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SPACE SHUTTLE DIGITAL FLIGHT CONTROL SYSTEM

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SUMMARY

The Space Shuttle digital, fly-by-wire, flight control system (FCS) presents an interesting challenge in avionics system design. In residence in each of four redundant general purpose computers (GPC's) at lift-off are the guidance, navigation, and control algorithms for the entire flight. (A fifth GPC houses a backup FCS.) The mission is divided into several flight segments: first-stage ascent, second-stage ascent; abort to launch site, abort once around; on-orbit operations, entry, terminal area energy management (TAEM); and approach and landing. The FCS is complicated in that it must perform the functions to fly the Shuttle as a boost vehicle, as a spacecraft, as a reentry vehicle, and as a conventional aircraft. The crew is provided with both manual and automatic modes of operations in all flight phases including touchdown and rollout.

INTRODUCTION

The Shuttle vehicle configuration is shown in Figure 1. It consists of the orbiter vehicle, the orbiter external tank (ET), and two solid rocket boosters (SRB's). During Shuttle ascent, control authority is provided by thrust vector control (TVC) of the three orbiter main engines and each of the two SRB's. Orbit insertion and on-orbit control are accomplished by combinations of 46 reaction control jets plus two gimballed orbit maneuvering engines (orbit maneuvering system or OMS). A blend of reaction control system (RCS) jets and the aerosurfaces is used during entry; all aerosurface control is used during TAEM and approach and landing. The aerosurfaces (Figure 2) include the elevons, used in unison for pitch control and differentially for roll control; rudder panels, used in unison for rudder control and differentially as a speed brake; and a body flap. Primarily the body flap protects the main engines from entry heating. However, it also supplements the elevons for pitch trim.

The orbiter is a first step in design of a control-configured vehicle. It is statically unstable in both pitch and yaw over a large percentage of the flight envelope (up to 2 and 1/2 percent of the body length in pitch). This design philosophy has permitted extensive weight (and hence cost) savings because it has allowed wing, tail, and aerosurface sizes to be minimized.

DESCRIPTION OF THE SPACE SHUTTLE FLIGHT CONTROL PROBLEM

During Shuttle mated ascent, the FCS consists of a three-axis attitude command system (Figures 3 through 5). Five seconds after lift-off, commands are issued to accomplish the pitch-over and roll-to-flight-azimuth maneuvers. During regions of high dynamic pressure, a load relief system in both pitch and yaw minimizes air loads on the vehicle. The system is optimized with respect to weight savings (due to load reductions) versus weight penalties (due to added propellant caused by flight path dispersions arising from the use of the load relief system). The load relief function is accomplished by lateral and normal accelerometer feedbacks blended into the attitude command system starting at 25 seconds into the flight. After the region of high dynamic pressure passes, the load relief function is blended out (95 seconds). The guidance system commands an open loop pitch program versus time. The trajectory is shaped to minimize gimbal angle requirements and to balance the weight penalties associated with positive and negative air loads due to winds and gusts.

During SRB tail-off, which is sensed as an acceleration decay, the system is commanded to fly a pitch program versus time for proper SRB separation conditions. At staging, the control system is switched to the second-stage mechanization, which is a standard three-axis attitude command system (Figure 6). At a given time, which corresponds to a predicted dynamic pressure of 25 psf, the guidance loop is closed, and a form of linear tangent steering is used to guide the vehicle to the orbit insertion point. In addition to the automatic modes described, an augmented manual capability is provided in both first and second stages of flight.

The abort modes are not discussed in this paper.

In the on-orbit flight phase the crew is provided with the 13 manual and automatic control modes listed in Table 1. Two gimballed, 6000-pound-thrust (OMS) engines are used for large delta-V maneuvers. Various combinations of forty 900-pound-thrust reaction jets are used for attitude control and small delta-V translation maneuvers. In addition, six 25-pound-thrust RCS jets are provided for high accuracy vehicle pointing. The jet select logic provides attitude and translational capability with less than 7 and 1/2 percent cross coupling into adjacent axes. The number of jets is predicated upon the requirement for a fail operational, fail safe capability throughout a mission.

The entry flight control system (Figures 7 through 9) is a blend of RCS and aerosurface control effectors. During the early portions of entry an all-RCS control system is used (Figure 10). When a dynamic pressure of 2 psf is reached (sensed from vehicle accelerations), the elevons are activated to provide a pitch and roll trim supplement to the RCS system. When a dynamic pressure of 10 psf is reached, the elevons provide sufficient authority for roll control, and the roll jets are inhibited. When a dynamic

Table 1. On-Orbit Control Modes

Mode
Manual direct rotation acceleration command
Manual direct translation acceleration command
Manual direct translation pulse command
Manual direct rotation pulse command
Three-axis manual proportional rate command augmentation
Manual RCS rotation discrete rate command augmentation
Three-axis attitude hold mode
Three-axis automatic attitude command
Automatic RCS local-vertical barbecue attitude command
Three-axis automatic inertial barbecue command
Two-axis automatic RCS translation command
Automatic OMS thrust-vector control
Manual OMS thrust vector control command augmentation

pressure of 20 psf is reached, the pitch jets are inhibited, the yaw jets being retained for yaw stabilization and control. During the majority of entry the vehicle is statically unstable in yaw. However, the stick-fixed dutch roll mode is dynamically stable. The control system takes advantage of this stability in that the vehicle is permitted to oscillate within course dead bands in roll and yaw, thus avoiding an excess usage of RCS for yaw stabilization.

The heating rates and total heating load to the vehicle are minimized by flying the high-speed portion of entry (down to 8000 feet per second) at high angles of attack (approximately 30 degrees). At Mach 8, an angle-of-attack transition is initiated, ending at an angle of attack of approximately 10 degrees (roughly the maximum lift-to-drag condition) and a velocity of approximately 1500 feet per second. During this transition, the vertical tail and rudder become effective. At an angle of attack of 18 degrees (Mach 5) the rudder control is activated. By the time the vehicle reaches an angle of attack of 10 degrees the rudder is fully effective, and the yaw jets are thereafter inhibited. The FCS is switched to conventional aircraft control mode for the TAEM phase of flight. Manual and automatic modes during entry are similar, the only difference being the substitution of a guidance steering command in the auto system instead of the rotation hand controller output in the manual system.

The TAEM flight phase is initiated at a velocity of about 1500 feet per second during entry with a corresponding altitude of approximately 70,000 feet. This flight phase extends to the approach and landing interface at approximately 10,000 feet. During this period, the guidance system issues commands to control the dynamic pressure and energy state of the vehicle and to provide steering commands to arrive at the approach and landing interface in alignment with the runway (Figure 11). Three basic control modes are

provided to the crew: manual direct (MD), control stick steering (CSS), and automatic. The manual direct mode (Figures 12 through 14) is strictly a backup in which no augmentation is used (i.e., all feedback loops are open).

When the crew selects control stick steering, the basic mode of operation becomes a command augmentation system (CAS). It is implemented as an N_z command (normal load factor) mechanization in pitch (yaw is similar), and roll rate is commanded into the roll channel (Figures 15 through 19). Two submodes to CSS are available. One is attitude hold in pitch and/or roll (Figures 20 and 21). When the stick is out of detent, the CAS mode is operational; when the stick is returned to detent, the attitude function is initiated at the attitude existing at the time the stick was returned to detent. A second submode to CSS is an indicated air speed (IAS) hold (Figure 22). In this mode, the speed brakes are commanded to maintain the air speed commanded by the crew. When the IAS is not selected, speed brake control is a manual function.

In the auto-TAEM mode, N_z commands are issued from the guidance system to the pitch and yaw channel, and roll commands are issued to the roll channel. This is shown in Figure 23 for the pitch axis.

After the vehicle exits blackout during entry, a TACAN (tactical air navigation) navigation aid is acquired by the communication system for navigation update. The guidance system steers the vehicle to intercept a heading alignment circle to bring the vehicle to the approach and landing interface. As the vehicle rounds the heading alignment circle (Figure 24), its orientation becomes such that the antennas capture a microwave scan beam landing system (MSBLS) navigation aid. This will occur at an altitude of roughly 14,000 feet. When lock-on is verified, the flight phase switches from TAEM to the approach and landing. The same three basic modes are available to the crew as discussed for TAEM (i.e., manual direct, CSS, and auto). The manual direct and CSS modes are as described for the TAEM phase. In the autoland mode (Figure 18), the guidance system issues attitude commands to the vehicle to fly down a steep glide slope, which varies from 21 to 24 degrees depending upon the payload weight (Figure 25). The speed brakes are modulated to hold an air speed of 290 knots. At an altitude of 1800 to 2000 feet, depending on payload weight, a preflare maneuver is commanded to bring the vehicle exponentially to a 3-degree glide slope. A final flare is commanded at approximately 200 feet altitude, and the vehicle nominally lands with a sink rate of about 2 and 1/2 feet per second and about 4000 feet down the runway. When main gear touchdown is detected (by a squat switch), the normal and lateral acceleration feedbacks (in CSS) and integrator loops (in auto) are opened, and a pitchdown command is issued. Roll commands are driven to zero. Lateral steering is initially accomplished with the rudder. After the nose gear slapdown has been verified, and after the velocity is reduced to approximately 110 knots, nose wheel steering is engaged. The rest of the rollout is accomplished with the rudder and nose wheel steering. The autoland function is totally automatic with the exception of gear extension ($h = 500$ ft) and runway braking, which are done manually.

DESCRIPTION OF THE SPACE SHUTTLE FLIGHT CONTROL SYSTEM MECHANIZATION

The flight control problem just described is essentially controlling a large number of quite different flight phases, some of which include unstable vehicle dynamics. Thus, flight control is a flight safety-critical function that must have great flexibility. The concept chosen for flight control is an all-digital, fly-by-wire implementation that uses several general-purpose computers connected by serial digital data buses to remotely located multiplexer/demultiplexer units (MDM's). The MDM's in turn, are connected to the flight control sensors, effectors, and controls. The guidance and navigation problems are solved by this same mechanization (with the appropriate additional sensors). It is used for all flight phases and elements, including control of the SRB's during ascent. The block diagram of this configuration is shown in Figure 26.

Efficiency of presentation requires that the computer complex be described first, including the MDM's and data buses. Then the operating configuration of the flight control equipment will be described.

Figure 27 illustrates the internal configuration of the computer and associated elements of the central digital elements (collectively denoted as the digital processing subsystem or DPS). At the core of the DPS are five general purpose computers. Each GPC is a modified IBM AP-101 central processor unit and core memory with a special input/output processor (IOP) that interfaces with 27 serial digital data buses. The memory contains 64,000 32-bit words with a nominal one-microsecond cycle time. The IOP contains a master sequencer and 27 data bus control elements. Under overall control of the master sequencer, each data bus control element has the capability to send and receive data over its particular data bus. In addition to data, the transmittals include commands to other equipment connected to the bus. In addition to the ability to request and subsequently receive data on the bus, each data bus control element can monitor data on the bus resulting from other data bus control elements (associated with other GPC's). This monitoring capability is fundamental to the processing of flight control sensor data, as will be described.

Of all the data buses, those central to this discussion are the eight dedicated to guidance, navigation, and flight control and the five inter-computer data buses. Each of these buses is connected to all of the GPC's. Also, those eight buses dedicated to the guidance, navigation, and flight control functions are connected to four MDM's located in the forward end of the vehicle, another four located in the aft end of the vehicle, and various devices to interface with controls, displays, event controllers, and the main engines. Each flight control sensor and effector is connected to one of the aforementioned eight MDM's, and communication between all flight control elements is via these buses.

Data transmittal over this bus network is by time division multiplex techniques at a one-megabit data rate; each word is 28 bits with the first

three bits used for synchronization and distinguishing between command and data formats. The next five bits identify the address of the word destination or source, as appropriate. The rest of the word is devoted to command or data information, except for the last bit, which is a parity bit. Each bus operates in a half-duplex mode.

The function of each MDM is to interface between the serial data streams on the bus and the several elements connected to it. The interfaces between the MDM and the several elements may be analog, digital, or discrete and may generally be in either direction. Several hundred elements can typically be connected to the MDM, the exact number being dependent upon the specific mix of analog, digital, and discrete interfaces.

Data for use in a GPC are obtained by a request (under GPC software control) being issued through the IOP, over a data bus, to a specific MDM (or other interfacing element), and then to the particular device. The reply (usually data) follows the reverse path. Because of the monitoring capability of each data bus control element, each GPC can receive the data, even though only one requested it. This feature is used to advantage in the flight control system, as will be described. The intercomputer buses are used for exchange of data between GPC's.

The five computers are synchronized only at each minor cycle (40 milliseconds) and are then only synchronized close enough to ensure "sequence synchronization" between machines; i.e., all machines are on the same minor cycle except for a small interval near the beginning or end of a minor cycle.

With this summary description of the central digital processing subsystem, it is now possible to describe the mechanization of the flight control system. The system uses both rate gyros and lateral accelerometers as basic stabilization sensors; inertial measuring unit (IMU) gimbal data are used for certain attitude hold and attitude command modes. Three sets of rate gyros are used, two on the SRB's and one set on the orbiter. Each set consists of three axes, and each axis is triply redundant. There are two sets of lateral accelerometers, one forward and one aft on the orbiter. Each set senses the two axes orthogonal to the vehicle roll axis, and each axis is triply redundant. Each redundant instrument in a set is connected to a different MDM. Controls are generally triply redundant within each set, and most controls are duplicated between the left and right seats in the cockpit.

The effectors vary for each flight phase. Of primary concern are the effectors used during the so-called "critical flight phases," those during which a hard-over failure of an axis would lead to vehicle loss before the fault could be rectified by crew actions. Most of the nonorbital portions of flight, both ascent and return, fall into this category. The flight control system must be designed to tolerate any two failures and still permit safe vehicle and crew recovery. The flight control effectors used during these flight periods have multiple input ports at the hydraulic secondary-valve level. These multiple inputs are normally used in a force-fight mode. The multiple inputs are also compared, and any deviation of

one input from the others of a specified amount for a specified time results in that input channel being "kicked out" by the actuator itself. These "pseudo-voting" actuators protect against two different input failures. Upon occurrence of a first failure, the action is the same for both the ascent and return flight phases: the bad channel is removed, and operation continues with the remaining channels. Upon occurrence of a second failure, the response varies according to the flight phase. The thrust vector control actuators used during ascent have only three input ports, and, upon occurrence of a second failure, the actuator simply centers that axis on that engine. Because there are multiple thrusters (three orbiter main engines and two SRB's), loss of thrust vector control in one axis is not flight critical. The mission can continue after the loss of thrust vector control from one axis. During the return portions of flight, things are different. There is essentially only one of each basic flight control surface, and centering of a surface would generally lead to loss of the vehicle. Consequently, the flight control effectors during these portions of flight have four input channels. After the first failure, operation continues with the remaining three. After failure of the second, failure detection and isolation of the bad input channel is easily detected by conventional comparison techniques, and operation continues with the remaining two channels.

During these critical flight phases, the flight control system is configured as shown in Figure 28. At the beginning of each flight control computational cycle (denoted as a minor cycle and equal to 40 milliseconds), the i^{th} GPC requests data from the i^{th} sensor. Because the involved data buses are connected to all GPC's, each GPC receives the data from the i^{th} sensor, either directly (in the case of the i^{th} GPC) or via the monitoring capabilities of the other GPC's. GPC's 1 through 3 are involved in data requests; GPC's receive data nominally only through the monitoring of bus traffic, but can assume the data request function of a failed GPC. The result is that all GPC's have all of the sensor data.

Each computer selects a single set of data for use in the flight control computations. Until an instrument failure is detected, the computer simply selects the middle value of the three data values for each measurement. Upon detection and isolation of an instrument failure (by combined hardware and software tests), the computer simply averages the data from the remaining two instruments. Upon detection and isolation of a second failure (again by a combination of hardware and software tests), the computer uses the data from the remaining good sensors. Since all computers have the same input data and use the same data selection algorithms, all computers use the same specific sensed values in the flight control computations.

On the downstream side of the computer, each computer is assigned to a specific input port of each flight control effector. Thus, if a computer should fail, the effect at the system level is simply a small transient because of the momentary incorrect force fight within the actuator. If such a failure occurs during ascent, GPC 4 assumes the role of the failed machine, and the failure tolerance of the system is returned to the same as it had at

launch. During entry through landing, four machines are nominally connected to the four input ports of each control effector. However, during these return phases, no GPC is brought in to replace a failed machine, since the ability to tolerate two failures exists without such replacement.

Because all GPC's are connected to all buses, and thus to all MDM's, it would be possible to operate the flight control equipment in a "master-slave" configuration; that is, one GPC would command all actuator ports until detection of its failure, at which time another GPC would assume command. This concept was rejected because of difficulty in proving that the master GPC would be made to relinquish control in all possible failure conditions.

A difficulty of operating flight control equipment in the parallel string configuration just described is possible divergence between the parallel control channels. This problem manifests itself by the commanded control signal from each computer differing from those issued by other computers by ever increasing amounts. In the Shuttle configuration, this would quickly cause various control channels to "kick out," even though no failures had occurred. The problem is caused by "noise" getting into the control channel that contains integrators. The noise can be from traditional amplitude variations (which could result from different channels using different sensors) or it could be a result of timing differences between channels. With "noise" and integrators in the control channels (almost always present for reasons of stiffening control loops or providing automatic vehicle trim capabilities), the problem will occur even when the loops are closed by a common set of vehicle dynamics.

There are two basic ways of solving this problem. One is to provide appropriate channel coupling to stabilize the divergence; the second is to suppress data and timing variations between channels. For the Space Shuttle, the latter has been chosen. As mentioned, common sensor data are selected by identical algorithms operating on identical data sets in all GPC's, and the GPC's are sequence-synchronized to prohibit divergence from timing variations.

During noncritical flight phases, mostly on orbit, the flight control is operated as an active-standby system, with one GPC in entire control and a second available for takeover should the first fail. The effectors on the OMS engines are also operated in an active-standby mode. RCS jets are operated with somewhat conventional jet-select logic.

During all flight phases, the fundamental flight control computational cycle is 40 milliseconds. A number of flight control computations take place only every 80 milliseconds, and some take place approximately once per second. In addition to the basic 40-millisecond minor cycle requirement, the total delay between sampling of the flight control sensors and transmittal of the resulting command to the effectors is constrained to be no greater than 20 milliseconds.

Software is provided for a number of flight control modes, including manual direct (MD) (totally unaugmented), control stick steering (CSS) (significantly augmented), and automatic (guidance loops closed). On-orbit operations entail many more modes. Details of the digital control laws for the multiple flight phases and multiple flight control modes within each flight phase cannot be discussed within the space limitations of this paper.

CONCLUDING REMARKS

As stated, the Space Shuttle digital fly-by-wire, flight control system is a challenge in avionics system design. It is not because of the sophistication of the algorithms being implemented; rather it is because of the number of flight phases requiring completely different control algorithms and control effectors, the large number of control modes, both manual and automatic, and because of the sophisticated and complex techniques required for management of redundant systems. It is one of the most interesting development programs ever undertaken and because of that is one of the most rewarding.

SYMBOLS

α	angle of attack
β	sideslip angle
γ	flight path angle
θ	pitch attitude
θ_c	pitch attitude command from guidance
h	altitude
K_t	time variable gain
K_y	lateral acceleration gain
K_z	normal acceleration gain
N_z	normal load factor
N_y	lateral load factor
P	roll rate
q	pitch rate

\bar{q}	dynamic pressure
$\dot{\bar{q}}$	dynamic pressure rate
r	yaw rate
R	range
t	time
Y	cross range
ϕ	roll attitude
ϕ_c	roll attitude command from guidance
ψ	heading angle
ψ_e	heading angle from computer

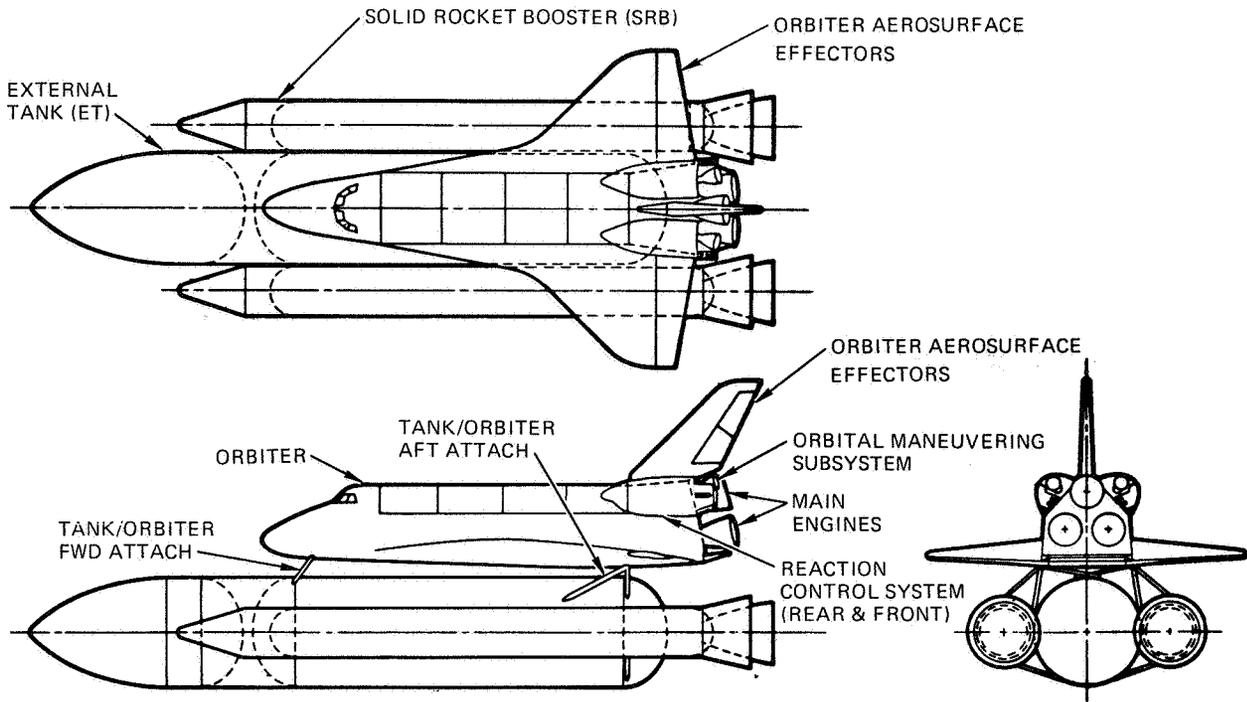


Figure 1. Shuttle Vehicle

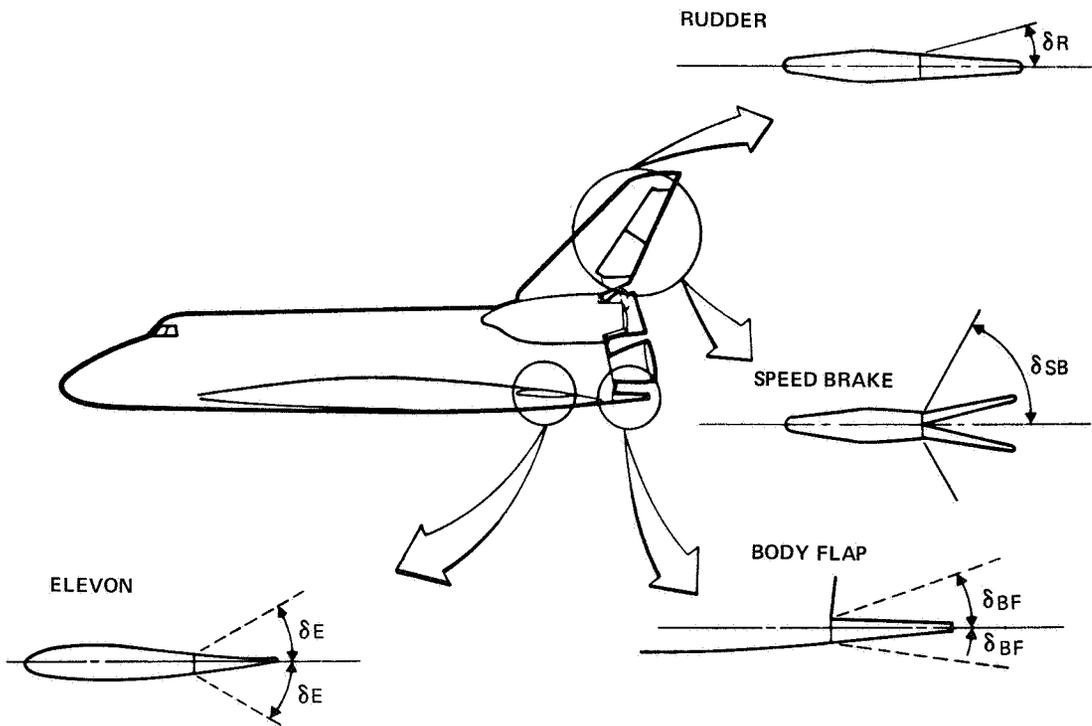


Figure 2. Aerosurface Configuration

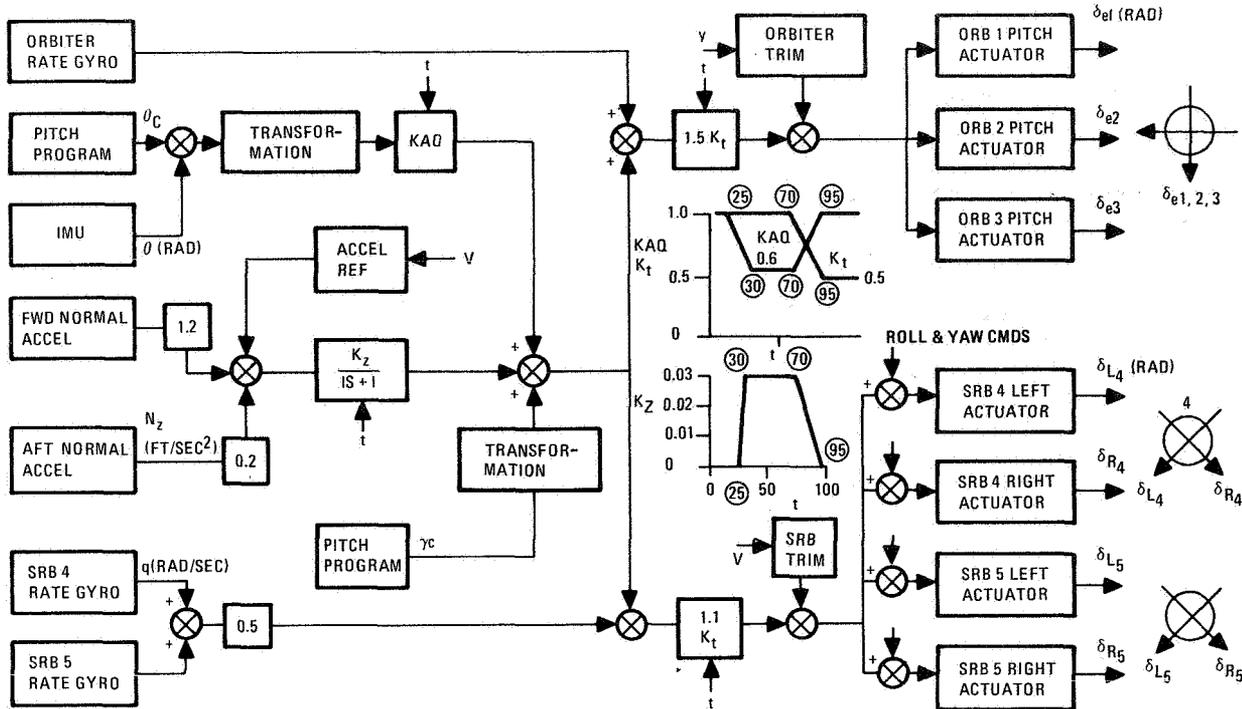


Figure 3. Mated-Ascent Pitch Axis Control

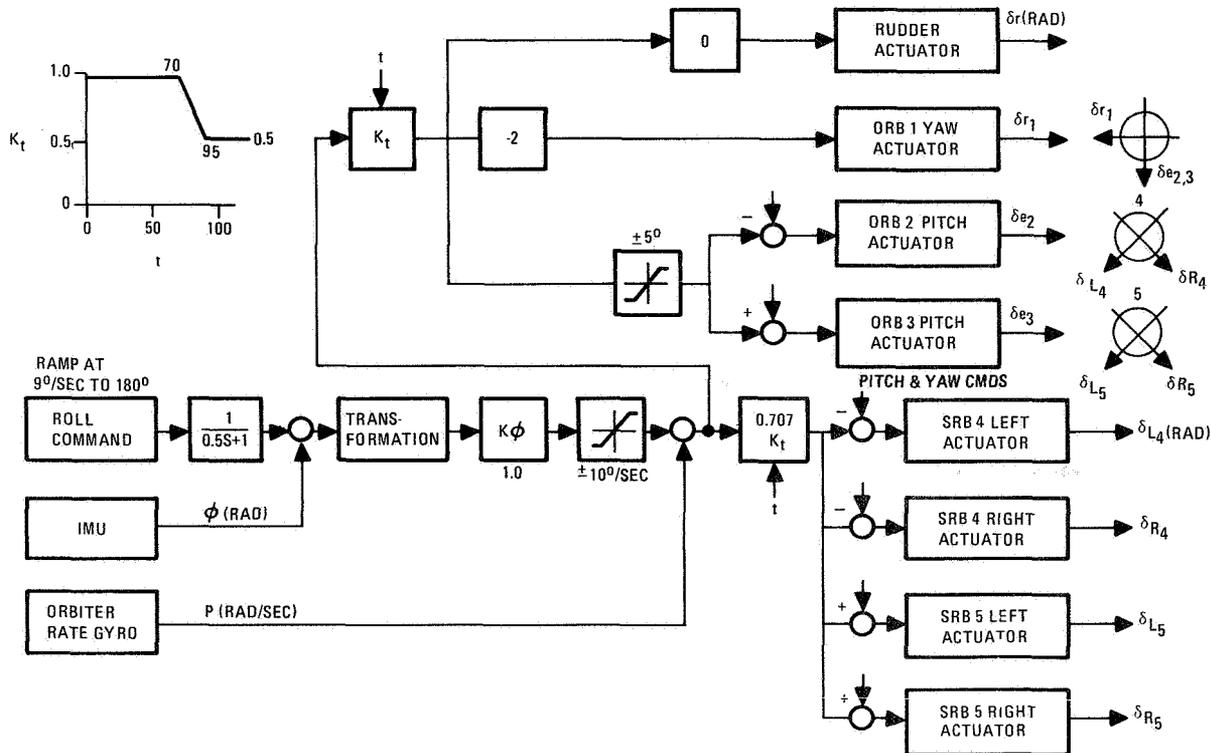


Figure 4. Mated-Ascent Roll Axis Control

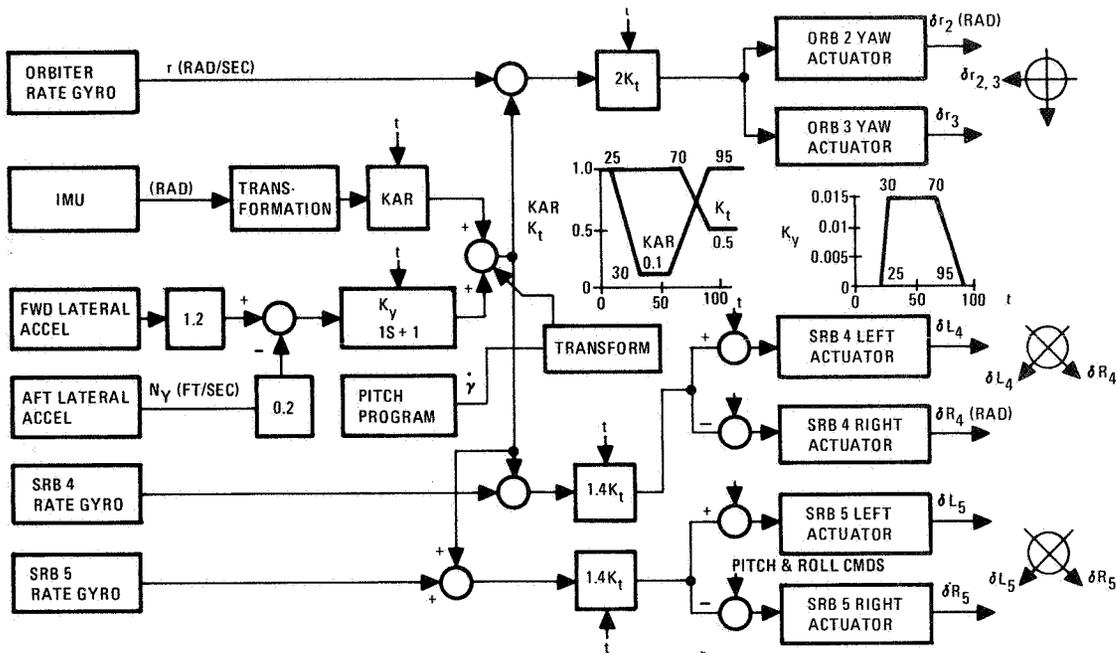


Figure 5. Mated-Ascent Yaw Axis Control

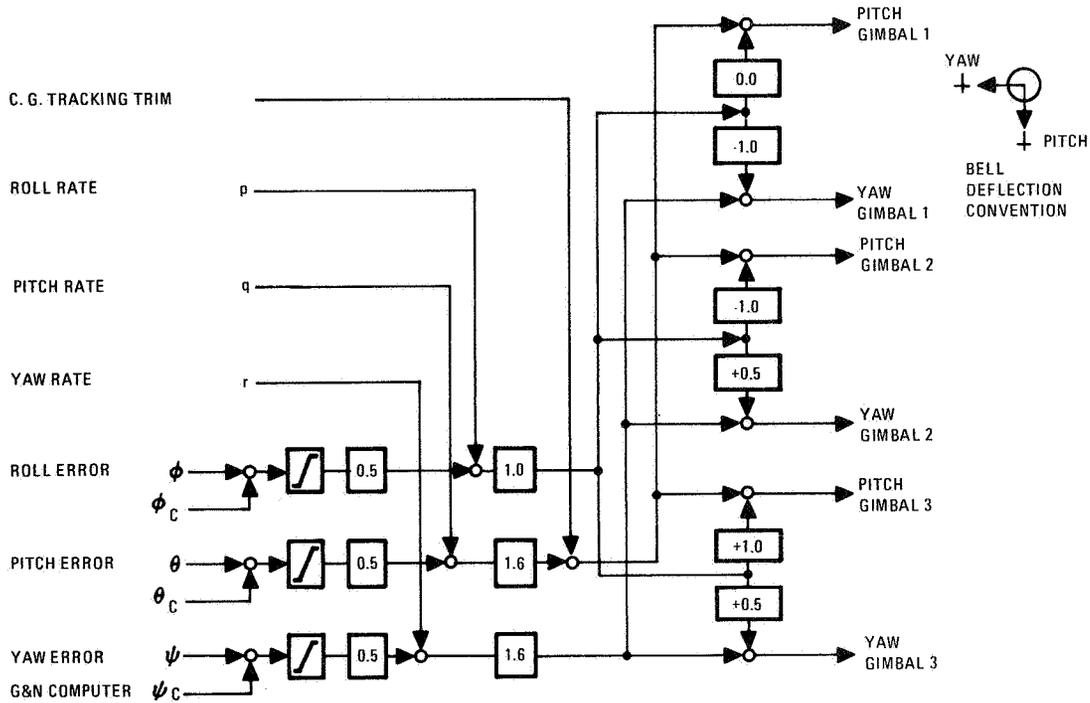


Figure 6. Second-Stage Flight Control Mechanization

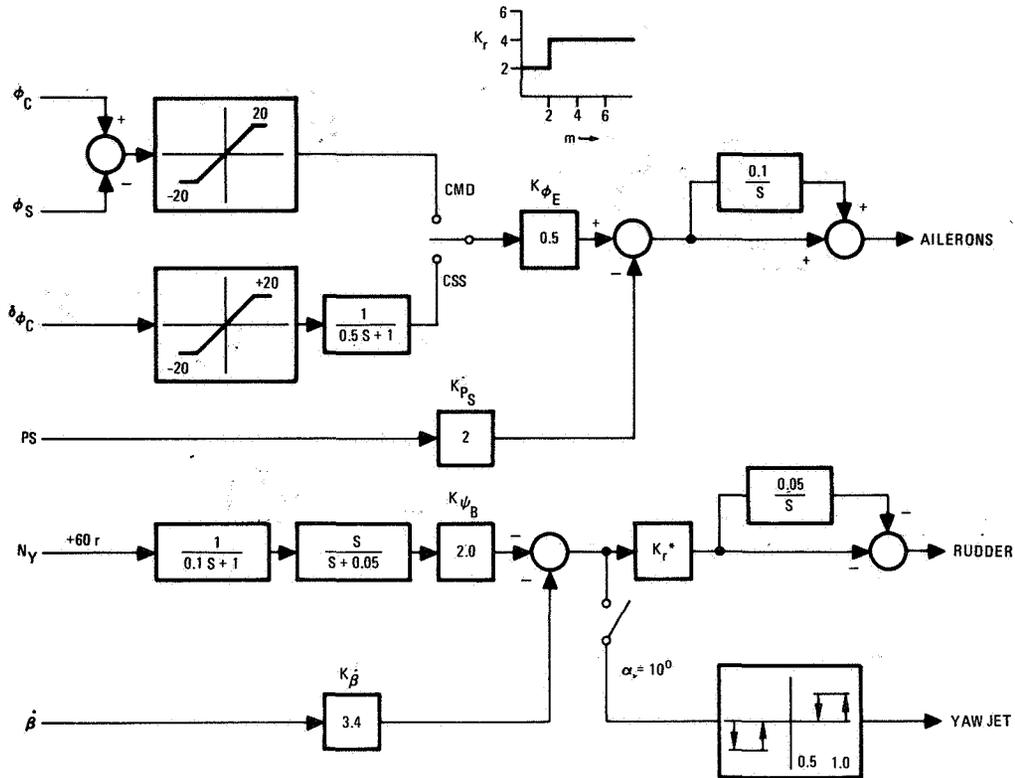


Figure 9. Final Entry Lateral Axis

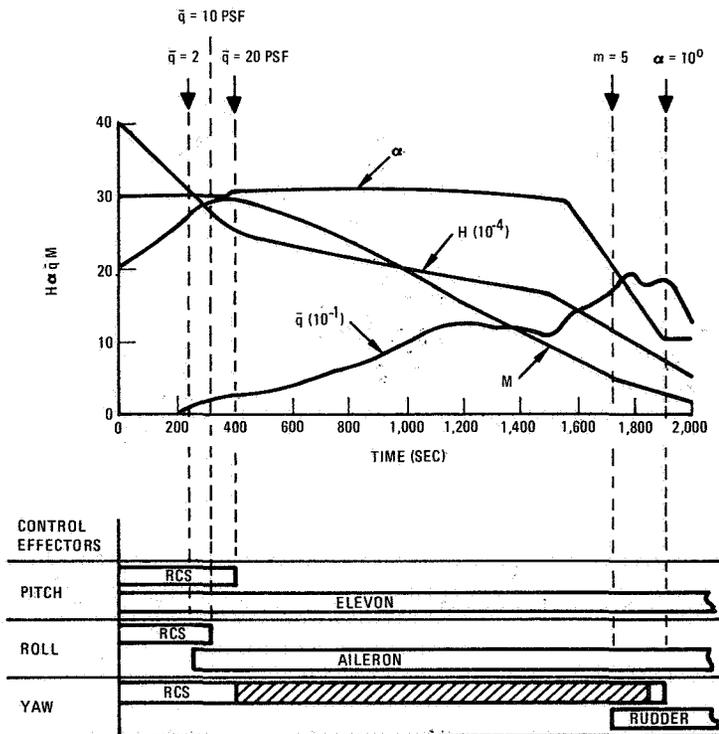


Figure 10. Entry Profile

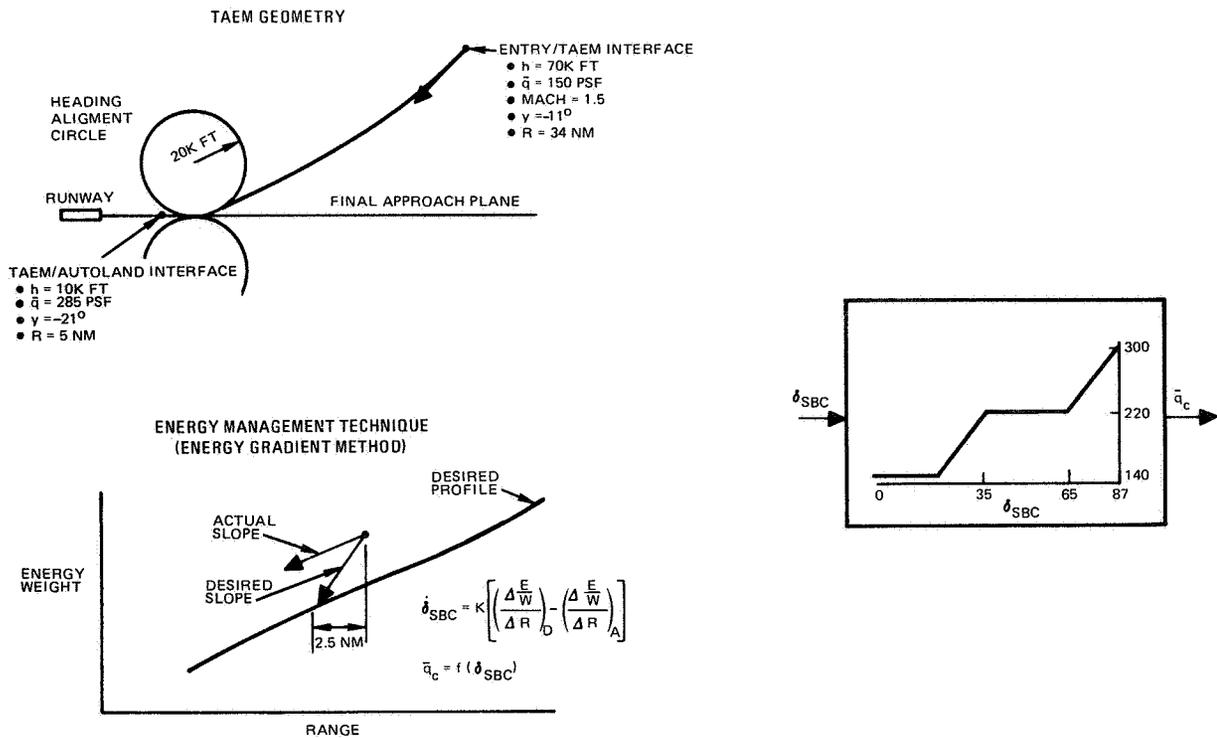


Figure 11. Terminal Area Energy Management (TAEM) Guidance System

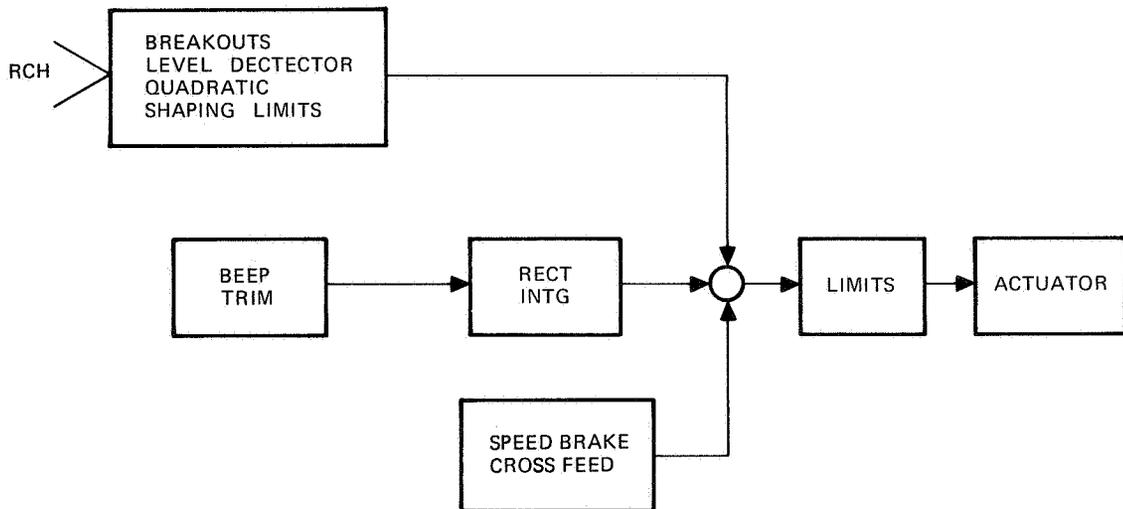


Figure 12. Pitch Axis—Manual Direct

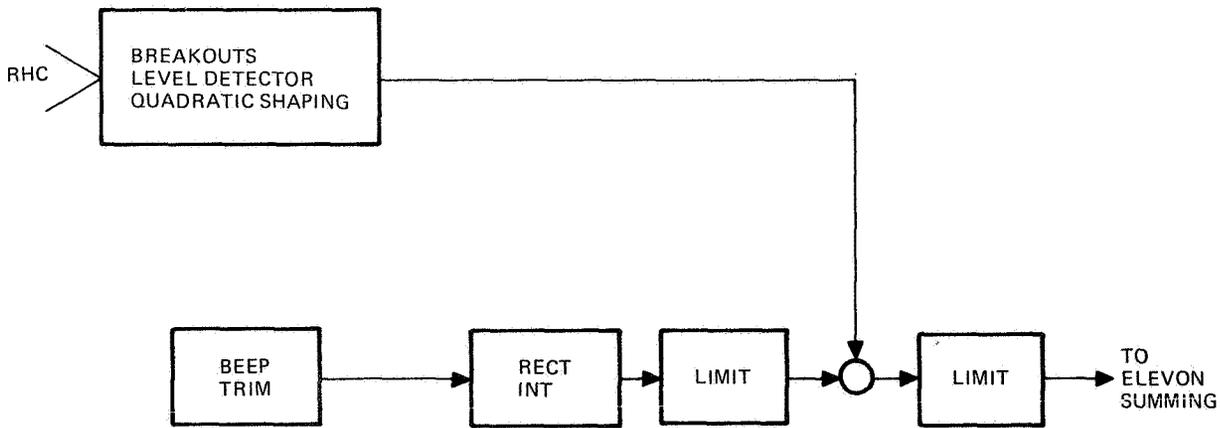


Figure 13. Roll Axis—Manual Direct

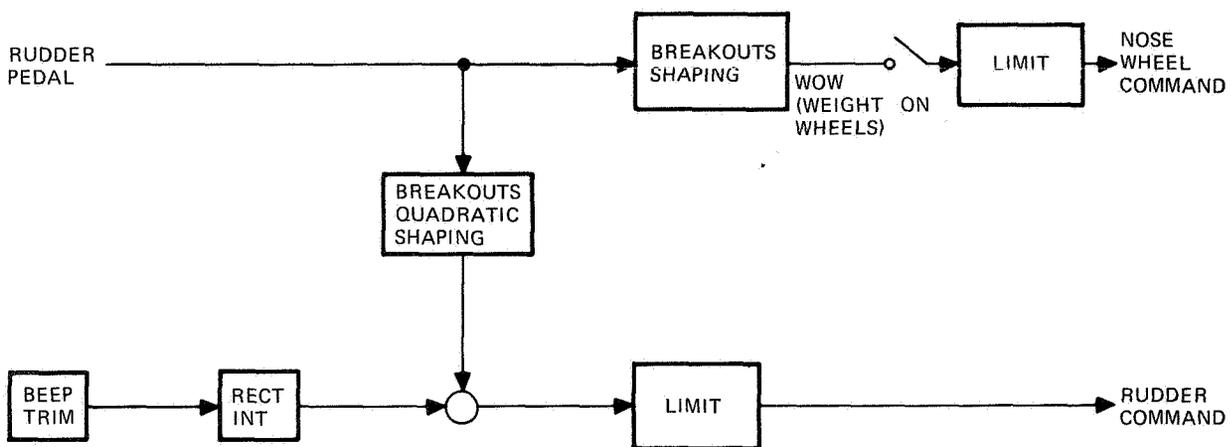


Figure 14. Yaw Axis—Manual Direct

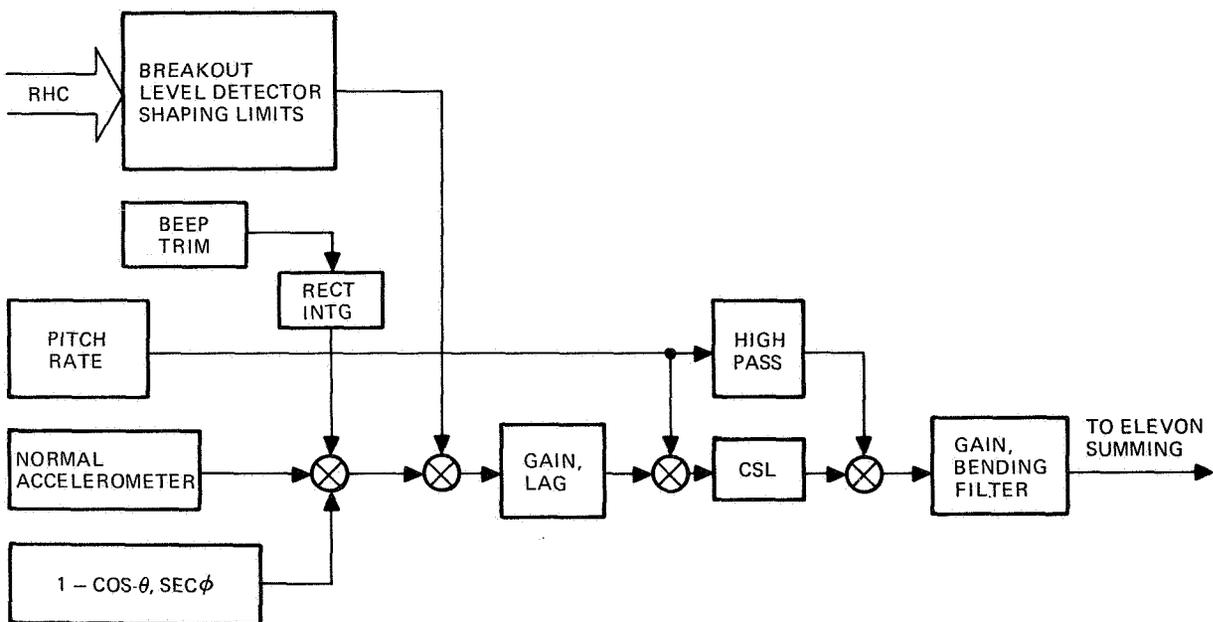


Figure 15. Pitch Command Augmentation

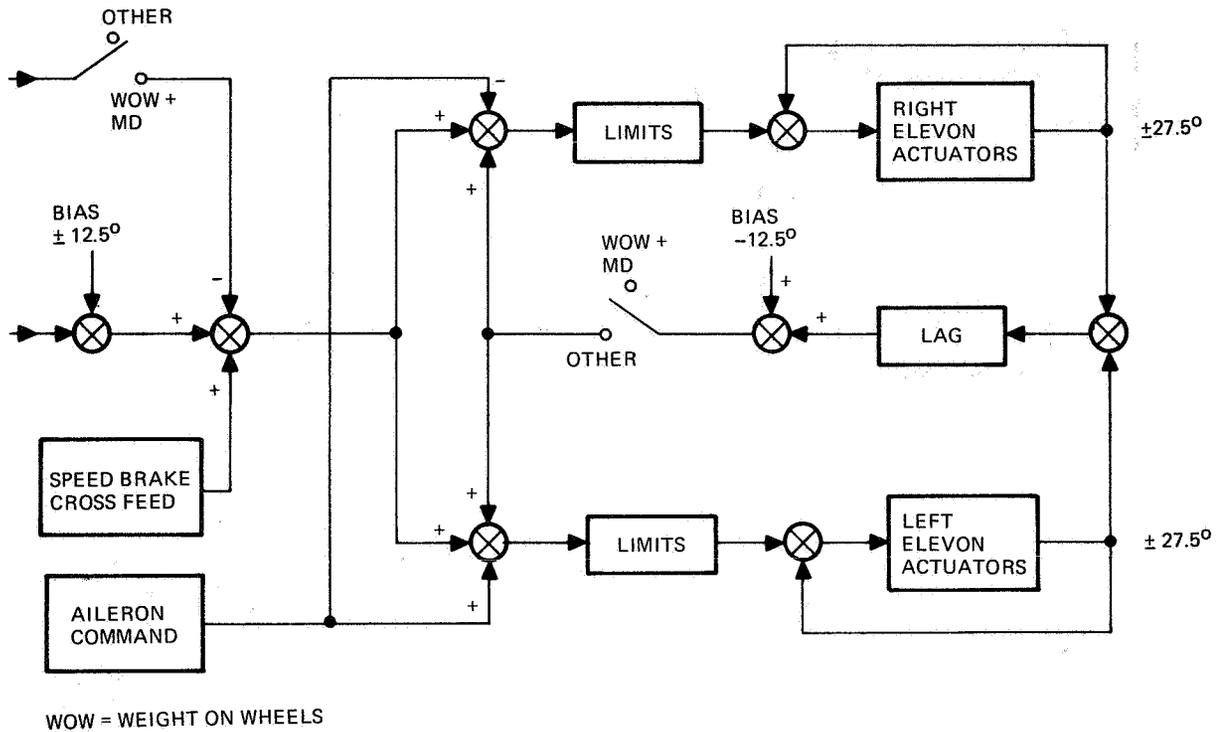


Figure 16. Elevon Summing

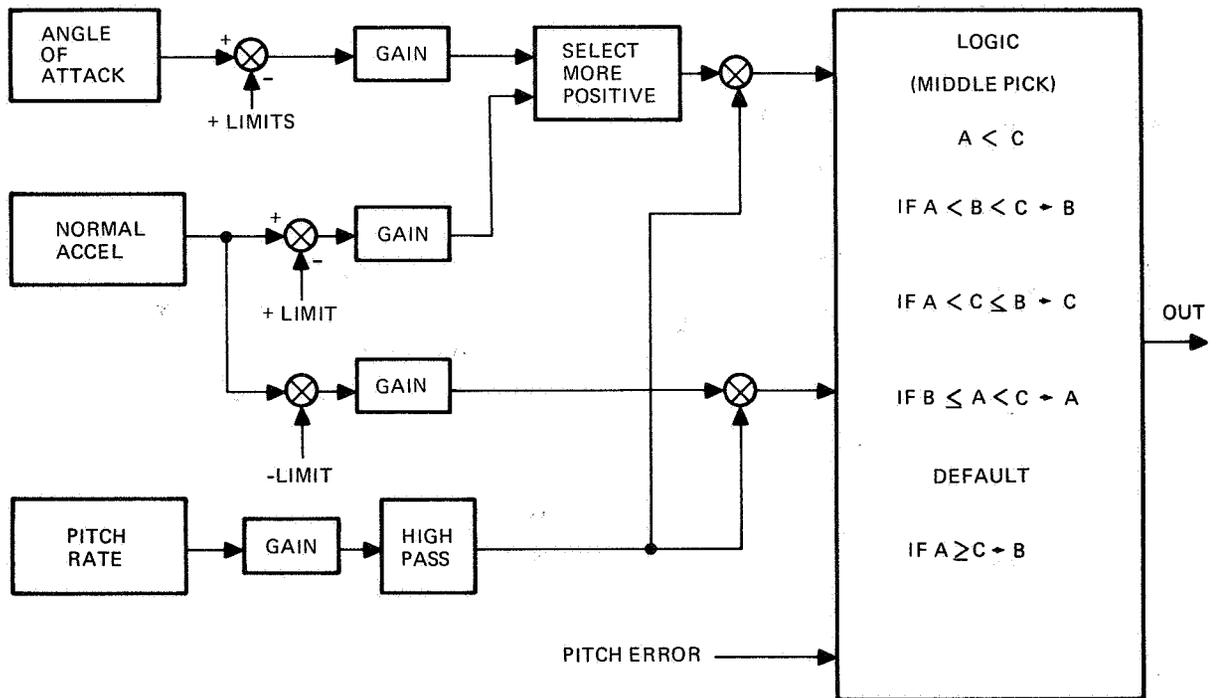


Figure 17. Command Signal Limiting

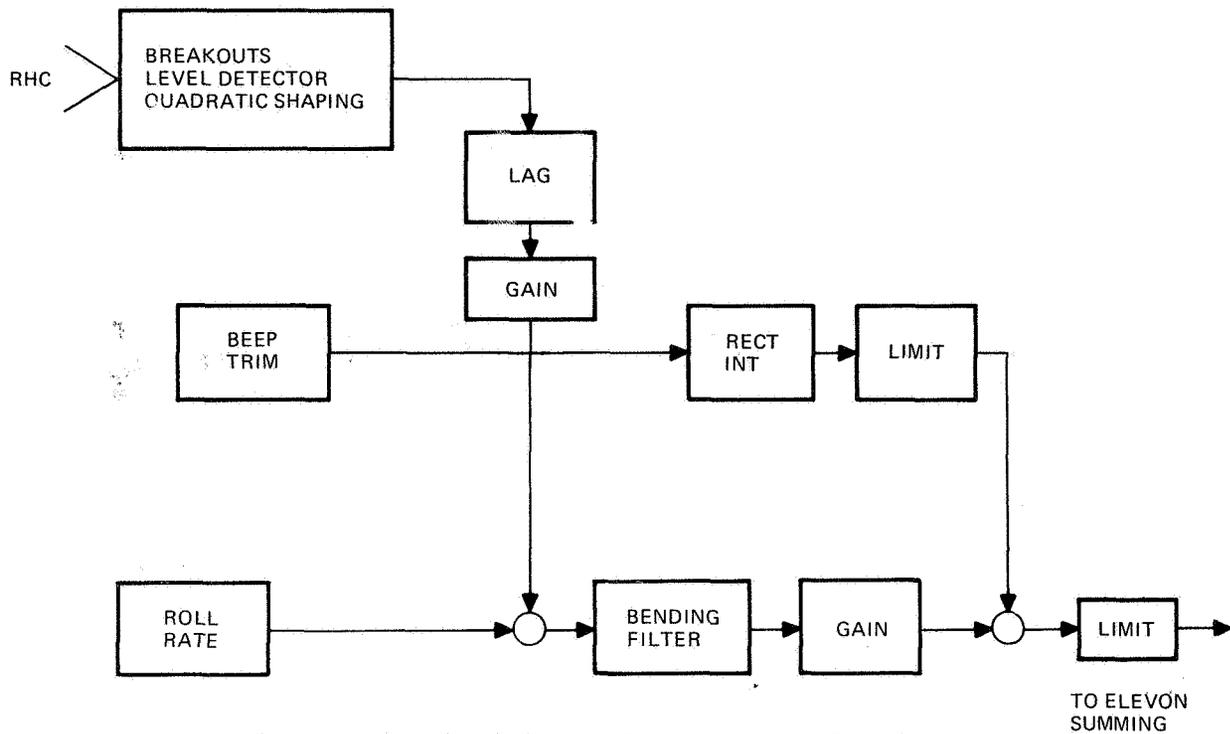


Figure 18. Roll Axis—Computed Air Speed

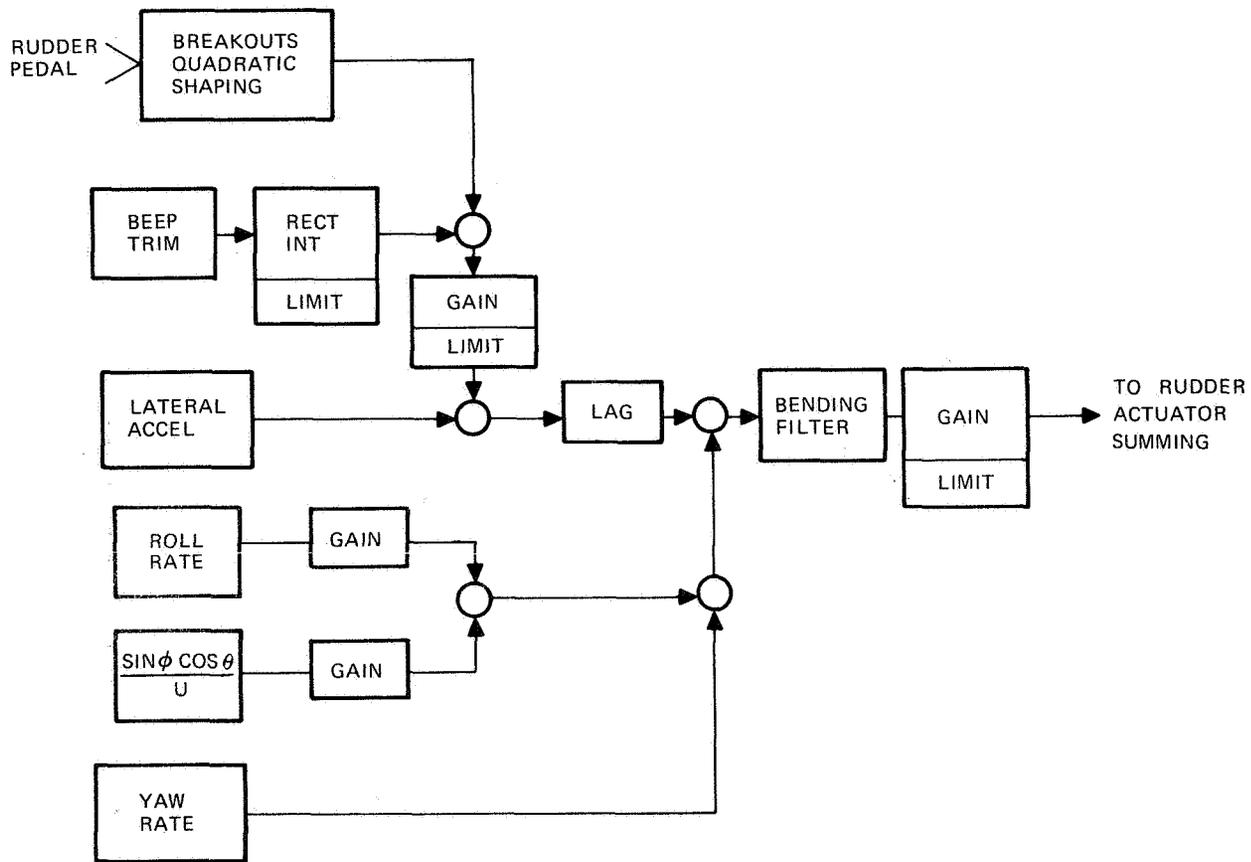


Figure 19. Yaw Axis—Computed Air Speed

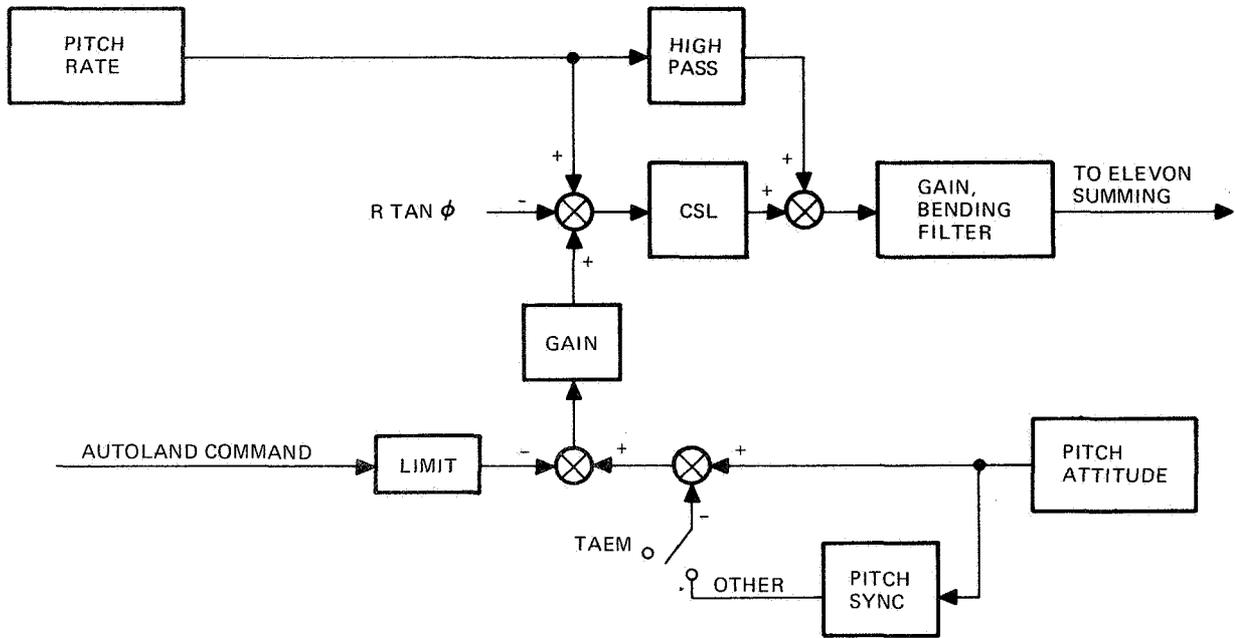


Figure 20. Pitch Attitude Hold and Autoland

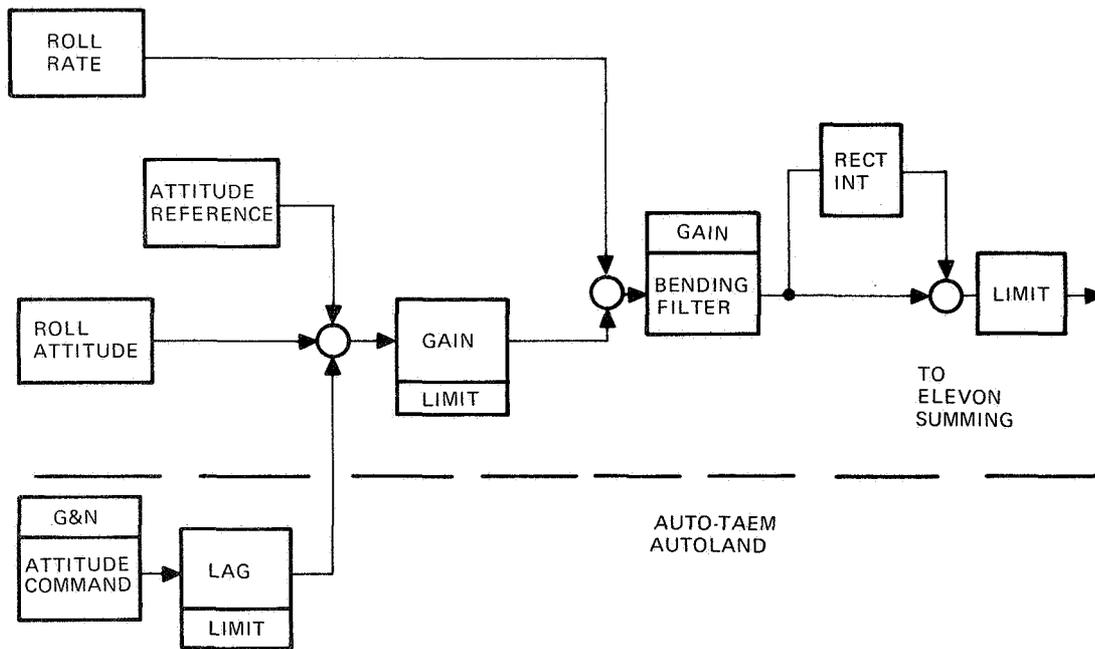


Figure 21. Roll Attitude Hold

C-4

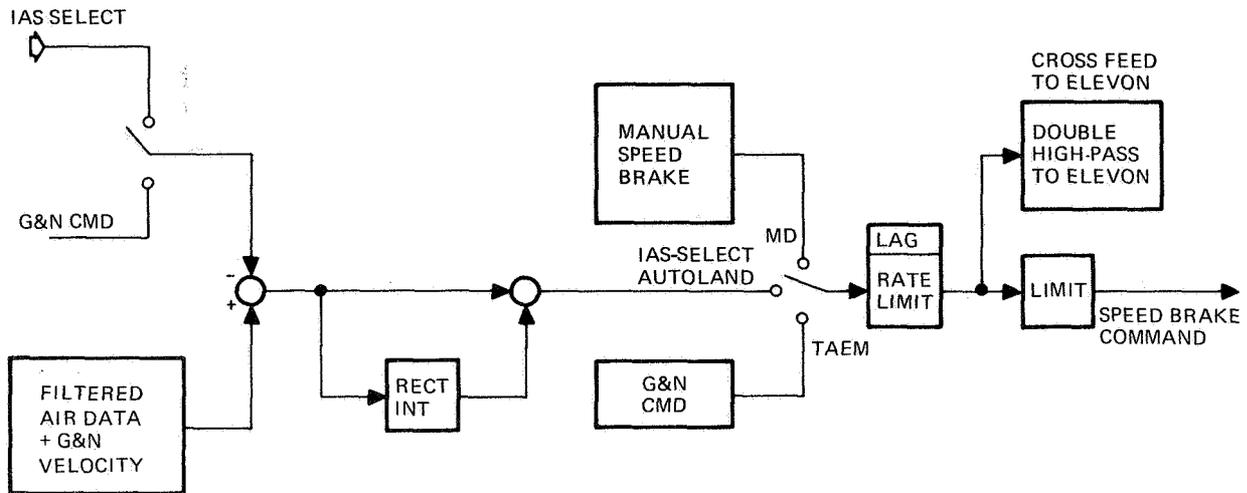


Figure 22. Indicated Air Speed (IAS) Hold

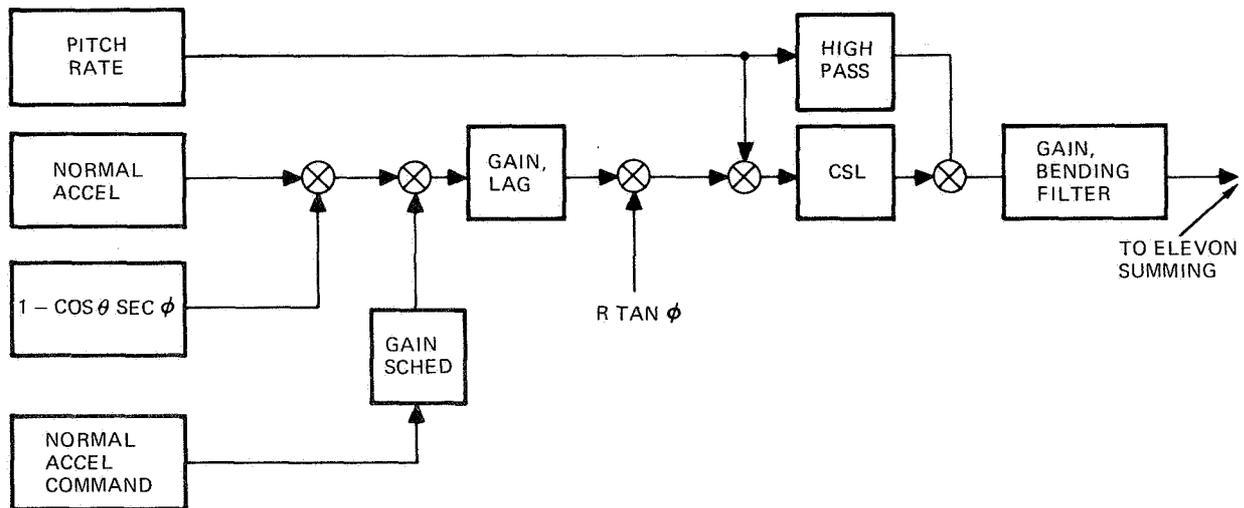


Figure 23. Pitch Auto-TAEM

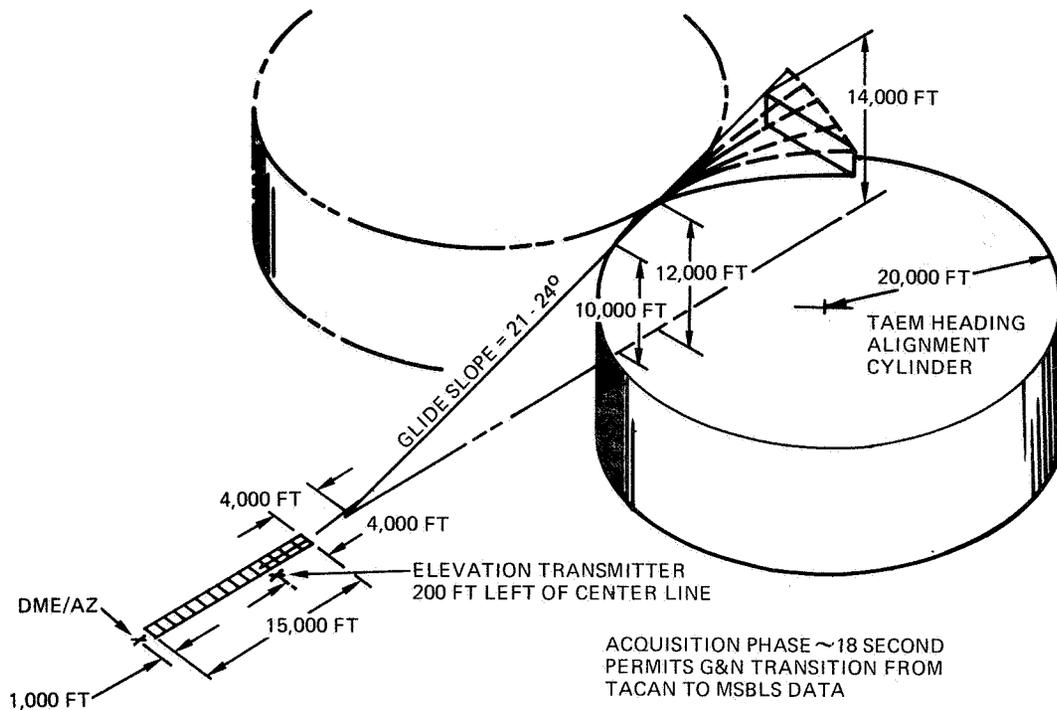


Figure 24. Microwave Scan Beam Landing System (MSBLS)

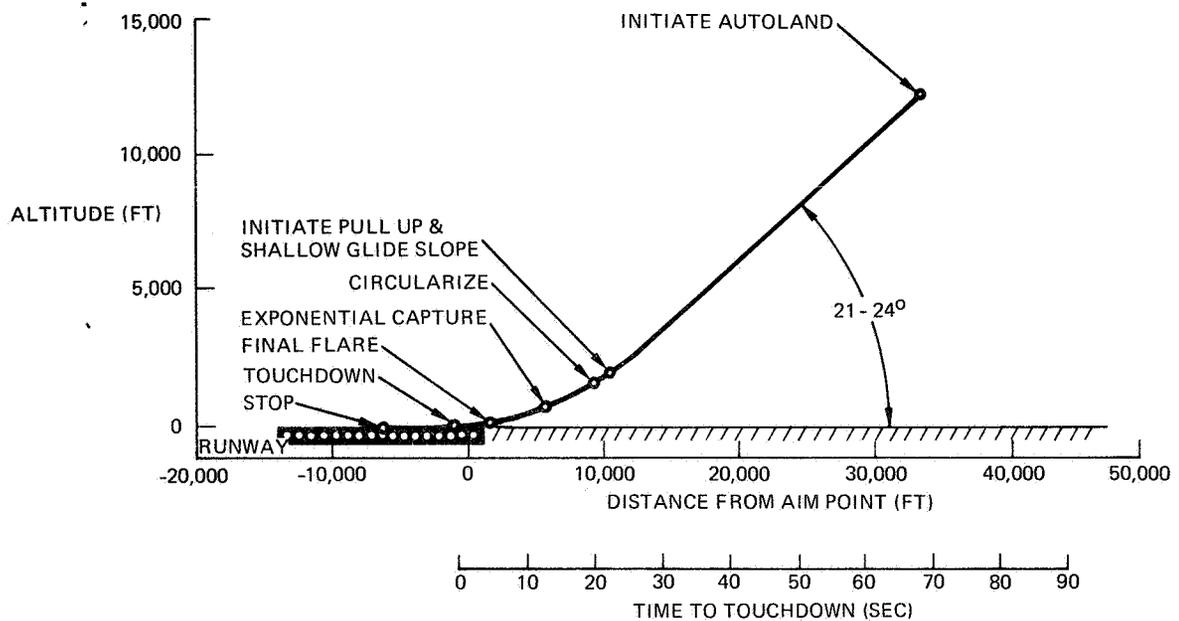


Figure 25. Autoland Trajectory

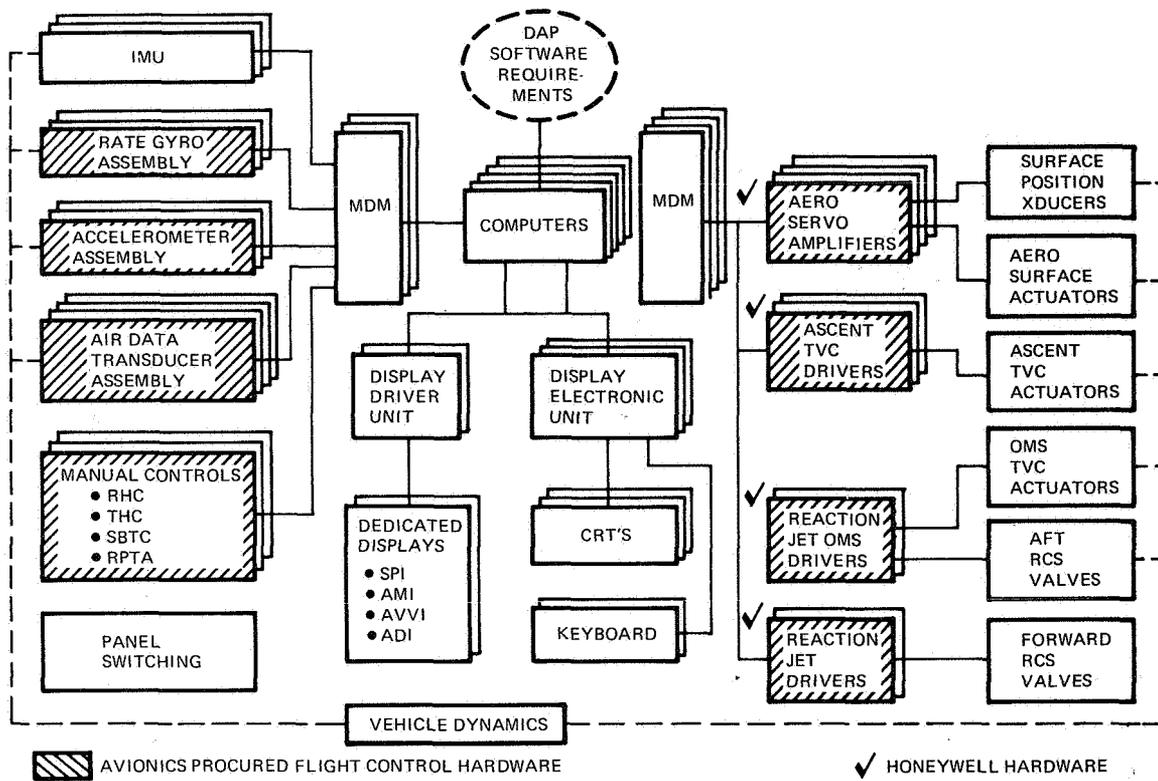


Figure 26. Flight Control System

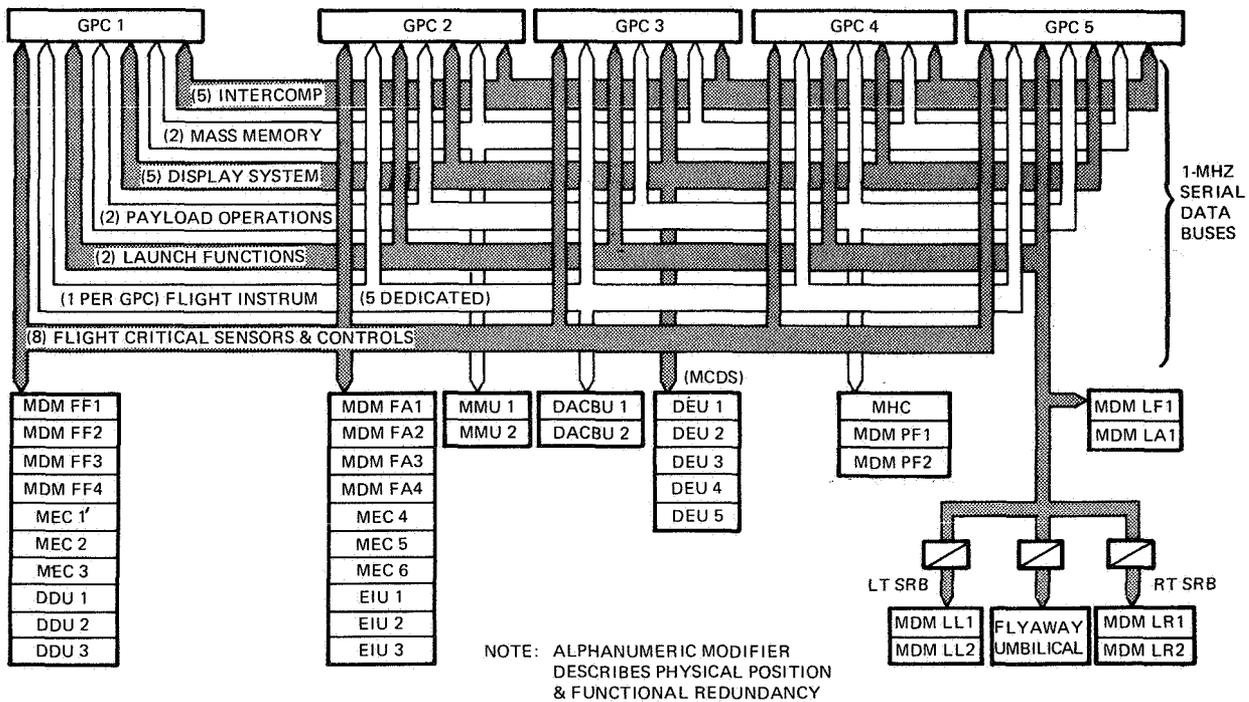


Figure 27. Digital Processing Subsystem Interface

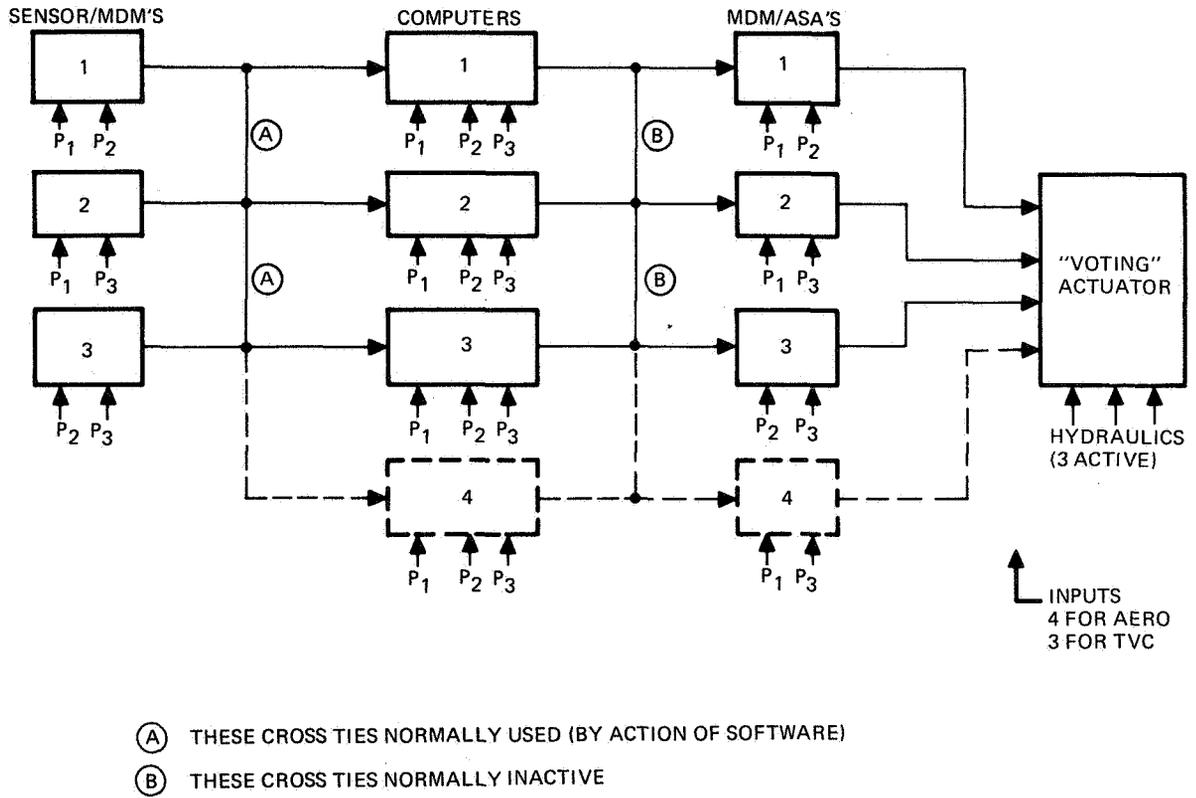


Figure 28. Basic Avionics Hardware Configuration for Critical Functions