[redacted]

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

Charles L. Seacord, M.S.
Assistant Chief Engineer
Aeronautical Division
Minneapolis-Honeywell Regulator Company

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.

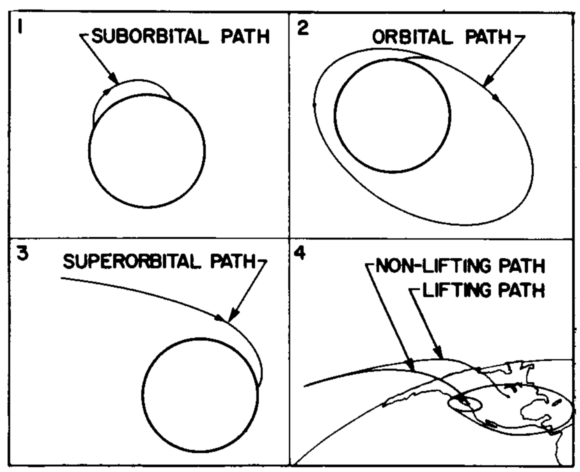


Figure 1. Types of Orbital and Re-Entry Vehicle Paths

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

[redacted]

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.

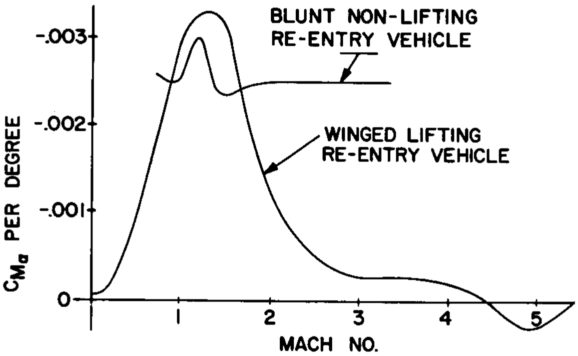


Figure 2. Comparison of Static Stability Characteristics of Nonlifting and Lifting Re-Entry Vehicles

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.

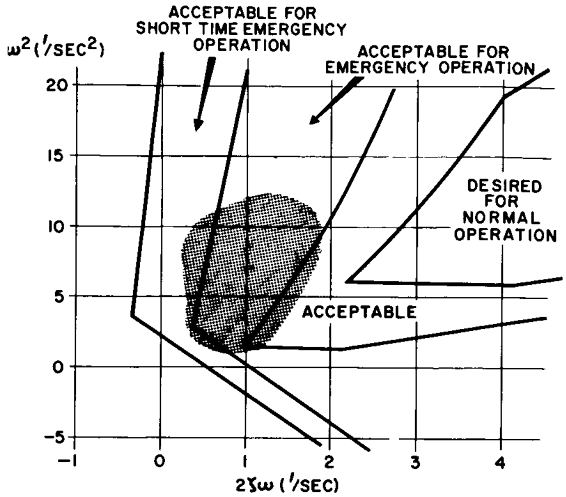


Figure 3. Handling Qualities in Pitch (ω = frequency of motion, ζ = damping factor)

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.

[redacted]

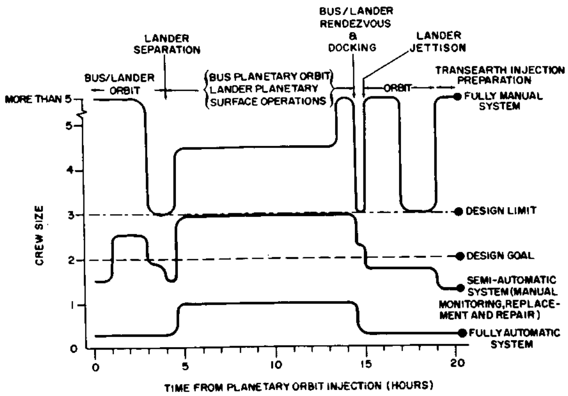


Figure 4. Effect of Crew Size on Automation Requirements

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

[redacted]

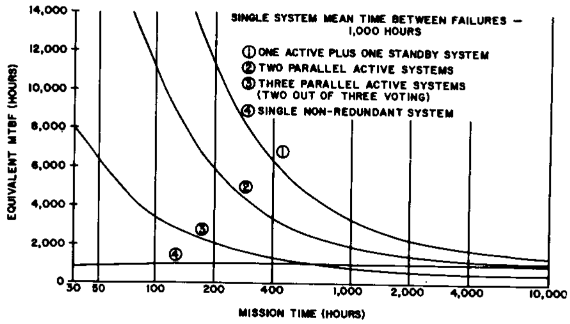


Figure 5. Effect of Redundancy on Equivalent MTBF

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 flightworthy 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,

[redacted]

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.

<u>Weight of Vehicle</u>

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.

[redacted]

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.

[redacted]

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 conductance 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.

[redacted]

[redacted]

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.

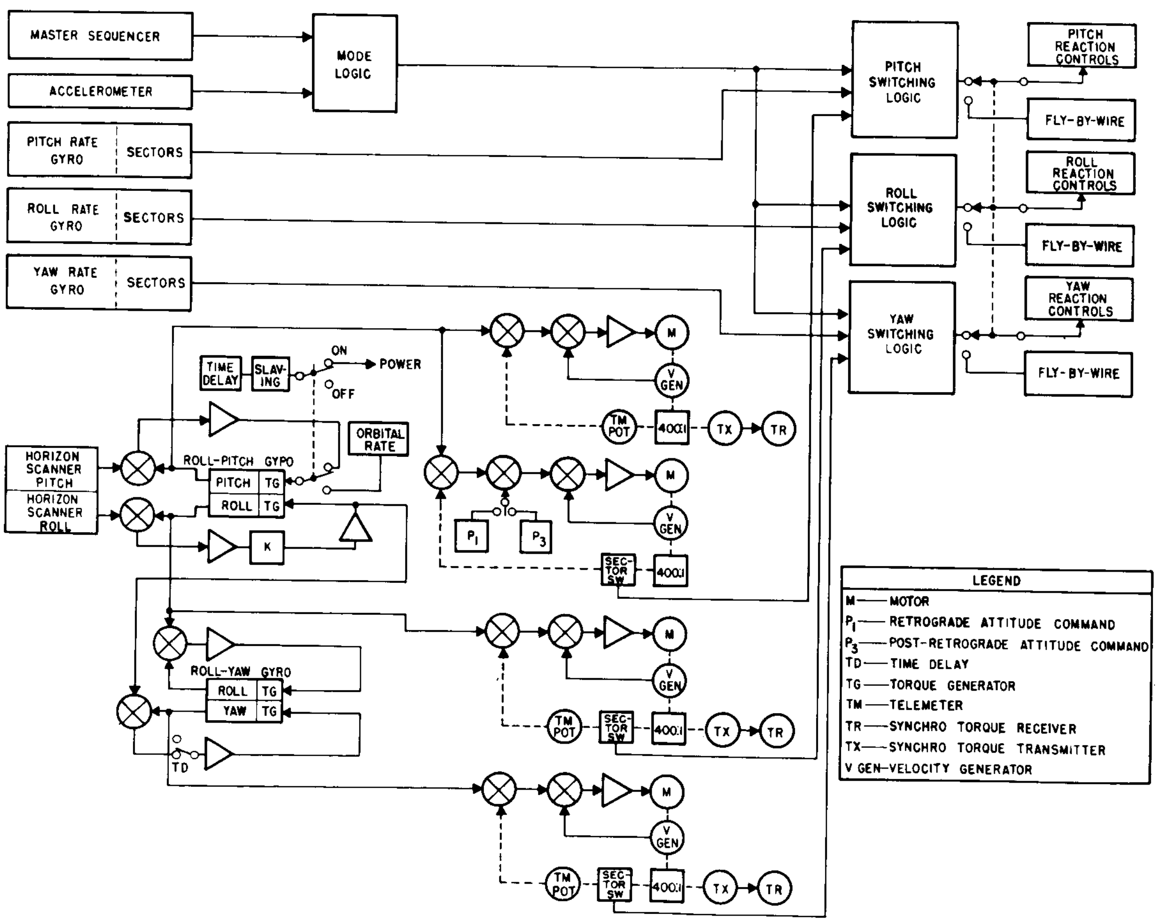


Figure 6. Mercury Automatic Stabilization and Control System

[redacted]

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 outgassing 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 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

[redacted]

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.

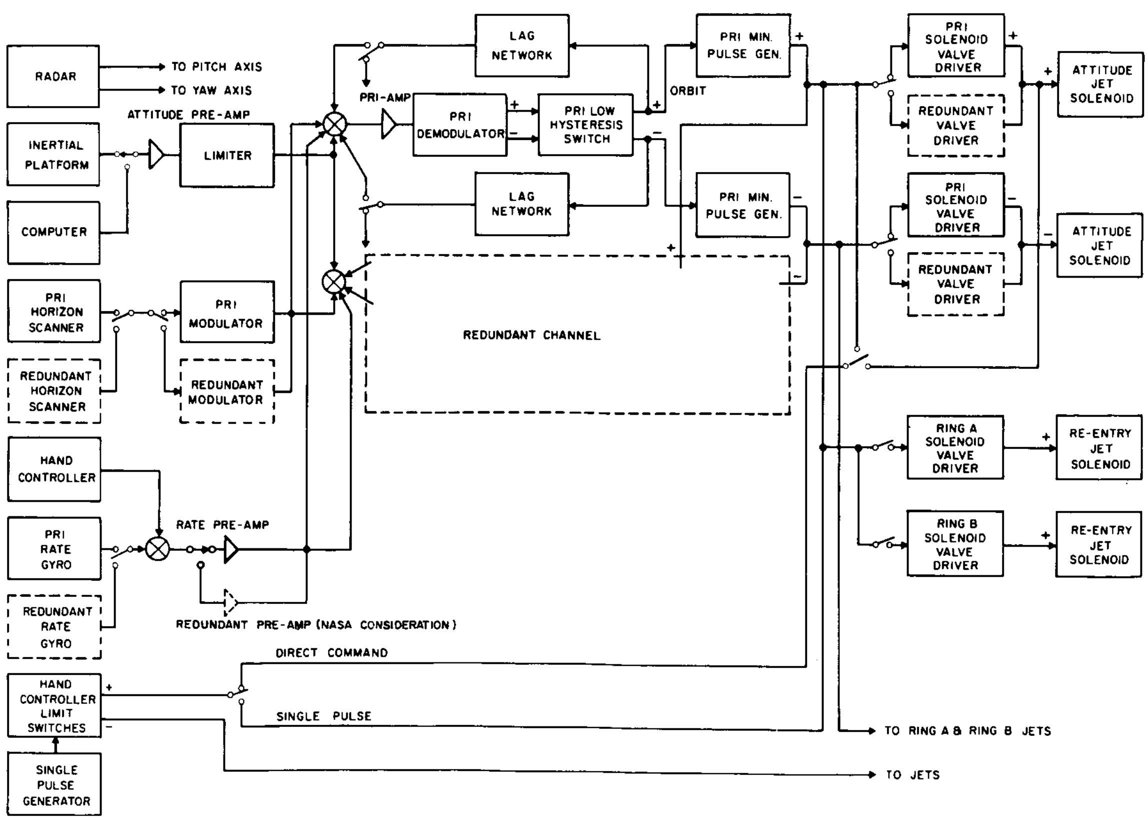


Figure 7. Gemini Roll Axis Control Diagram

[redacted]

[redacted]

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.

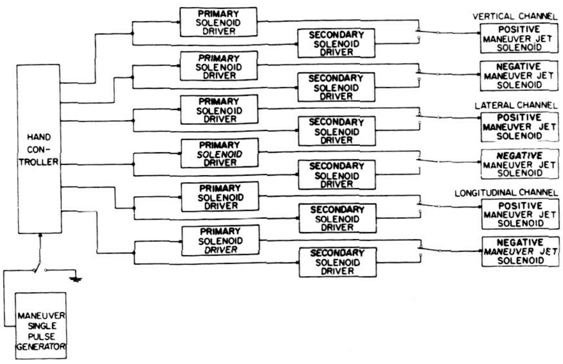


Figure 8. Gemini Maneuver On-Off Logic

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 possibly 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.

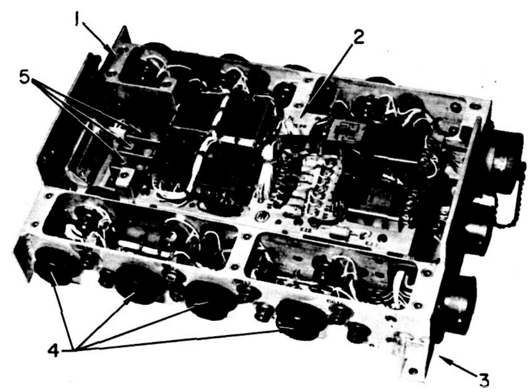
5. Solid-state switching in all signal circuits.

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 air-field chosen by the pilot within a circle of

[redacted]

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 elevon 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-

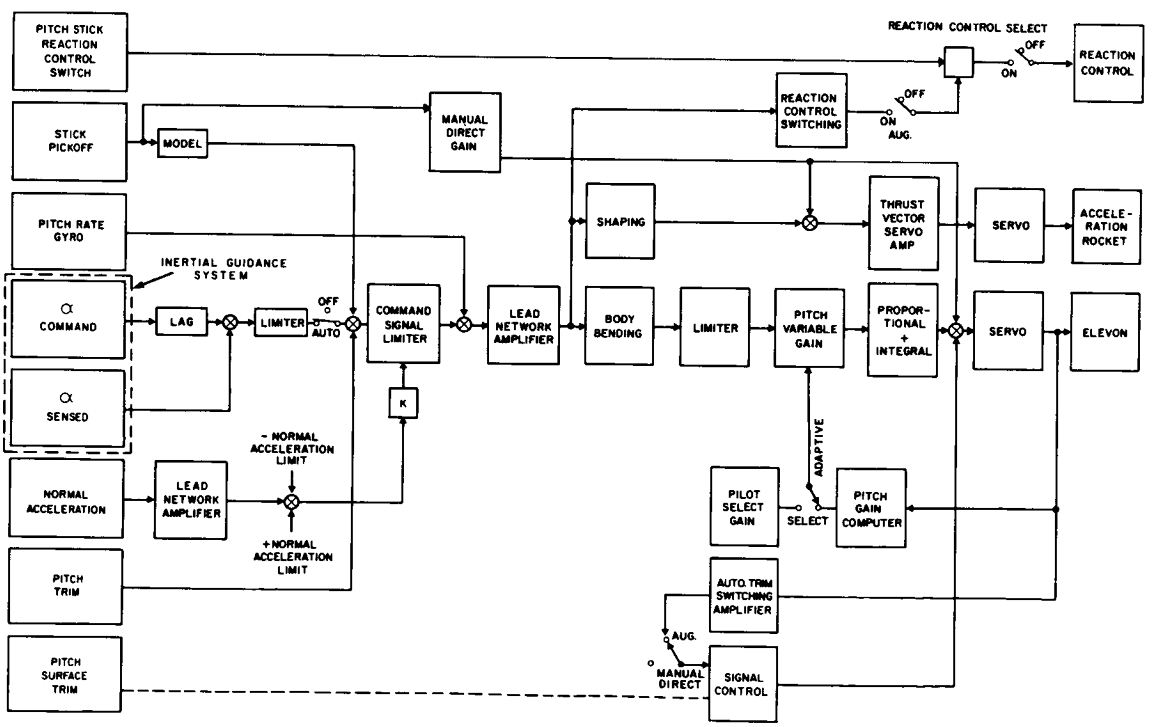


Figure 10. X-20 Pitch Axis Control Diagram

[redacted]

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

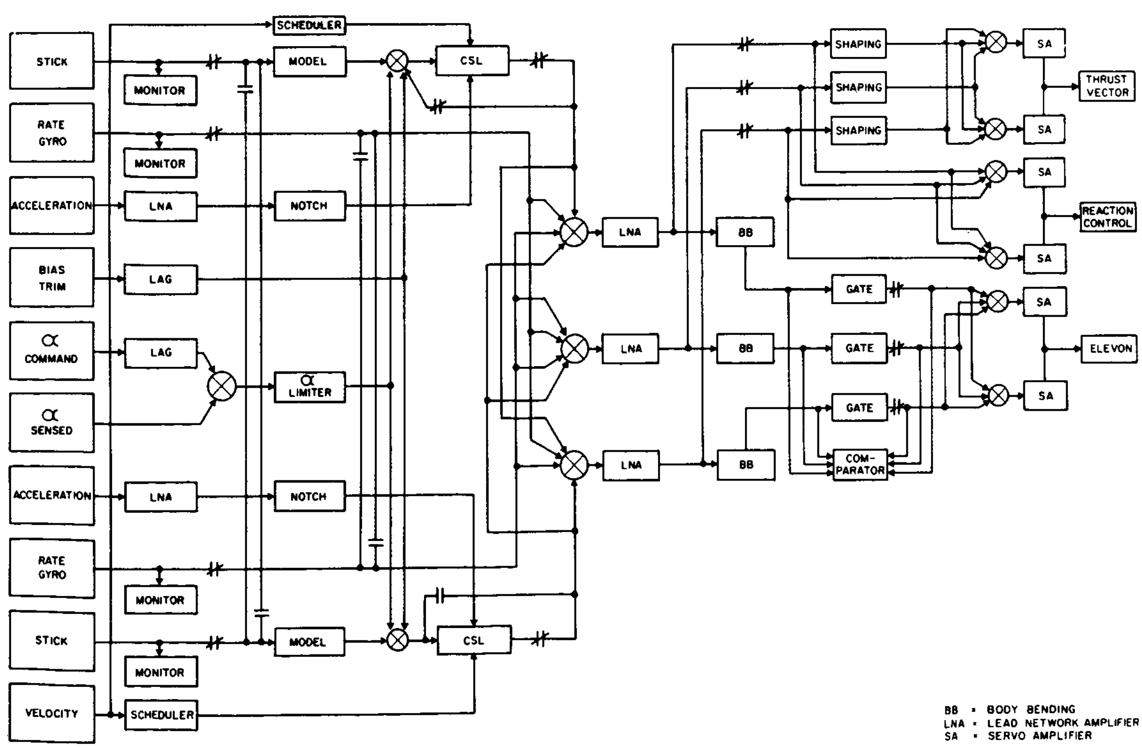


Figure 11. X-20 Pitch Axis Redundancy Mechanization

[redacted]

[redacted]

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.

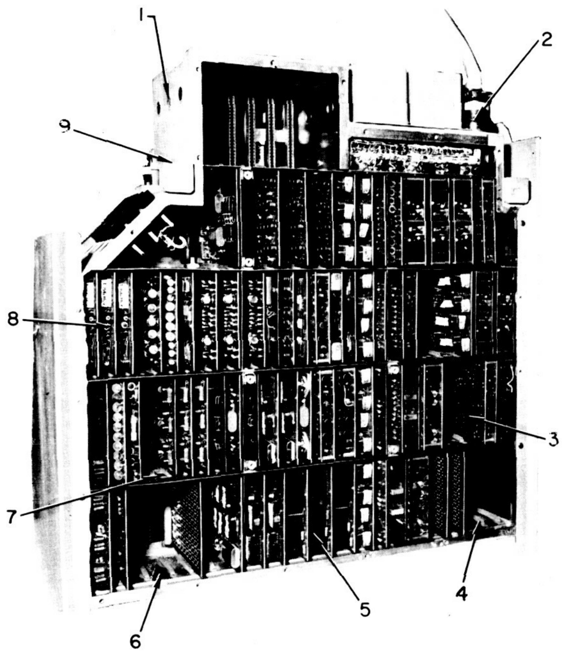
The adaptive system uses transistor differential d-c amplifiers as the basic electronic building blocks in summing amplifiers, active filters, and various other functions. These d-c amplifiers are about one-half the size and weight of a comparable magnetic amplifier and have better gain, band-width, 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

[redacted]

[redacted]

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.

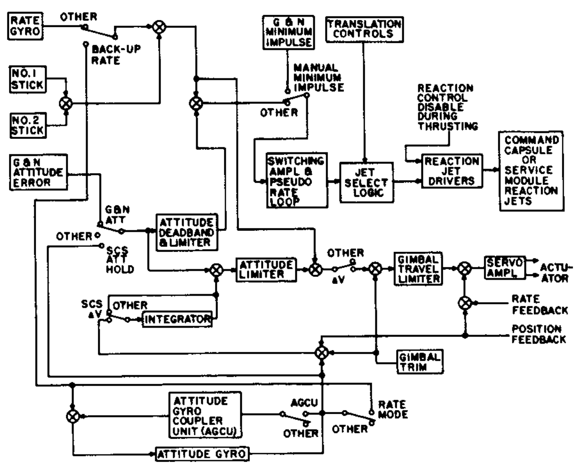


Figure 13. Apollo Command Module SCS Pitch Axis Control

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

[redacted]

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 vernier 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.

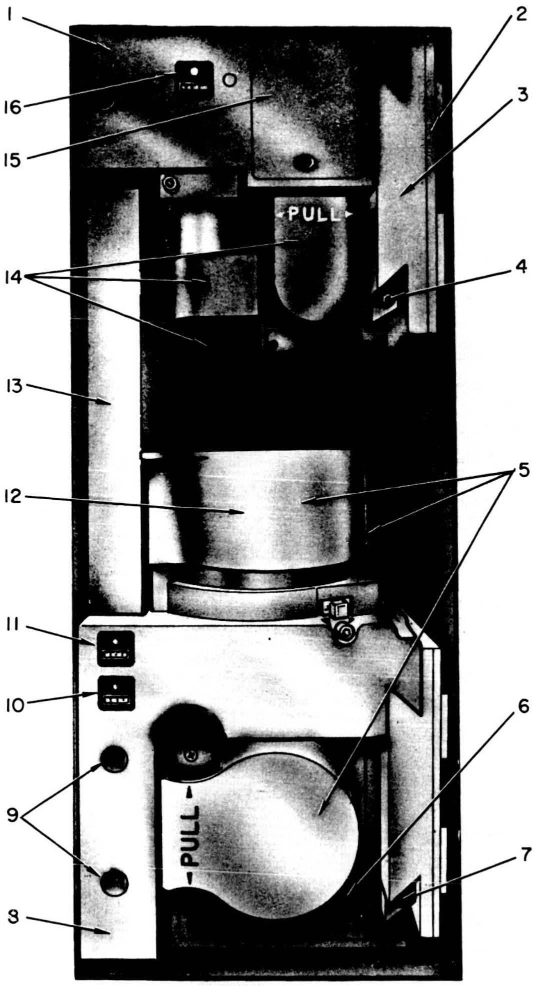
Figure 14 shows the present approach to inflight maintenance, as evidenced by the SCS rate and attitude gyros and accelerometer in the Command Module. The rate gyro package contains three orthogonally mounted rate gyros. Each gyro has a shroud containing an integral circuit connector. A quick-disconnect 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.

[redacted]



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.

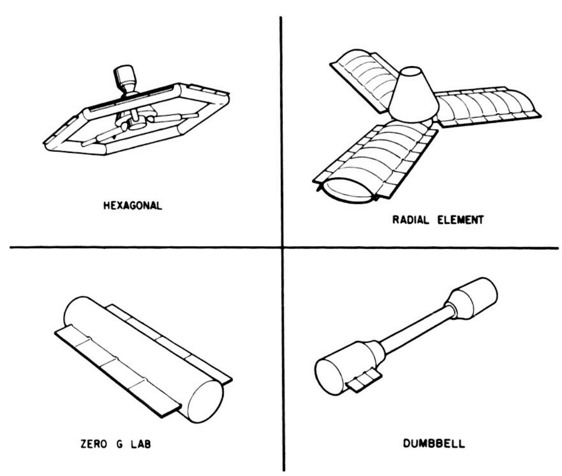


Figure 15. Manned Orbiting Space Station Configurations

[redacted]

[redacted]

<u>Control System Restraints</u>.- 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.

<u>Performance Requirements</u>.- 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):

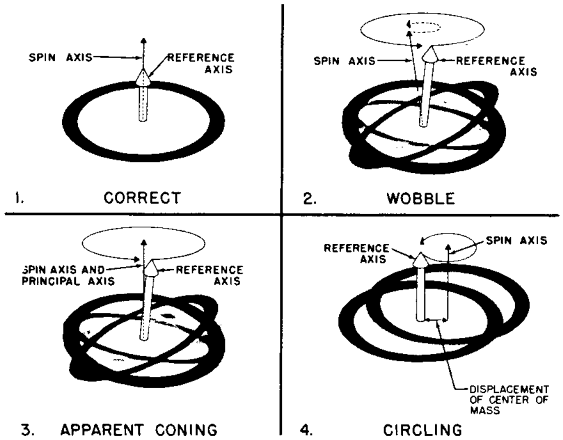


Figure 16. Space Station Modes of Motion

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

[redacted]

[redacted]

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.

<u>Qualification Testing</u>.- 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.

References

1. Yoler, Yusuf A.: "Dyna-Soar: A Review of the Technology," <u>Aerospace Engineering</u>, August 1961. Figure 3 is based on Figure 18, page 63, of that article.

2. Lindquist, O. H.: "The Design of Man-Machine Systems by Means of Quantitative Analysis Techiques of Human Factors Engineering," presented at the 7th Region IRE Conference and IRE-ISA Joint Technical Exhibit, May 24-26, 1960, Seattle, Washington.

3. Lindquist, O.H.: "Emphasis on Human Participation and Control," <u>Aerospace Engineering</u>, January 1962, pp. 48-90.

4. Mellen, D.L.: "The Development and Flight Test of an Adaptive Flight Control System for the X-15 Vehicle," Honeywell publication, February 21, 1963.

[redacted]

NASA-Langley, 1964 S-40