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 preselected 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.

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 onboard 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 shirtsleeve 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 reentry 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 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_DA, 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.

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

$$h_{local} = \frac{q_l}{T_o - T_w}$$

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

q_l = 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, CO_2 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 CO_2 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 re-pressurization 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, 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.

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 programing.

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° per second. Since .1° 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 programed 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 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.

TABLE 1

GEMINI SPACECRAFT

MISSION OBJECTIVES	14 DAYS IN ORBIT	CONTROLLED LAND LANDING	RENDEZVOUS AND DOCK	SIMPLIFIED COUNTDOWN	MAN'S PERFORMANCE
New Design Features Resulting	· Long Mean Life · Fuel Cells · Cryogenics · Space Radiator · PCM Telemetry · Equipment Adapter	· Offset Center of Gravity · Inertial Measuring Unit · Computer · Digital Command System · Paraglider · Landing Gear · Ejection Seats	· Orbital Attitude · Radar · Inertial Measuring Unit · Computer · Digital Command System · Docking Mechanism · Equipment Adapter	· Modular Concept · Accessible Equipment · Built-in Test Points · Automatic Checkout · Integrated AGE	· On-Board Decisions · Manual Control · Partially Removable Space Suit · Egress to Space

OTHER FEATURES GENERALLY CORRESPOND TO MERCURY.

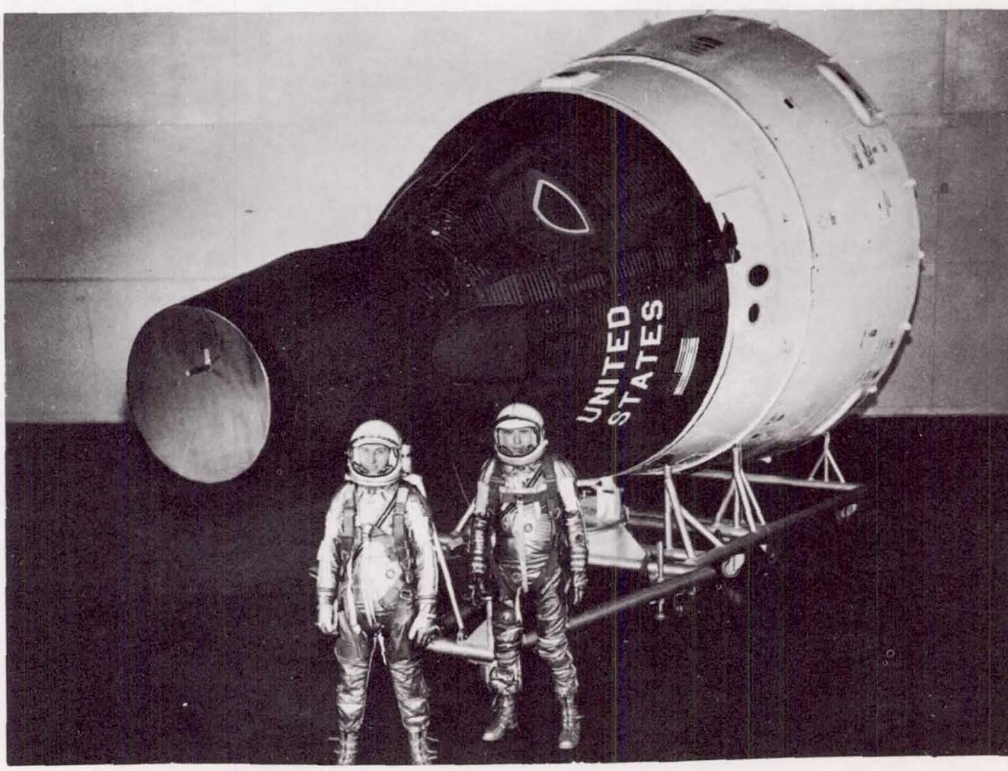


Figure 1. - Mock-up of Gemini spacecraft.

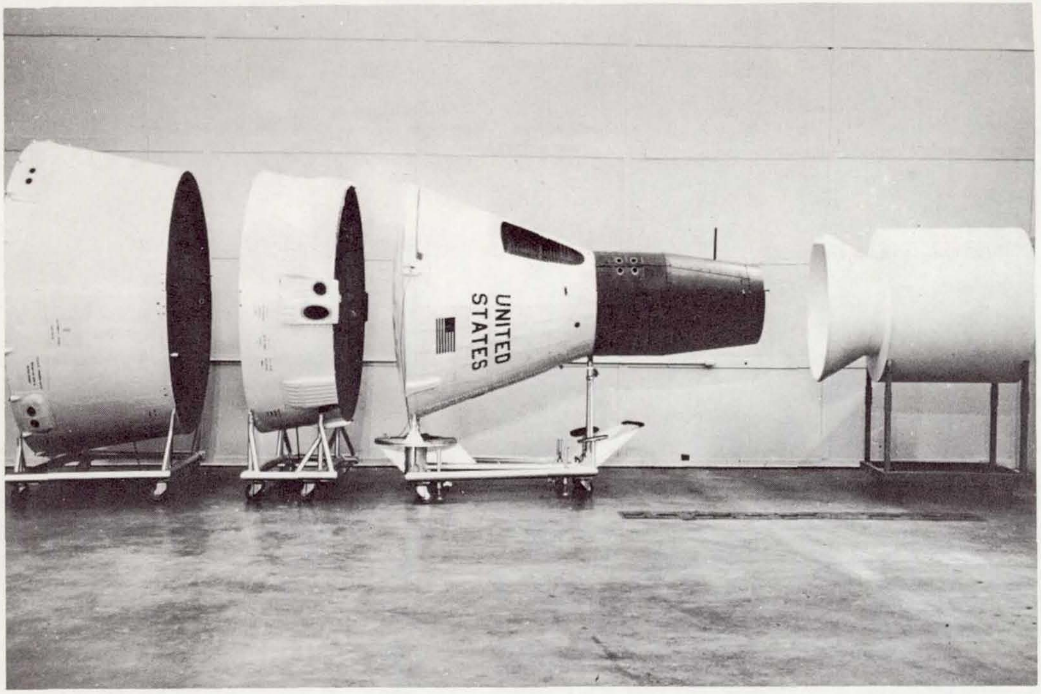


Figure 2. - Division points between major structural assembly points of Gemini spacecraft.

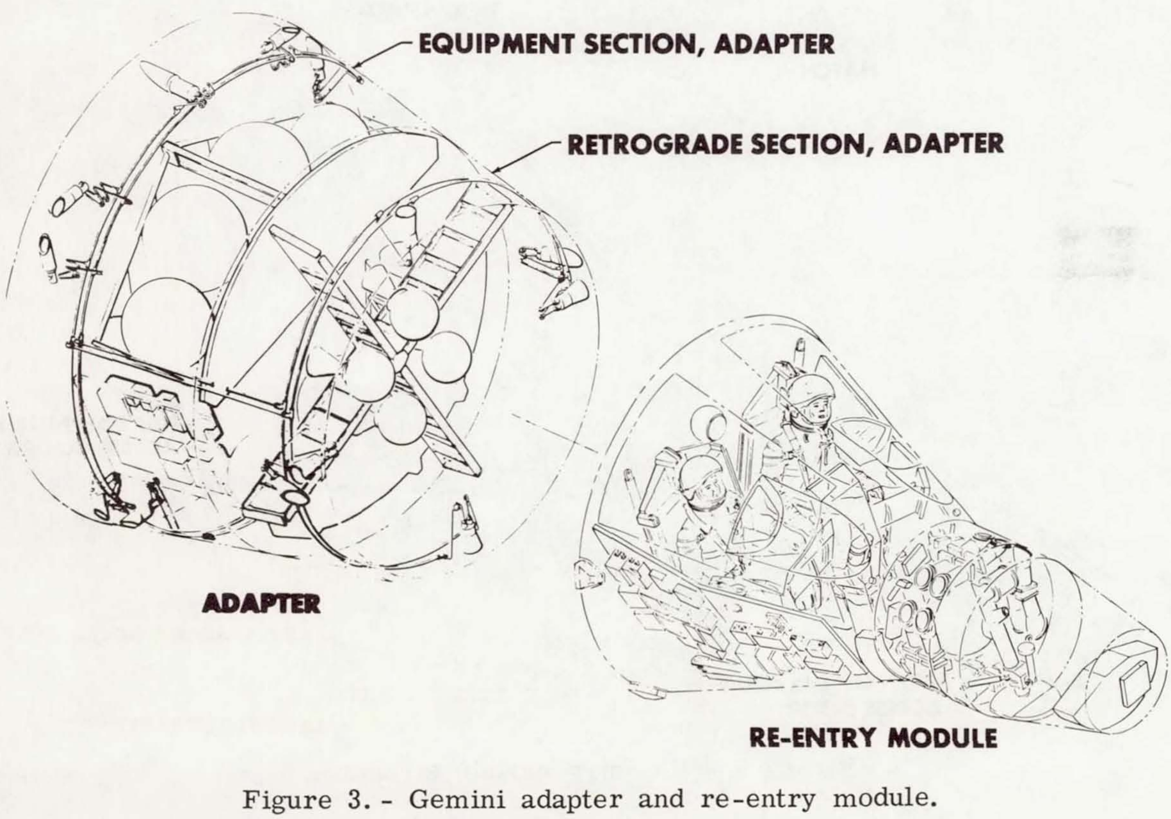


Figure 3. - Gemini adapter and re-entry module.

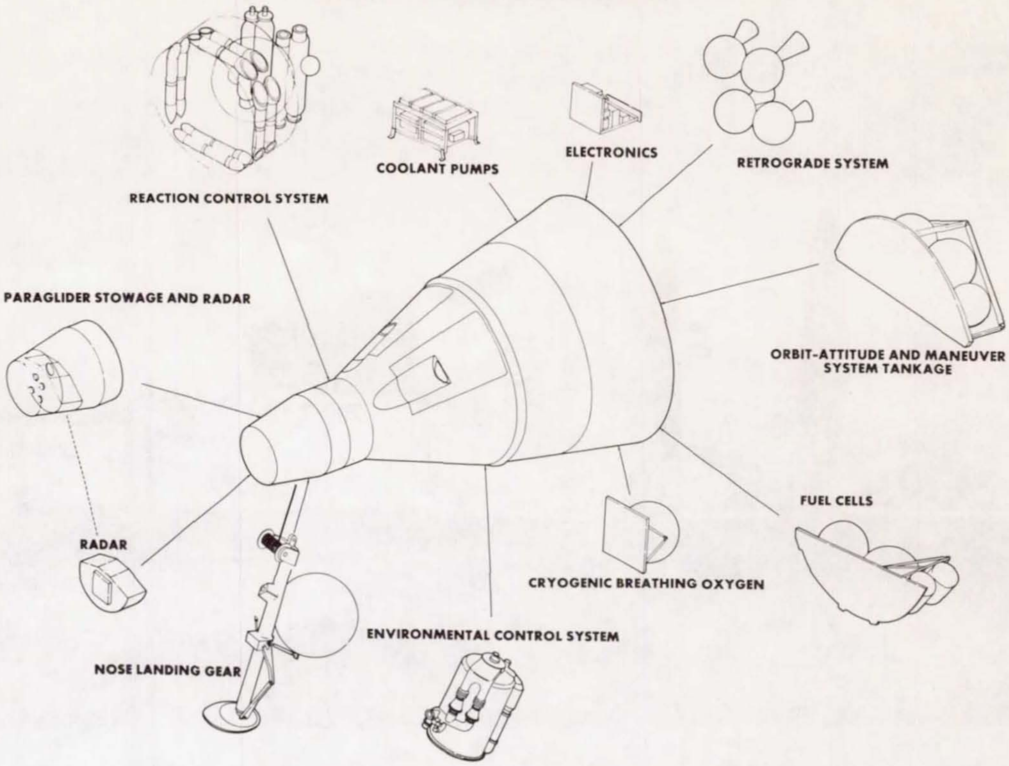


Figure 4. - Subassembly concept.

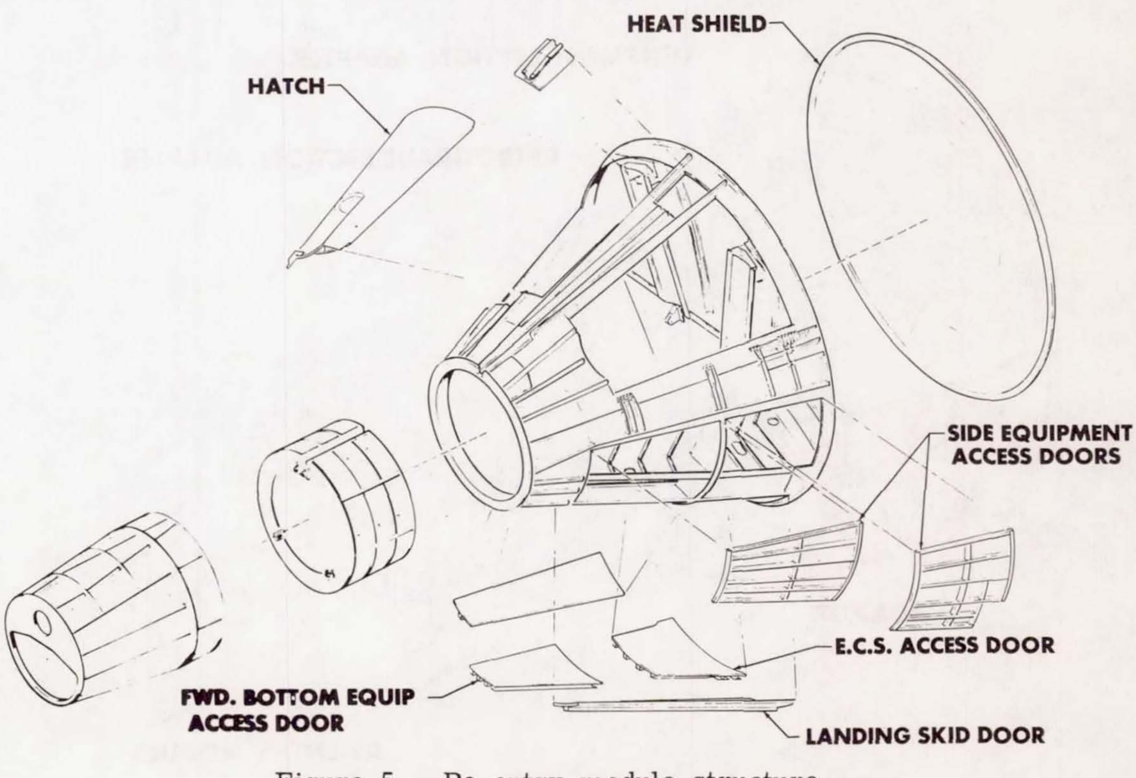


Figure 5. - Re-entry module structure.

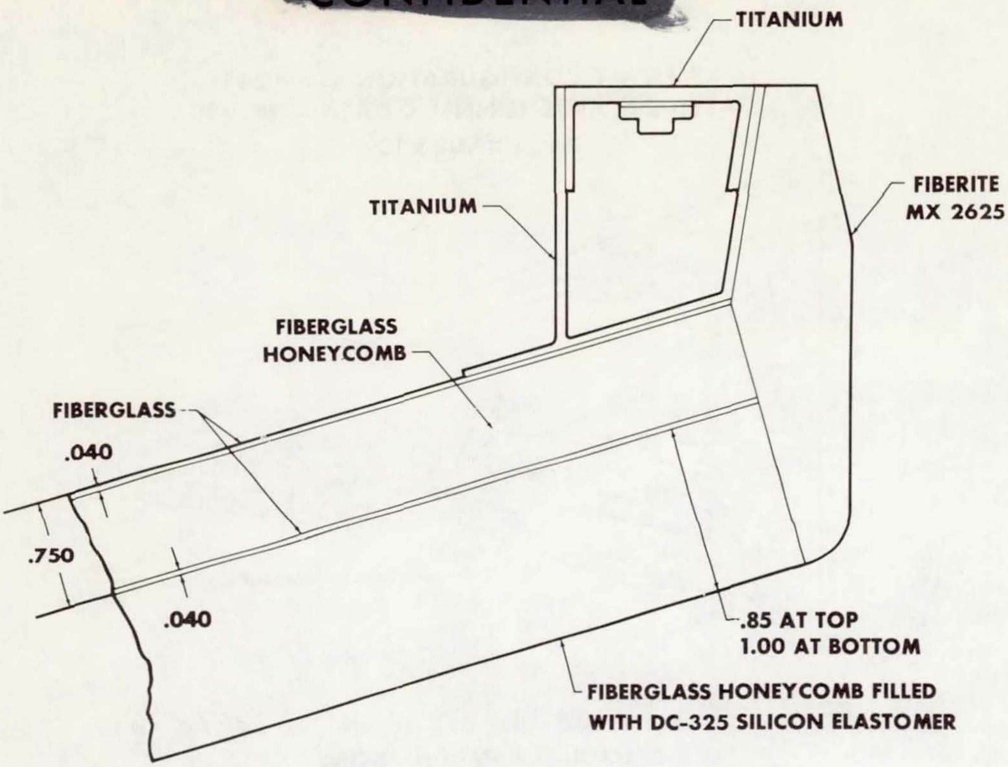


Figure 6. - Ablation shield.

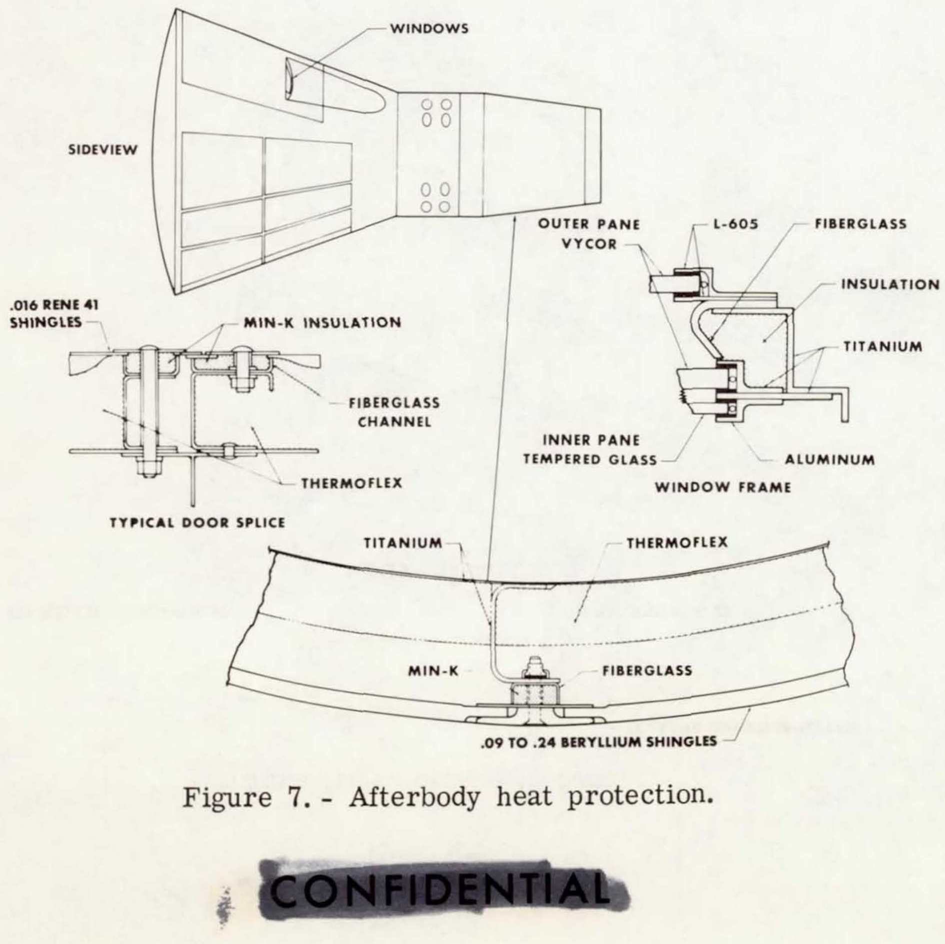


Figure 7. - Afterbody heat protection.

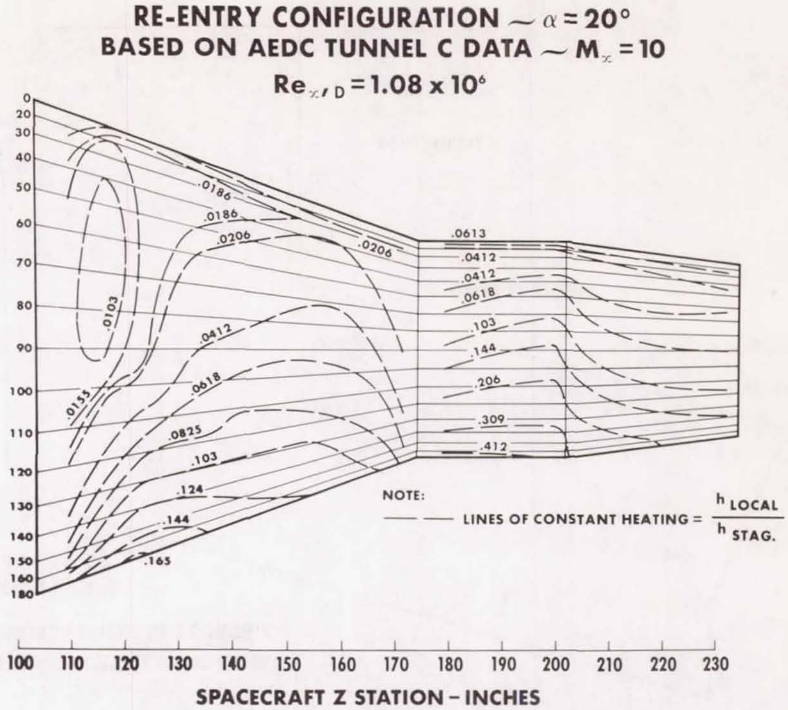


Figure 8. - Gemini afterbody heating distribution.

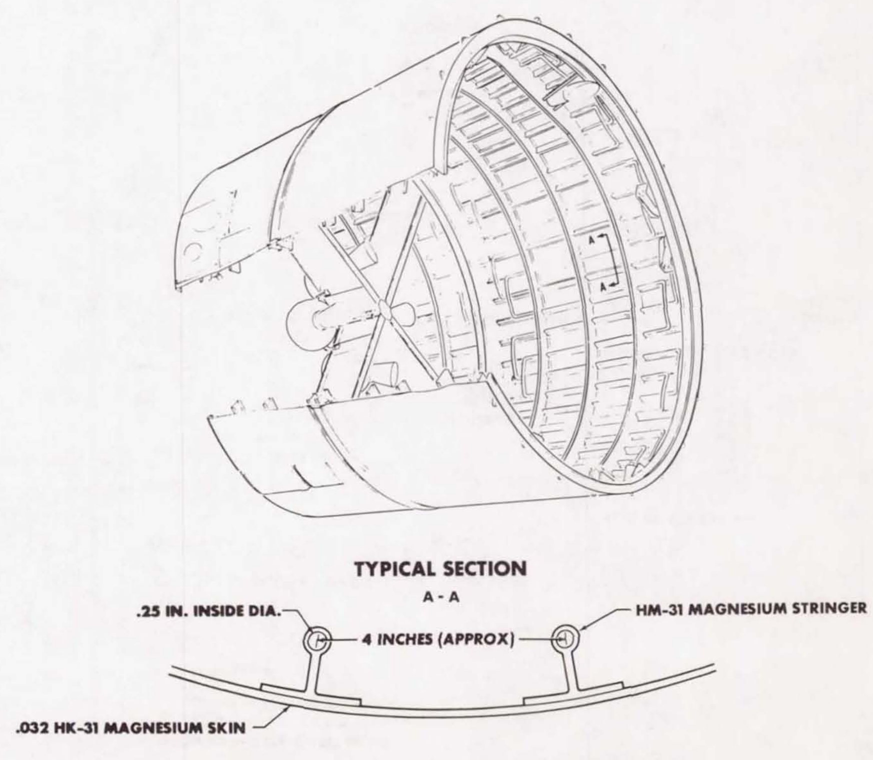


Figure 9. - Adapter structure.

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. H_2O 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.**
- **CO_2 REMOVAL BY LITHIUM HYDROXIDE BED**

Figure 10. - Environmental control system.

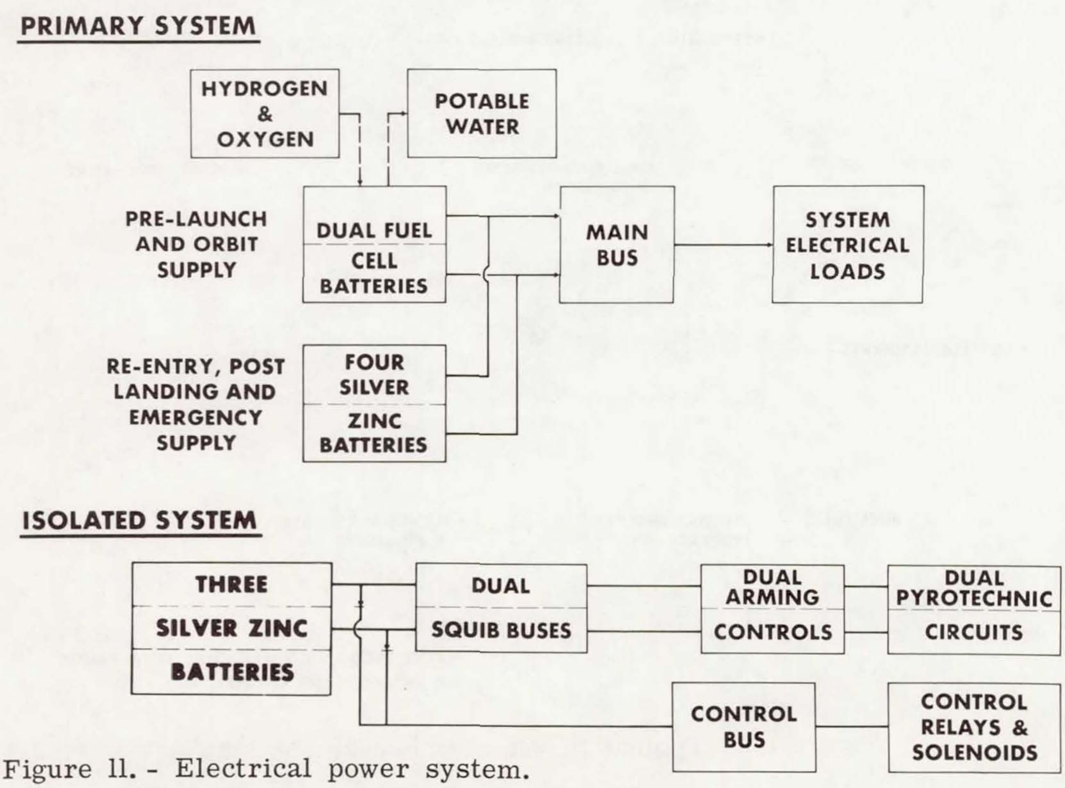


Figure 11. - Electrical power system.

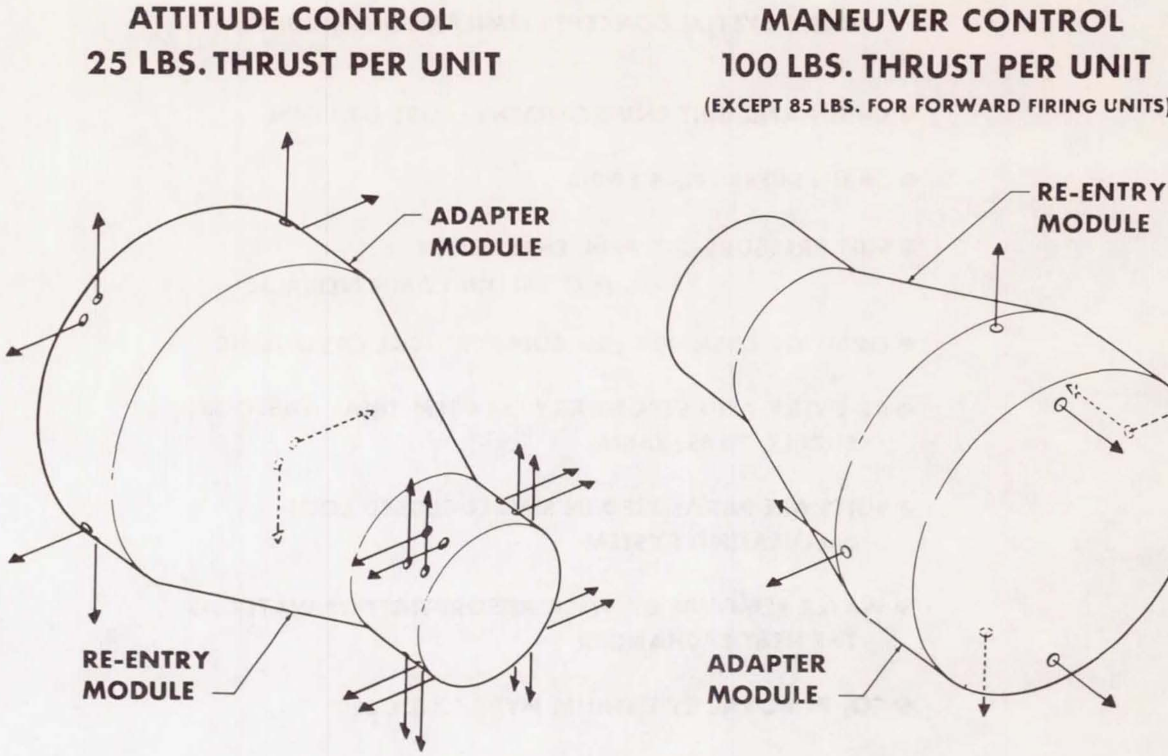


Figure 12. - Thrust chamber arrangement.

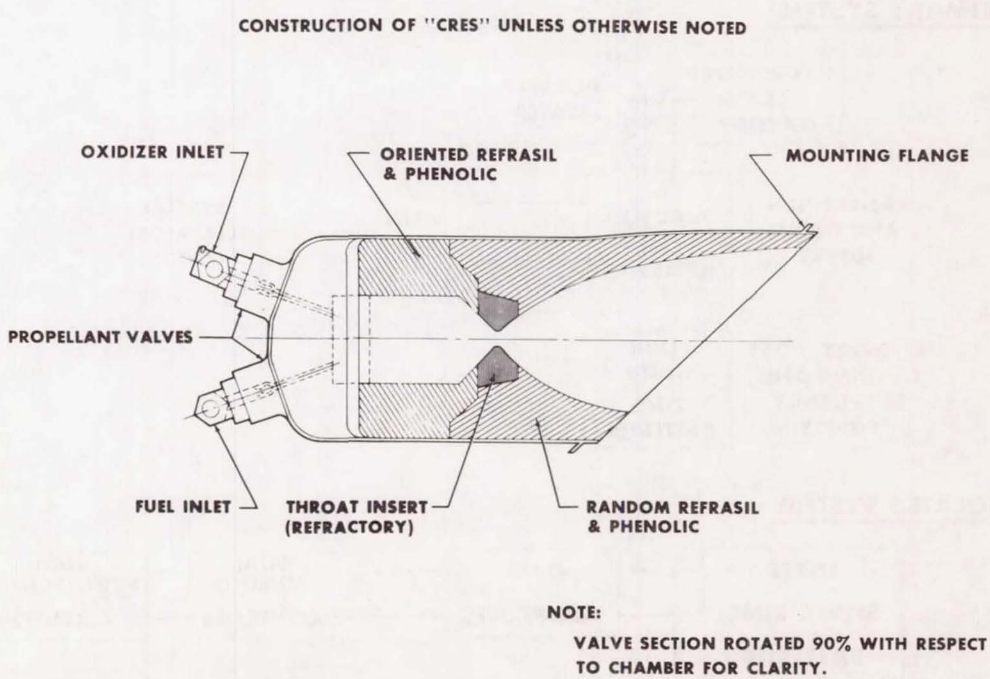


Figure 13. - Typical thrust chamber 25 lb. OAMS

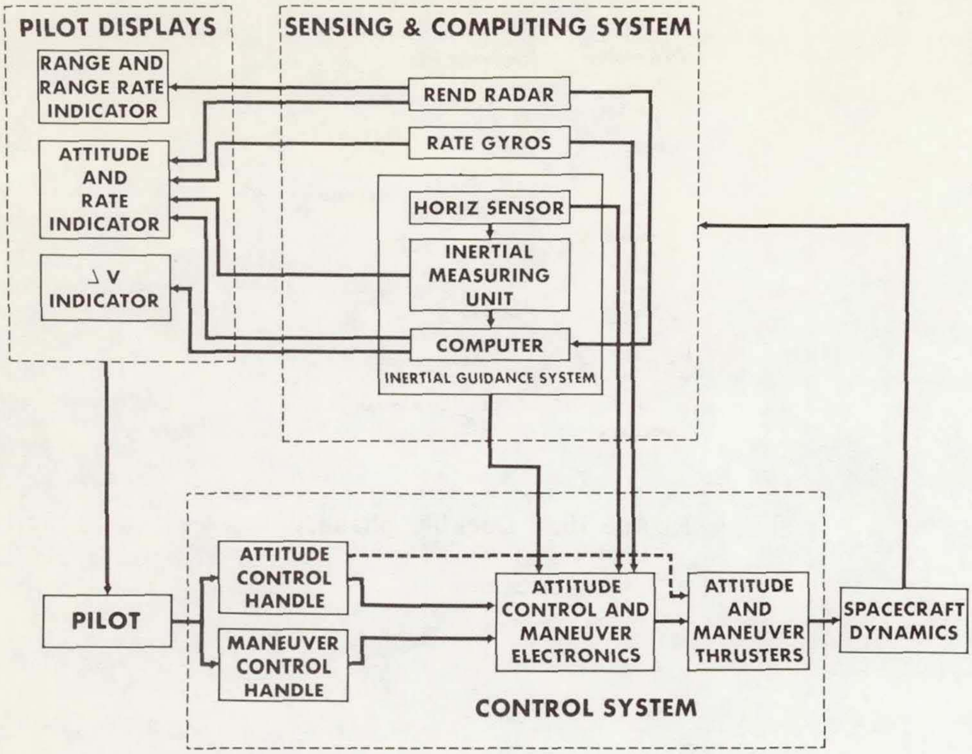


Figure 14. - Gemini guidance and control system.

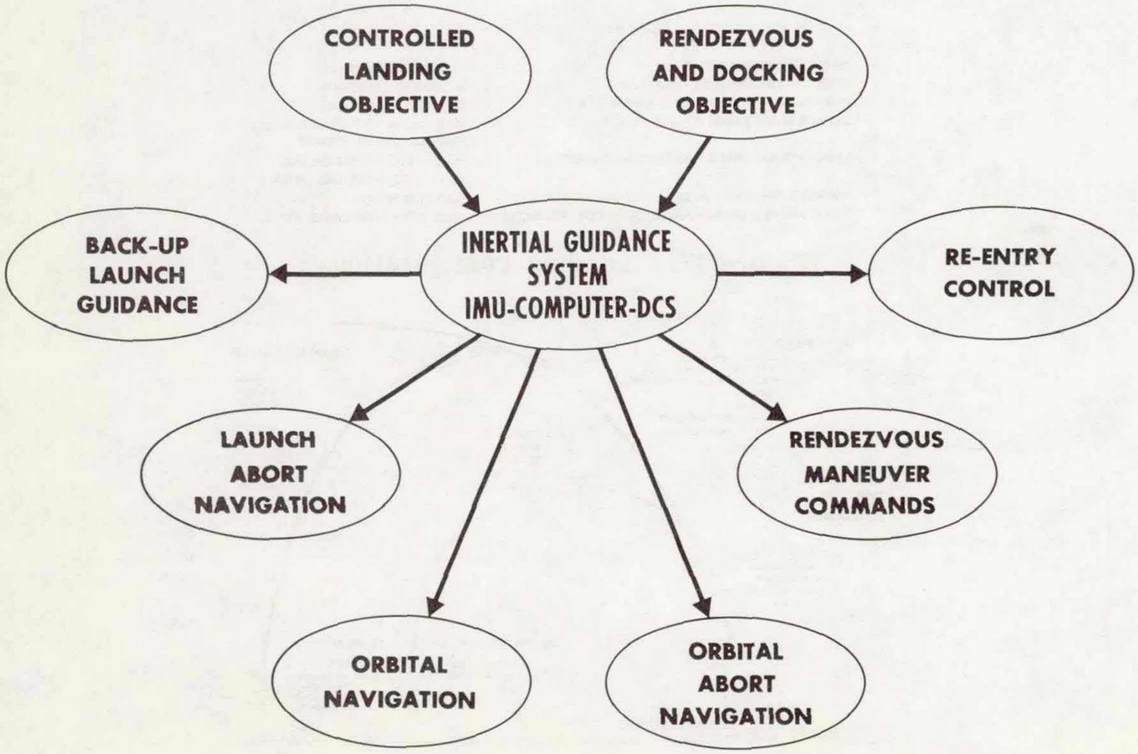


Figure 15. - Inertial guidance system mission application.

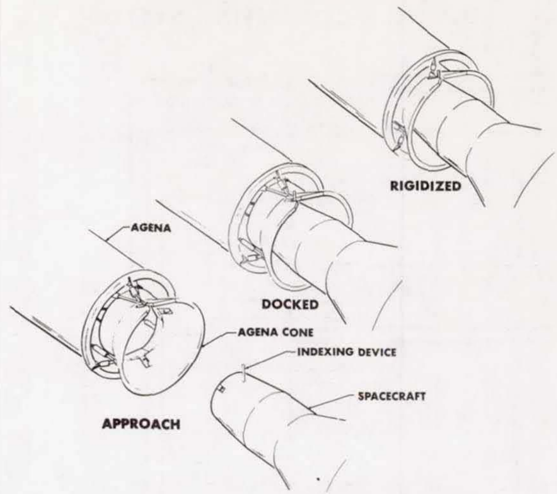
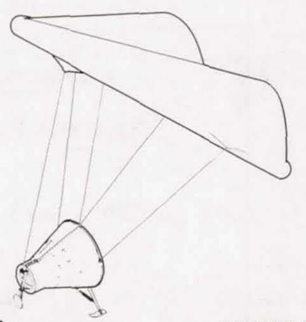


Figure 16. - Docking phase.



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.

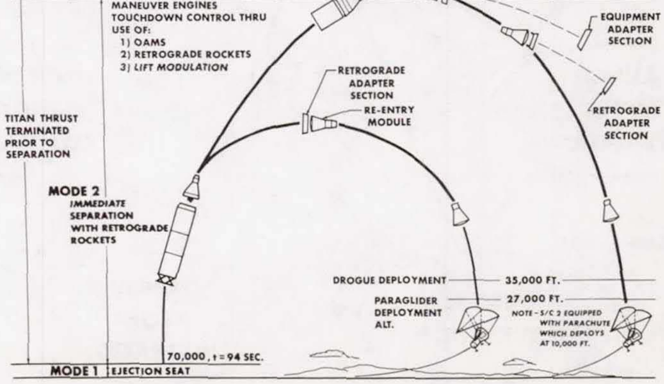


Figure 18. - Launch escape modes.