunar surface, the takeoff from the moon, and the LEM rendezvous with the parent craft left in orbit will not be described in this paper. While on the moon for several hours or up to several days the two men will perform the limited exploration and scientific examination which constitutes the goal of project Apollo.

Finally, back in lunar orbit, the three men set up and inject into a transearth trajectory using the service module propulsion and leaving the LEM in orbit. The trip back to earth will be similar to the outgoing leg. Guidance and navigation will control to the desired reentry corridor by application of several velocity corrections.

Just prior to reentry, the service module is staged and the guidance system is prepared to control the reentry path. This control is performed by steering the direction of the lift, available from the aerodynamic characteristics of the command module, such as to achieve a safe reentry to a prepared landing site.

In this mission we see two distinct modes of spacecraft operation and a corresponding configuration and requirement on the guidance and navigation equipment. First, during boost, translunar insertion, midcourse corrections, lunar orbit insertion, etc. the vehicle assembly is operating under thrusting conditions with requirements on the G&N to provide steering signals for guidance to the required velocity change. Second, during the majority of the time Apollo is in free fall motion following paths determined by the gravity pull of earth and moon. During this time, the G&N must navigate to determine position, velocity, and any velocity corrections required to accomplish the next target.

These operations of guidance and navigation are illustrated in Figure 3. The steering function of guidance operates on angular velocity and acceleration sensed by inertial instruments. The navigation uses optical line of sight angle measurements on which to base the determination of position and velocity. The two functions are interrelated as shown. Part of the navigation function is to provide information on initial conditions and desired velocity changes for guidance purposes during vehicle steering control phases. The guidance, on the other hand, measures changes in velocity actually accomplished during thrusting in order to update the navigation process. (In the above discussion the lift and drag forces during earth atmospheric entry are considered in the same class as the rocket thrusting phases, i. e. non-gravitational forces.)

We now identify four major subsystems of the Apollo Guidance and Navigation equipment:

1. Inertial Measurement Unit: The primary sensor for guidance phases providing measurements of angular velocity and acceleration from inertial instruments.
2. Optics: The primary sensors for the navigation phases providing angle measurements between lines of sight to stars and near planets.
3. Computer: The primary data processor for both guidance and navigation computations.
4. Displays and Controls: The communication interface between the navigator and the rest of the equipment.

Section 2. G&N Phenomena

The use of the equipment identified in the previous section depends upon application of physical phenomena, some of which are well known and understood and others which are unique to the Apollo G&N.

For steering control, the use of gyroscopes, accelerometers, and clocks as measurement devices in inertial guidance is well documented in applications to ballistic missiles control. Nothing will be said here about principles or theory, other than a description of actual hardware in a later section.

Use of optical instruments for space navigation, on the other hand, is not so familiar and indeed some of the phenomena utilized in Apollo are quite new. The basic principle of position determination from observations of heavenly body directions by an earthbound observer is not new. A mariner (or winged aircraft navigator) measures the angle of the sun or star above his local horizon with his sextant. An astronaut away from the earth also may use the horizon usefully or its near equivalent the local earth vertical or direction from him to the earth. Also he may use any identifiable landmark on the earth. Any of these would serve.

The earthbound mariner, from his star elevation data, the time of observation and the navigation tables, determines a line of position on the earth. Anywhere on this line an observer would measure the same star elevation. A second star sighting leads to a second line which intersects the first at his indicated position.

The astronaut would interpret an angle between the earth's direction and a known star as defining a conical surface of position. Anywhere on this cone he would expect to obtain the same angle measurement.

Figure 4 shows a hypothetical situation for this method of space navigation. From his spacecraft the navigator measures the angles from a particular earth landmark to the star Fomalhaut. This places him somewhere on the small cone shown which has its axis in the direction of Fomalhaut and whose half angle is equal to his measurement. A second sighting to the same landmark and to the star Deneb defines the second cone - very flat in this case because the measurement angle was near 90°. These two cones inter-

sect in a line containing the landmark and somewhere on which he is assured to lie. (The earth-bound mariner could stop here because his third coordinate was known explicitly by the fact that he was bound to the surface of the earth.)

The astronaut could complete his fix by utilizing a second earth landmark separated from the first and any star. This would work well in the vicinity of the earth but accuracy degrades as the apparent size of the earth gets small. So the third sighting shown in Figure 4 is with respect to the moon. In this case the moon's horizon or limb is used rather than a lunar landmark. The third cone, defined by this sighting of the elevation angle of the star Antares above the moon's horizon, intersects the previously determined line of position at the indicated location of the spacecraft. Actually the three cones have four mutual intersection points. The wrong three could be discarded easily in a practical situation.

By a technique such as this it is theoretically possible for the space navigator to determine a fix of his position with respect to the earth-moon system. Similar measurements repeated at some later time in his trajectory would provide data to determine velocity and the free fall path describing the spacecraft trajectory. The method described implies that the three angle measurements could be made simultaneously. Practically this would put too much of a burden on the navigator and/or equipment design to be considered for Apollo.

The navigation measurements for Apollo are the angles between the planets and stars, as described above, and the time the measurements are made. These measurements are taken in time sequence separated from 15 minutes to several hours apart according to an optimum plan. The details of the Apollo navigation scheme are described elsewhere.¹ Some of the important features follow.

The navigation measurements are used to determine position and velocity on any free fall trajectory - such as earth or moon satellite orbits or the transearth or translunar phases.

A measurement schedule is determined prior to the trip for approximate time of sighting, identity of planet, and identity of star such that the greatest enhancement of navigation accuracy occurs for the astronaut's effort under assumed accuracy of measurements and other existing limitations on the navigator, his equipment, and available celestial objects. For a normal flight, about 40 sightings in midcourse, each way to and from the moon, are anticipated.

Each sighting is used by the on-board computer to improve all six components of position and velocity in an optimum manner. The computation scheme also keeps an estimate of the uncertainties in its determination of position and velocity.

The system will accept navigation measurements of any form, such as ground track data or time of star-moon occultation as well as the planet-star angle measurements described above.

Velocity corrections to improve target conditions are made only when the knowledge of the required correction is sufficiently accurate and

large enough to make the rocket start and expenditure of maneuver fuel worthwhile. Approximately three corrections are anticipated for each mid-course leg of the trip. The level of fuel expenditure for either the outgoing or incoming leg is equivalent, roughly, to 100 feet per second rms velocity change.

Planet to star angles during earth-moon or moon-earth midcourse phases will be measured in Apollo with a visual sextant instrument capable of an rms accuracy of 10 arc seconds (about 0.05 milliradians).

Angle sightings, with respect to the moon, can be taken either to lunar landmarks or the horizon. An examination of good lunar photographs show an ample supply of distinctive landmarks on the near side and it may be safely assumed that, in the coming years, satisfactory marks may be mapped for the far side. The illuminated lunar horizon or limb is quite distinctive against the dark sky. Consideration of the shape and motions of the moon, altitude of the landmarks, and mountains on the limb must be included if the best accuracy is to be obtained. However, the problem is only one of obtaining the data, maps, and charts. A particular sighting is limited only by the systematic illumination of the moon by the sun.

The situation with earth referenced sightings is not so clear cut because of the effects due to the atmosphere. Cloud cover might obscure a particularly desirable landmark and the horizon seen from space shows no distinctive edge against the sky.

One attack on this problem has investigated earth-direction determination using longer wavelength radiation. The use of visual techniques, however, have so many advantages for manned Apollo that the problems associated with earth atmospheric effects at optical frequencies has received considerable attention.

Weather generated cloud cover over landmarks occur with a frequency which varies over the earth. Some areas are usually free, others may be usually covered. The problem of how many of the good distinctive landmarks are available at any time is clearly amenable to statistical analysis using local weather history for data. Work in progress² shows no reason why landmarks cannot be used as an excellent reference for earth-direction measurements most of the time. If good landmarks all become obscured, recourse to the horizon is possible.

The use of landmarks in sextant operation is illustrated in Figure 5. The figure is made from an accurate photo mosaic simulation* of the San Francisco Bay Area and hypothetical clouds as seen from 2500 miles with a 1.80 field telescope. This 28 power optical instrument will also have a second, displaced line of sight to pick up a known star and superimpose it onto the scene. The Apollo sextant instrument and its use will be described in more detail in later sections. By controlling the aim of the instrument and the offset angle of the optical axis for seeing the star the astronaut can superimpose the star, shown as a

* The reproduction process for this document severely limits the resolution available on the original simulation.

white dot, onto a particular landmark for which he has the geographical coordinates. The navigation measurement consists of the measured angle between the lines of sight and the time at the instant of superposition.

Use of the illuminated earth's horizon is illustrated in Figure 6. The observer out in space above the atmosphere sees, on the sunlit side, the earth-color blend into a brilliant white which turns toward sky blue and then gradually to the black sky as he scans to higher altitudes. The bright light is sunlight scattered in passing through the atmosphere. Light from an object on the horizon at sea level must pass through 23 atmospheres to reach an observer in space, whereas the light from an object straight below him passes through only one atmosphere. The object at the sea level horizon has its light scattered and attenuated such that it is invisible relative to the intense scattered sunlight.

In looking from space through the earth's edge at about 100,000 feet altitude above the sea level horizon, the observer sees the sky through one atmosphere. He should observe the same intense blue as is seen when looking straight up through the same amount of sunlit atmosphere from the ground. If the brightness of a little patch of sky at 100,000 feet is measured from space, one would expect to obtain a value very close to some standard value. This value could be computed on the basis of the sunlight aspect angle and would be only slightly affected by local sea level atmospheric pressure. At this altitude the brightness of the scattered sunlight decreases, due to a corresponding density variation, by a factor of two for each 17,000 foot increase in the altitude. Thus a measurement of absolute brightness to 10% should determine the altitude of the line of sight with an accuracy of approximately 2500 feet. An obvious advantage of working with line of sight measurements at this 100,000 altitude is that it is well above all common cloud types which would interfere with the measurement.

The instrument for this measurement includes an automatic star tracker and horizon photometer attachment in place of the sextant visual eyepiece. The navigator uses the second optical instrument - a low power telescope - to sense visually and then control the spacecraft attitude as required for making the above measurement.

On the dark side of the earth, the 100,000 foot atmosphere could be sensed by the refraction effect on the background stars, Figure 7. If two stars are observed - one setting near the horizon - until the apparent angular distance between them decreases by one arc minute vertical component, then the line of sight to the lower star is at some determinable point near 100,000 feet altitude where the density gradient is well known. The navigation measurement, in this case, consists of the time at which the one minute of arc change is complete. The earth's limb is now determined with respect to the background star - the setting star.

This measurement is similar to the occultation time of stars by the distinct moon's limb, a phenomena available to Apollo navigation. A closer analogy to occultation phenomena uses the photometer described earlier to sense the intensity

change, Figure 8, as the starlight sinks into the earth's atmosphere. The photometer would be set for the reference intensity of the particular star well before it is occulted. Once the attenuation reaches the preselected level, the time is recorded. This intensity change is predictable and is due primarily to two phenomena: The scattering of the light out of its path by the air and the light dispersion due to refraction in passing through the atmospheric density change.

These occultation measurements depend upon the existence of stars setting behind the planet's limb. This occurs very often while in earth or moon low satellite orbit and frequently enough in the cislunar trajectory to provide a useful source of navigation data.

The sextant operations of landmark-to-star or horizon-to-star angle measurements are excellent and natural operations on the part of the astronaut navigator with reasonable adroitness as long as the rates of change of the angles and directions are not excessive. This is the case during midcourse translunar and transearth operations for Apollo. Landmark sightings when in satellite orbit around either planet, however, must use instruments that can cope with the high rates involved and the short time that any particular landmark is in view. Fortunately, at these altitudes, the angular accuracy required for the landmark direction is considerably relaxed. In 100 mile altitude orbit, accuracies of the order of a milliradian or so (corresponding to 0.1 mile error) are sufficient. Thus, for orbital navigation, the high magnification available from the sextant is not used. It is replaced by a single line of sight, low power, wide field telescope whose optical direction with respect to the spacecraft, when on target, is compared with the orientation of the inertial guidance stabilized member. Of course, the stabilized member had been previously aligned to the stars with the same instrument. These data allow the computation of landmark direction with respect to the stars as limited by the inertial guidance stable member alignment and drift and the accuracy of the angle transducers reading the telescope directions and the stable member orientation. The use of this wide field telescope for orbital navigation is described in more detail in a later section.

The choice of navigation measurement techniques for use by Apollo has been primarily predicated on the requirement for completely on-board capability. This is necessary, certainly, on the far side of the moon out of reach of earth tracking or communications. However, earth tracking information, when available to the astronaut navigator and when of accuracy which is judged capable of improving the on-board navigation, would certainly be used. The on-board computer will be able to accept ground based data as well as the astronaut's sightings and make a proper weighting of their estimated accuracies in influencing the computed trajectory. The use of earth based tracking becomes primary in the event of failure of the on-board optical equipment.

In this same vein, cooperative land targets could be considered. Many points on the earth are cloud free practically all the time but unfortunately have no distinctive features. The African desert, for instance, might be a logical place to

install a flashing high intensity light during the mission to provide an almost certain landmark during the local night.

Section 3. Equipment Description

This section will give a physical description of the G&N equipment. Later sections will describe the modes of use and astronaut operation.

Figure 9 shows a cutaway view of the Apollo command module with the major elements of the guidance and navigation equipment shown in their approximate location.

During stress periods the three astronauts will be protected by their couches (the third couch shown dotted) in front of the main display panel where necessary operation of guidance and navigation can be performed. These periods, when all the crew is confined to the couches, are limited; immediately before and during earth launch, possibly during translunar injection, and during earth re-entry. The thrust levels during the rest of the mission are small and acceleration that is felt is of the order of $1g$ or less. The figure shows the center couch - for the navigator - with the couch knees folded so that he may make sightings at the navigation station while in earth orbit prior to translunar injection. Before starting translunar injection, he may go back to his couch for protection during the rocket burning phase. After this the couch is removed, folded up, and stored under the pilot's couch on the left. This provides considerable floor area for other crew tasks and allows operation at the navigation station in a standing position. This configuration is maintained until just before earth atmospheric re-entry when the center couch must be again installed for the coming stress.

The navigation station, which contains most of the guidance and navigation equipment, is located in the area called the lower equipment bay. Starting from the top in Figure 9, the first item identified is guidance and navigation display and controls, D&C. The sextant, SXT, is the two line of sight instrument for midcourse navigation angle sightings. The scanning telescope, SCT, with its two eyepieces is the single line of sight low power unit for earth and moon orbital sightings and provides general viewing. The IMU is the Inertial Measurement Unit used for inertially measured attitude signals and velocity changes. The Apollo Guidance Computer, AGC, is the central data processing, general-purpose, digital computer. Special power supplies, servo amplifiers, and miscellaneous electronics are contained in the Power Servo Assembly, PSA. The junction box and cabling complete the guidance and navigation hardware in the lower equipment bay.

Figure 10 is a photograph of a full-scale installation mockup of the Optics and IMU in the lower bay with the display panels removed. The optics, without the eyepieces installed, appears above the spherical IMU. Both are mounted on a rigid framework, called the Navigation Base, used so that angle measurements can be referenced between the two instruments. Space for the miscellaneous electronics of the PSA is shown below the IMU. The computer is installed in the space just underneath the mockup.

The display and control panels are shown installed in the mockup of Figure 11. Details and operation will be described in the following sections.

Figure 12 shows a cutaway diagram of the wide field, low power, single line of sight scanning telescope, SCT. Figure 13 is a cutaway of the other optical instrument: the narrow field, high power, two line-of-sight sextant, SXT. The significant details and use of these instruments will be described in Section 7.

The inertial measurement unit is shown schematically in Figure 14. Three gyros and three accelerometers are carried conventionally in a three degree of freedom gimbal structure. The outer axis of gimbal freedom, OGA, is mounted parallel to the re-entry control wind axis so that the high angular rates, during reentry roll control of lift, are "unwound" by the outer gimbal. This places the outer gimbal axis 33 degrees from the spacecraft symmetry axis. The inner gimbal, or stable member, carrying the inertial components, is aligned prior to each use of the IMU such that the inner gimbal axis, IGA, is normal to the plane of any planned trajectory or attitude turning maneuvers. Thus in orbit, for instance, the inner axis would be placed normal to the orbital plane so that the relative spacecraft rotation caused by keeping a fixed attitude with respect to the local vertical will not cause gimbal lock since it is "unwound" by the inner gimbal. By aligning the stable member in this fashion before each mission phase the three degree of freedom gimbal structure avoids danger of gimbal lock without the weight, size, and operation penalty of a fourth degree of freedom. However, unusual maneuvers of the spacecraft could bring the outer axis around into parallelism with the inner axis where the inertially fixed orientation of the stable member would be lost and re-alignment would have to be performed again.

Operations with the IMU are described in more detail in Section 5.

Figure 15 is a photograph of the stable member of a display model of the IMU. The three $2\frac{1}{2}$ " diameter gyros, 25 IRIG, and two of the three 1.6" diameter accelerometers, 16 PIPA, are shown. The inter-gimbal assemblies on each end contain slip rings, bearings, servo torque motors, and electromagnetic resolvers. Figure 16 shows a higher stage of assembly of this model. The gimbals are not conventional rings but are pairs of hemispheres of thin section aluminum. The device at the bottom on which the model rests is one of a pair of blowers which is used to circulate air for heat transfer. Figure 17 shows the complete assembly.

Figure 18 shows the package of miscellaneous support electronics called the Power Servo Assembly or PSA. Figure 19 shows a photo of the computer mockup. Both are constructed with removable trays on which are plugged modules. The modules are replaceable for inflight repair. One tray of the PSA and one tray of the computer carry spare modules. The design incorporates multiple use of common modules to gain maximum use of carried spares. Characteristics and operation of the computer are described in a later section.

Section 4. Operation Modes

The purpose of this section is to give a brief description for each of the various modes of operation of the utilization of the hardware previously described. This will provide an over-all picture before more detailed descriptions of operations are given in the following sections.

Major Subsystems

Figure 20 identifies the major subsystems of the guidance and navigation system. The left-hand column of boxes in the figure depicts the input sensing devices of the system. Similarly, the center column depicts the control and data-processing devices. The right-hand column lists the other spacecraft functions of direct concern to the guidance and navigation functions.

The data sensors of the G&N system are the radar, scanning telescope, sextant, and inertial measurement unit. The latter three are mounted on the "navigation base" in the command module of the spacecraft so that angle measurements can be related to a command rigid structure representing the spacecraft.

(The radar, the first sensor represented in Figure 20, is utilized in lunar landing operations not covered in detail in this paper.)

The G&N system performs its control and data processing by the astronaut using: display and controls, the computer, the coupling display units, and the power servo assembly shown in the second column of Figure 20.

The Apollo guidance computer (AGC) is the data-processing center of the guidance and navigation system. It is a general-purpose digital computer having a large quantity of wired-in memory and programs and sufficient erasable memory to meet all requirements. (See Section 6.)

The coupling and display units (CDU) are used to transfer angular information among the IMU, the computer, and the spacecraft autopilot, as well as to display various angle parameters to the astronaut.

The power servo assembly (PSA) is a support item. It provides various types of d-c and a-c power to the rest of the G&N system and also serves as the location of various other support electronics - in particular, the servo control amplifiers for the IMU and optics drives.

Three spacecraft functions outside the G&N system and part of the spacecraft stabilization and control system are of direct concern to the G&N system and are shown on the right of Figure 20. The attitude control system, the first, determines spacecraft orientation during non-accelerated phases and affects the ability to make optical sightings for navigation and IMU alignment purposes. The second is the equipment for control of propulsion-rocket thrust magnitude - starting and stopping these engines and modulating their thrust level when appropriate. The guidance system sends signals to initiate these functions. Finally, the autopilot function of the stabilization and control system receives the guidance steering error signals during the accelerated phases to

direct and control the rocket directions (or lift forces during reentry) so as to achieve the desired trajectory.

The use of these subsystems in carrying out the guidance and navigation functions during the important phases of the Apollo mission will be explained using block diagrams in the same format as Figure 20.

Guidance and Thrust Control, Figure 21

The G&N system here controls rocket thrust during the powered or accelerated phases of a mission and controls reentry lift during the reentry phase. The IMU is the only sensor used in this phase. It produces two outputs: velocity increments, which go to the computer (AGC), and spacecraft attitude, which goes to the coupling display units (CDU). The velocity increments are measured by the accelerometers in the IMU stabilized framework within which the computer determines the steering signals that it sends to the CDU. These increments are then compared within the CDU with the spacecraft attitude measured by the IMU gimbal angles, in order to generate attitude errors. The autopilot acts on these attitude errors and controls the rocket-motor thrust direction (or re-entry lift direction), causing changes to the spacecraft attitude so as to bring these errors to zero. Meanwhile, on the basis of these velocity measurements on which the steering signals are based, the computer also determines the rocket-engine cutoff and, when appropriate, modulation of the thrust. The display and controls (D&C) provide monitor functions to the astronaut. He can take control, of course, in various secondary modes to enhance mission success.

In order to carry out properly this guidance phase, the stabilized member of the IMU must be realigned with the appropriate fixed coordinate frame. There are two phases of this alignment: coarse and fine.

IMU Coarse Alignment, Figure 22

Neither the sextant, the scanning telescope, nor the radar are involved in the coarse alignment of the IMU. From the action of the stabilization and control system, the spacecraft has an expected or estimated attitude. This would be determined by the free-fall attitude control constraints for the vehicle. Based upon this orientation, the astronaut can use the computer to determine the desired IMU gimbal angles that would place the IMU stabilized member in the desired orientation for its next control use. These angles can be fed automatically to the CDU, which compares them with actual gimbal angles and generates error signals giving the difference between actual gimbal angles and desired gimbal angles. These error signals go to the IMU gimbal servos and rapidly move the stable member around to the orientation required. This coarse alignment results in an alignment accuracy on the order of one degree except as limited of course by the knowledge of spacecraft attitude as determined by the spacecraft stabilization and control system.

IMU Fine Alignment, Figure 23

The IMU fine alignment, as contrasted with the IMU coarse alignment, depends upon optical measurements. The sextant is the primary sensor and is used for tracking with its articulating line of sight the direction to a star that is to be used as the orientation reference. The scanning telescope, with its wide field of view, is used for acquisition and to check that the correct star is being sighted. The astronaut, through the display and controls, puts the sextant cross hairs on the star, thereby generating the star direction angles with respect to the navigation base. The IMU gimbal angles with respect to the navigation base are then measured, using the CDU to feed these angles to the computer. There a comparison between the actual and required gimbal angles is made. If the gimbal angles are not appropriate, gyro torquing signals are sent to the gyroscopes on the stabilized member of the IMU to drive the gimbals to the orientations that match up with the requirements for the IMU fine alignment. The accuracy of this fine alignment is of the order of a minute of arc. Since a single star direction can give only two degrees of freedom of orientation reference, a second star sighting is then necessary to complete the three-degree-of-freedom fine alignment of the IMU stabilized member.

Midcourse Navigation, Figure 24

The principal sensor used in midcourse navigation is the sextant with its two lines of sight. In its field of view, the star and the landmark are superimposed by the astronaut through the use of the sextant controllers. The navigator astronaut can also look through the scanning telescope for acquisition and identification as required, using its wide field of view. When the two targets are superimposed, the sextant feeds to the computer the angle between them. The computer uses this information to update its knowledge of free-fall trajectory, so that it can provide, at any time, information on position, motion, and trajectory.

The sextant has only two degrees of articulation with respect to spacecraft. Since there are two lines-of-sight, however, each requiring two degrees of freedom, additional freedom is required. This is obtained by control of the spacecraft attitude pitch and roll on signals from the navigator.

Orbital Navigation, Figure 25

During navigation phases in which the spacecraft is in orbit close to the moon or the earth, angular measurements do not have to be quite as accurate, but angular velocities are rather extreme. In this case, the scanning telescope is used as a single-line-of-sight instrument to track a landmark. With the IMU prealigned to a star framework, it is simultaneously giving spacecraft and navigation base attitude with respect to that framework while the scanning telescope gives landmark angles with respect to the navigation base. From these two subsystems, accordingly, the landmark direction with respect to the aligned space direction of the IMU is obtained. The computer receives this information to update the trajectory parameters of the orbit, and can supply to the navigator - by means of the display and

controls - position, motion, and trajectory information. Again, attitude control is necessary here, mainly to provide suitable conditions for tracking with the scanning telescope.

Rendezvous and Lunar Landing, Figure 26

Figure 26 can be interpreted as representing equipment in the Lunar Excursion Module for rendezvous and lunar landing. The sextant will not exist in the LEM, and the SCT will be a modified version of that in the command module. The radar and optical tracking devices provide the computer, AGC, with landing point or mother craft coordinates relative to the LEM. The IMU input to the computer provides a measurement of velocity. These data are processed to modulate and steer the rocket thrust appropriately.

Section 5. IMU Operation

The primary use of the IMU is in the measurement and control of the specific forces from the rocket thrust or atmospheric drag and lift. Figure 27 is a simplified block diagram showing the control loops used during the thrusting phases of vehicle operation. The spacecraft orientation, position, and motion are a result of the rocket thrust and rocket angles commanded to the engine gimbal servos. The spacecraft autopilot section has rate gyro feedback to the autopilot servo for rate stabilization. Spacecraft orientation and acceleration is measured by the guidance and navigation equipment using the IMU mounted on the navigation base attached to spacecraft structure. Based upon these acceleration or velocity changes measured with the pulsed integrating pendulum accelerometers, PIPAs, the Apollo guidance computer, AGC, generates steering attitude commands as angular rate signals which are integrated and summed with present attitude in the Coupling Display Units, CDUs. The outputs of the CDUs are steering attitude errors which are sent to the spacecraft stabilization and control system for response by the autopilot.

Based upon the measured acceleration history the computer generates an engine cutoff signal when the desired velocity change is achieved.

Before the IMU can be used for such control purposes the stabilized member carrying the accelerometers and stabilizing gyros must first be aligned to a particular inertial orientation relative to the desired trajectory. This introduces a number of different modes of IMU operation. Figure 28 shows a detail photo of the IMU control panel and the CDU panel. The meter provides the astronaut with indication of existing attitude error in three coordinates. He may choose to have the computer and its program operate the various IMU modes or do this himself depending upon which position he sets the transfer switch. If the navigator operates the IMU he uses the six button matrix shown. The first button "zero encode" drives the CDUs to null so that the computer can empty its CDU angle registers and start from zero. This is the first action after applying power to the IMU.

The second button "Coarse Align" sets the IMU gimbal angles to those matching angles set into the CDUs by the computer.

The "Fine Align" button is used in conjunction with star sightings made with the sextant to orient the IMU, via computer gyro torquing, to the angles desired by the computer.

The "Manual CDU" button provides for manual CDU operation with the hand slew switch and vernier thumbwheel on the front of the CDUs in case the computer is failed. The manual align button in this mode drives the IMU to the set CDU angles.

"Attitude Control" is the normal mode for providing steering and attitude errors to the spacecraft. During atmospheric entry, the button "entry" increases the slew capabilities in roll to provide the fast attitude changes about the wind axis to modulate the lift.

The bottom three CDU are associated with corresponding axes of the IMU. The top two CDUs are used with the two degrees of freedom of optics articulation as will be described in a later section.

Figure 29 shows the interconnections among the IMU, CDU, AGC, and spacecraft to accomplish the modes described.

Section 6. Computer Operation

Only general features of the Apollo Guidance Computer (AGC) will be given here since details of the logical organization are covered elsewhere.³ This section will stress more the operations of information transfer with the other spacecraft equipment and the astronauts.

The Apollo computer is a general purpose, versatile, digital computer in the usual understanding of the term, but is very specifically organized for the requirements of Apollo space-flight data handling and computation. Basic word length in the parallel operations is 15 bits with an added bit for parity check with routines for double and triple precision operations as required. Single precision additions have a 20 μ sec instruction time while double precision multiply subroutine is 800 μ sec.

Programs and fixed data are stored in a 12,000 word core rope memory. Variables are stored in a 1000 word coincident current Ferrite matrix erasable memory. Memory capacity can easily be almost doubled by eliminating the feature of the computer providing storage of its own spare replaceable modules within its basic case.

Use of the computer, for the purposes of this report, are best described by the interfaces with other hardware. The following is not a complete listing of these input and output data transfer features but will serve to help understand computer capability.

Discrete inputs are of several kinds. A simple contact closure, for instance, telling the computer that the astronaut has turned on power to the optics subsystem or that the CDUs are operating with the IMU in a particular mode, are simple input bits appearing on separate lines which the computer can examine under program. More imperative data, like the detection of an emergency

failure of the IMU, or the pushing of the computer keyboard buttons by the astronaut, cause interrupts to the existing computer operations so that early action, as required, is accomplished. The computer handles a number of programs at once with instructions being carried out in each in order of programmed priority, with less urgent programs getting their instructions handled after the more urgent are attended.

Discrete outputs are also of several kinds. Computer determination to turn on main rocket engines is signaled by the existence of a train of high frequency pulses on the particular lines to the engine control. The computer can change mode of operation of the various G&N subsystems by closing relays, under permission of the astronaut given either by the operations of the G&N controls or the computer keyboard.

Output variables are governed by the controlled number of pulses - or average pulse rate - sent on appropriate lines. Each of the five CDUs associated with the IMU and optics subsystems can have their shafts controlled by the computer in this fashion. Engine thrust level is similarly controlled when operating with a throttleable engine.

Input variables arrive as a sequence of single pulses representing increments (or decrements) in the variable and go to counters in the computer. Incremental encoders on each CDU shaft provide shaft angle data of this nature. Velocity increments from the Pulsed Integrating Pendulous Accelerometers mounted on the IMU stable member provide the sensed motion input from the IMU as a train of pulses.

Contents of particular registers in the erasable memory are arranged into words with appropriate identifying code for serial delivery to the telemetry system. After completion of the transmission of each word to the ground, a new word is assembled with new data under program or keyboard control.

For ground checkout on the launch pad, the checkout gear can transmit serial words to the computer through the umbilical which are decoded into the same format and treated exactly, by the computer, as are computer keyboard data to be described.

The communication between the computer and the astronaut is accomplished by the computer 21 digit character display and 12 button keyboard control as shown in Figure 30.

The three, two-digit displayed numbers labeled "program", "verb", and "noun" utilized a code which is listed for the astronaut prominently on the front of the G&N/D&C panel (see Figure 36). "Program" refers to the major operation mode of the computer such as "translunar injection", "midcourse navigation", or "entry". The "verb" and "noun" are taken together to give numerous possibilities of meaningful imperative sentences requiring only a limited vocabulary of verbs and nouns. Examples of verbs and nouns are listed below in acceptable pairs:

<u>Verb</u>	<u>Noun</u>
Display Value	Position
Display Uncertainty	Velocity
Compute	Abort Velocity
Read In	Star-Planet Angle
Change Program	Lunar Orbit Insertion

Paired verbs and nouns which are meaningless or not in the computer program repertoire will not be accepted by the computer through the keyboard and the astronaut is so informed by the "illegal order" error light at the top of the panel of Figure 30.

A verb is inserted by the astronaut by first pushing the verb key and then the two digit verb code. The display then lights up with the verb accepted by the computer. Then the noun is pushed in, similarly. If data also must be inserted, this is punched in with the numbers appearing as they are accepted. The computer takes no action on the verb, noun, and data until the astronaut is satisfied with the received sentence and pushes the enter button. If he sees a mistake, he pushes "Clear" and starts over.

When the computer wishes to communicate to the astronaut a request for data or signify an alarm, the verb and noun numbers flash at 1.5 cps until the astronaut takes action.

Detected failures within the computer are displayed on the lights at the top of the panel. If the error reset button does not correct the problem, various levels of diagnostic procedures have been worked out to identify what replaceable module is at fault. This capability for in-flight repair increases mission and safety probabilities by a tremendous factor.

The computer display and control panel of Figure 30 is located at the command module lower equipment bay next to the rest of the G&N equipment. A slightly abridged version operating in parallel with this panel is mounted on the main display area between the center and left astronauts.

Section 7. Optics Operation

The sextant, telescope, and associated support hardware of the optics subsystem are used for a number of measurements:

1. Star - earth landmark midcourse angle measurement
2. Star - moon landmark midcourse angle measurement
3. Star - earth illuminated horizon angle measurement
4. Star - moon illuminated horizon angle measurement
5. Star - earth dark horizon refraction time measurement
6. Star - earth dark horizon attenuation time measurement
7. Star - moon occultation time measurement
8. Earth landmark direction measurement
9. Moon landmark direction measurement
10. Star direction IMU alignment measurement

Only measurements 1, 3, and 8 will be described in this report to show the general methods available in the Apollo optics subsystem configuration.

Figure 31 shows optical schematics of the two instruments shown in more detail back in Figures 12 and 13. The sextant landmark sight line is fixed to the spacecraft along the shaft axis; the sextant star sight line has shaft axis and trunnion axis articulation as does the scanning telescope line of sight. These motions alone are not enough to provide the necessary operations of the instruments. First of all, the limited unobstructed field of view requires at least some spacecraft orientation control just so that the objects can be acquired. The sextant use is more constraining since the landmark line is rigidly fixed to the spacecraft along the shaft axis requiring that the shaft axis be aimed at the landmark, within the field of view, by orientation of the spacecraft.

Figure 32 shows the relationships along spacecraft roll, pitch, and yaw axes, the attitude control jets, and the optics instruments shaft axes. From this figure the motions of images in the optics fields resulting from spacecraft roll, pitch, and yaw motions can be inferred.

Figure 33 shows these motions within the field of unobstructed view of the instruments. Three sets of contour lines show directions of local image motions in a field identified in polar coordinates corresponding to the shaft and trunnion angles. The three sets of contours correspond to roll, pitch, and yaw spacecraft motions.

The sextant landmark-line, along the shaft axis, is in the center of the figure with the 1.8 degree field shown. Spacecraft pitch motion causes images to move vertically (in the normal sense of the observer astronaut) while roll motion causes "across" image motion. Note that yaw could also be used for "across" control but is less satisfactory as far as curvature of local field motion is concerned, and also requires more attitude fuel burning due to the larger yaw axis inertia. Thus the landmark line can be aimed by logical and easily interpreted controlled motions of spacecraft roll and pitch.

The star-line of the sextant is displaced from the landmark-line by the trunnion angle in a direction determined by the shaft angle. Trunnion angles are limited to within 50 degrees or so because of line-of-sight interference with local spacecraft structure. The star image would normally be moved in the 1.8 degree field by controlled motions of the shaft and trunnion. Controlled spacecraft motions, in order to keep the landmark in the field, cause roughly parallel motions of the starline - the variations increasing for the larger trunnion angles. The operation of the sextant, then, during the final measurement is roll and pitch, controlled periodically to keep the landmark in the field, and shaft and trunnion control to achieve the required superposition of the star on the landmark.

The scanning telescope can be made to look along the shaft or to follow the same shaft and trunnion angles as the sextant. With its much

wider field of view - 60 degrees at unity power - it, then, is used as a recognition and acquisition aid for the sextant.

The control of spacecraft orientation and optics articulation is diagrammed in Figure 34. Three control sticks for the navigator's use are shown on the left.

The top stick controls single impulse bursts from the appropriate attitude jets. An up motion of the stick causes one small torque impulse burst from the positive pitch jet causing a positive pitch angular velocity change of 1 minute per second for the light vehicle to something much smaller than this for the fuel- and LEM-heavy configuration. A resulting motion of the landmark in the down direction follows. Letting the stick return to center and pushing up again causes a second downward velocity increment of images in the field of view. Pushing the stick to the left and right cause corresponding increments in "across" velocity of the images by use of small roll impulses. This stick is used for vernier control of spacecraft motion and as a corresponding fine control to hold the landmark in the 1, 8 degree field of view of the sextant.

The bottom stick in Figure 34 is used for coarse control and slew. This stick is normally mounted on one of the couch arm controls but is moved below to the navigation station during navigation operations. A flexible cord from the stick allows use at either station. This stick commands roll, pitch, and yaw spacecraft angular velocity. With this portable hand controller, the navigator will bring the spacecraft to the sighting orientation. After this he will use the single impulse stick to stop residual motion and perform fine control. While under single impulse control the normal spacecraft attitude control system is disabled and only the single impulses may occur in response to navigator commands.

The center stick in Figure 34 is used for control of shaft and trunnion of the optics. A resolver is shown which may be selected to give up-down and left-right control instead of the shaft centered, polar motion resulting from by-passing the resolver. The cosecant attenuation on the shaft drive signal changes the shaft control gain as a function of trunnion angle so that shaft motion gain from the stick in the field of view is independent of the size of the trunnion angle. The stick sends angular velocity command signals to two small CDU velocity servos (physically identical to the IMU CDUs of Section 5), where the corresponding shaft and trunnion commands are integrated and displayed on dials. The commanded angles are here encoded on an incremental encoder for summation in the digital computer. The provision for zeroing of this encoding system is not shown.

The sextant shaft and star-line trunnion follow precisely the commanded angles. Necessary accuracy is obtained on the sextant trunnion transmission by use of a multipole, ultra precise, resolver-transmitter which provides a 64 speed electrical signal while its rotor operates at one speed on the sextant trunnion. The corresponding receiver system in the command servo has a normal precision one-speed resolver-receiver geared to 64 speed and located close in the gear train to

the readout dials and the encoder.

The scanning telescope follows the command shaft angle all the time. The telescope trunnion drive may be set (1) to follow the sextant trunnion command, or (2) may be set to zero to look along the shaft, or (3) may be set to 25 degrees offset. This third position is advantageous for sextant target acquisition, as will be shown.

Figure 35 shows the area of interest of the displays and controls mockup used in operation of the optics. The initial acquisition with spacecraft orientation is done by the navigator with the right hand while he is looking through the scanning telescope. After this his left hand is used with the optics control stick while his right hand can provide, periodically, the necessary small impulses from the impulse control stick. When a satisfactory alignment is controlled with the left hand and observed through the sextant, the right hand is available to punch the "mark" button which causes the computer to record the time and appropriate angles.

Star-Earth Landmark Midcourse Angle Measurement

The general situation for a midcourse navigation sighting is illustrated in Figure 36. This shows the acquisition orientation of the spacecraft with the optics shaft axis and sextant landmark-line pointed to the desired feature on the planet. This operation may be accomplished by the use of the wide field of the telescope with its trunnion set on zero. Just prior to this time, the expected star-landmark angle may be set into the sextant trunnion as shown.

After initial rough orientation of the spacecraft with the telescope trunnion on zero, the 25 degree offset can be set which would cause a view through the telescope, for example, as shown in Figure 37, where the earth is seen from 50,000 miles. The small circle 25 degrees from the center then is along the shaft axis and represents what would be seen in the 1, 8° landmark field in the sextant. The navigator can periodically control small impulses to keep the landmark in the small circle while he slews the shaft to acquire the star. The star should come up on the scale, shown in the reticle, at the expected trunnion angle. The wide field of view provides ample neighboring stars to assure recognition of the navigation star being used. After the shaft is controlled to put the star nearly on the index line, with the trunnion of the sextant preset to the expected value, and with the landmark inside the small circle, the navigator is assured that both the landmark and the proper star images will appear within the superimposed fields through the sextant.

What he sees now, when he changes over to look through the sextant, will be as shown in Figure 38. The landmark, in this case, might be a distinct pointed peninsula on the Isle of Pines off Cuba. With the small impulse control stick he will keep spacecraft motion such that this target drifts slowly across the field. If necessary, near the edge he can reverse its motion to drift back. Meanwhile, with his optics control, he attempts to achieve superposition of the star on this landmark - or, lacking this, to set up so that the two objects are equidistant from any one of the array of parallel "M" (for measurement) lines shown.

The exact control at this point is worth more careful study. The spacecraft, if it has any angular velocity about a random axis, can move the landmark and star in the field in any combination of three modes. (1) It could make the star and landmark move together in the field along M lines; (2) It could make them move together across M lines; or (3) it could make them separate or come together in a direction along M lines. A fourth possibility, having them separate or come together across M lines, cannot happen due to spacecraft angular velocity because of the purposeful feature of this sextant - or any sextant - which prevents an acceptable measurement situation from being affected by rotations of the instrument as a whole. This counter motion across M lines can be controlled only by trunnion angle changes and will change independently only as the direction to the landmark changes with respect to the stars. The landmark angular velocity will be the result of the spacecraft linear velocity component across the line of sight. Values of the order of 1 milliradian per second or less are typical of the midcourse situation.

The most precise operation, then, appears to consist of setting up a situation with the trunnion command held stationary such that the images are coming together as the star-landmark angle is changing. The shaft control alone can be used to keep the two images close together along the M line direction. As the two images pass over each other the navigator pushes the "mark" button which records the existing precision measurement angle and records the time of the event. Experience may show that tracking "on the fly" may be entirely satisfactory, however. Accuracy of 10 arc seconds is typical. This corresponds to almost 5 arc minutes in the 28 power field of view.

Star-Earth Illuminated Horizon Angle Measurement

This measurement utilizes the atmospheric scattering of sunlight phenomena described in Section 2, Figure 6. Because the eye is so poorly adapted to making absolute brightness estimates, an automatic eyepiece is substituted on the sextant for the visual eyepiece. This eyepiece has a rotating wedge star tracker which sends tracking error signals to the optics CDU drives positioning the articulating line of sight of the sextant to the chosen star. The landmark-line is pointed by spacecraft attitude control commands toward the horizon. The intensity of the horizon is sensed in the automatic eyepiece. The specific controls for this mode of operation are shown on Figure 35 labeled NVE for non-visual eyepiece. The intensity level is preset according to the sun aspect angle. The NVE level meter indicates unity when the detector sends a "mark" to the computer. Special procedure is necessary to assure that the horizon is directly below the star being tracked.

Earth Landmark Direction Measurement

The navigation situation for orbital operations is illustrated in Figure 39. The technique is equally applicable in lunar or earth satellite orbit. The spacecraft orientation is shown with the roll axis forward and horizontal. Other orientations are possible but this attitude has what is judged to be the best features.

The landmark is chosen to be reasonably close to the orbit ground track so that it will pass close to underneath the craft. The target is tracked with the scanning telescope to achieve a measurement.

The view in the telescope during this orbital navigation is shown in Figure 40. Acquisition consists of first picking up the target as it comes into view from the horizon by gross roll motion and forward trunnion setting. A period for recognition and acquisition of about 30 seconds or so is expected. Finally the trunnion shaft is used to track along its path by controlling the image to stay at the center of the reticule. During acceptable tracking, the navigator pushes the "mark" button which records time, the telescope trunnion angle, and IMU gimbal angles. The IMU was previously aligned, of course, with star sightings. The computer uses these data to improve its orbital navigation knowledge.

Section 8. Astronaut Operations

In the previous sections a number of operations associated with G&N hardware were described in which the astronaut was involved and had direct control and choice. This section will complete the description of design features concerning the operation by the navigator.

Information on standard and emergency procedures, diagnosis and repair, star charts, earth and lunar maps, etc. are displayed on the map and data viewer, Figure 41. This projection system takes film cartridges and displays data with high resolution on a 42 square inch screen. It is estimated that five of these cartridges would be carried on a lunar landing mission. This would correspond to about 9000 frames with high information density. Each cartridge can be removed and inserted with any frame in projection position. Motor slew of the film drive is provided.

To the right of the viewer in Figure 41 are condition lights informing the navigator of detected subsystem errors. Error detection at critical points throughout the equipment monitor error signals which are combined by logical "or" into groups of master error detection signals: "IMU fail", "accelerometer fail", etc. The ones which would sense emergency conditions are sent as discrete bits to the computer which, at astronaut option, can be instructed to take the appropriate emergency action. In any event, the computer displays the condition on the subject lights (and a corresponding set at the main panel). If the computer is not operating, the top light in the series "error detect" will be lit if any error is detected anywhere by the error monitors. The multitude of monitor points which make up the failure detections can be sampled individually by the spacecraft in-flight test system in order to localize the failure. Repair consists of replacing the failed module with a spare. A minimum of spares can back the many modules due to the purposeful design constraint of minimizing the number of different modules.

If failure occurs, each of the major subsystems can be individually turned off. The design is such that the remaining operating equipment can be usefully utilized in back-up modes of

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operation. The spacecraft stabilization and control system can be used by the crew utilizing ground track information via voice radio to provide backup for complete G&N failure. The chances of these failures is small due to the extensive reliability provisions now being used for qualification of manned and unmanned space flight hardware.

It is this ability for making in-flight repairs and operating in alternate and backup modes by which the astronauts enhance the operations of the mission.

Other capabilities of man not easily instrumented are utilized in Apollo. Specifically, the remarkable ability to recognize star and landmark patterns from charts and maps is a unique asset possessed by the astronauts. Another is man's judgement in determining proper operation of his equipment and optimum course of action.

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In summary, we have described a flexible system for manned operations. Almost every function can be accomplished automatically to relieve strain and tedium on the navigator, but he is given information in displays and command in controls to take over usefully at his discretion to enhance the probabilities of mission success and crew safety. We see a balance between complex and high speed measurement and data processing of the automatic equipment operating with the wonderfully adaptable sensors and judgement of man in a difficult task: the guidance and navigation for a moon trip.

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2. W. E. Toth, "Visual Observation of Landmarks," MIT Instrumentation Laboratory Report E-1067, (Monthly Technical Progress Report Project Apollo Guidance and Navigation Program, Period August 11, 1961 to September 13, 1961).
3. A. L. Hopkins, Jr., R. L. Alonso, and H. Blair-Smith, "Logical Description for the Apollo Guidance Computer," MIT Instrumentation Laboratory Report R-393, March 1963.

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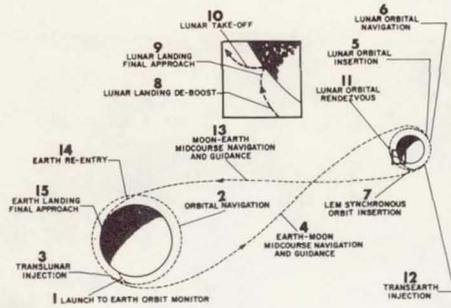


Fig. 1 Mission phase summary

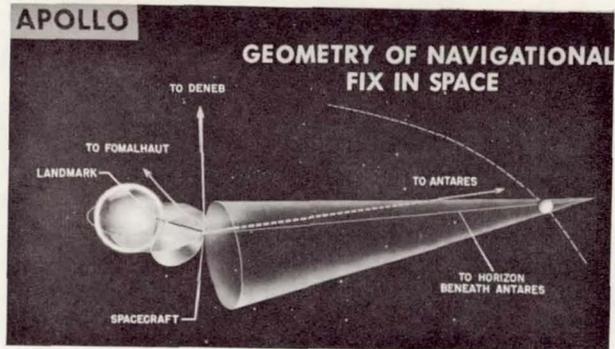


Fig. 4 Geometry of navigational fix in space

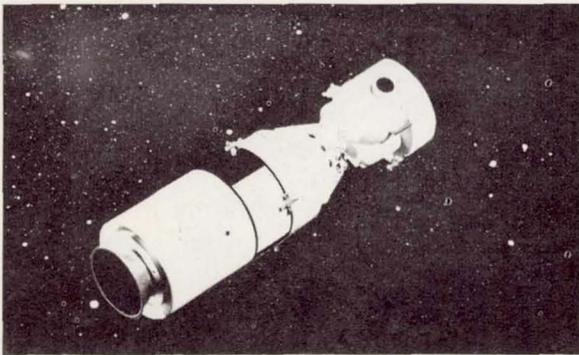


Fig. 2 CM, SM and LEM with translunar configuration

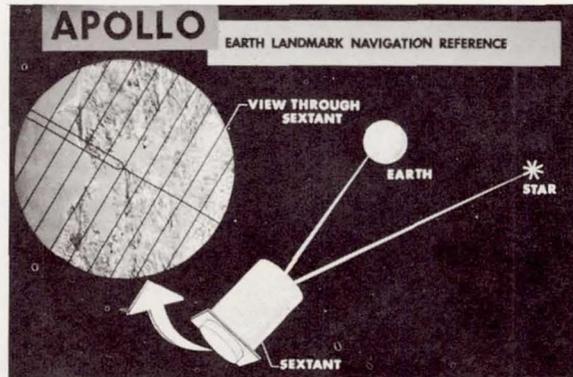


Fig. 5 Earth landmark navigation reference

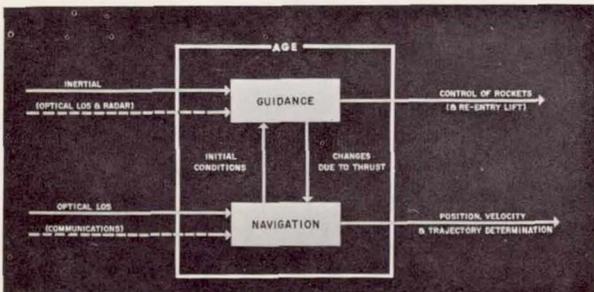


Fig. 3 G and N diagram

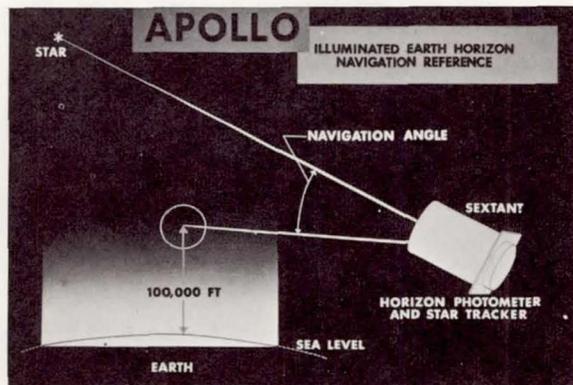


Fig. 6 Illuminated earth horizon navigation reference

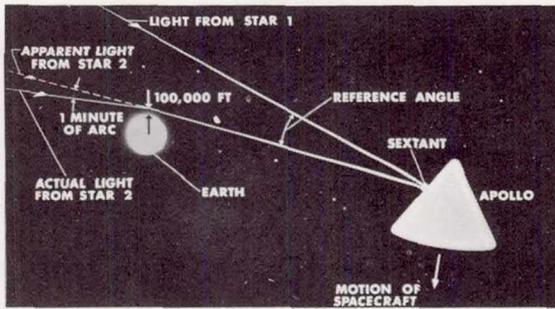


Fig. 7 Star refraction--earth horizon navigation reference

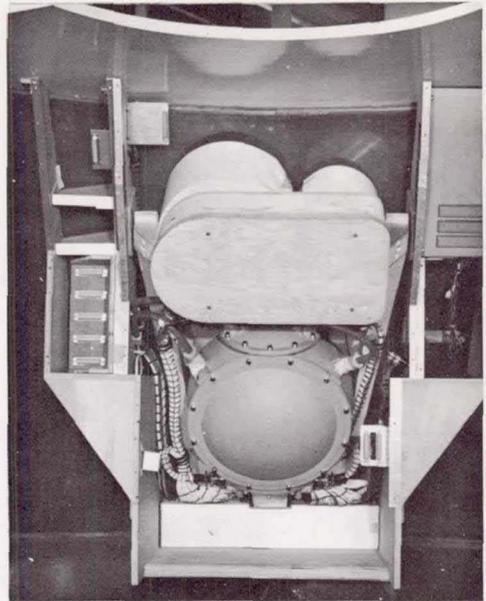


Fig. 10 Installation mockup

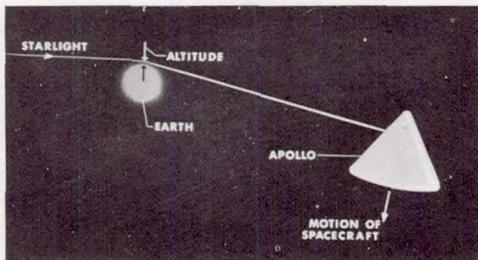


Fig. 8 Star attenuation--earth horizon navigation reference

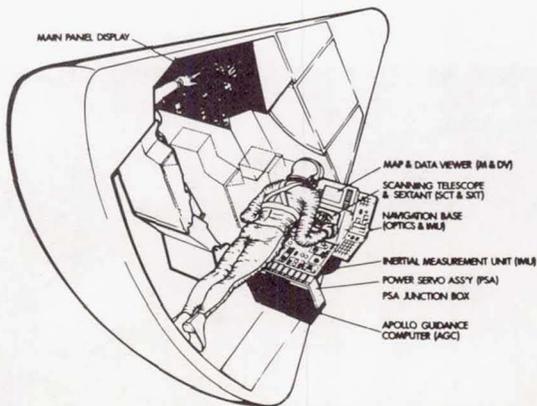


Fig. 9 Command module cutaway

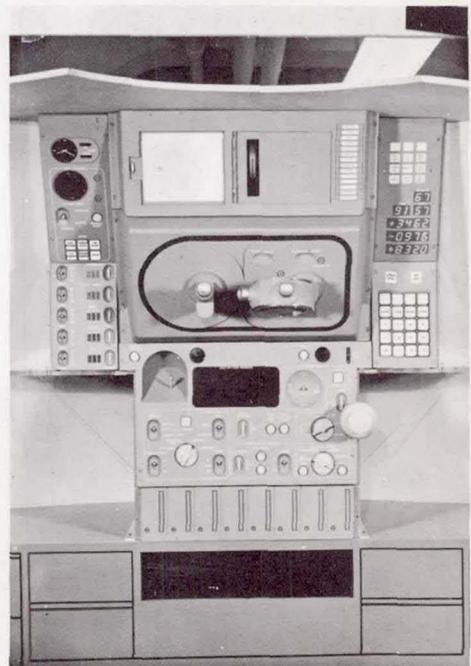
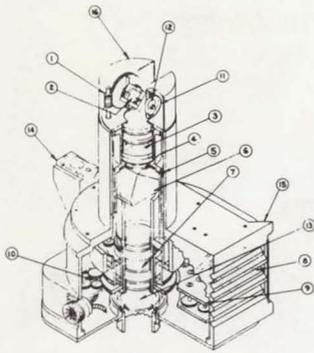


Fig. 11 Display and control mockup



- 1- TRUNNION DRIVE WORMSHAFT
- 2- MIRROR MOUNT ASSEMBLY
- 3- OBJECTIVE LENS ASSEMBLY
- 4- RETICLE
- 5- RETICLE ILLUMINATION
- 6- PRISM PRISM
- 7- RELAY LENS ASSEMBLY
- 8- HEAT EXCHANGER
- 9- SHAFT DRIVE GEAR BOX
- 10- TRUNNION DRIVE GEAR BOX
- 11- SPRING AND CAM FOLLOWER ASSEMBLY (ANTI-BACKLASH)
- 12- CAM
- 13- WINDOW
- 14- MOUNTING PAD
- 15- OPTICAL BASE
- 16- COVER

Fig. 12 Scanning telescope cutaway

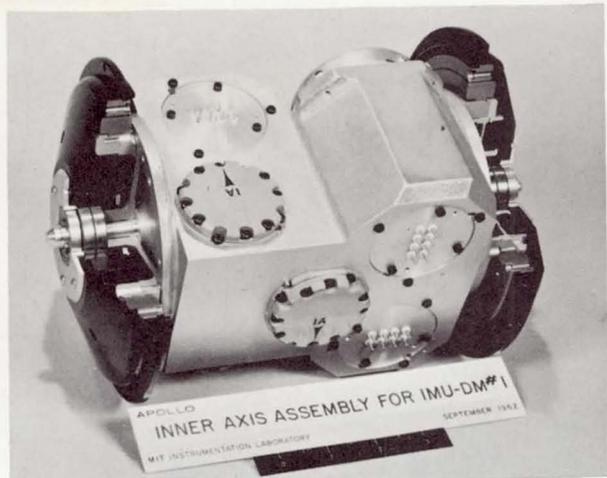
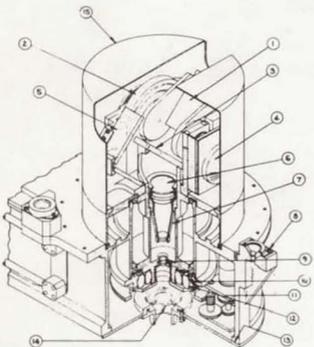


Fig. 15 Inner axis assembly for IMU-DM #1



- DESCRIPTION
- 1- ROTATING MIRROR
 - 2- TRUNNION AXIS RESOLVER
 - 3- BEAM SPLITTER
 - 4- TRUNNION DRIVE GEAR BOX
 - 5- FIXED MIRRORS
 - 6- OBJECTIVE LENSES
 - 7- INTERMEDIATE LENSES
 - 8- MOUNTING PAD
 - 9- RETICLE
 - 10- RETICLE LAMP
 - 11- SHAFT DRIVE GEAR BOX
 - 12- SHAFT AXIS RESOLVER
 - 13- PANEL
 - 14- EVERBEE SOCKET & SEAL
 - 15- COVER

Fig. 13 Sextant cutaway



Fig. 16 Outer axis assembly for IMU-DM #1

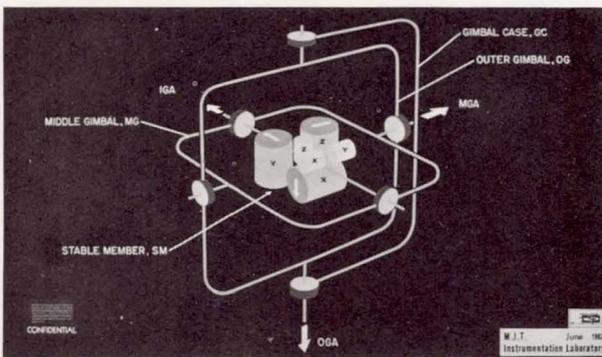


Fig. 14 IMU schematic



Fig. 17 IMU-DM for Apollo G and N equipment

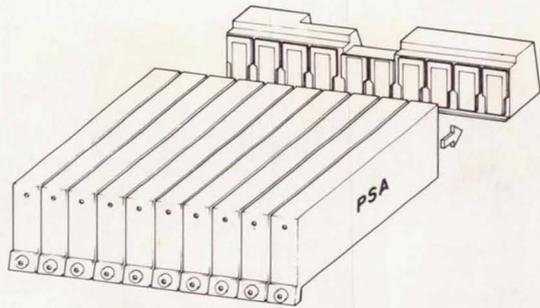


Fig. 18 Sketch of PSA

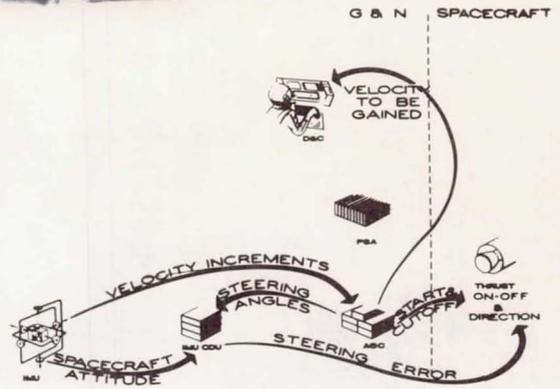


Fig. 21 Guidance and thrust control

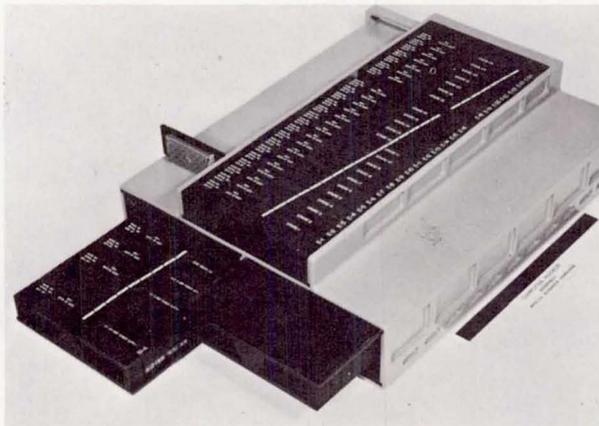


Fig. 19 Computer mockup

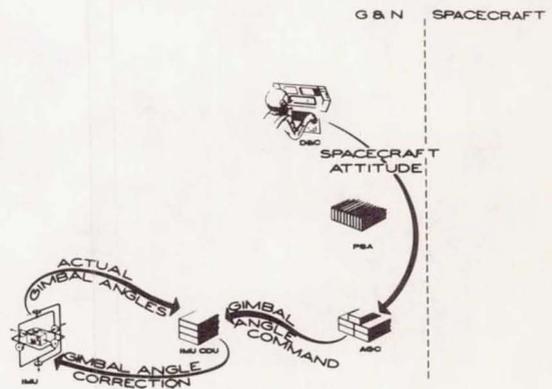


Fig. 22 IMU coarse alignment

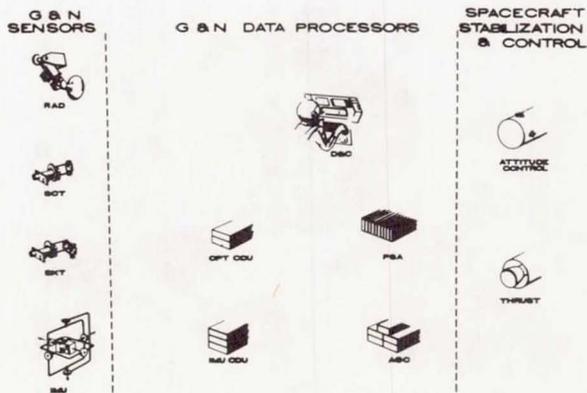


Fig. 20 Major subsystems

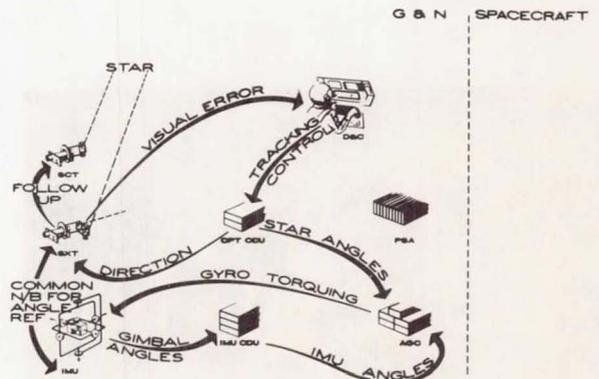


Fig. 23 IMU fine alignment

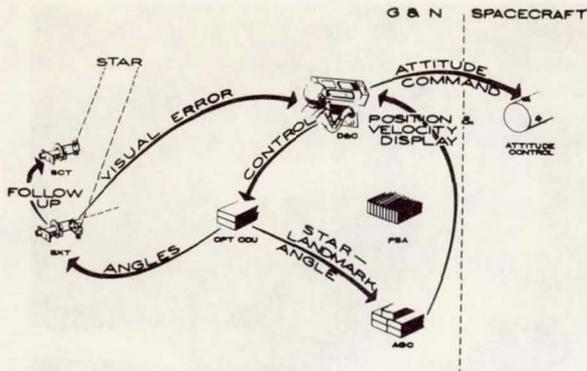


Fig. 24 Midcourse navigation

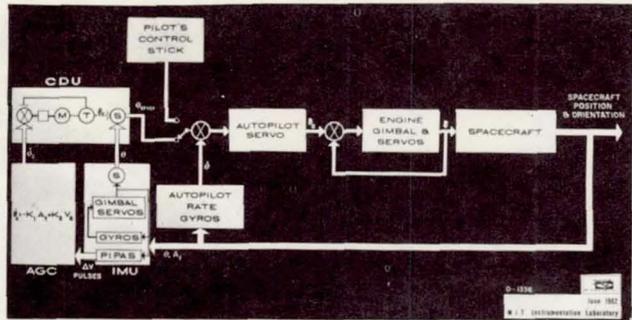


Fig. 27 Guidance and steering control

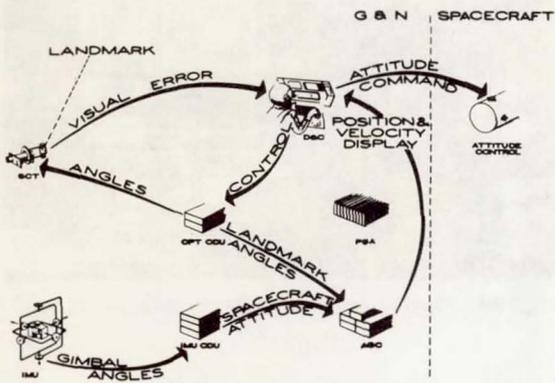


Fig. 25 Low orbit navigation

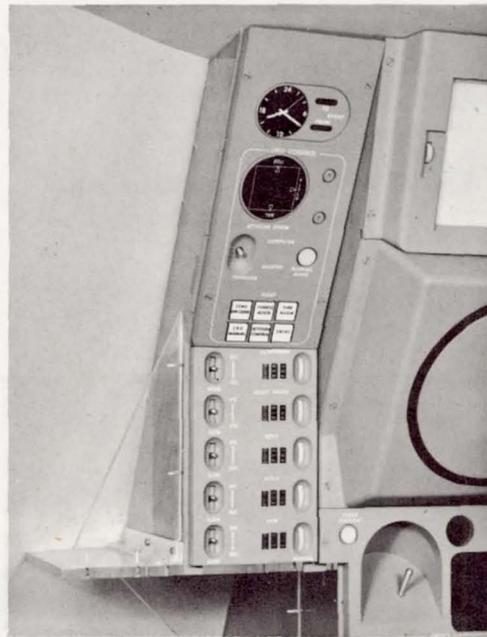


Fig. 28 IMU control panel and CDU panel

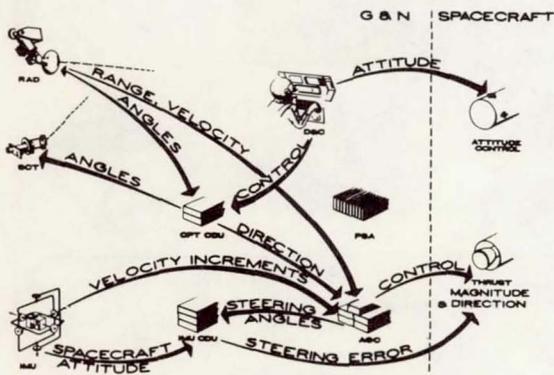


Fig. 26 Lunar landing and rendezvous

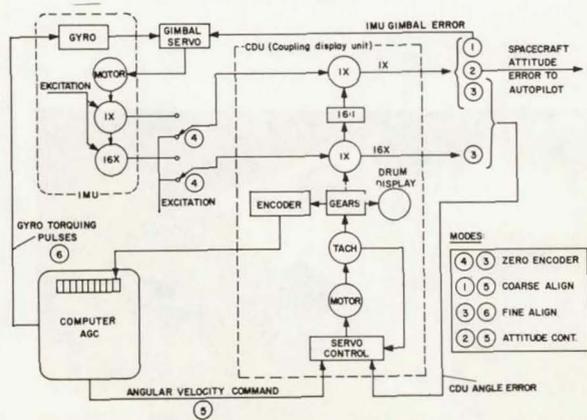


Fig. 29 Single axis schematic--IMU, CDU and AGC operation

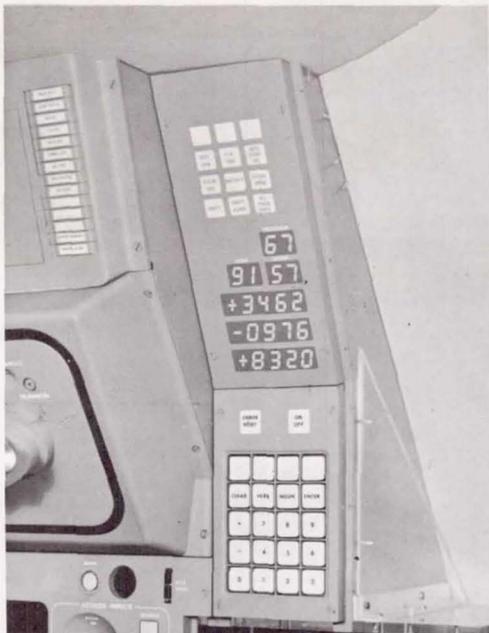


Fig. 30 Computer display and control panel

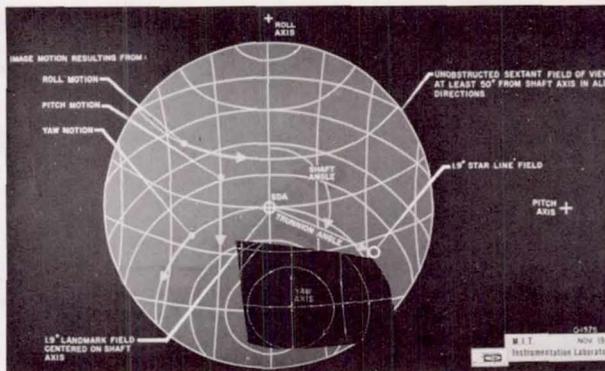


Fig. 33 Sextant field notions

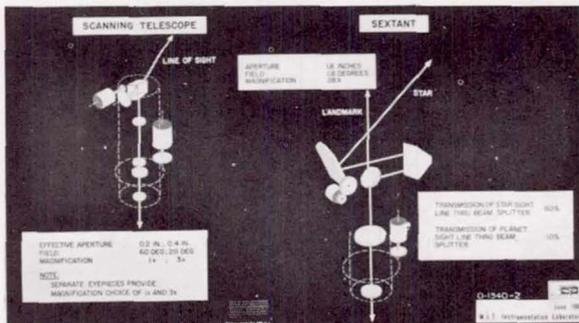


Fig. 31 Optical schematics

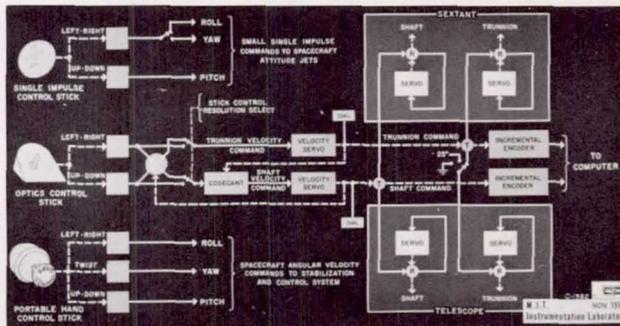


Fig. 34 Optics control instrumentation block diagram

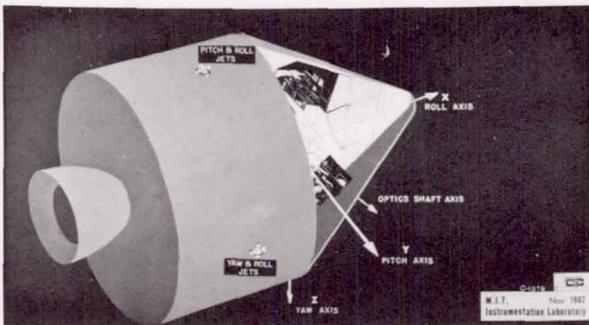


Fig. 32 Spacecraft and optics axes

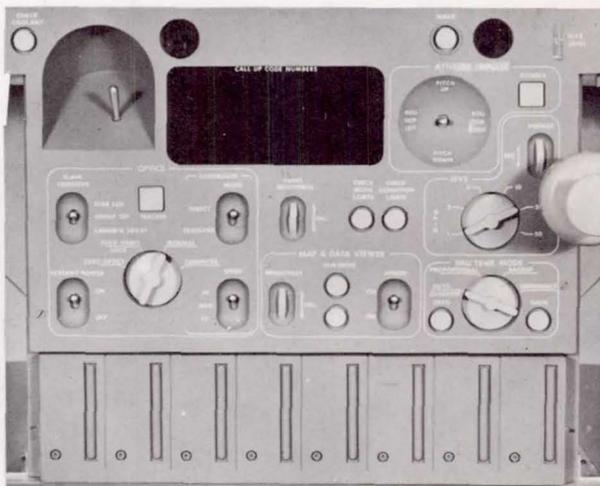


Fig. 35 Optics control panel

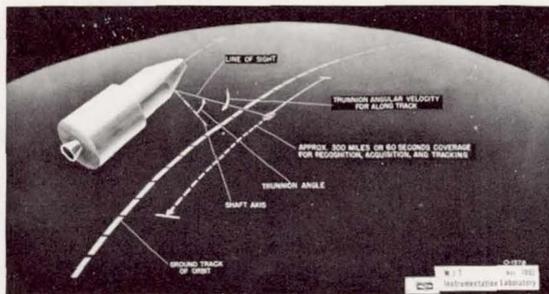
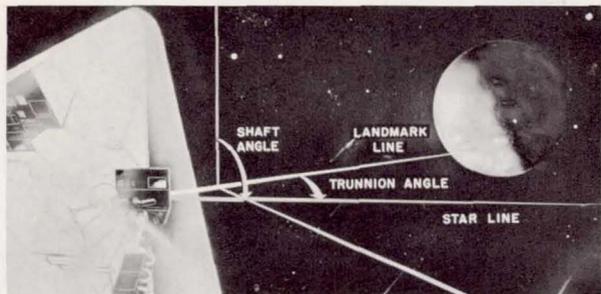


Fig 36 Spacecraft orientation midcourse navigation sighting

Fig. 39 Spacecraft orientation--orbital navigation sighting

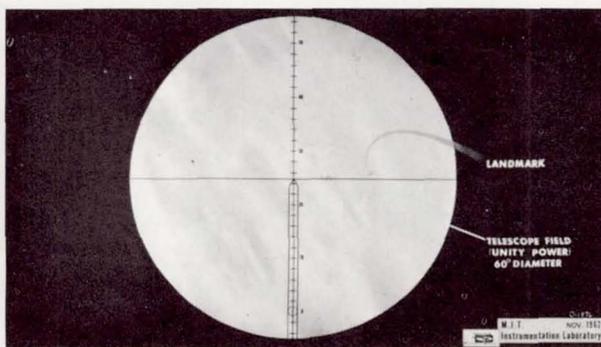
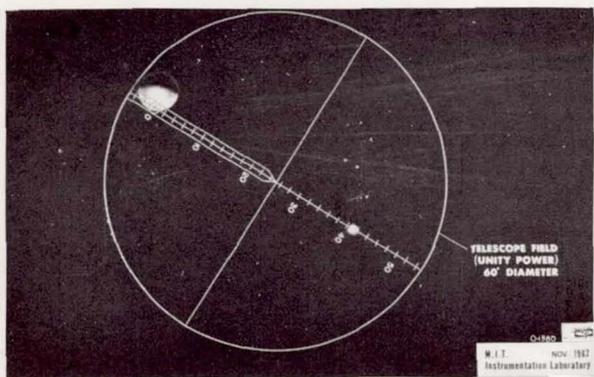


Fig. 37 Telescope view--midcourse navigation

Fig. 40 Telescope view--orbital navigation

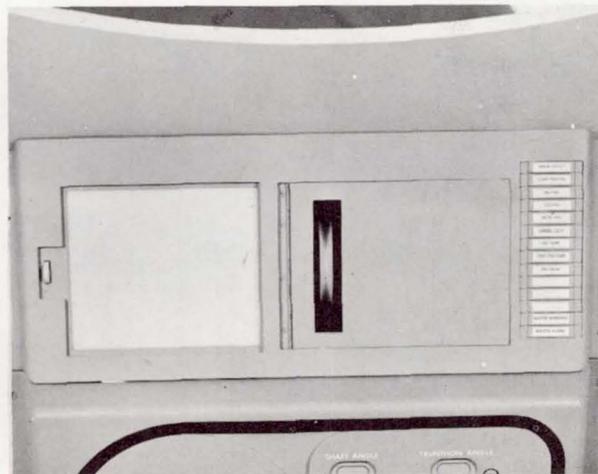
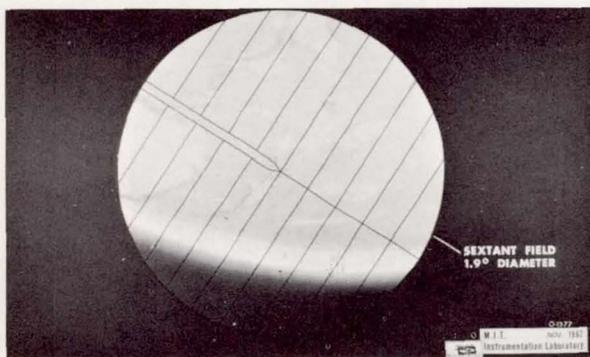


Fig. 38 Sextant view--midcourse navigation

Fig. 41 Map and data viewer mockup

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FACTORS AFFECTING THE DESIGN OF FLIGHT STABILIZATION AND CONTROL SYSTEMS FOR MANNED SPACECRAFT

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Summary

Six factors which have important influence on the design of flight stabilization and control systems for manned spacecraft are discussed. The factors considered are: type of vehicle, size of crew, time of mission, weight of vehicle, purpose of mission, and equipment thermal control concept. Following the discussion of the general influence of each factor, descriptions of flight stabilization and control systems for the current manned space programs are presented and some important effects of the various factors are noted. Block diagrams of the several systems and significant photographs of flight control hardware are presented.

Introduction

In common with all other complex devices, flight stabilization and control systems for manned spacecraft are the result of a myriad of compromises, each of which can be traced to some recognizable factor or design requirement. All these factors are probably not recurrent and thus will differ for each system considered. Therefore this paper will not attempt to consider all the factors which may affect a flight stabilization and control system design; rather, a set of six factors has been selected on the basis that each of them is of some importance in all instances, and further that these six factors will largely determine the functional and hardware design concepts.

This paper is divided into two major sections. The first section discusses the six selected factors and presents generalized examples of their separate influences; the second section contains a description of each U.S. manned spacecraft flight stabilization and control system and points out features in their design which are attributable to these six factors. (Hereafter, "flight stabilization and control system" is frequently abbreviated to "control system.")

Discussion of Influential Factors

Type of Vehicle

One of the most basic factors affecting control system design is the type of vehicle to be controlled. Manned spacecraft can be classified according to the type of flight regime, that is, suborbital, orbital, or superorbital. (See Figure 1.) However, study of the correlation of control requirements with these three regimes indicates that very little correlation exists. For example, a vehicle of the X-20 (Dyna-Soar) type will have much the same control requirements regardless of whether it is launched into a suborbital or

superorbital flight path. Also, a Mercury capsule can re-enter from a superorbital path with the same control system that was used in the first U.S. manned suborbital flight. Conversely, however, there is a marked difference between the control requirements for the X-20 and Mercury regardless of the flight path specified.

On this basis, then, the vehicle exterior geometry (and to some extent structural characteristics) will influence control system functional design for both free space, exit, and re-entry mission phases. The vehicle geometrical configuration and center of gravity location will determine whether the flight within a sensible atmosphere will be ballistic or aerodynamic and whether the vehicle will be statically or dynamically stable. The structural characteristics will of course determine whether there is a problem of structural frequencies coupling with the control system.

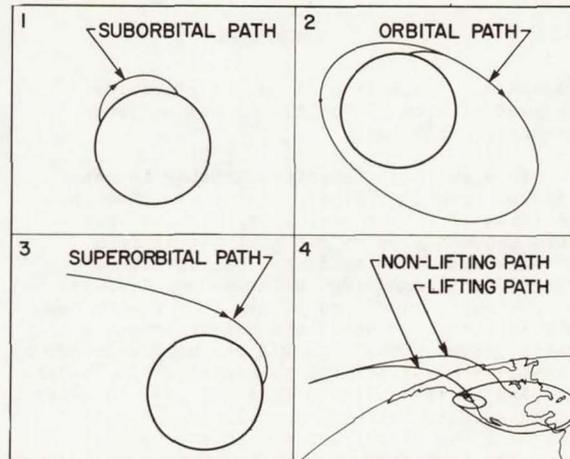


Figure 1. Types of Orbital and Re-Entry Vehicle Paths

In general it can be said that the problem of re-entry control increases in complexity as the L/D (lift-to-drag) ratio is raised from 0 to 2 or 3 and as the configuration changes from a blunt body of revolution to a winged, airplane-like shape. Several items contribute to this increase in complexity; for example, a nonlifting body does not necessarily need roll attitude control, but roll attitude must be controlled or modulated in a lifting body in order that the impact or landing area can be even approximately predicted. In like manner, pitch and yaw attitude control requirements are much less stringent on the nonlifting body

because of the lack of changes in transverse forces with angle of attack.

Figure 2 presents an example of the manner in which the geometric configuration affects the vehicle stability characteristics. Typical static stability plots are given for a low L/D (< 0.5) blunt body and for a high L/D ($2 < L/D < 3$) re-entry configuration. It is readily apparent that the blunt body has much less variation in static stability over the Mach number range and thus will require a less sophisticated control system; in fact it is probable that a satisfactory re-entry could be accomplished with a rate damping system alone and that a safe re-entry could be made in an emergency without even the damper.

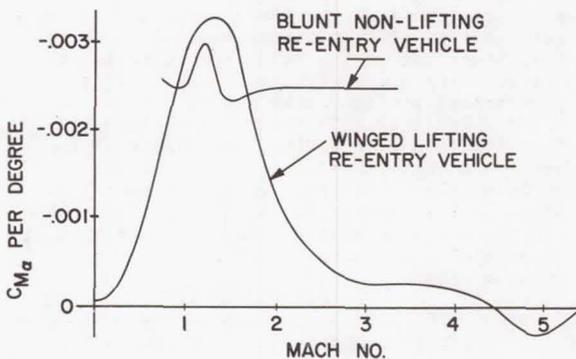


Figure 2. Comparison of Static Stability Characteristics of Nonlifting and Lifting Re-Entry Vehicles

When over-all vehicle stability is considered from the pilot's viewpoint, that is, in terms of flying qualities, the need for more augmentation on the high L/D vehicle becomes even more evident. One version of longitudinal handling qualities requirements is presented in Figure 3. In this figure the shaded area represents the characteristics which unaugmented, high L/D re-entry vehicles exhibit for various flight conditions. It can be seen that there is a definite need to alter both the frequency and damping in order to move all the flight conditions represented into the desirable area. (See reference 1.)

In addition, the basic fact that the high L/D vehicle generates an increasing amount of lift (until $L = W$) during re-entry means, as mentioned above, that the magnitude and orientation of the lift vector must be closely controlled. This in turn requires that the pilot or the control system must hold roll and angle of attack (or perhaps pitch attitude) within close tolerances in order to follow a given flight path and prevent the onset of dangerous aerodynamic forces or heating.

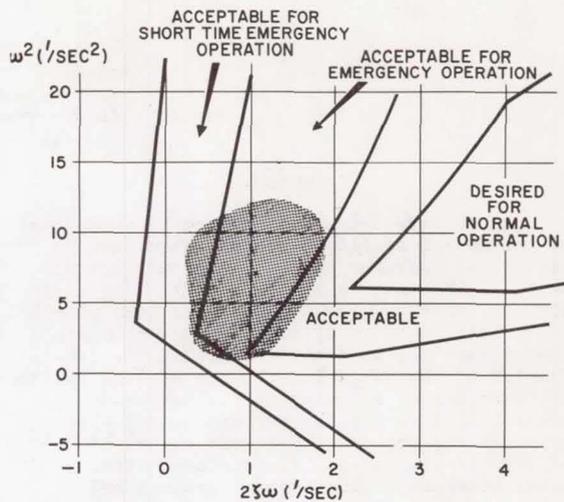


Figure 3. Handling Qualities in Pitch (ω = frequency of motion, ζ = damping factor)

These considerations lead to the following conclusions:

1. A nonlifting or low L/D vehicle will usually require only simple fixed-gain damping and low-precision attitude control. This control can and usually must be supplied by on-off reaction jets which allow the use of simple driving electronics.

2. A high L/D re-entry vehicle must have variable-gain damping and precise three-axis attitude control. Control is usually obtained by means of proportionally actuated aerodynamic surfaces. The control and actuation requirements generally call for the use of complex and precise electronics. The vehicle may be uncontrollable without automatic control so that great emphasis must be placed on high reliability. Such reliability will generally require parallel active redundancy (as indicated below under Time of Mission), which will further increase the electronic complexity.

Size of Crew

The effects of crew size on control system design can be illustrated by the summarized results of a human factors study of a planetary exploration vehicle based on the bus and lander concept. The study is based on the methods outlined in references 2 and 3.

The curves shown in Figure 4 represent the various crew requirements assuming different levels of system automaticity, for a planetary orbit phase of a planetary landing mission. The number of active crew members is plotted against the time from planetary orbit injection.

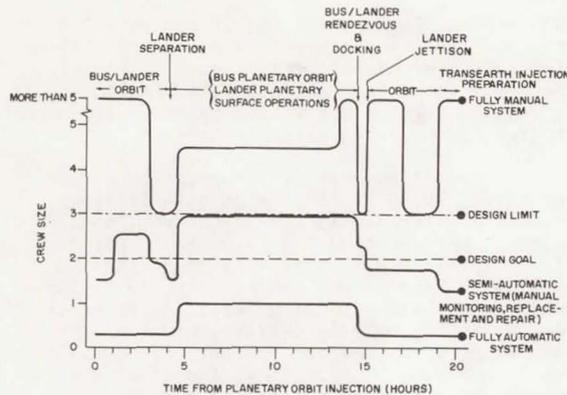


Figure 4. Effect of Crew Size on Automation Requirements

The "three-man" level is shown as the vehicle design limit. This three-man crew limit assures active participation of the crew at all levels of system performance: decision making, dynamic control, monitoring, checkout, replacement, and repair.

The design goal line at the "two-man" level represents the crew requirement for a semi-automatic system in which failures do not occur. This reflects a system philosophy of active crew participation at such a level that the equivalent of one operator as a "human spare" is available to achieve the necessary total system reliability.

The remaining curves are based on the crew tasks that are anticipated for the planetary orbit phase:

1. The commander of the vehicle is primarily concerned with command decisions, orienting and stabilizing the vehicle, stabilization and control system checkout, communications, equilibrium and dynamics monitoring, and planetary surface operations.
2. The navigator is occupied with subsystem alignments and gathering data for navigational position and orientation when he is part of the crew of three. He is also occupied with orbital correction, system monitoring, and communication when he is alone during orbit.
3. The systems engineer will be responsible for subsystem monitoring, trouble-shooting, and maintenance tasks.

The execution of all these tasks has been plotted against time in the upper curve of Figure 4 to indicate the number of crew members needed to carry out the work in the case of a hypothetical fully manual system. The requirement of a crew in excess of five men is evident during four periods of the orbit. This occurs because the execution of complete manual checkout procedures of all subsystems is very time consuming, and therefore many men are required to complete these tasks within the

allotted time. Other tasks, such as star sighting, position, and position error calculations, would also be time prohibitive without the benefit of a high-speed digital computer.

At the other extreme, the fully automatic system with a crew requirement of one man is plotted in the lower curve of Figure 4. This curve represents a hypothetical system with automatic monitoring and control so that the single operator is more of a passenger than a participator in system functions. His indicated partial activity at either end of the plot represents near-body observations, communication with earth, and a low level of system monitoring activity. The operator's full activity in the central portion of the plot represents his scientific and exploratory activities on the planetary surface.

The middle curve of Figure 4 represents the crew requirement for a system which is believed to represent a practical compromise. This realizable concept does not have the drawbacks of the excessive number of crew members of the fully manual system, nor is it as technically prohibitive as the fully automatic system. Rather, it is structured to utilize the intelligence and unique adaptability of the crew members working integrally with the advanced automatic subsystems which are designed to complement the crew's possible contributions and thus maximize mission success probability. This semi-automatic system plot is a composite of the proportion of each crew member's total capability which is required for the particular tasks assigned to him during this mission phase. This plot includes manual control of the orbiting bus and the lander as well as monitoring, trouble-shooting, and subsystem maintenance.

During a portion of the planetary orbit as sole occupant of the complex bus, the navigator will play a triple role by spending his waking time in continuous monitoring and maintenance of his system, supervising vehicle control, and solving his customary navigation problems. Meanwhile, the descent, planetary operations, ascent, rendezvous, and docking of the lander fully occupy the abilities of the pilot and systems engineer.

One conclusion that can be drawn from such studies is readily apparent in a gross sense, namely, that crew size can be decreased as automaticity is increased and crew work load is consequently decreased. This factor, however, is interdependent with others. For example, the cost and development time for a fully automatic control system might dictate the semi-automatic approach even though the required reliability could be attained in the automatic system.

Time of Mission

The design mission duration becomes an important factor in the design of flight control systems because of the interrelation of mission duration with the probability of successful operation of any of the various vehicle subsystems. Figure 5 presents four

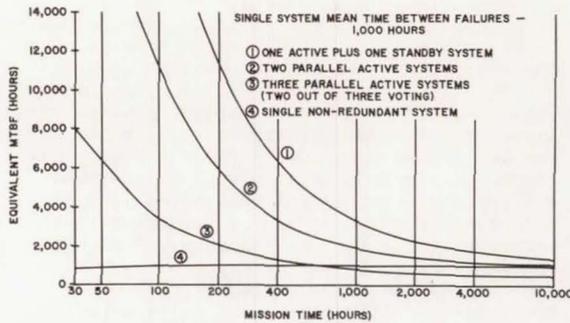


Figure 5. Effect of Redundancy on Equivalent MTBF

curves for various control system configurations ("configuration" here meaning the type and extent of redundancy employed). The curves are drawn with an ordinate of equivalent mean-time-between-failure (MTBF) and an abscissa on a log scale of total mission time. "Equivalent MTBF" as used here for redundant systems is that MTBF which would be needed in a non-redundant system to achieve the same reliability for a given mission time. The four curves represent:

1. A redundant system having one active channel with another identical active channel being maintained in standby condition (curve 1). In considering this system it is presumed that the pilot will be able to detect a failure of the active system and manually switch to the standby system.
2. A redundant system having two parallel active channels each equipped with independent monitors that can determine and switch out a malfunctioning channel (curve 2).
3. A redundant system having three parallel active channels equipped with comparators which conduct a continual two-out-of-three vote and switch out any disagreeing channel (curve 3).
4. A single channel non-redundant system having a mean-time-between-failure as determined by piece-part failure rate of 1,000 hours (curve 4).

Some interesting general conclusions can be drawn from an examination of these curves. First, it becomes evident that for long mission times, particularly above 1,000 hours, the efficacy of redundancy in increasing the equivalent MTBF is sharply reduced. In fact, configuration 3, the two-out-of-three voting system, actually exhibits a lower equivalent MTBF than the single non-redundant system for all mission times above 693 hours. Secondly, the greater effectiveness of the active-standby arrangement of configuration 4 indicates that it is by far the most effective approach whenever this arrangement is feasible from a safety standpoint (that is, where the pilot will have time to detect and switch out the malfunctioning channel).

Looking now at the low end of the abscissa scale, it can be seen that any of the three types of redundancy shown

contributes large increases in equivalent MTBF for short mission times. In fact, the numbers indicated for mission times below 50 hours become quite large and in effect almost eliminate a redundant flight control system as a probable cause of mission abortion.

By recalling some of the characteristics mentioned above in connection with lifting re-entry vehicles, it can be inferred that either configuration 2 or 3 would be particularly applicable to this type of vehicle because of the severe controllability problems which might occur while a pilot was detecting and switching out the failed control system channel. This need for instant switch-over would probably be a critical factor in the choice of a control system for a lifting re-entry vehicle even though the mission length might be sufficient to severely limit the equivalent MTBF obtainable. One solution to this problem would be to combine the active redundant and the standby arrangements in such a way that during extended orbital or deep-space flight the system would function as an active-standby system, but during re-entry it could be converted to an active two or three channel system.

Returning now to the high end of the mission time scale, it is evident that as the mission time becomes appreciably greater than the single-channel MTBF, all forms of redundancy lose effectiveness. It thus appears that missions with lengths measured in months and years rather than hours and days will require onboard repair or perhaps a much more conservative approach to the design of both moving-part mechanisms and active electronics in order that the MTBF values may approach the numbers associated with current telephone or utility equipment.

In any event it can be seen that the mission duration and the feasibility of in-flight component replacement combine to almost dictate the type of redundancy approach to be used. The only prospect of altering this situation will be through the use of flight-worthy components which have reliability increased by one or two orders of magnitude.

Purpose of Mission

The mission purpose of a manned space vehicle will influence chiefly the functional design aspects of the control system. For instance, consider Mercury and Gemini. Project Mercury provided an orbital vehicle which could carry a man for a limited number of orbits. Gemini has a broader mission purpose. In addition to the orbit phase, which is considerably longer than that for Mercury, Gemini is also required to accomplish orbital rendezvous. It is the addition of the different purpose, namely rendezvous, that causes the functional design of the Gemini control system to differ appreciably from that for Mercury. This is not to say that all internal functions of the control system are handled in a similar manner in the two systems and that the only differences are due to the rendezvous requirement. This is not the case. Gemini employs all solid-state signal switching,

does not use sector switches on sensors, and in general uses more advanced mechanization techniques. These differences, however, are not due to the functional requirements as created by the mission purposes, but rather to the advance in the state of the control art from the time the Mercury program started until the time the Gemini program started.

Extending the comparison further we can look at Apollo and Mercury. Apollo does have orbital flight as part of its mission purpose. However, orbital flight for Apollo represents only a small portion of the many flight conditions that must be encountered, and thus the portion of the Apollo control system that is necessary for orbital flight comprises only a small portion of the entire Apollo stabilization and control system. The larger portion is concerned with coasting attitude hold, velocity corrections, and rendezvous maneuvers. Each of these mission requirements creates the need for some additional hardware to fulfill the function and thus the complexity of the mission has a rather direct effect on the complexity of the control system.

If we look now at a vehicle of a basically different type, such as the X-20, we notice even more marked differences. Superficially it may seem that the mission purpose of the X-20 is quite similar to that of Mercury in that both are intended to go into orbit for a short period of time and then accomplish a safe re-entry. Both are intended to be controllable by the human pilot but both are also designed to accomplish a completely automatic re-entry. Here the similarity stops. Mercury accomplishes its re-entry along a ballistic and almost uncontrolled path utilizing a blunt body and heat shield to survive the aerodynamic heat encountered. The X-20, on the other hand, is to accomplish its re-entry by gliding into the atmosphere as a winged vehicle and thus it is subject, as described above, to all of the stabilization and control problems common to low aspect ratio high-speed aircraft. In addition it must follow a fairly narrow descent corridor in order to avoid intolerable aerodynamic heating. Thus it is in the differences of the mission purpose in regard to re-entry that Mercury and the X-20 differ; insofar as orbit phase is considered, the control systems for each are functionally somewhat similar.

As a final example, let us consider the problem of a manned orbiting space station. Here the purpose of the mission is not merely to accomplish manned orbital flight and re-enter safely but to provide an orbital laboratory in which men may work productively for weeks or months at a time. This change in purpose--from a short duration mission with a pilot aboard to control the vehicle to a long duration mission in which the vehicle is largely expected to control itself and thus allow the crew to conduct experiments--calls for a completely different functional design of the vehicle stabilization and control system. As mentioned below in the section on manned space stations, there are three or perhaps four completely different control functions required for an orbiting

laboratory as compared with a Mercury type vehicle.

Thus far some illustrations have been given of how the control system functional design must incorporate all the features necessary to allow the vehicle to fulfill its mission. In the reverse sense it is equally important that the control system, and for that matter all other systems, be designed to accomplish the mission purpose and nothing more. The reason for this is fairly obvious. Every pound put into orbit or accelerated to escape velocity costs hundreds of thousands of dollars, and to needlessly add a few pounds of weight to a space station control system in order to make it hold attitude closer than required is to needlessly spend several hundred thousand dollars for each vehicle placed into orbit.

Weight of Vehicle

The effect of vehicle weight on the flight control system design is perhaps an inverse type of factor. That is, a heavier vehicle does not necessarily require a heavier or more complex control system, but rather the heavier vehicle may permit the use of a heavier control system. A comparison of the ratio of vehicle weight to control system weight for the current manned space vehicles shows for Mercury - 80:1, for Gemini - 200:1, for Apollo - 128:1, and for the X-20 (Dyna-Soar) - 112:1. A consideration of the reasons contributing to the differences in this ratio brings out the following items:

1. The two vehicles with the most similar missions are Mercury and Gemini. Here the decrease in relative control system weight can be attributed almost completely to the use of more advanced sensors and electronic components. In the case of Mercury, as is described below, it was necessary to use existing state-of-the-art components in order to meet the time and reliability requirements of the program. Gemini came almost three years later and, while it too is a program not allowing extensive new component development, the advance in the state-of-the-art since the beginning of the Mercury program allowed the Gemini control system to weigh only slightly more than one-half the Mercury control system. The increase in Gemini system complexity caused by the added rendezvous mission requirement was probably largely offset by the reduction in automaticity compared to Mercury.

2. Looking now at the ratios for Apollo and the X-20, it can be seen that they are reasonably close together. The proportion of control weight to vehicle weight is about two-thirds of that indicated for Mercury. Inasmuch as both the Apollo and X-20 are considerably more complex than the Mercury system, it is apparent that the smaller relative weight of the control system must be due to the larger vehicle gross weight and the more advanced components and packaging techniques used in the Apollo and X-20 control systems.

3. If the weight of the Apollo control system is compared to the total gross weight of the translunar vehicle rather than to that of the Command Capsule alone, the ratio will be almost 700:1. This illustrates an important trend for future vehicles, namely that as the vehicle gross weight increases, the relative control weight decreases and thus becomes a less critical factor in buildup of vehicle weight. This will allow greater use of redundant channels and derated components, thus making possible the reliability that will be required for deep-space voyages.

The over-all effect then of an increased vehicle weight (or a decrease in control weight due to more advanced components) will be to allow more freedom in the functional design of the control system. This freedom will undoubtedly be used in improving performance and, even more important, in employing advanced multiple-channel redundancy techniques to improve mission reliability.

Equipment Thermal Control Concept

General Considerations.— The choice of a thermal control concept for the control equipment in a manned spacecraft is quite often determined by the seemingly unrelated factor of equipment location. That is, is the equipment located in a pressurized or unpressurized area? This is quite important because, with the current trend toward a comfortable, air-conditioned, shirtsleeve cabin environment, equipment in the pressurized area can operate under what is often referred to as room temperature laboratory conditions. Thus air is available for removing electronic equipment waste heat as long as the vehicle remains pressurized. Such waste heat can be added to the air by forced convection through the devices.

If for some reason the air pressure is lost, equipment waste heat must be dissipated to the equipment mounting structures and surroundings by conduction and infrared radiation. Unless equipment power levels and duty cycles are extremely low, excessive piece-part temperatures can result and equipment life may be severely reduced or terminated. Many devices can survive indefinitely under conditions of mounting surface conduction and infrared heat transfer if they are provided with external package surface area proportional to the internal heat generation rate. For example, neglecting conduction into the vehicle air frame, on the order of 10 watts per square foot can be dissipated from the surface of a device without exceeding 180°F component temperatures (for 140°F ambient).

For equipment with greater unit area heat flux, piece-part temperature may become excessive after loss of pressure so that operating life will be reduced. For earth orbiting spacecraft this condition need not be catastrophic because the thermal capacity of the equipment package and its mounting can absorb enough heat to prevent immediate damage. For a well-designed package, an operating time of 30 to 90 minutes is usually available after depressurization, and during this period the spacecraft can leave orbit, re-enter, and land.

If for various reasons the equipment is located outside the pressured area, it must usually be provided with a heat sink into which heat can be discharged by conduction. There can be an appreciable amount of cooling by radiation alone, but this is sufficient only for very low power dissipation devices. The amount of radiation occurring will not usually be sufficient for the average device and care must be taken to ensure adequate heat flow from all components to eliminate hot spots. The heat sink is usually a metal-to-liquid heat exchanger to which the chassis is attached. The hot liquid is either circulated through an external space radiator where heat is radiated to space (Gemini and Apollo) or the liquid may be ejected overboard (Mercury).

The liquid heat exchanger approach eliminates the problem of equipment heat dissipation during depressurization and also may have advantages during normal vehicle conditions. Studies show that most manned space vehicles in near-earth orbits or greater than approximately 0.8 astronomical units from the sun will require heating to maintain air temperatures between 70 and 80°F. Thus, it may be necessary to obtain heat from electronic equipment and add it to the air in various compartments where it is lost through the vehicle walls.

Selection of Component Packaging Scheme.

When the factors affecting thermal design of the equipment have been defined and constrained, a component packaging scheme must be selected which is compatible with the other aspects of equipment design, such as electronic performance, vibration, and shock. Selection of the thermal packaging scheme is based on realizing component temperatures commensurate with mission reliability and minimum package mass. Detailed calculations are made for component temperatures, based on the thermal environment and component heat dissipation. Digital and analog computer techniques can be employed for prediction of component temperatures. These analyses show problem areas which must be resolved by design modification.

In convection-cooled electronic equipment, problems occur with components whose internal heat generation is large compared with envelope area available for heat transfer. Additional metal must be used to spread waste heat over greater area. Heat transfer coefficients on the order of 10 BTU per square foot-degrees F are attainable in convection-cooled packages at one atmosphere air pressure. For a typical power transistor, the resulting thermal impedance between the envelope and the air stream is approximately 12°C per watt. If this impedance is too great, the component must either be mounted on a metal chassis or must be attached to a separate finned assembly. The latter approach is less desirable because it requires addition to the package mass without increase in the package structure. In the case of large complex packages it is often necessary to employ a "cut and try" approach in order to obtain desirable component temperatures with a minimum of cooling air flow.

The internal design of packages cooled by conduction to a heat sink involves sizing of thermal conduction paths from components to the package mounting surfaces; however it is also important to consider internal infrared radiation from the components. For example, a 4 by 3-inch circuit board spaced 3/4 inch on each side can dissipate approximately two watts with components at 180°F and surroundings at 140°F. Many switching and logic circuits have heat dissipation within two watts and thus no conduction heat transfer paths are necessary to prevent excessive piece-part temperatures.

There are several general approaches to be considered in the design of conduction-cooled packages. One approach is to sort out the piece-parts with high internal generation (such as power transistors, resistors, zeners, diodes) and mount these directly to the metal chassis. The remainder of the components could then be mounted directly on epoxy component boards or in open or potted welded modules.

In circuits where the majority of piece-parts generate a large amount of heat (one watt and up) and are also of large size, epoxy card mounting is generally undesirable for structural and thermal reasons. In this case, metal chassis mounting is the best approach.

In circuits where piece-parts generate between zero and 1.5 watts and are of small size, it is possible to mount all components in open or potted welded modules which are attached to composite aluminum and epoxy boards. During equipment operation in high vacuum (greater than 10^{-4} torr), heat conduction across interfaces is greatly reduced unless interface pressures are kept high (greater than 30 to 50 psi). Bolted, welded, or glued joints must be used in packages designed for steady-state space operation.

One interesting general conclusion can be drawn from Honeywell's experience in thermal design of hard-mounted electronic equipment. For either convection-cooled or conduction-cooled packages, stress and shock considerations, not thermal considerations, determine cross-sectional areas and surface areas of metal chassis parts. As a rule, therefore, good thermal design can be added to a package with little or no increase in package weight or volume.

Examples of Current Manned Spacecraft Control Systems

Mercury Automatic Stabilization and Control System

The first United States manned spacecraft program was conceived and carried out in an atmosphere of urgency, with no background of direct experience, and with deep concern for flight safety. Under such circumstances, the Mercury Automatic Stabilization and Control System (ASCS) was the result of conservative and proven design principles to minimize operating risks and development time.

A major portion of the ASCS was designed by Honeywell under contract from McDonnell

Aircraft Corporation. Certain components of the ASCS, such as the horizon scanners and the reaction jet system, were developed by other companies under McDonnell contracts.

Functional Requirements.- Because man's ability to perform in space was not completely understood before the Mercury flights, the ASCS had to be fully automatic, that is, capable of performance throughout the entire mission profile without astronaut assistance. Reliability was therefore the important design objective, since the ASCS is the primary system for Mercury capsule attitude control. Other major design constraints were minimum weight, minimum power consumption, and maximum use of previously developed and proven hardware.

The ASCS (Figure 6) consists of attitude reference components, rate sensors, logic electronics, and suitable displays. It is designed to sense spacecraft attitudes and rates and send signals to the control jets to maintain the desired attitude or to change from one attitude to another. Automatic, semi-automatic, and manual control may be selected for any or all of the three axes, and simultaneous operation of manual and automatic control is also possible. The functional requirements of the ASCS are best described in terms of six operating modes:

Rate Damping - Reduce pitch-yaw rates from 50 degrees per second (or less) to 0.8 degree per second within five seconds. Reduce roll rate from 10 degrees per second (or less) to 0.8 degrees per second within five seconds.

Orientation - Perform 180-degree yaw maneuver and position capsule in pitch to commanded attitude of 14 degrees. Hold commanded attitude in each axis within five degrees.

Orbit - Maintain attitude in each axis within five degrees.

Retrograde - Position capsule to retrograde pitch attitude of 34 degrees.

Post-Retrograde - Position capsule in pitch to re-entry attitude (one degree down) and maintain attitude in each axis within five degrees.

Re-Entry - Upon sensing 0.05-g deceleration, maintain pitch-yaw rates of less than 0.8 degree per second. Establish and maintain constant roll rate of 10 to 12 degrees per second.

Mechanization.- Two unfloated two-degree-of-freedom displacement gyros are used for attitude reference. The roll-pitch gyro is used as a vertical gyro with its spin axis aligned to local vertical. The roll-yaw gyro is used as a directional gyro with its spin axis aligned perpendicular to the orbital plane. The vertical gyro gimbals are slaved to periodic horizon scanner signals for long-term vertical reference. When the horizon scanners are not energized, a signal proportional to orbital rate is used to orient the vertical gyro in pitch.

Three rate gyros are provided in the system, each having outputs at discrete rates rather than proportional rates. These gyros are used for control in the damper and re-entry modes and are used with attitude error signals to command the switching logic in the orientation, retrograde, and post-retrograde modes. The rate gyros are not used during orbit mode.

The major electronics unit of the ASCS, known as the amplifier-calibrator, contains four major sections: mode logic, gyro slaving loops, attitude repeater servos, and control logic. The amplifiers and logic systems use solid-state devices throughout and approximately 500 diodes and transistors are required.

The mode logic responds to input commands and places the ASCS in an appropriate mode of control. The attitude repeater servos take the attitude gyro output signals representing pitch, roll, and yaw angles and drive multiple outputs: sector switches for control logic, potentiometers for telemetry; and synchro repeaters for attitude indication to the astronaut.

The control logic, which is mechanized by transistor and diode circuits not critically dependent on voltage, receives the step function outputs of the attitude repeaters and the discrete rate signals from the rate gyros. Using these step indications of attitude and rate conditions, along with the output of the mode switching logic delivered by the current phase of the mission, "decisions" are made which result in actuation of appropriate reaction control valves.

The attitude and rate gyros are examples of previously developed hardware which was adapted on short notice for use in Mercury. The gyros were originally designed for operation in autopilots of high-performance aircraft. To meet Mercury requirements, the vertical gyro was equipped with a heavy metal rotor to decrease drift rate by increasing rigidity. By minimizing gyro drift rate, the number of horizon scanner slaving periods could be reduced, thus conserving spacecraft power. Special high-temperature lubricants, wire, and insulation had to be provided in the attitude and rate gyros to ensure operation for extended periods at zero pressure without benefit of external cooling.

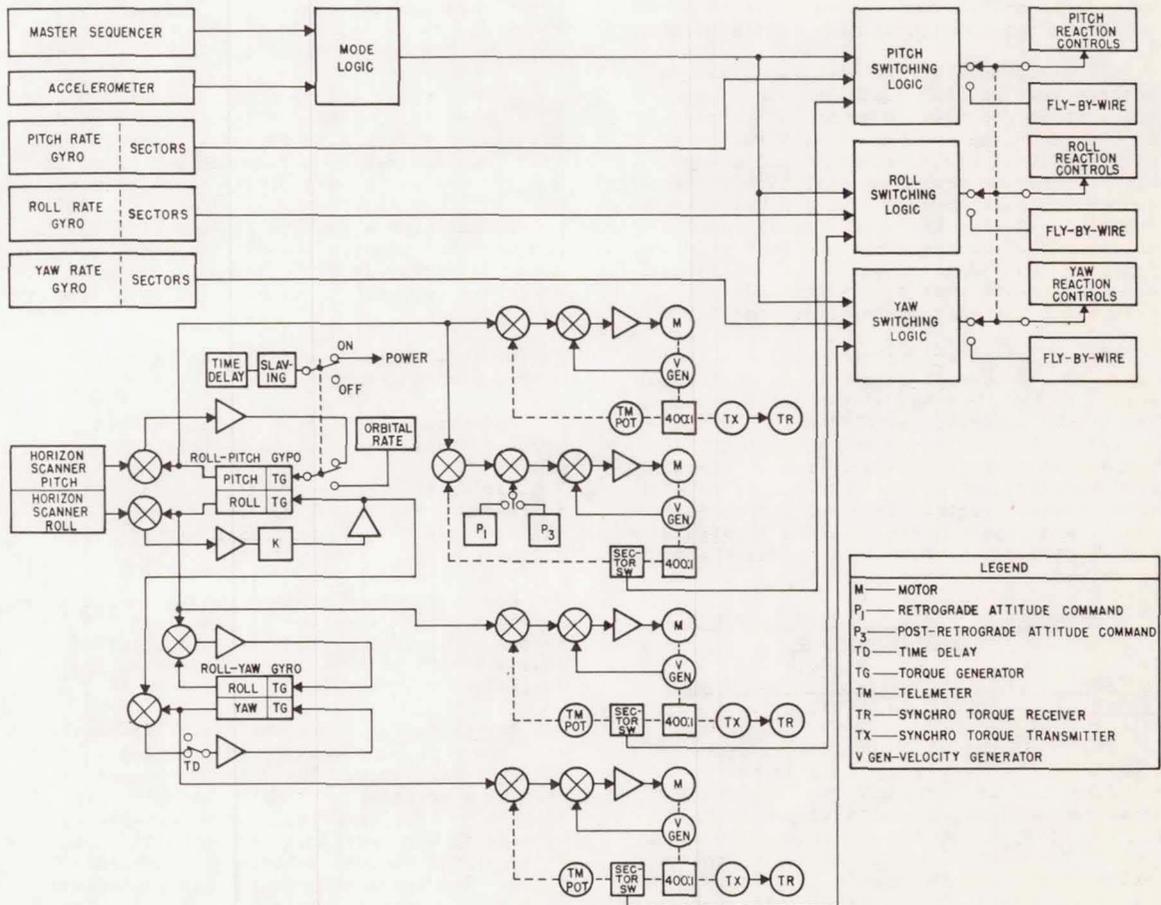


Figure 6. Mercury Automatic Stabilization and Control System

Although weight, space, power, and development time all prevented the use of functional redundancy in the ASCS, several design considerations are worth noting:

1. The digital nature of the control logic provides a degree of redundancy because the orbit attitude is maintained within desired limits by a series of five sector switches for each axis. Each switch backs up the previous one so that failure of any single switch will result in only minor variations from the normal limit cycle.

2. The various modes of operation are also arranged to back up other modes. Thus, if for any reason orbit mode cannot be maintained, the system switches into orientation mode. This has actually happened on several flights because of malfunctions of some of the small jets used for orbit mode control.

3. Another form of redundancy is shown by the use of both horizon scanners and attitude gyros. Early flight tests indicated that the horizon scanners, although performing reliably, sometimes mistook high altitude clouds and hurricanes for deep space and therefore provided an erroneous attitude reference. These effects are not serious when the gyros are slowly torqued to the scanner reference, but could be annoying if the erroneous signals were used directly for control logic information. Later design changes have improved the horizon scanner's operation.

Environmental Factors.- Extensive out-gassing precautions were observed because the ASCS equipment is located in the capsule with the astronaut. The paint and varnish used in all ASCS components was specifically designed to meet rigorous nontoxicity requirements. An epoxy coating which is nontoxic under conditions of high temperature and low pressure was developed for humidity and salt spray protection. Special nontoxic hookup wire is used throughout the Mercury equipment.

The 100 per cent oxygen atmosphere requirement necessitated the enclosure of all components with switching contacts and special selection of materials which are inert to oxygen.

Launch vibration and acceleration presented no difficult problems to the ASCS design since similar gyros and electronics had performed well under severe aircraft testing. All electronics except the attitude repeater circuitry is hard-mounted in the capsule.

No special heat transfer methods are provided in the Mercury capsule for ASCS equipment. To ensure operation under the zero pressure requirement, the equipment is designed with a maximum number of conducting paths from heat generating elements to minimize hot spots and to use the entire package structure as a heat sink.

Reliability of the ASCS has been exceptionally good on all flights with no control system failures to date. This result has thus verified the wisdom of the conservative design approach for the Mercury program.

Gemini Attitude Control and Maneuver Electronics

The primary objectives of Project Gemini are (1) to provide early manned rendezvous capability by development of rendezvous techniques and (2) to provide long-duration manned flight experience to evaluate man's performance capabilities under prolonged periods of weightlessness. These objectives are clearly different from Project Mercury, and the design of the Gemini flight control system reflects this difference.

McDonnell Aircraft Corporation determined the Gemini control system functional design, and Honeywell implemented and mechanized the functional design of the Attitude Control and Maneuver Electronics (ACME).

Functional Requirements.- Mercury experience has demonstrated that man is highly capable of exercising control techniques in an orbiting spacecraft. The Gemini control system is therefore not fully automatic. Selection of control modes is required of the astronaut since a programmed sequence of modes will not be used. Because the Mercury control system was designed for automatic operation about particular set points, it is limited to particular attitudes which can be maintained. The Gemini control system is much more versatile because it has a pseudo all-attitude hold mode with capability of holding attitude rate to less than 0.1 degree per second.

Mission durations up to two weeks dictate the heavy emphasis placed on low power consumption, light weight, and high reliability in the design of the control system. The study of rendezvous techniques places an additional heavy emphasis on control system performance.

The ACME functional design requirements are:
Automatic Attitude Hold - Maintain spacecraft attitude within one degree of the attitude reference supplied by the inertial platform, radar system, or computer. Maintain spacecraft rates at less than 0.25 degree per second.

Horizon Scanner Orbit Control - Maintain spacecraft roll and pitch attitudes within five degrees of the infrared horizon sensor reference. Provide for manual control of the yaw axis.

Rate Command Control - Maintain spacecraft angular rates in response to astronaut hand controller commands in conjunction with rate gyros. Maintain capsule rates within 0.1 degree per second of the commanded rate during orbit and within two degrees per second during re-entry.

Manual Control - Convert attitude hand controller signals to continuous or discrete (20-millisecond) commands to the attitude reaction jet system. Accept maneuver hand

controller signals to fire the maneuver reaction jets continuously or for discrete periods (250 milliseconds).

Mechanization.- The Gemini roll axis control diagram is shown in Figure 7. Attitude error signals originating in the computer, inertial platform, or radar system, are presented to the attitude control electronics for summing with rate information from the rate gyros. Proportional attitude hand controller signals are also presented to the electronics for processing. According to the commanded mode, the attitude control electronics selects the proper input signals and establishes the required gains for signal processing. The input error signal is then amplified, demodulated, discriminated, and compared to a reference switching level. When the error signal exceeds the reference switching level, an ON command is sent to the attitude or re-entry reaction jet solenoids or, for translational thrusting, to the orbit attitude maneuver electronics.

Power consumption in the Gemini ACME system in the orbit mode, using rate gyros, is about one-fourth that of the Mercury system in the same mode. This is accomplished through the use of very low current circuits. For instance, the low hysteresis switch, which converts the analog attitude information into on-off commands to the solenoid drivers, operates at only three microamperes of input

current. Hysteresis is so low in this switch that special laboratory equipment is required to detect it.

The Gemini control system is also capable of operating in the orbit mode with attitude signals from the horizon scanner alone, using pseudo-rate for damping. The system power consumption is then only three watts, 1/25th that of the Mercury control system in the orbit mode. This is made possible in part by pseudo-rate circuitry which provides rate signals without the use of rate gyros and their attendant power consumption. Other important factors contributing to efficient use of power are the use of de-energized relays in orbit mode, transistorized amplifiers, switches and gain-changing circuits, and optimization of the power supply for orbit mode loads.

The Attitude Control and Maneuver Electronics is required to meet extremely high reliability figures. For a two-week mission, the control system probability of success is 0.99721, and for a two-day mission, the figure is 0.999347. To attain this kind of reliability the system incorporates high-reliability parts, extensive redundancy, and derating of all components. Figure 7 shows the general areas of redundancy. The rate gyros are redundant and can be individually selected by axis. The switching amplifiers and logic are also redundant and can be individually selected.

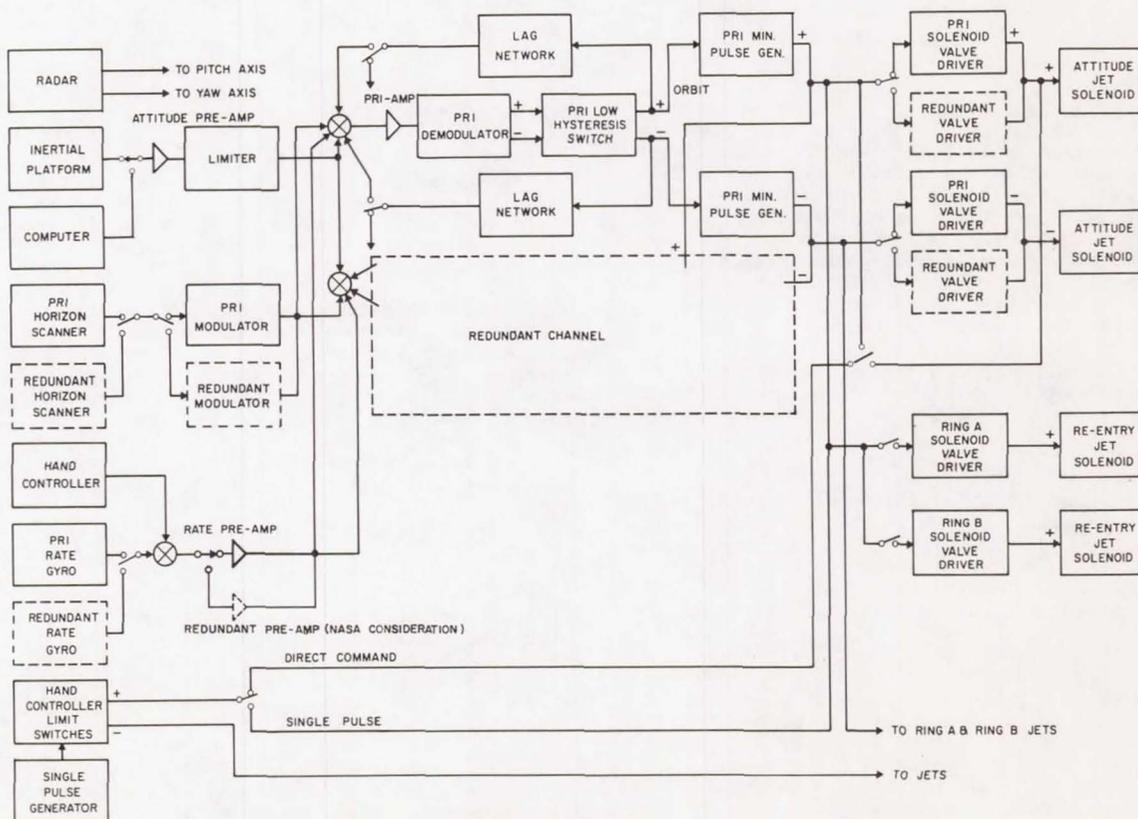


Figure 7. Gemini Roll Axis Control Diagram

Figure 8 shows the maneuver on-off logic and the redundant reaction jet solenoid drivers. These can be selected on a primary or secondary basis.

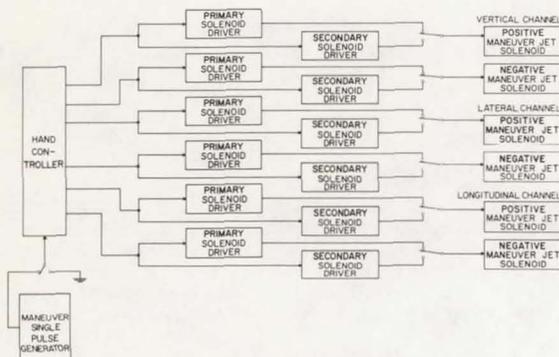


Figure 8. Gemini Maneuver On-Off Logic

In spite of the redundant circuitry and increased capabilities of the Gemini control system, the entire ACME weighs only 37 pounds compared to 52 pounds for the Mercury control system.

This light weight is made possible by use of:

1. Magnesium for the power inverter and rate gyro package castings.
2. Minimum gage sheet metal as determined by extensive stress analysis.
3. Miniature components assembled into "cordwood"-type welded modules.
4. Potting compound used only in electronic modules requiring special thermal considerations.
5. Solid-state switching in all signal circuits.

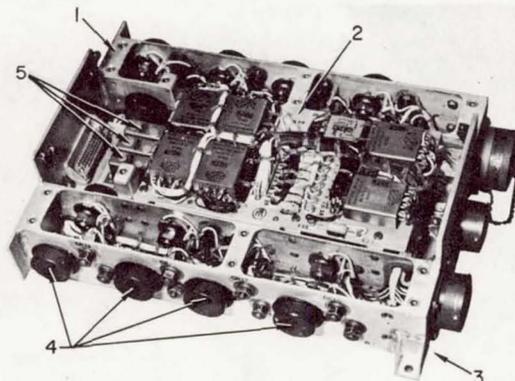
Environmental Factors.- Since the ACME equipment is not located inside the crew compartment, as in Mercury, operation is required in a vacuum environment. Circulating fluid heat exchangers, or coldplates, are provided for equipment mounting. Two approaches were used for thermal design: In the attitude control electronics package, it was possible to sort out the piece-parts generating most of the heat and mount them on the chassis for conduction of the heat to the coldplate. The remaining piece-parts are mounted on epoxy cards since they have such low heat dissipation that infrared radiant heat transfer to the package walls is adequate. In the orbit attitude and maneuver electronics, inverter, and rate gyro packages, all significant heat generating piece-parts are chassis-mounted. Figure 9 shows the method of mounting switching transistors on the aluminum channels and the broad base used for maximum coldplate mounting surface.

The use of aluminum channel chassis

design not only provides extensive heat conducting paths, but also affords a rigid truss-like structure for vibration resistance.

Each electronic module card is coated with an epoxy compound for protection against high humidity and salt fog atmosphere.

Maintainability.- Maintenance problems are greatly simplified in the Gemini control system. All adjustments, alignments, and calibrations are permanently accomplished at the factory. Complete interchangeability of all removable parts, sub-assemblies, and components is assured. Vehicle maintainability is also improved. The Mercury equipment is installed in layers within the one-man compartment, while the Gemini equipment is housed in bays around the outside of the vehicle. The increased ease of checkout and equipment maintenance places manned space-flight on more of an operational basis with advantage to both military and non-military applications.



- Legend:
1. Chassis--extruded aluminum channels with welded end caps
 2. Aluminum plug-in relay board
 3. Capsule coldplate (under chassis)
 4. Redundant output switching transistors
 5. Redundant maneuver solenoid switching relays

Figure 9. Gemini Orbit Attitude and Maneuver Electronics

X-20 (Dyna-Soar) Flight Control Subsystem Electronics

The X-20 (Dyna-Soar) manned orbital re-entry vehicle is designed for research of lifting re-entry and equilibrium glide flight problems. The X-20 flight control subsystem electronics is being produced by Honeywell under contract from The Boeing Company for the Air Force.

Functional Requirements.- The X-20 delta-winged orbital glider must be able to re-enter the atmosphere and land at any suitable airfield chosen by the pilot within a circle of

maneuverability over a thousand miles in diameter. Its range of speeds extends from over 15,000 miles an hour in orbit down to a landing speed of less than that of some of our present combat aircraft.

The self-adaptive concept of flight control is being used in the X-20 because of the widely varying flight conditions encountered during its mission. The direct forerunner of the X-20 control concept is the self-adaptive flight control system which has been proven in the No. 3 X-15 vehicle. Since the X-15 and X-20 must function both as aircraft and as spacecraft, many of their design problems are similar. The self-adaptive control system for each vehicle results in uniformly satisfactory performance over an extremely wide range of flight conditions without dependence upon air data scheduling of system gains.

The flight control subsystem is composed of rate and acceleration measuring devices, computing electronics, and control element driving devices to (1) augment the glider's natural aerodynamic stability, (2) compensate for undesirable control characteristics, (3) control the glider through pilot or guidance system commands, and (4) keep the forces acting on the glider within tolerable limits.

Mechanization.- The X-20 flight control electronics is actually three separate systems, one controlling each of the aircraft axes. The pitch axis is illustrated in functional form in Figure 10. This diagram shows the way input and feedback signals, sensed on the left, are

combined, shaped and used to drive the three control elements on the right. Inputs to the system come from three sources: pilot stick commands, vehicle motion sensed by gyros and an accelerometer, and angle-of-attack commands from the inertial guidance system. These signals drive three control elements: the elevator surfaces, a servo-driven rocket nozzle set, and the reaction control jets.

The pilot has four modes of flight control operation available to him:

Manual-Direct - In the manual-direct mode, the pilot uses his control stick to command vehicle movement through the flight control electronics. He may command control surface position, rocket motor thrust vector position, or reaction control operation. No augmentation is provided in the manual-direct mode.

Pilot-Selectable Gain - In this mode the three-axis stability augmentation system is activated in place of the manual-direct control. The augmentation system controls the aerodynamic surfaces, rocket motor thrust vector, and reaction jets in response to gyro and accelerometer commands. Pilot command provides commanded aircraft rate for stick displacement, instead of commanded control movement for stick displacement as in the manual-direct mode. The system loop gains are selected by the pilot for the Mach range through which he is flying.

Manual-Augment - The manual-augment mode is identical to the pilot-selectable gain mode except that the system loop gains are auto-

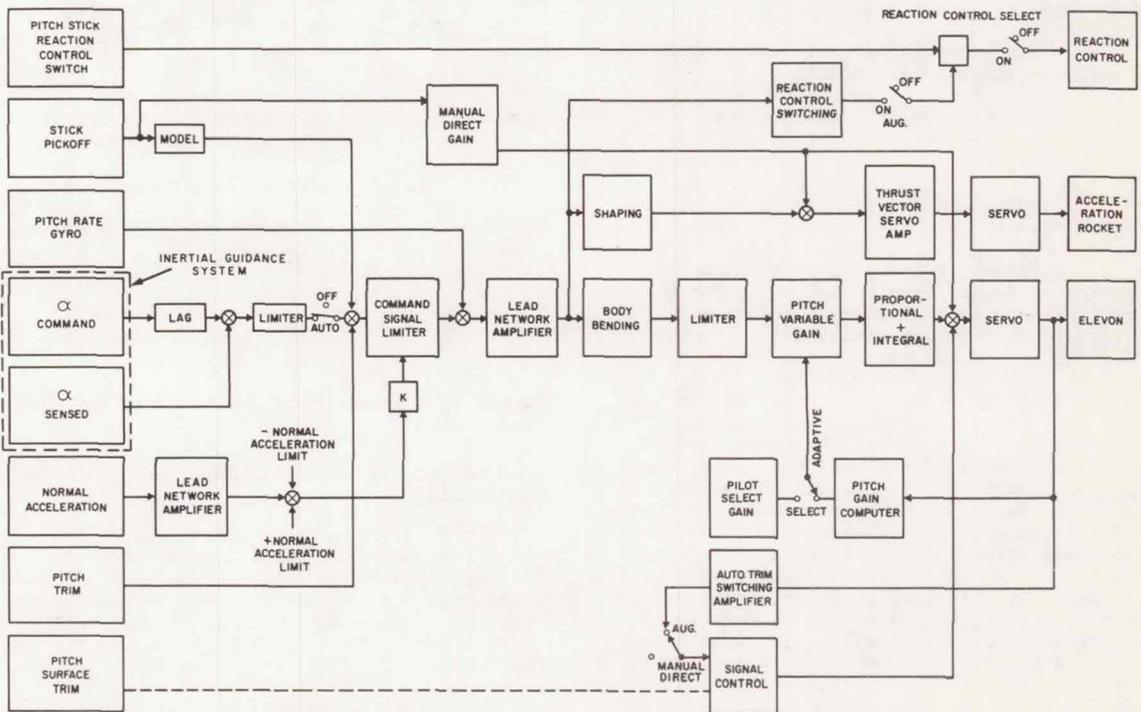


Figure 10. X-20 Pitch Axis Control Diagram

matically computed by the flight control electronics instead of being selected by the pilot. (The Honeywell self-adaptive concept used for this is described in reference 4.

Automatic - The automatic mode is identical to manual-augment except that outer-loop signals are accepted from the inertial guidance system to control angle of attack, sideslip angle, and roll angle. These three parameters are programmed for an automatic re-entry, and the flight control electronics automatically directs the vehicle to follow the programmed guidance system commands.

The command signal limiter (see Figure 10) is designed to limit the pitch commands from the guidance system or pilot's stick to values which will not endanger the vehicle.

Extremely high mission reliability is a requirement of the X-20. The flight control electronics must have a 50,000-hour mean-time-between-failure for a two-hour mission in the manual-augment mode. In addition, neither manual nor augmented performance shall be lost by a single failure. No component replacement is permitted in flight.

The high flight control reliability is achieved by the combined techniques of redundancy, monitoring, and crossfeeding. The flight control redundancy is based on two ground rules:

1. The system will tolerate any single

failure without loss of function or performance.

2. The system will automatically disengage itself as a result of any second failure which can cause a dangerous condition.

Figure 11 shows that the control system sensors and servos are each dual redundant while the electronics is triple redundant. The dual sensor outputs are monitored and then crossfed to the system electronics, and the outputs of the electronic channels are monitored and then crossfed to the servo amplifiers. The dual-redundant servo loops are monitored and the primary servo loop operates the control actuator under normal conditions while the secondary servo loop remains on standby.

Under the above ground rules, it was necessary to make the system electronics triple redundant. During normal operation the electronics output may be positive hardover, negative hardover, or any value between. Therefore, if one electronic channel fails, it will not have an output unique to a failure. A voting mechanism, or monitor, determines which channel differs from the other two and disengages that channel. This satisfies the first ground rule. If either of the remaining channels fails, the voting monitor senses a disagreement between the two channels and disengages the axis of control. This satisfies the second ground rule.

Dual redundancy is provided for the sensors

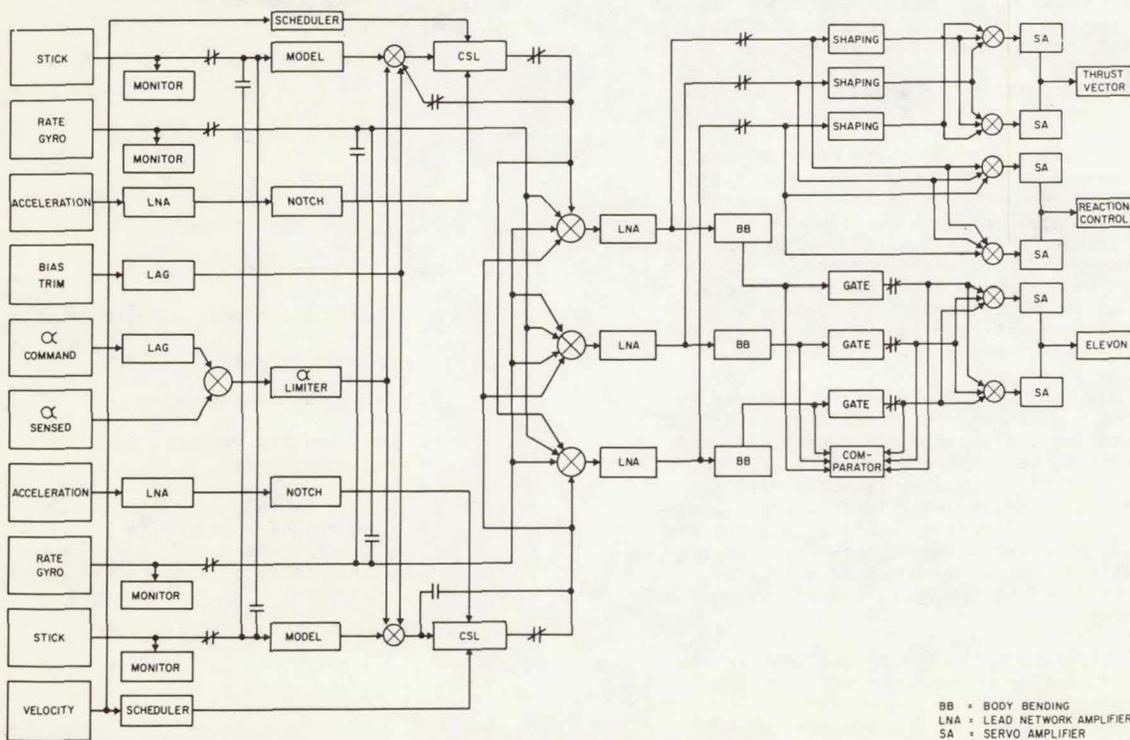


Figure 11. X-20 Pitch Axis Redundancy Mechanization

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because unique indications of sensor failures, such as a gyro open or hardover, can be monitored. A spinmotor rotation detector is also provided to detect gyro motor failures.

The servo system is also dual redundant, but the failure detection monitor employs a triple channel arrangement similar to that described for the system electronics. The monitor contains a servo-loop model which is an electronic analog of the other two loops. By comparing the outputs of the primary and secondary servo loops, and also the output of the servo model, the monitor detects which of the channels has sustained a failure. A failure of the primary loop results in transfer of control to the secondary loop.

The adaptive system uses transistor differential d-c amplifiers as the basic electronic building blocks in summing amplifiers, active filters, and various other functions. These d-c amplifiers are about one-half the size and weight of a comparable magnetic amplifier and have better gain, bandwidth, and drift characteristics. Extremely low drift rates are obtained by using high reliability, matched transistor pairs manufactured from a single silicon chip.

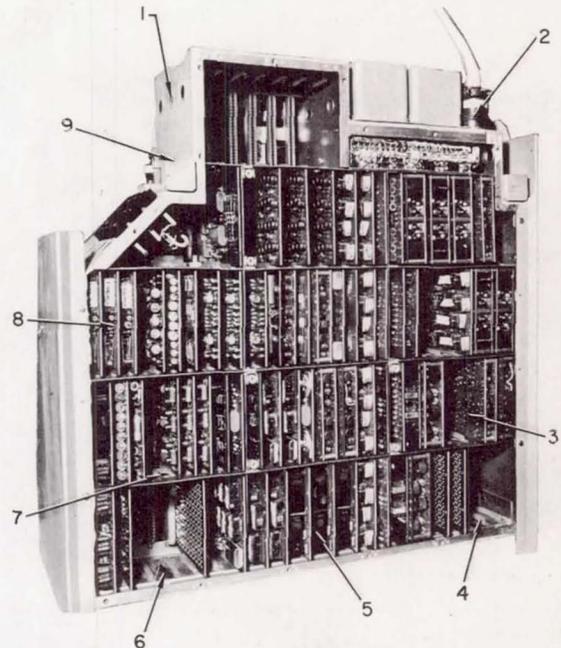
Environmental Factors.— Because the flight control computer (Figure 12) is hard-mounted and subjected to high vibration levels, special care has been taken to ensure a rugged design. The chassis is a formed, half-hard aluminum shell with side covers of honeycomb aluminum sandwich material to provide structural stiffening at a minimum weight penalty. The internal shelves and structural members are half-hard aluminum sheet. The front side of the chassis contains 79 plug-in electronic circuit cards, while the hard-mounted components - power supply transformers, relay cartridges, and bench level test connectors - are accessible from the rear side.

The circuit cards slide into the shelves between nylon guides and engage the mating connector at the rear of the card pocket. Each card is firmly held in position at its four edges: top and bottom by the nylon card guides, at the rear by the card connector, and at the front by silicone rubber bumpers attached to the chassis side cover. The rubber bumpers provide a positive pressure on the card to ensure reliable connector mating.

The plug-in cards are approximately four inches square and contain potted assemblies, cordwood-packaged unpotted assemblies, and individual components mounted on printed circuit cards. In general, each card is associated with a specific function: One card contains four servo amplifiers, another four demod amplifiers, and so on. This grouping of functions creates system flexibility by allowing easier incorporation of design changes.

In contrast to Mercury and Gemini, the primary method of heat removal from the computer is by forced convection. The coolant enters the bottom of the chassis through 135 0.059-inch diameter holes and absorbs heat from the components as it rises through each

level of the computer. The coolant is discharged through the screened air vents near the top of the computer. The configuration of the card assemblies within the chassis offers a chimney effect to facilitate the coolant flow. The air inlet holes in the bottom of the chassis as well as the air passage holes in the shelves of the computer are located for maximum utilization of the coolant. Under emergency conditions without coolant, the computer is capable of operating for two hours with only slight degradation of performance by using the chassis and mounts as heat sinks.



- Legend:
1. Screened outlet air holes
 2. Redundant connectors
 3. Plug-in electronics
 4. Nylon circuit card guide
 5. Welded electronic modules
 6. Air inlet holes
 7. Air passage holes
 8. All circuits at least dual redundant
 9. Dual beam chassis construction with welded shelves and stringers; honeycomb aluminum cover bolted to chassis for rigidity.

Figure 12. X-20 Flight Control Computer

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Apollo Command Module Stabilization and Control System

The complexity of factors affecting the Apollo Command Module Stabilization and Control System (SCS) design is a direct result of the most ambitious mission ever attempted by man. The combined requirements for the multi-phased mission - earth orbit, translunar injection and coasting, midcourse corrections, lunar orbital injection, rendezvous and docking, transearth injection and coasting, earth entry orientation, and re-entry - impose a great variety of design tasks. The Command Module SCS is being developed by Honeywell under contract from North American Aviation for NASA.

Functional Requirements.- Although the detailed (SCS) performance requirements are too extensive for adequate discussion here, the following items indicate some of the factors which have been considered in the functional and hardware design.

1. The SCS is actually a three-in-one system which must interface with Command Module reaction jets, Service Module reaction jets, and Service Module thrust vector gimbal actuators. Each interface requires compatibility matching and different performance requirements.

2. The system shall be capable of controlling rates during limit cycle operation to 0.02 degree per second or less. This severe requirement is necessary to allow accurate navigational sightings and to conserve fuel during coast periods.

3. The reaction system must provide both small amplitude limit cycle and efficient maneuvering operations. During maneuvering the SCS must provide constraints on command rates which will conserve fuel but will not compromise the maneuvering capability.

4. Since the Apollo vehicle must be capable of rendezvous and docking, the SCS jet selection logic must provide simultaneous rotational and translational control.

5. The SCS must be able to effect precision control of velocity corrections in order to meet the narrow entry window from a transearth trajectory at superorbital velocity.

6. The Command Module is a lifting vehicle during earth entry with a L/D ratio of 0.5. The symmetrical shape of the capsule minimizes any aerodynamic cross-coupling, therefore greatly simplifying the entry stabilization problem.

7. The Apollo earth entry problem involves essentially a single axis control of roll attitude with only rate damping required in pitch and yaw. In general, the Command Module represents an optimum design yielding minimum earth entry stabilization problems.

8. The Apollo vehicle has a variable configuration. The SCS must perform initially

with the Command Module plus the Service Module and the Lunar Excursion Module, a combined weight of about 45 tons. On the final segment of the return trip, the vehicle consists of the Command Module alone at about five tons. The variation in vehicle configuration and fuel load results in a wide range of vehicle inertias and center of gravity positions which must be considered in system analysis. Fuel slosh and vehicle bending add to the stability problems.

9. A 0.995 probability of successful SCS operation is desired for a 14-day mission.

Mechanization.- The flight control subsystem of the SCS contains the inertial sensors and electronic computer assemblies which provide both attitude and rate stabilization and control. The flight control hardware consists of (1) a three-axis rate gyro package, (2) an attitude gyro and accelerometer package for both three-axis attitude sensing and longitudinal axis g sensing, and (3) electronic computer assemblies for amplification, shaping and integration of signals, mode switching, jet selection logic, reaction jet solenoid drivers, thrust vector servo control, attitude reference computation, and velocity increment computation.

The SCS pitch axis block diagram is given in Figure 13. Rate gyro signals are summed with limited attitude error signals to provide maneuver stabilization. Manual control inputs are introduced by summing the outputs of two hand-operated rotational controllers with the rate signal. During manual control inputs the attitude errors are synchronized and a rate response proportional to command is obtained. In case of a rate gyro failure, the attitude gyros can be operated in a rate mode if control is required before the rate gyro can be replaced.

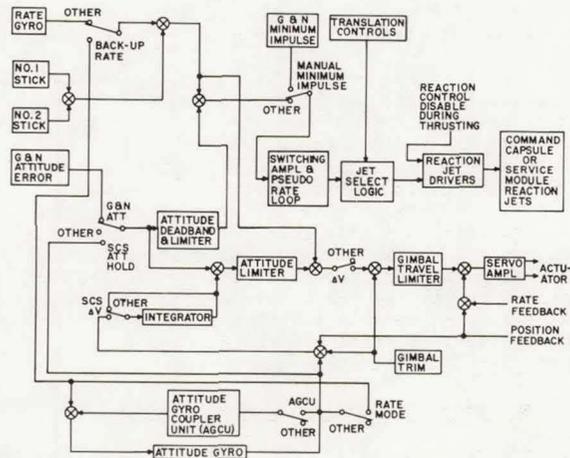


Figure 13. Apollo Command Module SCS Pitch Axis Control

The SCS attitude reference comprises three strapped-down precision integrating gyros specifically developed to meet Apollo

performance and high reliability requirements. The attitude gyros may be operated to provide three functions:

1. For attitude hold, the gyro outputs are used directly as attitude error signals.
2. For rate damping, the gyro output is fed back into the gyro torquer to provide immediate backup rate gyro capability.
3. For attitude reference, the gyro outputs are synchronized through a three-axis attitude gyro coupling unit (AGCU) to provide Euler angle reference information for display and command purposes. The outputs of the AGCU are compatible in reference orientation with the Guidance and Navigation (G and N) system signals.

Attitude error signals generated by either the G and N System or the SCS attitude gyros are fed through a deadband and attitude error limiter. The deadband provides a wide deadband limit cycle for the noncritical coast phases of the mission. During these phases a unique pseudo-rate feedback is used which causes limit cycle operation well within the extent of the rate gyro deadband. In addition the width of the deadband itself can be varied by the crew in order to further minimize reaction jet fuel consumption in those periods of the flight when close attitude control is not necessary. The attitude error limiter acts as a rate command limiter to conserve fuel during extensive automatic maneuvers. Rate signals are summed with the limited attitude error and are fed through the jet select logic, to the switching amplifier and to the reaction jet driver amplifier which provides the power to drive the reaction jet solenoids.

In order to provide the Apollo crew with a rerender rate control in each axis for precision navigational sightings, a minimum impulse command technique may be selected to cause very small vehicle rate changes by pulsing the reaction jet solenoids.

Thrust vector control is based on a rate-plus-displacement technique. In this mode an attitude command is summed with attitude and fed into the control loop. An attitude limiter acts as a rate command limiter, and a gimbal travel limiter prevents the actuator from running against its position stops. Prior to thrusting, attitude hold in all three axes is provided by the reaction jet system. At thrust initiation, the pitch and yaw control is transferred to the thrust vector control loops, and the pitch and yaw reaction systems are disabled. Roll reaction control continues throughout the thrusting maneuver.

Environmental and Maintenance Factors.- Coldplate mounting of the electronics and sensors requires efficient thermal conduction paths. At the same time, the reliability requirement demands standby redundancy, as indicated in Figure 5, which is provided by inflight replacement of gyros and electronic circuit subassemblies. To solve both the coldplate mounting and maintenance problems, special hardware packaging designs have been

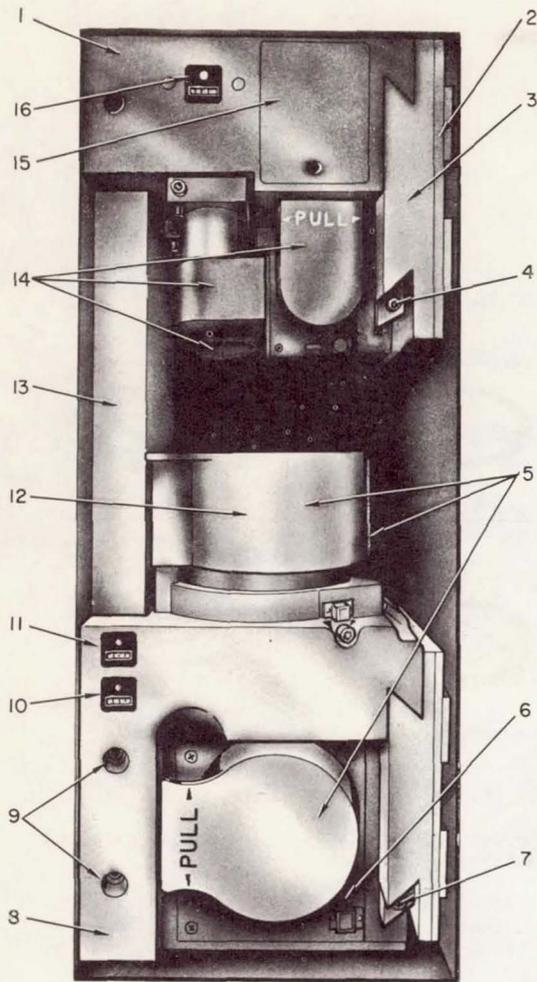
developed which will provide positive mounting contact and convenient package removal by an astronaut wearing his pressure suit and gloves and working under zero g conditions. The hardware must also pass rigid outgassing, humidity, and oxidation requirements.

Figure 14 shows the present approach to inflight maintenance, as evidenced by the SCS rate and attitude gyros and accelerometer in the Command Module. The rate gyro package contains three orthogonally mounted rate gyros. Each gyro has a shroud containing an integral circuit connector. A quick-disconnect clamping mechanism is used to secure each gyro in place. Each gyro and also the gyro electronic module is easily replaceable by an astronaut. Positive, accurate alignment of the gyros to the spacecraft axes is assured by precision surfaces and clamping techniques so that no inflight alignment procedure is necessary. A color indicator at the gyro clamping device shows the astronaut when positive locking is achieved.

The attitude gyro and accelerometer package contains three orthogonally mounted rate integrating gyros and a hinged pendulous accelerometer. Each sensor has a thermally insulated shroud with an integral connector. These sensors, like the rate gyros, may be readily replaced without alignment necessity. Any rate or attitude gyro may be replaced under shirtsleeve conditions without removing the mounting package from the hardware compartment. Even under pressure-suit conditions, the package design permits an astronaut to perform any necessary maintenance.

Inflight replacement of circuits is also required so special consideration was given to the need for packaging all piece-parts together in a replaceable subchassis. Within each subchassis, small piece-parts are packaged in potted, welded modules which are thermally connected to the subchassis. Larger piece-parts are mounted on brackets formed on the subchassis. Each subchassis is clamped in place in an assembly which mounts on the spacecraft coldplate.

The nature of the Apollo mission demands that the control system design must have a high inherent reliability; parts must be of tested and proven high reliability; the techniques of reliability analysis must be valid; and quality control must be rigorous. In addition all parts must tolerate long exposure to high humidity and 100 per cent oxygen without any change in characteristics or release of toxic fumes. To obtain the required reliability and still keep onboard spares at a minimum, it is necessary to use parts which in many cases exceed Minuteman standards. The use of such parts assures the highest inherent reliability. Reliability beyond this level is a direct result of reliability and design teamwork throughout the system development process. The value of this factor to control system performance is of the highest importance in manned space programs.



Legend

1. Rate gyro package
2. Spacecraft coldplate
3. Package mounting plate
4. Package clamp
5. Removable attitude gyros (accelerometer at rear)
6. Quick disconnect gyro clamp and indicator
7. Package clamp
8. Attitude gyro and accelerometer package
9. Interface connector jackscrews
10. Accelerometer elapsed time indicator
11. Gyros elapsed time indicator
12. Thermally insulated gyro shroud and integral connector
13. Spacecraft wiring channel
14. Removable rate gyros
15. Removable electronics
16. Elapsed time indicator

Figure 14. Apollo Command Module SCS Inertial Sensors in Mounting Compartment (Conceptual View)

Manned Orbiting Space Stations

The primary factors affecting design of a control system for a manned orbiting space station naturally result from definition of the configuration and the mission requirements. At this time no specific mission requirements have been defined for manned orbiting space stations and hence no unique configuration has been developed. However, considerable effort has been expended in studying possible mission requirements and applicable configuration designs. Of the basic configurations, four specific concepts have received the most attention. These are illustrated in Figure 15: A rotating hexagonal wheel or radial element configuration providing a simulated gravity effect in the rotating areas and a zero-g laboratory in the nonrotating hub; a non-rotating cylindrical configuration providing zero-g conditions; and a spinning dumbbell configuration consisting of a living module connected to a counterbalancing mass by cable or semirigid tube. Much of the material discussed below is based on the results of a recent joint North American Aviation-Honeywell study.

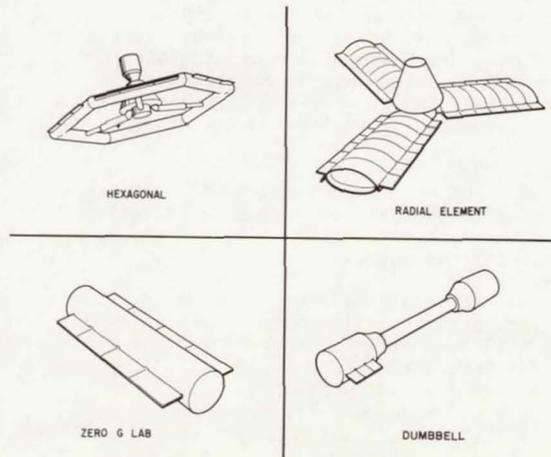


Figure 15. Manned Orbiting Space Station Configurations

Control System Restraints.- For any space station configuration, the major factors affecting control system design stem from operational considerations, such as one to five year life, onboard maintenance requirements, and orientation toward the sun for efficient utilization of solar energy. In addition, a space station would probably require periodic resupply of food, propellant, and other expendable items. This would be provided by a manned or unmanned resupply vehicle which would rendezvous with the station and dock for equipment transfer.

The above factors combine to impose restraints on control system design such as:

1. Reaction jet systems must be designed so that no hazard is introduced by transferring hypergolic propellant components in the resupply operation. Preferably, a complete self-contained system would be transferred from the resupply vehicle and automatically affixed to the outside of the station.

2. Inasmuch as the basic purpose of the personnel aboard is to conduct experiments, vehicle control should be completely automatic. Personnel would serve as monitors of system operation but must also have the authority and provisions to assume complete control when desired or in the event of system malfunction.

3. If a space station is to be developed in the near future, it is probable that solar cells would be used as a source of energy and this would require that one station axis be continuously directed at the sun.

4. In each space station configuration, the size of control elements becomes a significant parameter in studying control system mechanization. For example, a large station may require control moment gyros five feet in diameter with an angular momentum of 30,000 slugs-feet per second.

5. Very few existing control elements can be expected to perform without wearout failure for a three to five year period. Gyros, accelerometers, reaction jets, and any element with moving parts must be designed so that ready replacement can be effective in event of failure. System modules must be designed so that spares can be transferred to the station and installed under zero g environment.

6. Any maintenance which the crew could be expected to perform must be carefully considered in the design of tools and component packaging.

7. Efficient management of energy dissipation for orientation control and rate damping will be a primary restraint on control system design and may be a more significant parameter than system weight.

Performance Requirements.- Control system performance requirements for the nonspinning zero gravity laboratory will not be significantly different from requirements for other space vehicles. Rate damping about three

axes will be necessary. Attitude control in either two or three axes, depending on the requirements for solar orientation and antenna pointing, must be provided. In addition, command control of an unmanned resupply vehicle may be necessary for rendezvous and docking.

For spinning configurations, some new approach to control logic and control element utilization may be anticipated. For example, consider the modes of motion of a spinning vehicle (Figure 16):

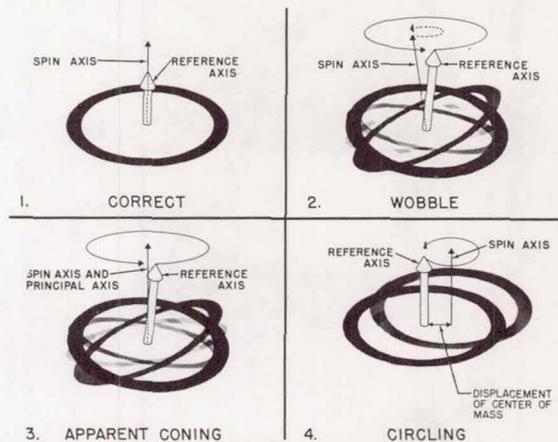


Figure 16. Space Station Modes of Motion

Correct Mode - The body reference axis and the spin axis coincide.

Wobble Mode - There are several equivalent definitions and characterizations of this mode of motion. The simplest form of wobble is the response of a radially symmetric spinning station to an impulsive torque. If the motion is undamped, the "tip" of the reference axis travels at a fixed rate and describes a "circle" in inertial space. Body rates and angular accelerations vary in a cyclic manner, and sensors measuring orientation show an error of either constant or cyclically varying amplitude depending upon the body's mean orientation. Wobble can be damped by reaction jets or, more efficiently, by momentum exchange devices such as reaction wheels or control moment gyros.

Apparent Coning Mode - Mass imbalance out of the station spin plane causes a misalignment of the spin axis and the body reference axis. The "tip" of the reference axis travels at a fixed rate and describes a "circle" in inertial space as it does in simple wobble. However, the rate is always the station spin rate, all body angular accelerations are zero, and all body rates are constant. Momentum exchange devices can very effectively counteract out-of-plane mass imbalance.

Circling Mode - Mass imbalance in the station spin plane causes spin about an axis parallel to but not coincident with the body reference axis. This is a difficult mode to

sense because it produces no inputs to gyro and celestial orientation type of sensors. Body rate about the reference is constant, the other rates are zero, and all body angular accelerations are zero. Circling can be eliminated by deployment of station masses to put the center of mass on the reference axis.

Vehicle attitude must be controlled by orientation of the spin axis. Reaction jets or magnetic torquers are most effective in this role. Reaction wheels are not effective in control of attitude, but would provide efficient control of apparent coning and wobble damping. Control moment gyros could be used in place of reaction wheels.

For both spinning and non-spinning configurations, the most significant source of external disturbance torques will probably result from gravity gradient across the station. This torque results from the fact that the configurations are not symmetric and the differences in the principal moments of inertia will be fairly large. In order to control attitude against the influence of the gravity gradient torque, a significant amount of energy will be required. If reaction jets are used to supply this energy, approximately 1000 pounds of fuel per month could be expended for some configurations. The character of the torque is such that it can be effectively unidirectional for periods as long as 40 to 50 days. The influence can be a significant factor in control system design.

Qualification Testing.— A final consideration which must influence system design is that the character of the system and size of the control elements may require a new philosophy of system qualification testing. For some space stations being considered, it would be impractical to develop a full scale space station simulation to check out and qualify the control system in the manner used for the development of present systems. Lack of a zero gravity test environment and the large size of possible control elements required will complicate the design of adequate tests, and this factor must be considered in the initial stage of system design.

Speculation on the Future

Speculation on the future of a technology advancing as rapidly as that of spacecraft design is about as risky as trying to guess as to which way a woman driver is going to turn. There are however, a few observations which, at least at present, seem fairly safe.

For future vehicles it is likely that the weight and volume of stabilization and control equipment (with the possible exception of reaction jet tankage) will become a minor factor while the stronger emphasis will be placed on high reliability and adequate performance. This statement is made because future control equipment will inevitably become considerably smaller and lighter due to the increased use of microminiature electronics. At the same time it is likely that vehicle weight will increase particularly for scientific exploration vehicles, at least

to the level represented by the Apollo translunar vehicle. The cost of the control system for scientific exploration vehicles will probably be of secondary importance because it, like the weight, will be quite negligible compared to the cost of the entire vehicle. These circumstances will allow control system designers much greater freedom in choosing the functions to be included and the mechanization by which the function will be accomplished.

It is very probable that digital mechanization will play an important part in future space vehicle control, and in fact the identifiable separate control elements may be reduced to sensors and torque producing devices with all computation and signal shaping taking place in a central digital computer. For this millenium to be attained one certain requirement is the development of digital computers with the required long-time reliability.

It also seems probable that a requirement will arise for space vehicles of a totally different type from the exploration vehicles. These will be military vehicles, perhaps of a satellite inspector or an interceptor type. These vehicles would necessarily be as small as possible in order to minimize launch cost. They should ideally of course also be as simple as possible, yet it seems probable that an operational military vehicle would have to have the ability to reach a reasonable choice of landing sites and thus will have to be of the lifting re-entry type. Again from an operational viewpoint such vehicles would need some form of automatic energy management system associated with the basic control system. This class of vehicles would probably present control problems somewhat similar to those now facing the designers of equipment for high-performance military airplanes, namely, a conflict between reliability and the required functional complexity, a conflict between cost and both reliability and performance, and finally one problem (familiar to those who have worked with manned aircraft control systems) providing handling qualities that will please all the pilots.

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