Shuttle Vehicle Configuration Impact on Ascent Guidance and Control

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This paper discusses the factors leading to the design of the Space Shuttle ascent flight control system. Vehicle constraints on performance, loads, heating, software, and engine gimballing formed the basis for control system requirements and led to configuration and design decisions. Topics discussed include the selection of a thrust vector control capability on the Shuttle solid rocket booster, the control requirements of the solid rocket booster thrust mismatch and misalignment, the Orbiter flight and pad orientation, flight control sensor placement, the ascent load relief system, the elevon load relief system and performance improvements by reduced engine bias angles.

Nomenclature

APU	= auxiliary power unit
Ex, Ey, Ez	= altitude error roll, pitch, and yaw, deg
FPR	= flight performance reserve
GN&C	= guidance, navigation, and control
h	= altitude, ft
IMU	= inertial measurement unit
MDM	= multiplexer/demultiplexer
MPS	= main propulsion system
NY	= lateral acceleration, ft/s^2
NZ	= normal acceleration, ft/s^2
$\dot{p},\dot{q},\dot{r}$	= vehicle angular acceleration, deg/s^2
P_c	= chamber pressure, psia
PRL	= priority rate limiting
$\bar{q}$	= dynamic presure, psf
$\bar{q}\alpha$, $\bar{q}\beta$	= squatcheloid parameters, psf-deg
RM	= redundancy management
SRM	= solid rocket motor
t	= time, s
T_{sep}	= time of solid rocket booster separation command event
TMM	= thrust mismatch
Vr	= relative velocity, ft/s
WAZ	= wind azimuth (direction from which the wind is blowing)
α	= angle of attack, deg
β	= angle of sideslip, deg
δ_ϵ	= elevon deflection, deg
ΔP	= pressure differential, psi
η	= squatcheloid clock angle, deg

Introduction

THE selection of a parallel burn concept for the Shuttle ascent configuration has left its mark on the ascent guidance and control (G&C) configuration. The solid rocket booster (SRB) and its parallel mount configuration has been proven a dominant feature in the design of the first-stage flight control system (FCS). The decision to provide SRB thrust vector control (TVC) and the design of the system to accommodate SRB thrust misalignment and mismatch tolerances were significant. The need to be able to boost the winged Orbiter through ascent environmental conditions without seriously impacting the Orbiter design resulted in implementation of a load relief system for ascent first stage. Also, an elevon load relief system was required to avoid subjecting the Orbiter elevons to hinge moments beyond entry design limits. The purpose of this paper is to focus on detailed features of the ascent G&C system that have been driven by the Shuttle vehicle [Orbiter, SRB's, and external tank (ET)] configuration.

SRB Thrust Vector Control

The early Shuttle configuration featured fixed SRB nozzles (Fig. 1). Ascent control authority was derived from Orbiter Space Shuttle main engine (SSME) TVC and Orbiter aerosurfaces—elevons used as ailerons and rudders used for roll control. The SRB nozzles were canted 11 deg in the yaw plane to reduce control disturbances. The fixed SRB nozzles minimized the complexity of the boosters but led to other complications. The SRB's had thrust termination capability and the Orbiter used auxiliary abort solid rocket motors (ASRM's) for first-stage abort in case of an SSME failure. With primary control power centralized in SSME TVC, an SSME failure in the atmosphere presented a significant control problem.

The ASRM's, thrust termination, and the SRB TVC were the subject of a series of configuration trade studies that were interrelated to a significant extent. The trade study evaluation of SRB TVC resulted in the decision to implement TVC on the SRB. The major considerations of this study are summarized below.

The first-stage ascent trajectory requires a significant roll maneuver following liftoff and a vertical rise to roll the vehicle from the launch pad orientation to the flight azimuth. Major disturbances during high dynamic pressure conditions are the result of wind, wind shear, and gusts. The SRB is also a major source of potential disturbance because of thrust misalignment or mismatch between the left and right SRB. Also, the Shuttle configuration has a significant roll/yaw coupling. The center of gravity is slightly above the SRB and ET centerline and toward the Orbiter. As the majority of the mass is the SRB and ET, Orbiter yaw corrections and aerodynamic side forces create a roll torque. Yaw disturbances from SRB mismatch and SRB misalignment also result in roll disturbances. The 11 deg cant angle of the fixed nozzle SRB is required to minimize the disturbance moments. By canting the nozzle toward the center of gravity the disturbance is reduced.

Roll disturbances are controlled by the differential deflection of the bottom two SSME's, yaw deflection of the top SSME, differential (aileron) deflection of the Orbiter inboard and outboard elevons, and Orbiter rudder deflection.

Submitted Oct. 6, 1982, revision received June 17, 1983.

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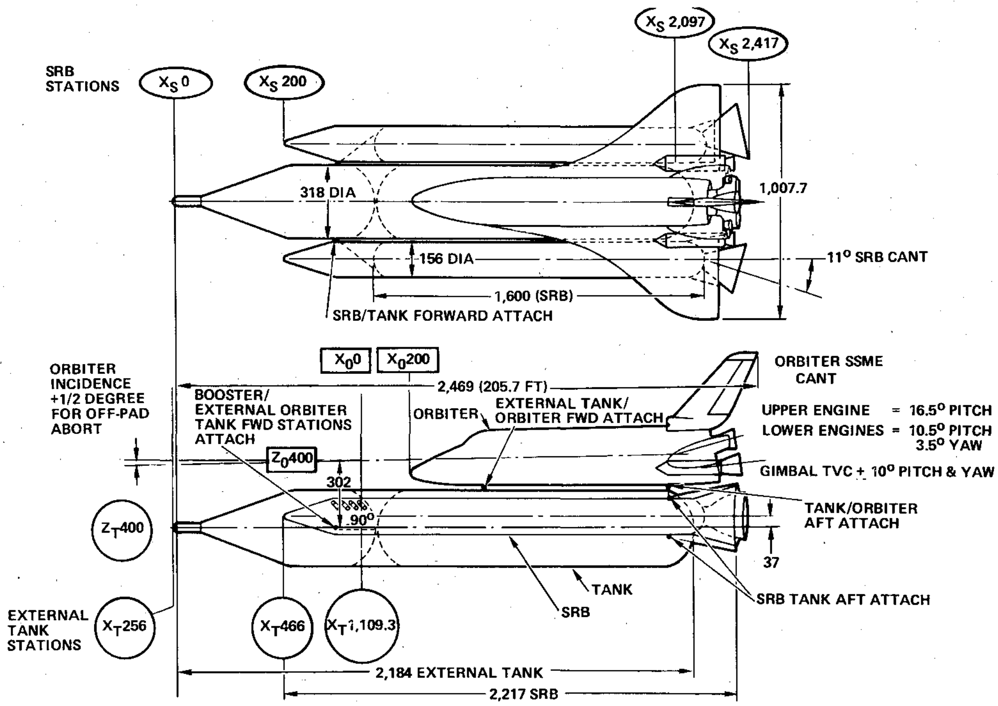

Fig. 1 Baseline integrated vehicle (August 1972).

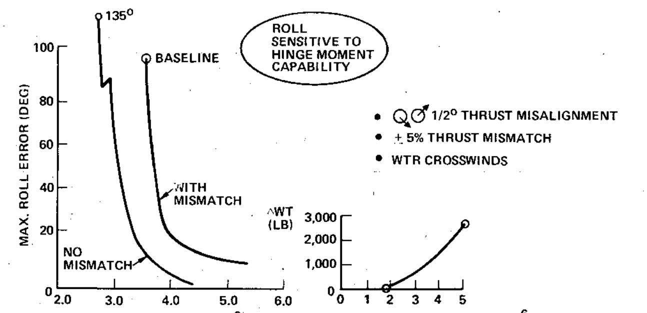

Fig. 2 Effect of elevon hinge moment capability, baseline flight control.

Figure 2 illustrates the roll control situation for the early configuration. Large roll attitude errors under disturbance conditions are indicative of lack of control authority. Ascent control also placed a structural load on the elevons that exceeded entry strength requirements. A vehicle weight penalty was required to meet ascent hinge moment and wing shear load requirements.

In addition, roll control effectiveness is also reduced due to aeroelastic effects at high dynamic pressure. The Shuttle ascent trajectory maximum dynamic pressure exists over a wide time range of the trajectory because SSME throttling and SRB thrust shaping have been designed accordingly. Thus, aeroelastic effects are significant for a large part of the ascent first stage.

A significant payload performance improvement was achieved by the incorporation of Shuttle SRB TVC. The effective thrust of the motor is the nozzle thrust reduced by the cosine of the cant angle. With incorporation of SRB TVC, the cant angle was first reduced to 4 degs. With this 4 deg cant, an axial thrust position could be obtained for separation. The reduction of lateral thrust forces at separation allowed reduction in booster separation motor size. A later change was made to eliminate the 4 deg cant angle; the result is today's configuration.

This cant angle reduction results in a lower gross liftoff weight (GLOW) required to perform reference mission payload requirements. The cost-per-flight savings of the size reduction was an overriding factor in the decision to implement SRB TVC. SRB TVC implementation costs were estimated at only 85% of the sizing reduction. Additional savings in GLOW and cost because of changes in abort modes can also be attributed to SRB TVC capability.

With fixed SRB nozzles, an SSME failure in the first 30 s of boost required the Orbiter to use auxiliary ASRM's in order to separate from the ET/SRB stack (thrust termination on the SRB's) and initiate a glide return to the launch site. Early glide return capability was designed to cover critical hazardous conditions and early SSME out conditions. Table 1 summarizes the ASRM requirements study that led to the deletion of the ASRM's. The incorporation of SRB TVC gave adequate control to continue ascent. The TVC capability, along with other redundancy and design changes, allowed the deletion of the ASRM's. Eventually, SRB thrust termination was also deleted, resulting in a simpler, more reliable, and less costly SRB. Thus, the cost tradeoff for SRB TVC (where sizing reductions more than paid for the TVC system) was regained several times over with the simplification of abort modes and SRB design.

The SRB thrust is a major source of possible disturbances. One common disturbance—SRB thrust misalignment—is due to dynamic thrust vector deviation, geometric misalignment of the nozzle, and SRB-to-ET alignment. For ascent control to function without SRB TVC, verification that the SRB misalignment is 0.5 deg or less was required. Experience with smaller scale motors indicated that the 0.5 deg could be met, but the Shuttle SRB is much larger than previous motors. Sufficient experimental testing proved that the 0.5 deg misalignment requirement was not feasible. A 1.0 deg misalignment criterion was established, consistent with test plans. Figure 3 shows the roll sensitivity to SRB thrust misalignment and presents verification requirements to assure thrust misalignment criteria.

Figure 4 shows the Shuttle configuration with SRB TVC at the conclusion of the SRB TVC trade study. Subsequently, several configuration changes were made to arrive at the current configuration. The 4 deg SRB cant angle was removed, since disturbances were not reduced significantly by the cant angle. The SRB was moved to the ET centerline,

Table 1 Summary of ASRM requirements study

FAILURE/ HAZARD	CREW EGRESS (PRIOR TO LIFT-OFF)	ASRM	NO SRB TVC CONTINUE ASCENT TO 30 SEC	SRB TVC CONTINUE ASCENT	COMMENTS	
LOSS OF THRUST OR CONTROL (1) SSME	—	✓	✓	$\boxed{\checkmark}$	• ABORT DESIGN CRITERIA	
LOSS OF THRUST SSME (2 OR 3)	—	✓	—	$\boxed{\checkmark}$	• PROVIDE TVC TO 30 SEC • PROVIDE THRUST TERMINATION	
LOSS OF CONTROL SSME (2 OR 3)	—	✓	—	$\boxed{\checkmark}$	• PROVIDE TVC TO 30 SEC • PROVIDE THRUST TERMINATION	
TIME CRITICAL ORBITER FAILURE	—	—	$\boxed{\checkmark}$	$\boxed{\checkmark}$	• MINIMUM FLIGHT TIME AFTER LIFT-OFF FROM 30 SEC IN ASCENT TO LANDING	
SRB TVC FAILURE	—	—	—	$\boxed{\checkmark}$	• NULL TVC & CONTINUE ASCENT	
SRB IGNITION FAILURE	✓	✓	—	N/A	• SENSE FAILURE & THRUST TERMINATION WITHIN 200 MILLISEC • THRUST TERMINATION PORT NOZZLE INSULATION • IGNITION REDUNDANCY • HOLD DOWN	THESE FAILURE CONDITIONS REPRESENT UNLIKELY EVENTS
PREMATURE SEPARATION SRB ORBITER ASRM	 — — —	 ✓ ✓ N/A	 — — ✓	 N/A N/A N/A	• DESIGN REDUNDANCY • ELECTRO/MECHANICAL INTERLOCKS • DELAY ARMING TO AFTER 30 SEC	
SRB FAILURE CAUSING THRUST TERMINATION	—	0–4.6 SEC GAP 4.6–30 SEC ✓	— —	— —	• DESIGN REDUNDANCY MARGINS • PRESSURE INCREASE OR BURNTHROUGH NOT CREDIBLE IN FIRST 30 SEC	
LEAK/FIRE	—	✓	$\boxed{\checkmark}$	$\boxed{\checkmark}$	• ISOLATE HAZARDOUS COMPARTMENTS • SHUTDOWN EFFECTED EQUIPMENT • WARNING TIME > 30 SEC	

NOT USABLE AFTER 30 SEC & LIMITED ABORT AFTER 89 SEC FOR IDENTICAL FAILURES

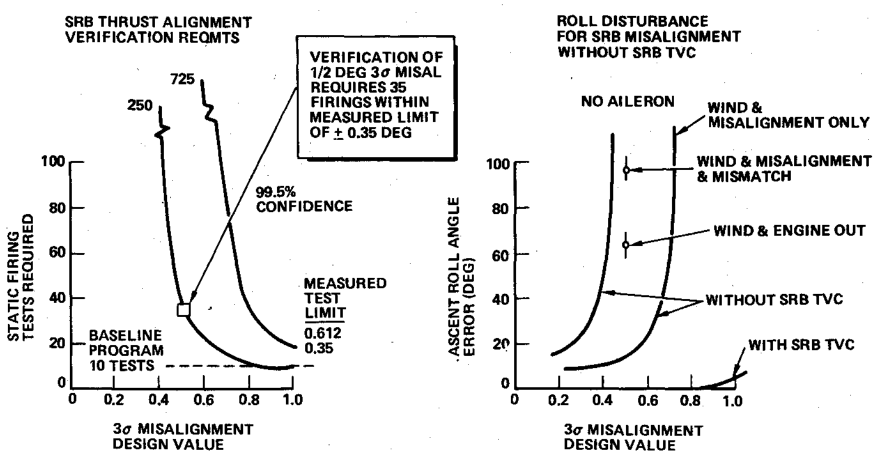

Fig. 3 SRB thrust vector alignment verification.

Orbiter SSME roll angles and deflection angles were modified slightly, and the SRB TVC actuators were moved from a 90 to a 45 deg orientation to maximize pitch and yaw nozzle deflection.

With SRB TVC capability in place, the ascent FCS design was assured of adequate control authority. The performance of the roll maneuver is smooth and precise (as demonstrated in flight), attesting to the roll control capability of the Shuttle. Abort capability was improved as SRB separation no longer had a lateral thrust force to overcome, and the control burden was relieved from the Orbiter aerosurfaces. The SRB design incorporated a gimbaled nozzle design, but thrust termination and a large fixed cant angle were eliminated. There were cost savings over the Shuttle program's life.

SRB Thrust Mismatch and Misalignment

SRB thrust mismatch and misalignment disturbances were a major factor in dictating SRB TVC. The thrust misalignment is defined as having a single motor magnitude of 1 deg that includes allowance for nozzle geometric alignment, actuator installation, and nozzle flow variations. The SRB-to-ET alignment is within 0.25 deg, giving a root sum square (rss) net motor alignment to within 1.03 deg. For a two-SRB operation, the misalignment of each motor is assumed to be in the same direction (pitch, yaw, roll, etc.), with a value of 0.75 deg (approximately $1.03 \times \sqrt{2}/2$.

The thrust mismatch is defined as the difference in thrust between the left and right SRB. Three time periods are specified: liftoff, steady state, and tailoff. Figure 5 defines differential thrust vs time for liftoff and steady state, and Fig. 6 defines the thrust differentials for tailoff. These thrust differential values were derived from the data base of Titan III C and D flights scaled for Shuttle conditions. Early tailoff mismatch values consisted of 450,000 lb, with revised criteria indicating a maximum of 710,000 lb; however, the same 4.5×10^6 lb-s total impulse during tailoff was retained.

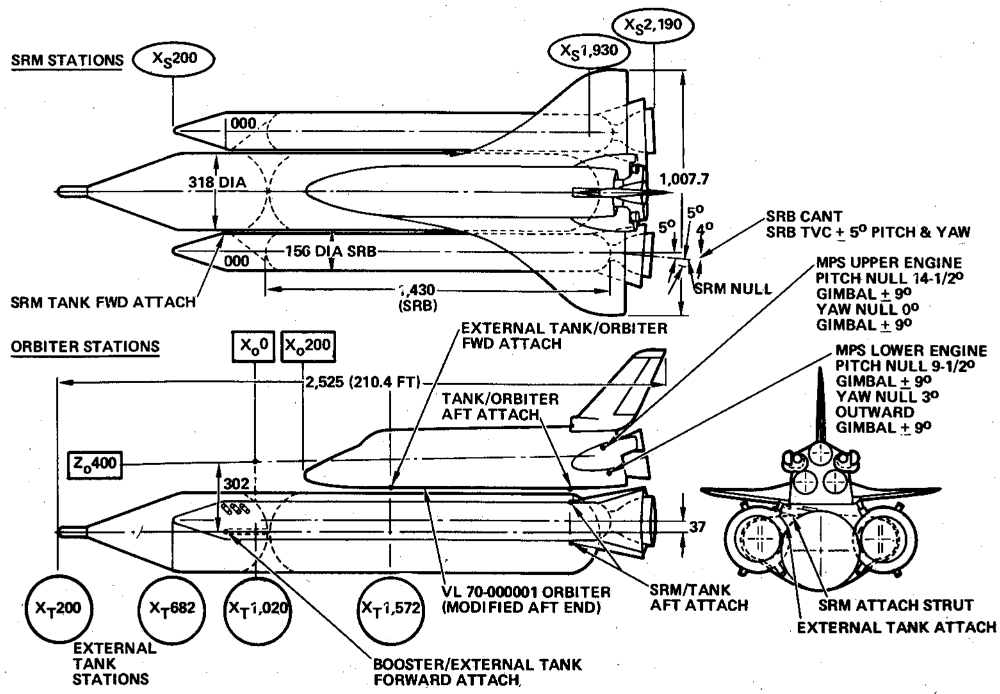

Fig. 4 SRB TVC configuration, integrated vehicle.

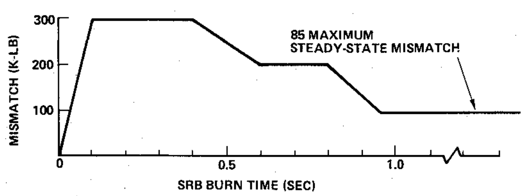

Fig. 5 SRB thrust differential during ignition.

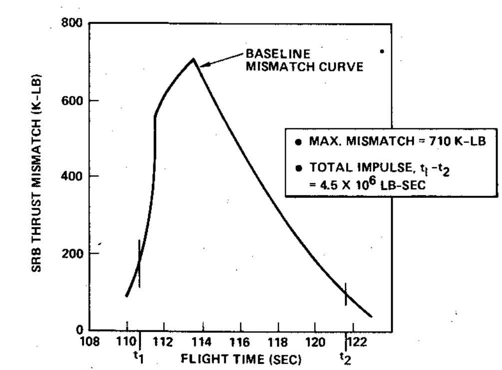

Fig. 6 SRB thrust differential during tailoff.

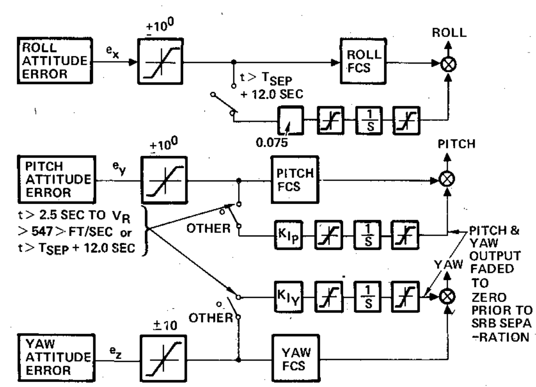

Fig. 7 Trim integrator configuration.

Misalignment and mismatch are the primary disturbances that affect liftoff clearances. Also, early misalignment results in attitude errors from commanded steering. Pitch and yaw trim integrators (Fig. 7) were added to the ascent FCS to compensate for these early misalignment errors. Roll, pitch, and yaw trim integrators are used in second stage to adjust the center of gravity trim. The first-stage pitch and yaw trim is initiated at 2.5 s and continues until a relative velocity of 547 ft/s; this is also the start of load relief.

The trim integrator establishes a value to compensate for steady-state thrust misalignment prior to the buildup of aerodynamic effects and then holds that value until fading out during SRB tailoff. The integrator remains fixed during the load relief period of the trajectory because load alleviation requires a buildup of attitude error in the presence of wind and gust disturbances. By balancing the misalignment effects prior to maximum dynamic conditions, the integrator reduces possible air load deviations. Table 2 shows a dispersion analysis of dynamic pressure times angle of attack ($\bar{q}\alpha$) and dynamic pressure times angle of sideslip ($\bar{q}\beta$). A spectrum of thrust, sensor, aerodynamic, and vehicle tolerances are examined individually for four wind azimuths (roughly headwinds, tailwinds, and crosswinds). These indicated dispersions are then combined statistically to derive an overall system dispersion. Table 2 shows dispersion determined with and without a trim integrator function. The trim integrator shows a reduction in the dispersion in the most significant azimuths.

Thrust mismatch during tailoff saturates the control authority of the system. Figure 8 (left) illustrates the thrust decay difference between motors and the resulting design thrust mismatch. The right-hand side of Fig. 8 depicts the SRB and Orbiter SSME actuator deflections required to trim the mismatch moment. The SRB TVC has the capability of trimming the mismatch up to approximately 122 s. The Orbiter SSME TVC capability becomes effective at approximately 120 s. These curves assume full control capability and instantaneous gimbaling. The challenge is to obtain response from the system to nearly utilize that capability.

Table 2 Dispersion analysis for ascent load flight conditions trim integrator effect, Mach 1.25

Source	Design tolerance of source, deg	WAZ = 35 deg		WAZ = 125 deg		WAZ = 215 deg		WAZ = 305 deg	
		$\Delta\bar{q}\alpha$	$\Delta\bar{q}\beta$	$\Delta\bar{q}\alpha$	$\Delta\bar{q}\beta$	$\Delta\bar{q}\alpha$	$\Delta\bar{q}\beta$	$\Delta\bar{q}\alpha$	$\Delta\bar{q}\beta$
SRB thrust misalignment without trim integrator	0.75	244	30	36	238	90	29	19	267
SRB thrust misalignment with trim integrator	0.75	65	25	4	16	15	104	59	7
Trim integrator reduction		179	5	32	222	75	−75	−40	260

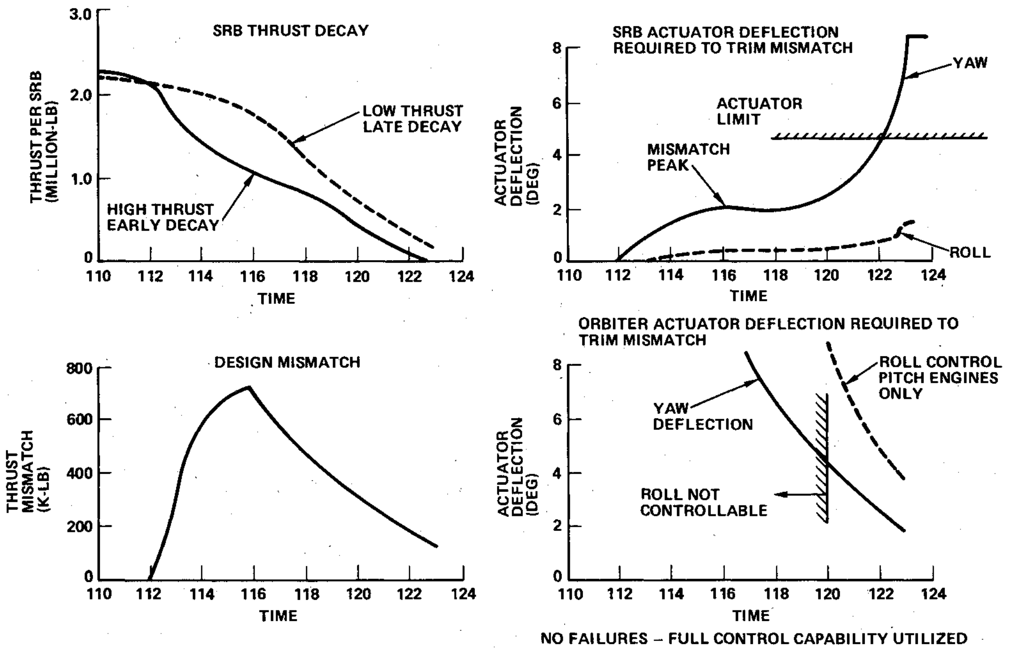

Fig. 8 SRB thrust mismatch at tailoff.

The ascent FCS uses SRB chamber pressure measurements to anticipate tailoff and adjust gains and times for mismatch control. Vehicle constraints for tailoff transients exist for both heating and separation. The trajectory prior to SRB separation is a critical region for heating conditions on the ET. The transient response due to tailoff can result in a maximum 11 deg sideslip condition. The ascent FCS mechanization during tailoff is defined in Fig. 9 which shows the events used to schedule the gain, mechanization, and trim changes. As SRB thrust decays, the SRB gains can be increased and still keep stability margins. The pitch trim that is programmed into the SRB gimbal for the majority of the trajectory is phased over to the SSME. The roll mixing logic for second stage is initiated and if SSME out conditions occur, the changes to the roll mixing logic are enabled (not required in first stage). Figure 10 shows the transient improvement for an earlier initiation of gain scheduling ($P_c = 450$ in place of $P_c = 400$). The requirement to reduce possible transients at tailoff comes from a sensitivity of ET heating from shock interference patterns between the forward ET attach strut and the Orbiter. The Mach number region prior to SRB separation produces these shock patterns. Dispersions in angle of attack and angle of sideslip increase heating rates and also spread the high heating area over a large tank acreage. There was a significant penalty in ET insulation weight, requiring a more durable super light ablator (SLA) insulation over a larger area.

The experience in early development flights showed that SRB mismatch is well below specification values. Figure 11 compares the flight-experienced tailoff mismatch to the specification limits.

Orbiter Flight and Pad Orientation

Orbiter orientation on the launch pad determines the extent of the roll-to-azimuth maneuver required of the FCS. Early launch pad configuration trades between tail north, tail west, and tail east showed no significant performance impact for the roll maneuver. The pad orientation, originally tail north and, finally, tail south, was decided based on launch facility design and constraint issues. The overriding advantage of this is to make maximum use of the Saturn V crawler and Launch Pad 39.

The Orbiter heads-down orientation during flight was selected primarily for performance and abort maneuver advantages. The performance gain was approximately 960 lb. The abort advantage consists of either a glide return to launch site (no longer an abort mode) or a powered return to launch site maneuver. For the turnaround maneuver, a single pitch action is sufficient. Other advantages of heads-down attitude are improved S-band antenna-look angles and crew view of the horizon.

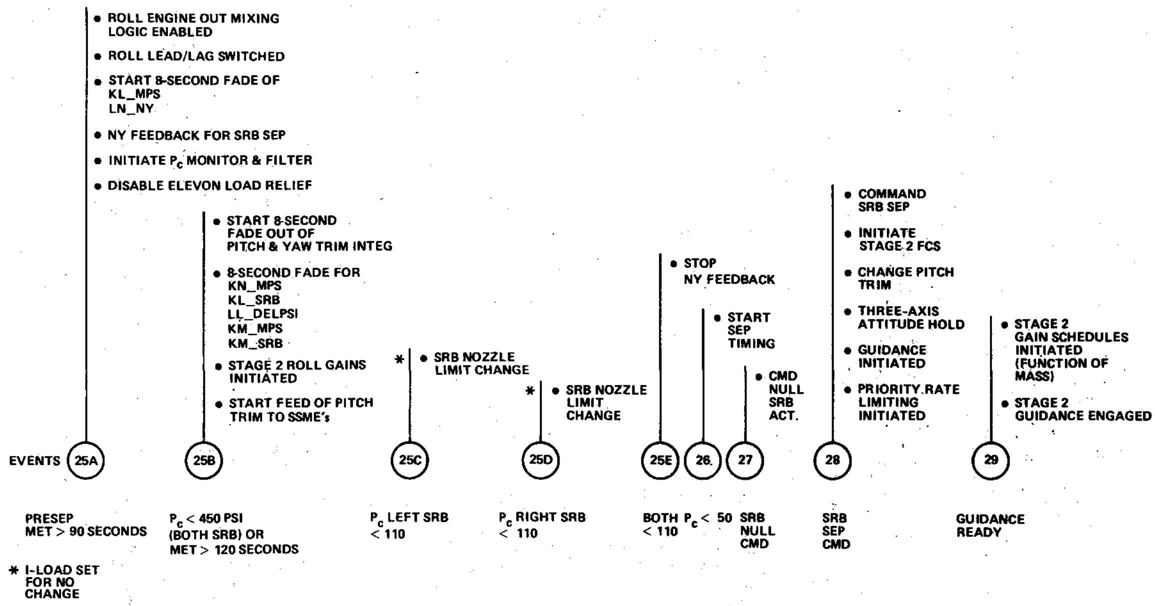

Fig. 9 Preseparation ascent FCS reconfiguration.

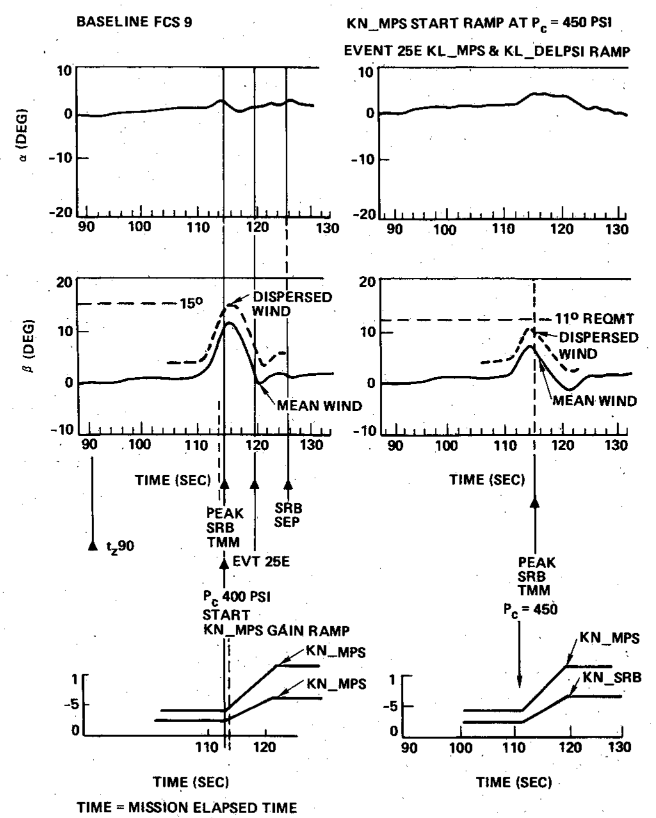

Fig. 10 Angle of attack/angle of sideslip improvement with revised gain schedule.

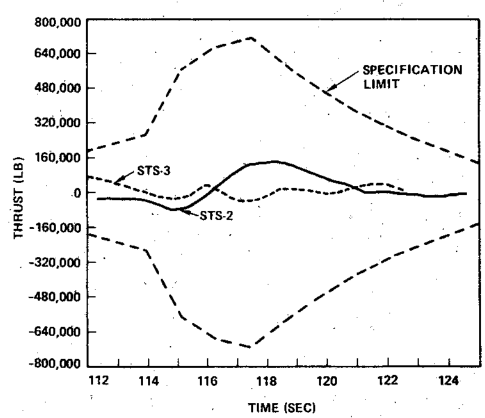

Fig. 11 Flight tailoff thrust mismatch compared to specification limits.

Flight Control Sensor Placement

Flight control sensors consist of three redundant inertial measurement units, four body-mounted normal and lateral accelerometers, four Orbiter rate gyros (pitch, yaw, and roll), three left and right SRB rate gyros (pitch and yaw), three left and right SRB chamber pressure transducers, and four left and right inboard and outboard elevon delta pressure transducers. The placement of these sensors and the rationale for placement are summarized in Fig. 12. The accelerometer and rate gyro sensors are placed to minimize structural bending impact on stability. The flight control sensors are located at the tail of the vehicle in the proximity of the primary control effectors. Originally two sets of accelerometers (forward and aft) were blended. The aft accelerometers were deleted for environmental reasons. The forward accelerometer is now mixed with a differentiated rate gyro signal. The SRB rate gyros originally were located in the aft skirt but were moved to the forward compartment for environmental reasons. The Orbiter rate gyros were moved from a side-mounted position on the aft bulkhead to a common centerline lower position. This was due to the fact that in the side-mounted position, pitch elevon motion caused yaw rate pickup due to structural coupling.

Load Relief

Normal FCS functions (to provide stabilization, control, and response to guidance commands) are not sufficient for the Shuttle during first-stage boost through the atmosphere. During this first-stage boost, the requirement is to keep design flight conditions within vehicle capability in the presence of a

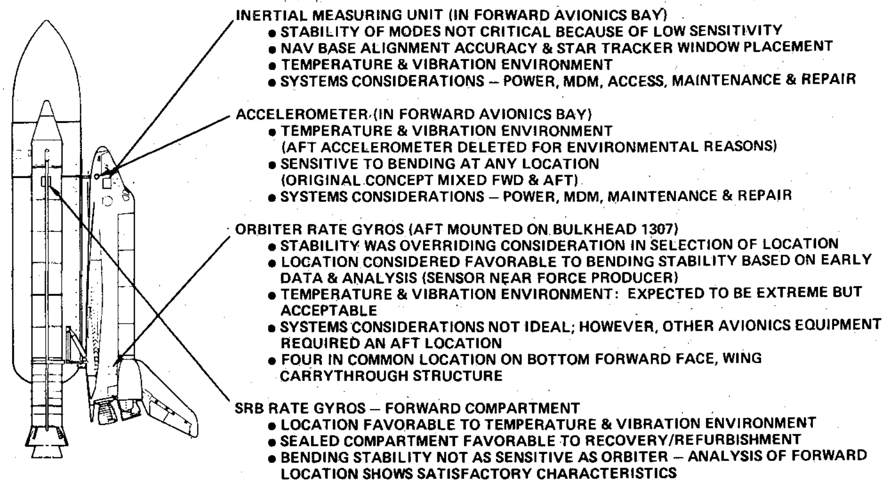

Fig. 12 Rationale for present flight control sensor locations.

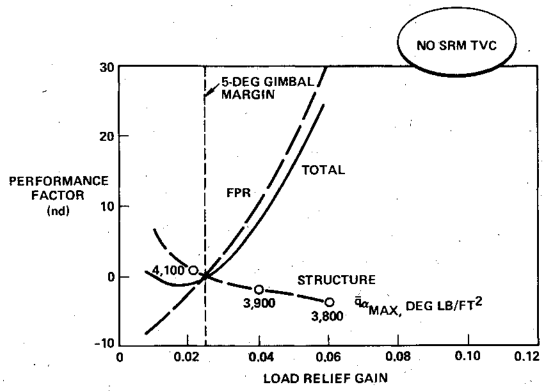

Fig. 13 Effect of load relief on performance.

design specification of natural environment. The impact on the G&C system shows up in the following forms:

1) Wind biasing. Steering software to bias the nominal trajectory to the mean wind.

2) Ascent load relief. Flight control corrections to the prelaunch steering based on sensed acceleration.

3) Elevon load relief. Programmed elevon with hinge moment override.

4) Sensor tolerances. Hardware and redundancy management.

5) Failure transient constraints. Actuator failure constraints and engine-out software adjustments.

The trajectory steering is established for the mean (monthly mean wind for the launch location), the mean wind steering providing reference acceleration, attitude commands, and SRB pitch trim. The load alleviation function of the ascent FCS produces error signals for differences between measured acceleration (normal and lateral) and reference acceleration. The actuator commands from load relief require attitude errors resulting in modifications to vehicle flight path, which reduce the aerodynamic loads experienced.

The wind criteria for ascent design consist of synthetic vector wind profiles. A statistical model allows construction of wind, wind shear, and gust on a monthly basis. The mean wind is used for nominal steering. To define the ascent loads constraints, a 95% profile with 99% shear and 9 m/s discrete gust must be constructed for selected wind azimuths, with the peak shear and gust occurrence placed at a selected Mach number for several discrete Mach numbers.

Early system trade studies evaluated fixed attitude control, 100% load alleviation, and a blended attitude and load alleviation control. A blended system was selected based on a balance between structural weight saving and trajectory performance penalty. Figure 13 shows the selection of level-of-load relief gain. The definition of load relief capability for vehicle loads and design required development of a unique analysis technique. The original definition of flight conditions was a single $\bar{q}\alpha$ span of ± 4000 psf-deg and $\bar{q}\beta$ span of ± 5100 psf-deg. A revised specification of $-3000 \leq \bar{q}\alpha \leq 4000$ psf-deg and $\bar{q}\beta$ span ± 3600 psf-deg was established later to reduce Orbiter structural loads. As loads analysis matured, it was determined that simple spans did not provide adequate definition. Combinations of $\bar{q}\alpha$ and $\bar{q}\beta$ presented critical loads; a simple headwind, tailwind, crosswind, or quartering wind was not sufficient. A system to define a continuous map of flight conditions was established. The $\bar{q}\alpha$ vs $\bar{q}\beta$ map has been termed "squatcheloid," a coined name based on the shape of the plot. A complete squatcheloid definition of flight conditions consists of the $\bar{q}\alpha$ vs $\bar{q}\beta$ map; angular accelerations $\dot{p}$, $\dot{q}$, and $\dot{r}$; dynamic pressure $\bar{q}$; inboard and outboard elevon deflection; altitude; and time (Fig. 14).

The squatcheloid approach is implemented by a flight control analysis of system response at a series of discrete Mach numbers, $M=0.6$ to 3.0. For each Mach number, a series of trajectory runs define response at eight wind azimuths around the squatcheloid. The $\bar{q}\alpha$ and $\bar{q}\beta$ points and corresponding data are used to define $\dot{q}$ vs $\bar{q}\alpha, \dot{p}$ and $\dot{r}$ vs $\bar{q}\beta$, δ_ϵ, h, t, and $\bar{q}$ curves. The latter variables are plotted vs an angle η which is the clock angle around the $\bar{q}\alpha$ vs $\bar{q}\beta$ curves (0 deg at 12 o'clock).

The loads analysis utilizes the squatcheloid information to survey vehicle loads at 30-40 points around the $\bar{q}\alpha$ and $\bar{q}\beta$ envelope. The critical loading conditions are not only the Orbiter wing elevon, but the Orbiter ET forward and aft attach fittings and the ET-to-SRB forward and aft attach fittings as well.

The sensitivity of ascent loads to ascent flight conditions makes the relationship between ascent FCS and vehicle loads constraints quite strong. The FCS response must be considered from two aspects. First, the dynamic response of the system will modify and change the flight conditions, i.e., $\bar{q}\alpha$, $\bar{q}\beta$, $\dot{p}$, $\dot{q}$, $\dot{r}$. This dynamic response is affected primarily by load relief, attitude and rate gains, and vehicle stability. The FCS mixing logic will also change the distribution of the control forces and moments between the Orbiter and SRB. The mixing logic will primarily affect the Orbiter, ET, and SRB attachment loads.

An example of the constraints on control system design because of loads impact consists of a proposed change that removed the load relief signal from the Orbiter TVC. The change was proposed on the basis of a bending stability requirement. The accelerometer feedback is a strong source of structural bending coupling with control effects. In this in-

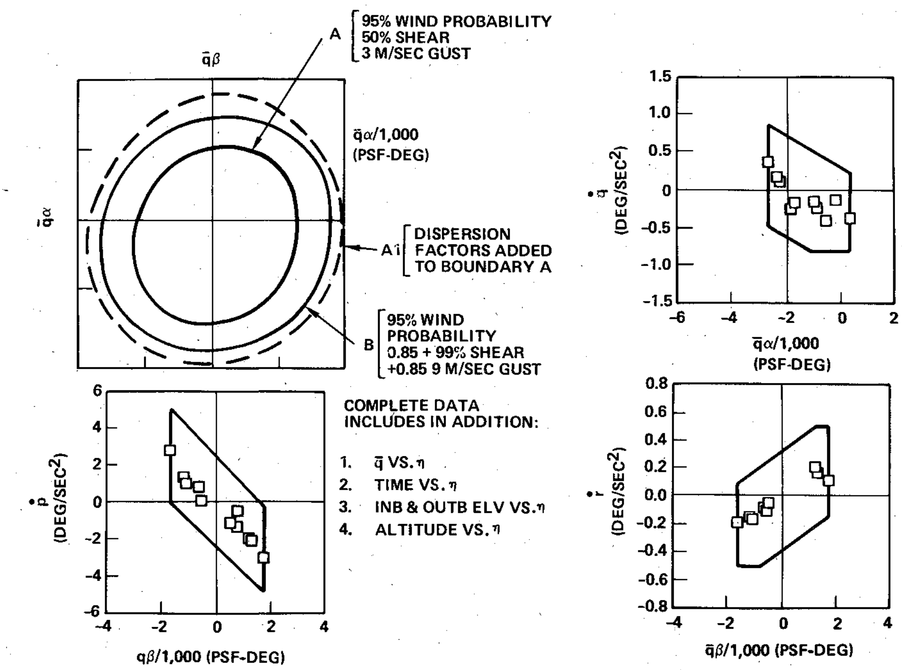

Fig. 14 Squatcheloid for Mach 1.05.

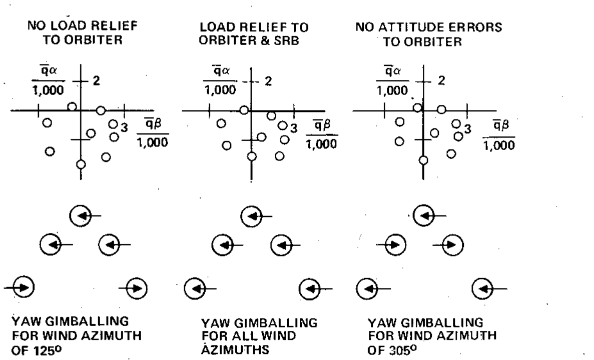

Fig. 15 Flight condition comparison of FCS mechanization changes.

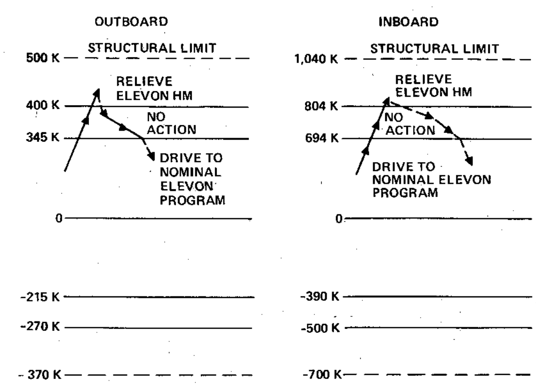

Fig. 16 Elevon load relief control logic boundaries.

stance, the destabilizing coupling of the Orbiter-mounted accelerometers from the SRB TVC was less than that from the SSME TVC.

The loads assessment of the three options showed an increase in loads between Orbiter and ET attachments and a decrease in SRB and ET attachments for the "no SSME TVC load relief" option. The comparison of flight conditions for these three options is shown in Fig. 15. The baseline system is shown in the center of the figure with load relief signals to both Orbiter and SRB TVC. The yaw gimbaling pattern is the same for all wind conditions, as shown in the diagram below the $\bar{q}\alpha$ and $\bar{q}\beta$ conditions. The alternate "no Orbiter load relief" option, shown in the left-hand side of Fig. 15, caused a slight increase in flight conditions and, for some wind azimuths, an opposing pattern between Orbiter and SRB yaw control is experienced. Another alternate option, shown on the right, consists of no attitude error control to the Orbiter TVC and yaw and roll commands to the top engine. This option caused an opposing control pattern but negligible change in fitting loads. The loads assessment of the system without load relief in the Orbiter TVC showed a significant redistribution of loads and corresponding impact on attach structure loads. The final control system logic option consists of load relief to both SRB and SSME, with bending compensation obtained by filters.

Elevon Load Relief

The use of Orbiter elevons for active vehicle control was deleted with the addition of SRB TVC. The elevons were fixed at zero until loads study showed that excessive elevon hinge moments would be experienced during ascent. A fixed position other than zero would keep hinge moments acceptable, but wing shear load and root bending moments then would be excessive. A programmed elevon deflection was designed which kept both the wing and elevon within acceptable loads. The analysis indicated that load margin existed for nominal aerodynamics, but that uncertainties in aerodynamic hinge moment coefficients could result in excessive elevon loads. A closed-loop elevon load relief system solved the aerodynamic sensitivity issue. The elevon secondary ΔP transducers measure the load experienced by the elevon and, therefore, these measurements are proportional to hinge moment. The system feeds back elevon ΔP measurements and moves the elevon away from its programmed position in a direction to reduce elevon loads. The inboard and outboard elevons have different load limits and operate independently. The left and right panels, both inboard and outboard, are slaved together. If either side exceeds the hinge moment limit, both are moved to relieve the overload conditions.

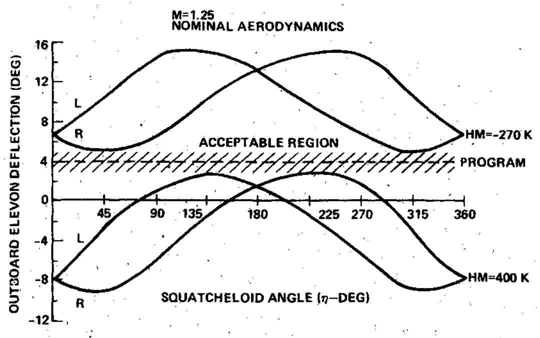

Fig. 17 Outboard elevon deflection vs squatcheloid angle for design hinge moment.

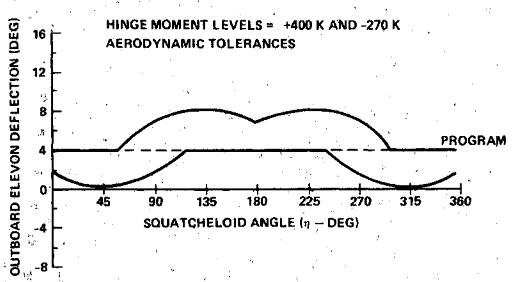

Fig. 18 Outboard elevon deflection possible to relieve hinge moment.

The elevon load relief system operates in overload cases only. Normally, the elevon deflection follows the program. If a ΔP measurement exceeds a plus or minus threshold, an incremental command to move the elevon (rate limited to conserve hydraulic power) is executed. Once the load falls below the relief limit, the incremental deflection is held constant. When the load falls below a return level, commands are given to drive back to the nominal program. This logic is illustrated in Fig. 16. The left and right outboard elevon deflection angles required to keep the plus minus design hinge moment values for the $\bar{q}\alpha$ and $\bar{q}\beta$ conditions around the squatcheloid are shown for Mach = 1.25 in Fig. 17. The program value is 4 deg, and the acceptable region for nominal aerodynamics is 3-4.5 deg. With aerodynamic tolerances the acceptable region disappears. Figure 18 shows the deflection angles over which the load relief system could move the elevons to maintain both left and right outboard elevons within hinge moments of 400 to −270 K in.-lb. The possible elevon angles established required wind tunnel test conditions.

Engines Parallel

The SSME orientation in the pitch plane is necessary to fit the engines within the Orbiter aft structure and to orient the

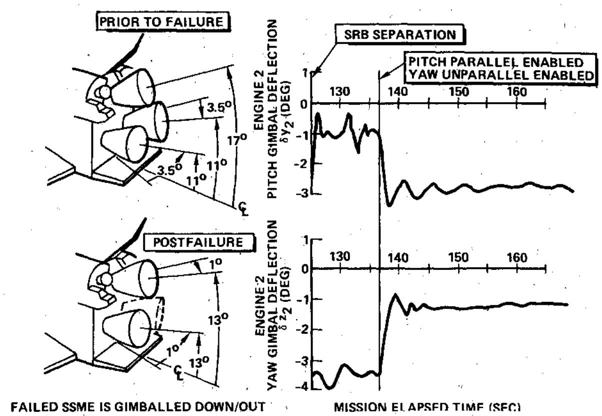

Fig. 19 SSME gimbal positions, pitch parallel and yaw unparallel.

null thrust vector an average of 12 deg toward the ET. The positive and negative 10.5 deg of SSME gimbal authority in pitch balance the conditions between SRB separation (requiring negative deflection) and main engine cutoff (MECO) (requiring positive deflection). In the yaw plane, the bottom engines are canted 3.5 deg. The performance effect of the engines not being parallel is significant. The FCS puts a yaw bias in these two bottom engines after thrust buildup. This yaw parallel is taken out if either of the lower engines has failed. After SRB separation, in the case of a lower engine failure, the upper engine and remaining lower engine are biased 3 deg toward each other to parallel the thrust vector in pitch. The thrust parallel performance gain for an abort critical mission is 600 lb. Figure 19 shows the pitch and yaw gimbal transient when the flight control reconfiguration is performed.

Conclusion

As the shuttle system design matured, the FCS was called on to accommodate vehicle constraints in order to avoid hardware or structural changes. The adaptability of the ascent G&C system succeeded in providing the Shuttle vehicle with stability, control, guidance, and navigation and, in addition, achieved total system solutions to configuration design.

Acknowledgments

This paper surveys the efforts of many people in several organizations. An acknowledgment to these contributors is recognized. Group participants in this effort included Rockwell International's Ascent Performance and Guidance, Navigation, and Control organizations; flight control development personnel at Honeywell, Inc., St. Petersburg; and other contractors and NASA centers under the direction of Johnson Space Center, Houston, Texas.