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Technical Challenges of Integrating the Space Shuttle

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ABSTRACT

The Space Shuttle is a complex flight vehicle comprised of four major elements: orbiter, external tank, main engines, and solid rocket booster.

Integrating the requirements, design, and verification requires resolution of challenging technical problems in flight performance, aerodynamics, aerothermodynamics, structural dynamics and loads, flight control, and propulsion.

The departure from typical cylindrical booster and spacecraft launch configurations complicates analysis and design. Techniques being used to identify and resolve technical problems encountered in integrating the Space Shuttle are discussed.

INTRODUCTION

The Space Shuttle is a unique flight vehicle because it is a hybrid airplane-spacecraft launched by a combination of three liquid propulsion engines and two solid rocket boosters. An appreciable advancement in state-of-the-art technology is not required; however, integrating primary elements of the Space Shuttle, each designed and manufactured by a different contractor, is formidable. Integrated analysis, design and verification are complicated by the unusual configuration asymetry—a drastic departure from the cylindrical configuration of most recent spacecraft and booster launch vehicles. Consequently, the detail required and the scope of conditions to be considered in the analysis to establish requirements for the design of the elements are very much greater than in previous programs. Major ground test programs are planned to verify the design of the integrated vehicle.

The Space Shuttle system will be capable of launching a variety of payloads into earth orbit from either the Eastern Test Range (ETR) at Kennedy Space Center or the Western Test Range (WTR) at Vandenberg Air Force Base. Maximum payload capabilities will be 65,000 pounds for an easterly launch from ETR and 32,000 pounds for launch into polar orbit inclinations from WTR. Nominal orbital altitude is approximately 150 nautical miles. The Shuttle also will be capable of retrieving payloads from such orbits and returning them to earth. In addition, the manned orbiter element of the Shuttle vehicle will be capable of functioning as a space laboratory for moderate duration missions. The orbiter provides accommodations and equipment for up to five mission specialists, as well as the normal flight crew of commander and copilot. An on-orbit stay capability of seven days is required, extended to 30 days during the operational phase of the program.

SHUTTLE VEHICLE/SYSTEMS DESCRIPTION

The Space Shuttle vehicle is comprised of four major elements: the orbiter, main engines (SSME), external tank (ET), and two solid rocket boosters (SRB). Overall vehicle configuration is illustrated in Figure 1. Figure 2 summarizes gross characteristics for each element and Figure 3 depicts a typical mission profile.

NASA's Johnson Space Center (JSC) is responsible for overall integration of the complete Space Shuttle flight and ground systems. The Space Division of Rockwell International is the prime contractor, supporting NASA in accomplishing the integration.

The Shuttle orbiter resembles a contemporary delta-wing aircraft. It houses the crew and payload and returns from orbit to a conventional horizontal landing. Three large (450,000-lb thrust) liquid oxygen/liquid hydrogen rocket engines mounted in the aft region of the orbiter provide propulsive thrust during ascent in addition to that provided by the SRB. Each engine is gimbaled in pitch and yaw to provide thrust vector control forces. Smaller rocket engines are also located in the aft region to provide final impulse for orbit insertion, orbital transfers or maneuvers and deorbit. Small rocket motors are located in both forward and aft regions for attitude control and stabilization. Aerodynamic surface controls include split elevons along the wing trailing edge; a split rudder along the trailing edge of the vertical fin, which also can be flared open to serve as a speed brake during descent; and a hinged body flap at the lower aft extremity of the fuselage to augment control during descent and landing approach. The entire external surface of the orbiter, except the windows, is protected by a reusable insulation to maintain acceptable structural temperatures under entry heating environments. NASA's Johnson Space Center is responsible for orbiter development. Rockwell's Space Division is the prime contractor to JSC to implement design, development, and fabrication of the orbiter.

The Shuttle external tank (ET) serves as the core of the launch vehicle and contains the liquid oxygen and liquid hydrogen propellants burned by the main engines during ascent. Liquid oxygen is located in the forward tank to maintain an acceptable center of gravity for the combined vehicle. A single large feed line (17-inch diameter) is routed from the bottom dome of each propellant tank into the aft fuselage of the orbiter to supply the main engines. A flight separation umbilical is located at the lower surface of the aft fuselage. The main engines burn out slightly before orbital velocity is achieved; then the system and structural attachments between the orbiter and ET are separated, and the ET follows an entry trajectory to impact in ocean areas where traffic is sparse. NASA's Marshall Space Flight Center (MSFC) is responsible for the external tank project and has selected Martin Marietta Corporation as the prime contractor for ET design, development, and fabrication. This effort is being accomplished largely at the Michoud Assembly Facility, New Orleans.

The solid rocket boosters (SRB's) provide the primary thrust during the initial portion of the ascent trajectory. The nozzle on each booster is gimbaled in both pitch and yaw to provide a portion of vehicle control forces during flight. The two SRB's are ignited after all three main engines have reached a satisfactory thrust and performance level; the vehicle then lifts off the launch platform. The SRB's continue to burn for approximately 122 seconds.

Nominal flight conditions at SRB burnout are an altitude of 143,000 feet, velocity of 4620 ft/s, and ground track of about 26 nautical miles down range from the launch site. The SRB's are separated from the ET by pyrotechnic release immediately after burnout; auxiliary rocket motors are ignited to ensure safe separation trajectories away from the ET and orbiter. A parachute system housed in the nose compartment of each SRB decelerates the SRB before water impact and ensures a controlled tail-down impact attitude. Expended SRB's will float and will be recovered and refurbished for subsequent use. MSFC is responsible for the SRB project and has selected Thiokol Corporation as prime contractor for the motors.

MSFC also is responsible for development of the main engines for Space Shuttle, with the Rocketdyne Division of Rockwell International as prime contractor. KSC is responsible for development of launch, landing, recovery, refurbishment and maintenance ground facilities and equipment. In addition to JSC and MSFC, NASA centers such as Langley Research Center and Ames Research Center have participated heavily in development of engineering data to support Shuttle vehicle design. This support effort has been concentrated on aerodynamics, thermodynamics, and structural dynamics.

The following paragraphs address some of the technical challenges in integrating the Space Shuttle.

POWERED FLIGHT ANALYSIS

Powered flight analysis is the analytical task which encompasses trajectory design and vehicle performance evaluation. Early in the program, the major concerns were optimization of individual element performance-related requirements (e.g., propellant loads, nozzle expansion ratio, thrust levels), configuration trade study support, and abort mode concept development. Now element designs have progressed and hardware fabrication is underway. The major concern of the trajectory designer is shifting to development of flight modes which recognize maturing understanding of the element and subsystem capabilities and limitations.

Finally, it is this function which continuously monitors and reports the performance capability of the system. Candidate changes in element design, element performance, flight modes or requirements which may either improve or degrade performance are also evaluated.

The challenge to successful total integration derives, first, from the multimission requirements (Figure 4). Two launch sites with substantially different energy requirements and significantly different design winds and natural environments are involved. Two trajectory constraints, maximum dynamic pressure of 650 psf and maximum load factor of 3g are also imposed. Furthermore, cost constraints required sizing the elements (SRB and ET) with minimum system performance margins. Intact abort requirements (one SSME thrust loss) for the orbiter dictate consideration of three different abort modes, depending on the mission phase. As in all programs, structural and heating constraints must be considered in trajectory design. The Shuttle program faces this problem also, with specific emphasis on the integration problem, namely, trajectory solutions to loads, heatings or performance issues must consider the constraints of four elements (orbiter, SRB, ET and SSME) and must provide the most equitable balance where design impacts cannot be avoided.

Ascent Profile and Abort Mode Integration

The requirement for total mission intact abort capability for the orbiter is a key driver in trajectory design, and must be a prime consideration in the design of ascent trajectories. In fact, as will be shown, a Shuttle reference trajectory is a set consisting of a nominal mission-completion trajectory and the abort-mode trajectories. Also, as will be seen on the energy-critical mission, it is the abort situation which determines the performance capability.

Figure 5 describes a trajectory set. A nominal trajectory and two abort modes are shown. The nominal trajectory accomplishes mission completion. The first abort mode available is a return to launch site (RTLS) and can be accomplished with one orbiter engine out, or with any other failure which does not either render the orbiter non-safe or degrades its performance. The second abort mode available is abort-once-around (AOA) and can be accomplished with no more than one orbiter engine out or any failure which does not cause a non-safe condition. An additional point to be noted is the effect of auxiliary propulsion system requirements on trajectory design and performance.

Each reference mission requires specific amounts of orbital maneuvering system (OMS) and reaction control system (RCS) propellants to be carried for orbit operations (Figure 4), in addition to propellant for ascent OMS burns, deorbit, and attitude control. OMS and RCS propellant may be utilized in engine-out abort maneuvers. With a large OMS load, as in reference mission 1, it may be easier to perform an abort than a nominal ascent. In mission 3A, the abort trajectory is critical for payload capability. Here, the abort performance may be further assisted by designing the ascent trajectory to favor abort at the expense of nominal performance. These are the key considerations in understanding the Shuttle ascent profile.

Figure 5 illustrates nominal ascent to a 100-n mi circular orbit. After launch and a brief vertical rise for tower clearance, the Shuttle rolls to an inverted attitude and executes an open loop tilt program to SRB staging.

The orbiter and ET continue powered flight for another six minutes (nominal) to reach a main engine cutoff (MECO) at near orbital speed. After ET separation and a short coast period for clearance, an OMS burn (3-5 minutes) raises orbit apogee to 100 n mi. A second OMS burn circularizes the orbit. Second-stage flight and subsequent maneuvers are guided by closed-loop, near-optimal guidance equations.

There are three intact abort modes. An early loss of an SSME requires a return to launch site (RTLS) using orbiter main propulsion to reverse the flight direction and place the orbiter in a position from which it can glide back to the primary runway. There is no post-MECO OMS usage. Modified entry conditions are within normal mission design capability. As the point of engine failure and abort initiation moves farther from launch, the RTLS maneuver becomes increasingly difficult because a larger velocity must be reversed with less remaining propellant. A point of last RTLS normally occurs about two minutes after staging, as noted in Figure 5.

Loss of an SSME after RTLS capability is exceeded requires an abort once around (AOA) in which a post-MECO OMS burn produces acceptable entry conditions during the first orbit. The reduced T/W with an engine out is compensated by running the remaining two engines at maximum power level, by using surplus OMS/RCS propellant pre-MECO, and in some cases by targeting to easier MECO conditions, although this will increase the severity of entry heating.

Additional intact abort modes are available depending on the mission and the point at which the abort maneuver must be initiated. Abort to orbit (ATO) attains nominal MECO using main propulsion and some OMS/RCS. The remaining OMS and RCS propellants may be sufficient to fly a multi-orbit alternate mission. Mission continuation (MC) also attains nominal MECO, but by the use of excess ET propellant that would be needed only for earlier engine failures. The nominal mission objectives are pursued after MECO.

Since intact abort capability is required continuously from liftoff through MECO, it is necessary to tailor maximum payload and trajectory design so that no gap exists between the last RTLS point and the first AOA point. In an abort-critical mission (e.g., mission 3A) these points will just meet at the abort mode boundary. Use of OMS/RCS propellant and use of dual MECO targets (easier for AOA) are techniques for maximizing performance. A third technique, more subtle in effect, is to purposely shape the trajectory from liftoff to mode boundary as if engine failure and AOA initiation are going to occur at the boundary. The effect is to loft the first part of the trajectory in anticipation of degraded T/W over the second part. When no failure occurs, the orbiter/ET continues on three engines from the mode boundary to nominal MECO conditions.

Baseline reference mission 3A affords an example of the application of these techniques in four steps, as follows:

	(1)	(2)	(3)	(4)
Trajectory shaping	Nominal	Nominal	Nominal	AOA
AOA MECO target	Nominal	Nominal	AOA	AOA
Abort propulsion	2 SSME	2 SSME + OMS/RCS	2 SSME + OMS/RCS	2 SSME + OMS/RCS
Resulting payload (lb)	26,300	30,200	30,300	32,000

The fourth step, fully enhanced abort performance, meets the mission payload requirement of 32,000 pounds. Without the intact abort requirement, the Shuttle could deliver 40,200 pounds to the same orbit.

Key Trades

Ascent performance analyses have evaluated trades as basic as the number of orbiter engines and details as fine as an engine startup delay. A quick summary will convey the scope of these efforts. A deeper discussion of recent performance problem areas will illustrate typical approaches to problem resolution.

Sizing and Design

- System performance margin requirements = 7000 lbs
- SSME mixture ratio = 6:1
- Liftoff T/W = 1.5
- Suborbital ET disposal

SRM Optimization

- Thrust shaping for performance optimization, loads, system flexibility
- Nozzle expansion ratio = 7.16 to minimize cost per flight

Trajectory Design

- MECO targeting to satisfy ET disposal requirements for land avoidance and low shipping density
- AOA MECO altitude = 55 n mi to optimize ET thermal protection requirements versus ascent performance

Control of Load Factor by First Stage Thrust Shaping

The first-stage thrust-time relationship must carefully balance performance against maximum dynamic pressure $\bar{q}$ and against excessive inertia loads which may occur in the high-g period prior to SRB burnout. The means to this end are the SRM grain design and throttling of the SSME's. The SRM grain design has the major influence on first stage performance because of the 4:1 thrust ratio compared to the SSME's. The general requirement calls for a drop in SRB thrust after about 20 seconds to control dynamic pressure and a gradual rise to a sensitive shoulder (maximum load factor of 3g) preceding grain tailoff (see SRB curve, Figure 6) to maximize performance.

During detailed evaluation of SRB thrust dispersions due to seasonal temperature variations, flight-to-flight variations, and development tolerances, it was seen that ET design loads incurred during the high load factor portion of first stage were exceeded based on the system baseline thrust curve current as of January 1975. Several solutions were examined, including ET structural beef-up, thrust curve redesign and reduced safety factors. It was shown that the thrust curve redesign coupled with SSME throttling (Figure 6) was the most attractive program solution considering cost, schedule, and system performance. The redesign involved shifting total impulse (lower thrust) from the high load factor area to the high $\bar{q}$ area (increased thrust) and then throttling the SSME's down to about 90 percent power level to control $\bar{q}$ to the specification value of 650 psf. A comparison of the original baseline and revised thrust curves is given in Figure 7.

Adopting SSME throttling as a standard first-stage procedure carries an additional benefit. In the mission planning phase, the throttle schedule may be adjusted to minimize dynamic pressure or to compensate for predictable T/W variations due to SRB batch dispersions (tag values), seasonal temperature variations, changes in aerodynamic drag and for payload weight.

Structural Load Constraints

A major integration activity concerns trajectory design to minimize element structural loads during the transonic flight regime. Three concepts are involved. First, the major structural constraints can be expressed as functions of $\bar{q}\alpha$ and $\bar{q}\beta$. These are shown as the boundary condition lines on Figure 8. Second, the capability of the flight control system to limit the exposure of the vehicles to maximum values of $\bar{q}\alpha$ and $\bar{q}\beta$ for combinations of design winds, gusts, failures (engine out) can be expressed by $\bar{q}\alpha$ - $\bar{q}\beta$ envelopes at Mach numbers of concern for each mission. Figure 8 shows typical $\bar{q}\alpha$-$\bar{q}\beta$ envelopes for the three reference missions at a specific Mach number. On the left side of Figure 8 are representative uncorrected trajectories, typically zero α and β.

Vehicle angular accelerations ($\dot{p}$, $\dot{q}$, $\dot{r}$) correlate well with $\bar{q}\alpha$ and $\bar{q}\beta$. The third concept is that of trajectory wind biasing. By providing steering commands for the no-wind trajectories to shift the centroid for each mission as shown, the design $\bar{q}\alpha$ and $\bar{q}\beta$ envelope is minimized and brought within the structural constraints.

The basic trajectories become the mission reference trajectories and are the basis for performance analysis, design trajectory development, and guidance logic development. The associated $\bar{q}\alpha$ - $\bar{q}\beta$ envelopes may be used directly in loads analysis, or six DOF trajectories, using the reference trajectories guidance laws, may be developed and utilized for loads development.

Flight Performance Reserve

To accommodate dispersions in vehicle systems and natural environment, it is necessary to provide extra propellant in the ET above that required for a nominal ascent. This propellant is flight performance reserve (FPR). Because of the 6:1 weight preponderance of LOX over LH_2 in the propellant load, it is beneficial to bias the LH_2 supply to reduce the probability of encountering a heavy LOX residual at cutoff, as may result from loading and mixture ratio errors. Taken together, requirements on FPR and fuel bias assure guidance cutoff (no premature depletion) in the presence of 3σ flight dispersions. Ascent performance studies determine FPR and bias by simulating as many as 40 independent dispersion sources and statistically processing the results.

The primary groundrule for Shuttle FPR is to accommodate dispersions on both nominal and intact abort trajectories. Thus, the payload critical mission 3A and the AOA trajectory mode are used as the basis for FPR calculations.

Current baseline values are 5,200 pounds FPR and 1,100 pounds fuel bias. The largest single contribution to FPR is the SRB web action time uncertainty. In total, SRB related dispersions contribute 37 percent, and main propulsion (including ET loading errors and fuel bias) contributes 51 percent. Uncertainties in aerodynamic properties, GN&C, winds, and inert weights contribute the remaining 12 percent. Periodic review of FPR dispersions focuses attention on big contributing factors and stimulates development of ways to reduce their effects.

Trajectory Design for Orbital Flight Test

The OFT program will demonstrate Shuttle flight worthiness and mission capability through a progression of six increasingly strenuous test flights. Early flights will maximize hardware performance margins by reducing requirements well below design levels, while later flights will expand requirements to levels representing design capability. In view of this approach, trajectory design for OFT missions assumes a different aspect from trajectory development for vehicle design, namely, how to provide necessary test conditions at minimum risk. Factors in trajectory design include seasonal factors (rain, wind, temperature) payload weight and cg location, entry and ascent aerothermal environment, and insertion orbit (tracking, deorbit, crossrange requirements). Maximum AOA abort capability is desired.

Current work on the first OFT flight illustrates a basic tradeoff: system margins cannot all be maximized at the same time. Rules for this initial flight stipulate a 100 psf reduction in max q (to 550 psf), but this can be obtained only by SSME throttling. Although throttling capability can be tested thoroughly prior to OFT it is an undesirable operational complexity for a first flight. Earliest AOA capability is desired, but this objective conflicts with the desired cool entry trajectory. Other proposed objectives and constraints pose similar conflicts which must be compromised in future trajectory design studies.

AERODYNAMICS

There are three major objectives in the aerodynamic development plan for the Shuttle integrated system.

The first is to support development of the overall system arrangement by establishing the aerodynamic and aeroload impacts of various arrangement candidates.

Another important objective is to provide continuing and maturing evaluation of the basic stability and control aerodynamic characteristics of the configuration for (1) the total launch vehicle, (2) the orbiter/ET and the SRB's independently just prior to SRB separation, then in various attitudes subsequent to separation, (3) the orbiter and ET just prior to ET separation and again in post separation attitudes.

The third important task is development of air loads data to support structural flight load analysis. The challenge is formidable since the complexity of the configuration renders conventional analysis methods of only superficial use and extensive recourse to the wind tunnel tests of very detailed models has been required to a greater extent than on previous programs, but accomplished with modest program funding.

Configuration Arrangement

Major considerations in the refinement of the baseline Shuttle configuration have been the relationship between c.g. travel, structural load paths, element weight impact, the resulting impact on the TVC requirements of the propulsion system and subsequent influence on the orbiter boat-tail design. These trade studies also entail optimization of the ET and SRB configurations. Natural characteristics of the baseline configuration provide a desired small positive stability margin from liftoff to SRB staging (Figure 9). Thus, aerodynamic key considerations in these arrangement trade studies was the influence of each option on aerodynamic performance in terms of drag, aero loads and aerodynamic effect on heating. Examples of these influences are shown in Figure 10.

The external tank nose design was changed from a conical shape to an ogive, reducing drag and SRB/ET interference. The SRB's were moved aft, reducing SRB/ET nose interference, thereby reducing drag and heating. The SRB's also moved circumferentially from 20 degrees to zero degrees above the horizontal, thus relieving orbiter wing aero loads and ET structural loads. The SRM nozzles were placed a minimum of 100 inches aft of the SSME nozzles to reduce plume impingement and heating problems. The design of the SRB skirt was influenced by weight versus drag considerations.

A study performed by MSFC was conducted to reduce the drag significantly from the present level. However, large fairings were required which are not attractive when evaluated in a trade study which considered not only drag, but also cost and weight.

Wind Tunnel Program

A key feature of the Shuttle wind tunnel program involves the cyclical acquisition of aerodynamic and aero loads data banks to establish a continuously maturing data base reflecting evolving details of element designs. Approximately 5,280 wind tunnel hours have been run to date. Approximately 2,660 more hours are planned in support of the integrated vehicle verification aerodynamic analysis cycle and loads evaluation. All wind tunnel tests are coordinated with, and approved by, the NASA Shuttle program management at JSC. The most important tests are run at Ames Research Unitary Tunnels. Significant support and supplementary tests have been provided by MSFC.

Past wind tunnel tests which have been run on the launch configuration simulated the full Mach number range. The models simulated only the major protuberances such as attach structure, external structural rings on the boosters, and propellant feed lines. Upcoming tests are being established to refine the aerodynamics and local flow field effects on air loads to account for all significant protuberances and to obtain' more data on Reynolds number effects, particularly on control surface hinge moments.

Results from recent power-off aerodynamic tests on the orbiter/ET and SRB separation determine force and moments on the elements alone and at various points along the separation paths. SSME plume effects (forces) on the SRB's during separation from the orbiter/ET and RCS plume effects on the ET during orbiter separation from the ET are among objectives of wind tunnel tests currently planned.

Determination of the power-on effects of the SRM's and SSME's on the base environment of the Shuttle launch configuration is complicated by the lack of vehicle axial symmetry and the multiple plume-flowfield interactions. To obtain an initial estimate of power-on base drag, data from several flight vehicles of different base configurations were utilized since no theoretical means was available to predict the base flow phenomenon. These data were combined and a total power-on curve was generated. The need for more accurate data compatible with the actual design became apparent because of extreme structural and performance sensitivity to variations in base AOA pressure loads and drag increments.

An extensive plume technology program was directed by NASA MSFC and a Shuttle plume test program was performed by Rockwell in 1973. Primary purpose of the technology program was to determine simulation parameters and methods of application to Shuttle-like vehicles to obtain useful base pressure data. Two simulation parameters were found to be significant to the design of Shuttle wind tunnel plume models. These are initial plume angle and plume shape. A method of applying cold flow test data to the full-scale Shuttle configuration was devised to implement the results of the plume technology program. Basically, a prototype plume for certain flight conditions is generated using the method of characteristics, then the plume shape and initial expansion angle are matched with a model nozzle design compatible with available wind tunnel facility pressure and mass flow limitations.

Data shown in Figure 11 represent a summary of current test results defining the power-on base axial force for each vehicle element in the presence of each other. The data has been corrected, using plume technology correlation methods.

AEROTHERMODYNAMICS

Shuttle vehicle elements (orbiter, ET, SRB, SSME) must survive hostile flight environments induced during ascent, such as skin friction from the surrounding airstream (aerodynamic heating), and convection and radiation from the propulsion systems' exhaust gases. The physical arrangement of vehicle elements dictates that they be treated in an integrated fashion because each element affects the induced environment of other elements.

At the outset of the Shuttle program, it was determined that the integration contractor would analyze the aerothermodynamics of the total system to obtain a unified analysis of flow field interactions between the elements and to eliminate redundant analyses and test programs among element contractors, enhancing cost effectiveness. To date, induced environment design criteria has been provided in support of all element preliminary design reviews (PDR's) and these criteria are being updated for element critical design reviews (CDR's).

Although the Shuttle flight regimes have been encountered on past spacecraft such as Apollo and Gemini, the geometrical complexity of the Shuttle vehicle and the propulsion system arrangement and characteristics make the prediction of induced environments a technically challenging task. Past flight experience on basic aerothermodynamic phenomena has been an invaluable aid in the analysis of the heating sources for the Shuttle vehicle. Analytical approaches to the solution of aerothermo problems are better known now. Nevertheless, the complex flow fields enveloping the Shuttle configuration pose problems of greater magnitude than for earlier spacecraft. As shown in Figure 12, surface flow patterns are quite complex, as recorded by the oil flow technique during a wind tunnel test at Arnold Engineering Development Center (AEDC) at M = 3.7.

Aerodynamic Heating

The prediction of aerodynamic heating for the Shuttle vehicle is highly dependent upon wind tunnel data to determine the complex flow field effects on heating and pressure distributions. These experimental observations are then correlated with well known theoretical analyses for simple shapes such as flat plates, cones, cylinders and spheres which are used to simulate various local regions of the elements. The wind tunnel data provides information as to how the simple theories should be externally factored to arrive at the observed heating level for the complex Shuttle geometry. These factors are commonly termed interference factors and include both overall proximity effects of combined elements and the localized protuberance effects on each element. Correlating wind tunnel data with theories proven by past flight experience permits the tunnel data to be scaled to flight conditions with an adequate degree of confidence. Distribution of maximum heating on the upper centerline of the ET barrel section for the design mission 3A trajectory is shown in Figure 13.

Tests have been conducted on the first stage configuration from M = 2.0 to 5.5 and on the second stage from M = 5.3 to 21. SRB staging occurs at approximately M = 4.6. Models used in these tests have been extensively instrumented to record details of the complex heating patterns. The primary aeroheating model is instrumented with over 1000 thermocouples. A key accomplishment in this area is the resolution of various interpretations of interference heating phenomena into a unified approach. This approach recognizes that increases in heating due to interference effects such as orbiter nose shock impingement on the ET and SRB are caused by both induced flow transition from laminar to turbulent state ($\dot{q}_{turb} \gg \dot{q}_{lam}$) and local pressure rise. Results of this analysis reduced predicted interference effects by a factor of 5, thus avoiding overly conservative design criteria.

At this point in the program the major aeroheating environments have been defined. Refinements to these data yet to be accomplished include:

1. Effect of spray-on foam insulation (SOFI) roughness on ET heating
2. Localized protuberance effects on ET, SRB and orbiter
3. Effect of revised trajectory shaping on Shuttle heating and thermal protection system (TPS) requirements

Plume Heating

The prediction of plume induced environments is also very dependent on experimental data to determine convective heating and pressure distributions over the base regions of the elements. In this case, there is a reversed flow caused by rocket plume interaction. It is very difficult to describe analytically and simple shape analogies are not available as with aeroheating. Consequently, no reliable general theoretical analyses with which to correlate the data are available. The test data are scaled directly to flight using relationships shown to be adequate for design by comparison with Saturn flight data.

Presented in Figure 14 is a photo of the 0.0225 scale model used to obtain plume base convective heating data. The SSME and SRB systems are hot-firing and use the same propellants as the full-scale vehicle. Tests with this model have been conducted at M = 2.9 to 4.5, and simulated altitudes of 80,000 feet to 140,000 feet for the first stage configuration and 140,000 feet to 170,000 feet for the second stage. For altitudes above 170,000 feet, a larger partial model of the orbiter and SSME's has been tested in vacuum chambers at simulated altitudes up to 360,000 feet. During this program, plume base convective heating testing techniques have been significantly improved, resulting in the highest quality short-duration plume heating data ever obtained. This has had a marked effect on the accuracy of design environments for the Shuttle vehicle and has resulted in elimination of potential thermal problem oversights. These tests have been run in facilities at MSFC, Calspan Corporation, and Lewis Research Center. Further tests will be performed at JSC.

In addition to base convective heating, the SSME and SRB plumes emit thermal radiation which affect the Shuttle aft end, primarily. Radiation from the test model plumes cannot be scaled to flight, therefore other approaches must be used. Correlations of Titan ground and flight test data are used to predict SRB radiation, and an analytical model has been developed to estimate SSME radiation. This SSME radiation model has been checked against Saturn S-II data at high altitudes, but no large oxygen/hydrogen-powered boosters have provided vehicle propulsion starting at, or near, sea level (as Shuttle does). Hence, verification of the SSME radation analysis at low altitudes remains to be done. Future flight hardware ground tests of the SSME and the SRB will provide radiation data with which to verify analytical methods now employed for design.

Shown in Figure 15 are the predicted heating histories due to various sources for the center of the ET LH_2 tank aft dome.

Important considerations remaining in this area include:

1. Verification of radiation math models with ground test data
2. Effects of launch pad/plume interactions on radiation heating at liftoff
3. Localized protuberance effects on SSME nozzles
4. Effects of reaction control system (RCS), orbital maneuvering subsystem (OMS), and SRB separation motor plume impingement on the SSME, ET, orbiter, and payloads

Development flight test instrumentation will provide data on both aerodynamic and plume heating that will be used to update the environment prediction math models for certification of the Shuttle vehicle for its operational envelope.

FLIGHT CONTROL AND SEPARATION

The flight control system (FCS) for the ascent flight of the Space Shuttle integrated vehicle utilizes thrust vector control of the three Shuttle main engines (SSME) and two SRB nozzles to control the vehicle. The control system responds to commands from the guidance system which are preprogrammed for first stage ascent and are determined from closed loop guidance for second stage ascent as required to reach the SSME burnout conditions.

The control system uses attitude information from the inertial measuring unit (IMU) and attitude rate information from the rate gyros to steer and stabilize the vehicle. During the high aerodynamic loading phase of the ascent flight; body-mounted accelerometers provide a load relief function to compensate for design wind and gust conditions. The control system incorporates gain variations based on time or velocity, wind bias programming and control effector mixing logic. These relationships and functions are described in Figure 16.

SRB separation is accomplished by pyrotechnic release of the four structural attach points, three aft and one forward on each SRB. The relative velocity between the orbiter/ET and SRB's is provided by forward and aft clusters of four booster separation motors (BSM's) on each SRB.

ET separation is accomplished under nominal flight conditions by pyrotechnic release of the three structural attach points and translation of the orbiter away from the ET using the reaction control system (RCS). For separation under return to launch site (RTLS) abort conditions, the orbiter/ET performs a powered pitchdown maneuver prior to separation to allow separation under aerodynamic conditions.

From the onset of the Shuttle program, the development of the ascent flight control system and the separation systems for the SRB and ET faced formidable technical challenges which included:

1. Asymmetric thrust causes high lateral drift which complicates liftoff and facility clearance
2. Mission requirements to roll through large angles shortly after facility clearance from the tail south launch configuration at KSC to the eastern ascent azmuths
3. Large center of gravity excursions in both "x" and "z" coordinates as shown in Figure 17
4. Complex loads during the high dynamic pressure portion of flight, requiring acceleration biasing and orbiter elevon position changes for load relief
5. SRB thrust mismatch and SRB gimbal limitations during tailoff which requires control logic mixing and makes switching from first stage to second stage control more critical
6. SRB separation without contact and prevention of separation motor plume impingement on the orbiter thermal protection system
7. ET separation following a powered pitch down maneuver with attitude and $\bar{q}$ constraints during the RTLS abort
8. Obtaining adequate control response and stability characteristics in the above situations including the effects of system dispersions, subsystem failures, and complex structural bending

Important aspects of several of the above challenges are discussed below.

Liftoff

Development of the system configuration resulted in an interface between the SRB aft skirt/nozzle and the launch facility as illustrated in Figure 18 where it is seen that the SRB nozzle is close to the support post. The nominal thrust SSME and SRB vector alinement during trimmed, vertical attitude liftoff results in a significant drift toward the posts on the north side of the SRB. Additional drift in this direction caused by vehicle dispersions and wind results in undesirable configuration compromises to maintain a safe clearance margin. Therefore, a trim function is added in the FCS logic to deflect the SRB nozzles in a direction which will reduce the northward drift sufficient to provide the clearance shown in Figure 18. This results in a mis-trimmed vehicle in pitch for about two seconds. The vehicle pitch rotation is minimal, and the trimmed vehicle control and vertical attitude are maintained with no problems.

Ascent to SRB Staging

Vehicle guidance during this phase has an open loop attitude command versus velocity history. Key functions of the FCS are to trim the large cg excursions (Figure 17), aerodynamic moments and system disturbances, and to provide steering commands to the TVC of both propulsion systems to follow the attitude versus velocity history. A very significant function is to limit the external air load environment ($\bar{q}\alpha/\bar{q}\beta$) to an envelope defined by integrated vehicle structural constraints under design wind and gust environments with specified vehicle dispersions (Figure 19). This is the first booster flight system to incorporate such a function, although analytical studies of load relief systems date back to the pre-Apollo period. The key issue here is that the loads condition to be controlled is the $\bar{q}\alpha$-$\bar{q}\beta$ envelope, together with correlated body rates. The vehicle senses the on-set of excessive $\bar{q}\alpha$-$\bar{q}\beta$ conditions through sensitive vehicle mounted accelerometers which sense pitch and yaw acceleration. The signals are blended with the attitude and rate commands as shown in Figure 20 which is a simplified pitch channel block diagram. As the critical load condition period is reached, the attitude command gain, K_θ, is reduced, and the load indicator gain, K_Z, is increased. The signals are carefully filtered to avoid exciting vehicle bending modes. The sensitive accelerometers may react, not only to external accelerations, but to vehicle modes or random vibration. The filters must be carefully designed to avoid passing these extraneous signals, but must be sure to pass the desired load indicator signals. Two additional issues are being dealt with by the FCS designer. The high gains required in the load relief channel tend to decrease stability margin. Gain levels to achieve high stability margins in the attitude and rate channels tend to defeat the load relief functions. The flight control system has been designed to optimize the balance between the load relief function and stability requirements so as to satisfy the required structural constraints with maximum flight control stability margin.

The filters must be carefully designed to avoid passing these extraneous signals, but must be sure to pass the desired load indicator signals. Two additional issues are being dealt with by the FCS designer. The high gains required in the load relief channel tend to be destabilizing. Gain levels to achieve high stability margins in the attitude and rate channels tend to defeat the load relief functions. The flight control system has been designed to optimize the balance between the load relief function and stability requirements so as to satisfy the required structural constraints with maximum flight control stability margin.

Elevon Load Relief

A significant ascent flight control design problem arises from aerodynamic hinge moments on control surfaces, most importantly the elevons. No single fixed elevon deflection was found to be adequate at all Mach numbers to maintain aerodynamic hinge moments within a range acceptable to the structural capability of the elevons, their actuators, and the actuator/wing attachments. Until recently, it was planned to deflect at least the outboard elevon panels according to a pre-programmed load-relief schedule during first-stage ascent and to include provisions for adaptive support if necessary. Recent studies have shown that aerodynamic and system uncertainties exceed the load-relieving capability of this approach. An adaptive feedback which will modify the deflection schedule in the event that sensed elevon hinge moments approach design limits will be implemented in the FCS.

SRB Tailoff Thrust Mismatch and Gimbal Limitations

Variations in thrust profiles between the two SRM's are expected, and the flight control system must compensate for the unbalanced thrust forces. This is particularly important during the SRM tailoff period when there could be a significant thrust mismatch between the two SRM's. Also during this period the flight control system must be ready to accept the loss of any one of the three main engines and must also make the transition to second-stage control.

The problem is complicated by the gimbal point shift of the SRM nozzle as a function of the chamber pressure. As the chamber pressure changes, the nozzle and gimbal point moves either forward or aft and extensions and retractions of the thrust vector control actuators result in variations in actual nozzle deflections. The flight control commands provide for the gimbal point shift by limiting gimbal deflection commands.

SRB Separation

The selection of the baseline SRB separation system involved system tradeoffs where impacts to the SRB, the ET, and the total system were evaluated. The structural methods included ball/socket fittings, pin-jointed links, and aft hinges. Techniques for providing relative orbiter/ET-SRB accelerations included pistons, aerodynamic surfaces, and rocket motors. The current baseline was easily the highest weight and most cost-effective of many options examined. The key issue, once this baseline was selected, was optimization of separation motor orientation, selection of propellant characteristics which would provide adequate separation impulse forces, minimize installation difficulties, and minimize separation plume damage to the orbiter TPS. Extensive propulsion tests were run in a high-altitude chamber (AEDC) of a range of solid propellants impinging on various TPS samples to evaluate the relative damage potential of various solid propellant additives (e.g., aluminum oxide) and propellant temperatures. Then parametric dynamics studies were accomplished to determine the impulsive energy-orientation combination which would satisfy safe separation requirements with minimum weight, and finally the thrust time curve of the separation motor was optimized together with iterations of the orientation to minimize the plume damage to the orbiter TPS. The final design requirements are shown in Figure 21.

ET Separation

The ET separation system is designed to provide safe separation for exo-atmospheric conditions associated with nominal and ATO and AOA aborts. In addition, safe separation at dynamic pressures up to 10 psf is required for RTLS aborts. Of prime importance in ET separation is clearance of the orbiter/ET fluid and electrical umbilicals. The umbilical separation planes are located approximately 3 to 5 inches inside the orbiter mold line. The exo-atmospheric separations are relatively passive, with disturbances generated only by the system itself.

However, during a RTLS separation, aerodynamic disturbances are present and the unstable characteristics of the ET are taken advantage of in providing the required separation motion. As depicted in Figure 22, the ET has a negative pitch rate as it moves down and aft during separation. Safe separation during this mode is possible for a wide range of initial conditions, which allows selecting target separation conditions, facilitating post separation orbiter recovery.

STRUCTURAL ANALYSIS AND DYNAMICS

The complexity of the Shuttle structural configuration—four bodies in an nonaxisymetrical arrangement with a high degree of aerodynamic sensitivity, together with two parallel burning propulsion systems including throttling capability—provides a challenge to the loads analysis community in terms of the detail of math models required and the scope of dynamic situations to be assessed.

In addition to the basic static and dynamic loads, pogo effect is of significant concern because of feed line lengths and soft structural interfaces. Analysis of acoustic environment and its impact on the flight vehicle and ground facility design demands attention because of the high acoustic levels anticipated and also because of the potential sensitivity of present and planned payloads to vibratory environment.

Besides the challenge offered by the baseline vehicle design, the mission profile results in a wide variety of structural loading conditions. A typical mission profile for Space Shuttle is given in Figure 3.

Engine ignition and liftoff produce severe acoustic levels (163 dB overall sound pressure levels over the aft region of vehicle). Significant dynamic transients result from the engine thrust buildup combined with wind gusts and the "twang" of vehicle release from the launch platform. Extensive dynamic analyses have been performed to determine vehicle modal response and structural loads resulting from the combined dynamic transient inputs. Stiffness of the launch platform and support pedestals to the SRB aft skirts was considered in these analyses, as well as flight vehicle stiffness characteristics.

The region of high dynamic pressure during ascent produces critical loading on portions of the orbiter fuselage and aerodynamic surfaces and on the interface attach structure between the elements of the flight vehicle. Air loads in pitch and yaw directions are imposed as a result of winds, wind shears, and gusts. The regime of high loading extends from approximately Mach 0.9 to Mach 1.5. Again, extensive analyses are required to determine an envelope of structural design loadings for the vehicle that will encompass the range of Mach number, angle of attack combinations in pitch and yaw, and thrust vector forces resulting from control system responses to atmospheric disturbances.

Shortly before SRB burnout, the vehicle maximum longitudinal acceleration of 3 g's will be achieved. This produces critical loadings on the ET propellant tanks because of the inertia head of the propellants and on the ET-to-SRB forward attach structure. Aerodynamic loads are relatively small at this time.

SRB staging produces thrust/inertia loading combinations that are critical on portions of the ET and orbiter fuselage and thrust structure. At this time aerodynamic loads are negligible; however, aerodynamic heating is significant on the ET and SRB's. Water impact loads on the SRB's are critical for design of the forward and aft skirt and nozzle assemblies.

Thrust and inertia loads during second-stage boost (orbiter and ET only) produce some critical loads for orbiter thrust structure and portions of the fuselage. There are no aerodynamic loads during this flight phase.

Loads on the orbiter during ET separation, orbit insertion, and orbital operations are relatively benign. In general, critical design loads occur only on local structural regions associated with support of auxiliary rocket motors, docking probes and hatches, and payload-handling mechanisms. However, the crew compartment, which is maintained to a sea-level atmosphere, will experience maximum pressure differential for a sustained period. Significant temperature gradients, some in excess of 300 F, may be induced on the orbiter structure during fixed-attitude holds associated with on-orbit operations. These temperature distributions establish initial conditions prior to entry that aggravate the severity of thermal gradients which must be considered in combination with flight loads during the descent and landing approach phases. It is also necessary, for adequate mission flexibility, to consider random orientation of the fixed-attitude hold; i.e., tail sun, top sun, bottom sun, etc., in establishing preentry temperature distributions.

Loads on the ET after separation are of some concern in that loading conditions resulting from high thermal loads during ascent and entry, together with internal pressure and air loads during ET entry, must be considered in predictions of ET break-up altitude and fragment dispersion analysis.

Very high aerodynamic heating flux is encountered on the orbiter during entry over the Mach 25 to Mach 12 regime; however, structural loads are relatively small during this portion of the unpowered descent. Surface temperatures on the thermal protection system will range from approximately 2600 F on nose and wing leading edges to 650 F in more sheltered regions. Thermal stresses in the thermal protection system (TPS) resulting from these temperatures and associated severe thermal gradients must be considered in the design and material selection for TPS components. Because of the insulating effectiveness of the TPS, the orbiter primary structure will experience maximum temperature considerably later; some regions will not reach maximum temperature until after landing.

Structural design conditions during the aerodynamic portion of descent flight are similar to those of conventional aircraft. This involves definition of a velocity/load-factor envelope to set limits of required structural capability; flight anywhere within this envelope is permissible, based on structural constraints. The load factor limits have been identified at +2.5 g's and -1.0 g; this permits adequate pullup from the entry trajectory and energy-management maneuvers required to ensure successful dead-stick approach and runway touchdown. This phase of the mission profile results in critical design loadings on orbiter wing, vertical tail, aerodynamic control surfaces, and portions of the fuselage. Significant thermal-induced loads must be considered in combination with aerodynamic and inertia forces for most structural regions.

The orbiter is designed for a relatively "hot" landing to minimize wing area and vehicle weight. Touchdown velocity is 180 knots, and maximum sink rate is 9.6 ft/s. Significant dynamic transients occur during touchdown and landing rollout, and extensive dynamic analyses have been performed to determine orbiter structural response and loadings for this phase. Critical design loads are induced on the forward fuselage and supports of major mass items in addition to the landing gear and its local support structure. Again, thermal-induced loadings must be considered in combination with landing dynamic loads.

Shuttle program task assignments require that element contractors develop design loads for those mission segments where the element is not operating with the integrated system (e.g., SRB water impact loads). The integration contractor (Rockwell International), is responsible for analyzing the integrated vehicle (i.e., from final assembly to ET separation). Therefore, the following discussion will emphasize loads and dynamics analysis associated with integrated vehicle operation.

Vehicle Modes and Loads

Idealized beam models and classical methods of analysis are inadequate to predict structural behavior accurately, considering the complexities of the mated vehicle configuration and orbiter structural arrangement. Therefore, a large-scale finite-element mathematical model of the vehicle serves as the heart of the structural analysis approach for Space Shuttle. It permits a detailed representation of local load paths, stiffness characteristics, and description of the basic three-dimensional characteristics of the structure.

ASKA (automated system for kinematic analysis), developed by Prof. Argyris and colleagues, is employed by Space Division, as well as other aerospace divisions of Rockwell International, as the primary finite-element analysis system. This analytic tool serves as the foundation for an integrated system approach to determine vehicle modal characteristics, structural response to dynamic transients, aeroelastic effects, and detailed internal load distributions for selected structural design or assessment cases. This integrated system, called MMLS (model-modes-loads-stress), is illustrated schematically in Figure 23.

A detailed finite-element "stress analysis" model is developed for each element of the vehicle based on current knowledge of basic configuration geometry, internal structure arrangement and sizing, and mass distributions. These characteristics evolve from mission performance requirements and preliminary design development. Figure 24 presents an example of this type of finite-element model for the orbiter, ET and SRB. Approximately 7,000 node points are contained in the orbiter portion of the model. The total model is substructured into individual nets to facilitate preparation, checkout, and execution of the large-scale computer program. This level of detail is required to develop accurate and directly usable internal load distributions throughout the structure; however, it is not necessary or practical to use this large a model to determine vehicle modal characteristics and dynamic responses. Therefore, the stress model is mathematically collapsed to a simpler dynamic model. The dynamic model for the complete vehicle contains 750 dynamic degrees of freedom. Stiffness and mass matrices for the dynamic model are used to extract natural mode shapes and frequencies of the vehicle. These modal results form the basis for evaluation of structrual loads resulting from dynamic inputs and support related disciplines of flutter, aeroelasticity, flight control, pogo, and other structural dynamic analyses. As many as 30 to 50 modes may be used, depending on the requirements of a specific analysis.

External loadings in terms of air loads, engine thrust loads, and so on, are applied as lumped forces at appropriate node points on the dynamic model. If the condition being analyzed is a dynamic event, a time-dependent description of these forces is required. The mass matrix provides "unit" inertia loads at each node point, and the actual inertia forces are determined as a function of vehicle rigid body and modal response to the external force system. The vehicle trajectory and flight control system characteristics affect the definition of the external force system and vehicle response, so it is necessary to integrate data from these disciplines, as well as aerodynamic pressure distributions, into the MMLS cycle.

The net forces at each node point are integrated by the computer program to develop conventional shear, axial force, bending moment, and torsion loading distributions. If a dynamic event is involved, these loading parameters are displayed as time-history summaries for selected stations. These results are used to select, from the large number of flight loading conditions considered, cases potentially critical for design of the various structural regions. Load data in this form are summarized for application to design of the ET and SRB's.

Internal loads for the element structures are developed for selected cases determined by the survey of external load results. The net external nodal forces determined from solution of the dynamic model are apportioned to the finer-grid node points of the stress model. Internal forces at each node point are determined by the ASKA system based on considerations of static equilibrium and elastic strain compatibility. This is equivalent to solving a structural distribution problem with several thousand redundants for each load case. Internal loads from temperature gradients over the structure also are significant; therefore, thermal models for temperature distributions are analyzed by specifying appropriate temperatures at each node point as input data. The ASKA program can determine the corresponding unrestrained thermal deflections between node points and solve for internal forces at each node to maintain static equilibrium and strain compatibility. The internal loads results for thermal and/or load cases serve as the basis for detailed structural strength and stability analyses and sizing of the individual structural members. The structural development of Space Shuttle is an iterative process involving major cycles of the MMLS system. Each update improves the fidelity of data as results of the preceding cycle are reflected in refinement and better definition of the structural characteristics. The same evolution is true for many of the external input parameters such as configuration geometry, aerodynamic data, and flight control system characteristics. External and internal loads recently completed and issued represent the fourth major MMLS cycle since the start of the contract.

The major virtue of the finite-element approach, i.e., the ability to describe detailed structural and loading characteristics, imposes major problems, however, because of the enormous amount of data, both input and output, that must be processed. The most recent MMLS cycle examined approximately 600 discrete loading conditions to ensure adequate coverage of structural design requirements over the entire flight mission profile. When this is considered in combination with several thousand nodes on the structural model, it is apparent that tens of millions discrete input and output data points are involved in a complete load cycle. It is obviously impractical to review or process this quantity of data manually, so computer programs for various portions of the cycle must be written to extract required data directly from magnetic storage. In addition, a number of auxiliary computer programs have been developed to automate preparation of input data, to search output data to determine critical conditions for additional evaluation or for presentation to the user, and to process output data to a format of direct use to the designer or stress analyst.

Pogo

One of the major problem areas experienced by large pump-fed liquid rocket launch vehicles has been a longitudinal instability termed pogo. This instability involves the participation of the fluid-feed system, the engine and the vehicle structure. An instability of the pogo loop may be illustrated in (Figure 25) as beginning with small vehicle accelerations which produce perturbations through the propellant tank, feedline and turbopump supports into the propellant feed system pressures and flow rates, which in turn cause thrust oscillations resulting in increased vehicle oscillations.

In the past, these phenomena have been given only cursory attention in the early vehicle design phases and thus the pogo susceptibility has not been evident until late in the vehicle's development or during the early flight test programs. The results of this approach induce potentially serious design, cost, and schedule impacts.

An overall approach has been adopted to define the Space Shuttle's pogo susceptibility early in the design phase so that suppression and verification testing can be incorporated in the basic vehicle design and development program. Specific pogo suppression accumulator-type devices already have been designed for the first flight article.

Prior rocket vehicles had axisymmetic configurations requiring only longitudinal degrees of freedom to be considered. The Space Shuttle vehicle (SSV) is an assymmetrical configuration which complicates the analytical solution in that coupling of motion in all three orthogonal axes must be considered within the region of potential pogo susceptibility of 1.5-40 Hz. This unique structural model includes the complexity of the LO_2 and LH_2 propellant tanks with tilted fluid surface effects, a three-dimensional hydroelastic phenomenon. Detailed definition is similarly required of the local thrust structure and attachments for the propellant feed system.

Historically, solid rocket motors have been dynamically modeled as a lumped propellant mass in an elastically supporting structure. Since the SRB's are an active propulsion system together with the SSME's during early powered flight, a more realistic description of the propellant mass as a viscoelastic or elastic system is necessary. It is predicted that approximately 80 vehicle system vibration modes—20 of which will have high gain—will exist within the anticipated frequency range of pogo susceptibility (1.5-40 Hz).

The long LO_2 SSME propellant feedline (approximately 120 feet) with multiple supports, branches, elbows, and internally tied bellows requires a level of analytical modeling heretofore not encountered in the prediction of boost vehicle pogo. Approximately seven fluid/structural vibration modes are within the frequency region of pogo susceptibility. Thus, a definite potential exists for coalescence with the high gain vehicle structural modes.

The pogo analysis problem for the Shuttle is magnified in that the SSV is the first boost vehicle to use both solid and liquid propulsion systems simultaneously during early boost. Interaction mechanisms between these propulsion systems must be isolated and dispositioned to assure system stability. Potential pogo-inducing mechanisms are oriented toward the common structure shared by the two systems. Mechanisms to be evaluated include vehicle structural vibration modes coalescing with solid-propellant motor-cavity acoustic modes and variations in motor burning characteristics due to vehicle induced environments, e.g., perturbations in propellant strain, local gas velocities, temperature, and pressure. The SSME's are liquid bi-propellant (LH_2, LO_2) systems with orbiter-mounted low-pressure turbopumps and engine-mounted high-pressure turbopumps. The orientation of these pumps permits relative motion between the turbopumps—a potential for increased flow perturbation into the engine.

Additional system complexities in evaluating SSV pogo susceptibility includes the potential coupling of the flight control and flutter sensitive modes.

Early in SSV development, a plan of action (Pogo Prevention Plan) was developed by the integration contractor in concert with the SSV Pogo Integration Panel, detailing analytical development and test verification requirements necessary to quantify the SSV pogo susceptibility. The panel chaired by the NASA, includes associate contractors, and consultants in specific technology areas. The basic philosophy of the plan is to incorporate a pogo suppressor into the baseline vehicle design, predict the pogo stability by analysis, verify the math models by component, subsystem tests, major ground tests (i.e., MPTA, MGVT), and analytically predict adequate stability margins for the first vertical flight. Final verification of pogo stability margins and the combined system stability model will be accomplished during development flight tests. This approach is mandatory, since a large rocket-propelled vehicle cannot be demonstrated to be pogo-free by ground test due to the complexity of interaction mechanisms and inability to duplicate flight boundry conditions.

Figure 26 outlines the analytical development tasks and associated tests scheduled for verification of the pogo models.

Consistent with tasks outlined in the Pogo Prevention Plan, analytical studies and development tests were conducted to select the optimal location, type, and size of suppressor. Preliminary pogo susceptibility and sensitivity analyses indicated the inlet to the high-pressure oxidizer turbopump on the SSME would effectively suppress pogo instability.

Active and passive suppressor concepts were tested by MSFC and Rockwell Space Division, and both suppression systems functioned satisfactorily. Due to the simplicity of design and cost effectiveness, the passive suppressor (accumulator) was selected as the baseline suppression system for application in the SSME design.

Analytical development and detailed test planning tasks are being performed. Implications of the refinements on the vehicle pogo stability are being assessed to support major program milestones (e.g., PDR, CDR and FMOF). Selected independent model analysis, test data analysis, and data interpretation are being conducted by the integration contractor to assure adequate element and system models are available for inclusion in the system pogo stability model.

PROPULSION

The Shuttle main propulsion system (MPS) is an integrated system which includes three elements: the external tank (ET), orbiter, and Space Shuttle main engines (SSME's). The integrated MPS system is shown schematically by Figure 27. The MPS presents a significant integration challenge, not only because of the physical arrangement of the systems, but also due to the fact that three elements are being designed and built by different contractors.

Liquid hydrogen and liquid oxygen propellants are contained in the external tank which also supplies propellant lines, gaging sensors, and overpressure venting.

The orbiter provides propellant lines, prevalves, pressurization control, hydraulic power, electrical power, engine commands, and engine mounting accommodations.

The Space Shuttle main engine (SSME), Figure 28, being developed by the Rocketdyne Division of Rockwell International, represents a significant advancement in rocket engine state of the art due to its high chamber pressure (3000 psi), its wide thrust range (50 to 109 percent of 475K nominal vacuum thrust), the engine mounted computer for closed loop thrust and mixture ratio control, and its reuseability capability. The SSME design was begun well in advance of the orbiter and external tank design to insure that preliminary flight certification of the engine was accomplished prior to initial orbital flight tests. As a result, SSME design interface criteria were based upon very early Shuttle vehicle configuration studies. This necessitated emphasis on SSME schedule and cost impacts during trade studies conducted for interface problem resolution. Some of the significant challenges met in designing the main propulsion system and resolving interface issues are discussed below.

Hydraulics

The SSME utilizes five hydraulically actuated valves for thrust and mixture ratio control. Hydraulic power for these valves is supplied by the orbiter hydraulic system, which also supplies hydraulic power for orbiter components such as elevons, rudder and thrust vector controls. Engine hydraulic pressure requirements are critical due to the rapid valve response required for engine control. A pressure transient analysis was performed using a mathematical model to describe the orbiter and engine induced transients. It was determined that large orbiter thrust vector control actuator deflections resulted in unacceptable engine interface hydraulic transients. The transients (supply pressure decrease coupled with a return pressure surge resulting in a low delta pressure available) caused the engine control system to sense the condition as a hydraulic supply failure. Since this condition was likely to occur in each flight, it was deemed unacceptable and corrective action was required. Trade studies were performed and solutions such as a dedicated engine hydraulic system, redesign of the engine control system, and supply and return line accumulators were evaluated. Accumulators were selected to minimize cost, schedule, and weight impact to the overall Shuttle system.

Electronics

Significant interface problems also were encountered in the electronics system. Engine control is by means of dual redundant programmable digital computers mounted directly on the engine. The orbiter is required to provide engine start, shutdown, and thrust level commands to the engine in the proper digital word format and with proper timing to insure desired engine responses. Since the orbiter general-purpose computer and the engine controller computer characteristics are not identical, an engine interface unit is used to convert input/output data into a compatible format. Continued coordination is necessary to insure software compatibility between the engine controller, engine interface unit and the general-purpose computers as the program matures.

Problems were also encountered in the sensitivity of the engine controller to transients in orbiter-provided electrical power. This problem was resolved with design changes in the orbiter to minimize transients and changes in the SSME controller to decrease the sensitivity.

Propellant Feed System Fluid Dynamics

The propellant feed system supplies liquid oxygen and hydrogen propellants from the ET through the orbiter to the SSME's. To satisfy SSME requirements, the propellants must be supplied at the proper flowrates within certain pressure and temperature limits.

The design of the Shuttle main propulsion propellant feed system has to account for a large number of variables and transient conditions. For example, SSME requirements have to be met while fully draining the ET to minimize residuals, and the pressure slumps and surges created by accelerating and decelerating the relatively incompressible propellants in the feedlines must be accommodated without adversely affecting the SSME start transient or exceeding structural limitations. To select a system design, the requirements and characteristics of all three elements were integrated into a systems analysis. The complexity of the analysis required the development of several computer programs. Results of the analysis were evaluated in conjunction with such practical considerations as minimizing weight and cost to determine an optimum configuration.

LH_2 Recirculation System

The LH_2 recirculation system chills down the SSME's and provides the required LH_2 temperature conditions at SSME start. Motor operated pumps within the orbiter draw LH_2 from the ET through the feedlines and pump it through the SSME's. The LH_2 is returned to the ET through the recirculation line.

The quantity of LH_2 recirculation required depends upon the temperature of the fluid extracted from the ET and the amount of heat input to the LH_2 from the ET, orbiter and SSME's. Each element has a different LH_2 system insulation configuration due to environmental and economical considerations. Also, since the LH_2 replenish flow from the facility is transported to the ET through the recirculation return line, back pressure effects created by the replenish flow tends to reduce the recirculation flowrate through the SSME's. A satisfactory design solution to the element interface requirements has been established.

Antigeyser System

The antigeyser system must suppress geysering to prevent LO_2 line damage under all operational conditions while also producing temperatures within the orbiter, SSME and ET LO_2 lines to support SSME requirements. This is accomplished by providing a 4-inch diameter line in parallel with the 17-inch main LO_2 feedline as shown on Figure 29. The antigeyser line provides a convection flow loop which maintains cold LO_2 in the feedline.

The antigeyser system has to accommodate the replenish flows introduced through the orbiter from the facility. The flow control ranges and the fluid temperatures delivered by the facility, plus the heat added by the orbiter lines, affect the system performance. Helium from the facility is bubbled into the bottom of the antigeyser line to augment the flow around the antigeyser loop. Adequate flow must be induced around the antigeyser loop while ingesting replenish flow from the orbiter such that sufficiently low temperatures are produced within the ET feedline to meet SSME requirements.

Pressurization System

The pressurization system provides ground-supplied gaseous helium to the LO_2 and LH_2 tanks prior to SSME start; vaporized propellants (gaseous oxygen and hydrogen) are provided from the SSME's subsequent to start to maintain the ullage pressure within required limits. The pressurization gases must be supplied within temperature and flowrate limits to satisfy both ET structural and SSME operating requirements.

To select a design which meets these limitations, an integrated study of the system, including both the transient conditions at engine start and the conditions during engine operation, was required. Due to the complexity of the study, a computer program was developed to analyze adequately the integrated system performance under all conditions, including the variance of SSME pressurant supply characteristics during throttling. A satisfactory system has been defined, using on/off flow control valves located in the orbiter which receive commands from transducers located in the external tank.

A primary concern in this analysis was to insure the flow control devices would not create a pressure imbalance which could cause SSME heat exchanger oscillation and possible subsequent damage. Another serious concern was compatibility of the SSME GOX pressurization gas temperature with ET level sensors and tank insulation bondline.

Ground Interface

Establishment of a compatible ground interface between the launch facility and the Shuttle vehicle involves integration of complex element, program, and facility requirements and analysis of the integrated vehicle and facility.

For example, the addition of spray-on foam insulation to the external liquid oxygen tank to prevent ice formation resulted in a problem in chilling the SSME's. The insulation reduced the tank boiloff rate and, in turn, reduced the LO_2 replenish flowrate. While this would normally be an advantage, the reduced flowrate resulted in warmer replenish LO_2 due to the additional time the LO_2 was exposed to heat transfer in the long facility transfer lines. With the Shuttle LO_2 system configuration (Figure 29), warm propellant from the facility was introduced to the SSME LO_2 inlets, and the maximum allowable SSME inlet temperature for satisfactory engine start was exceeded. The initial concept for solving this problem was to chill the LO_2 in the ground system by use of an LO_2/LN_2 heat exchanger. Subsequently, detailed analysis revealed that the SSME inlet temperature requirement could be met by slightly overfilling the LO_2 tank, terminating the tank replenish several minutes prior to liftoff and bleeding the cold LO_2 from the ET through the orbiter lines and SSME's. The LO_2 in the ET is maintained cold by keeping the tank vented to atmosphere and allowing the liquid to boil, reducing the bulk temperature. This procedure was implemented after the SSME contractor performed an analysis which resulted in revising the engine requirements to specify that the maximum temperature limit had to be met for only three minutes prior to start.

VERIFICATION

As in other recent space programs, the design of the Space Shuttle vehicle will be verified by a combination of analyses, ground tests, and flight tests. The first six flights, which will be conducted from KSC, are designated as design, development, test and evaluation (DDT&E) flights. The seventh flight is designated as the first operational flight.

With the emphasis on cost effectiveness in the Shuttle program, extensive studies have been conducted to achieve balance in the various approaches for performing verifications. Generally, the extent of ground testing planned is less than has been accomplished on some past space programs.

Many verification tests will be conducted independently on the orbiter, main engines, external tank and solid rocket boosters. However, several significant integrated ground tests are planned. Some of these major integrated tests are described below.

MAIN PROPULSION TEST

The main propulsion test program is a series of planned cryogenic tankings and static firings designed to integrate and evaluate the functional integrity and performance of the main propulsion system (MPS), which includes interfacing orbiter subsystems, clustered Space Shuttle main engines (SSME's), external tank (ET) and associated ground support equipment (GSE). The main propulsion test article (MPTA) is shown in Figure 30.

The test program will be conducted at the National Space Technology Laboratory (NSTL) in Mississippi using a modified Saturn S-IC test stand.

System/Subsystem Verifications

To adequately support verification of the main propulsion system and associated subsystems for flight readiness firing (FRF) and the first manned orbital flight (FMOF), it was decided that prime functional hardware used in the test be of flight configuration as much as possible. To implement this objective, a flight weight external tank (MPTA-ET) is to be mated in the test stand with an orbiter test article designated MPTA-Orb. The MPTA-Orb has a flight configuration aft fuselage structure with a substitute covering in place of flight TPS for ground test acoustic fatigue protection. The forward and mid-fuselage structure of the orbiter are not functional for this test and have been replaced with a substitute truss and interface section. Mounted in the aft fuselage is a flight configuration MPS with three flight configuration SSME's. The portion of the flight hydraulic system associated with SSME valve control and thrust vector control (TVC) servo-actuators for engine gimbaling is included in the aft fuselage, with hydraulic power obtained from GSE to drive these systems. One ground computer called Shuttle avionics test system (SATS) will perform the avionics control and monitor functions for the test and substitutes for the five orbiter flight computers. Also included on the MPTA-Orb is a flight purge vent and drain system in the aft fuselage which is required to operate during tankings and firings for aft compartment conditioning.

The MPTA-ET will be a complete flight weight tank with provisions for auxiliary drain, vent and pressurization systems which are required for safety reasons. The structural connections between the tank and MPTA-Orb will be flight hardware, except that pyrotechnic devices used for in-flight separation will not be included.

In planning the test program, as much useful verification data as possible is obtained from all areas of the test, such as facility and GSE activation, test article assembly, mating in the test stand, cryogenic tanking tests, and static firing tests. The static firing portion of the test program consists of a total of 15 firings, using flight nozzles (expansion ratio of 77.5:1) on the engines initially, and subsequently using "stub" nozzles (expansion ratio of 35:1) to achieve full throttling of the engines. Full throttling can not be attained at sea level using the flight nozzles due to flow separation caused by the high expansion ratio.

The MPTA program is designed to satisfy two principal test objectives:

1. Demonstrate main propulsion system performance and compatibility with interfacing elements and subsystems

2. Investigate off-nominal conditions and verify design changes

These two overall objectives break down into many more specific objectives describing each element and subsystem contribution to the test program.

The test program starts with a LN_2 tanking with 10-20 percent levels in both LO_2 and LH_2 tanks. This test insures the functional compatibility of the facility, GSE and test article with no risk due to hazardous propellants. This test is followed by a LO_2/LH_2 tanking which will fully demonstrate objectives such as structural integrity of MPS-related flight structure, propellant fill techniques, engine conditioning (LO_2 bleed, LH_2 recirculation), aft compartment purging, geyser suppression, pre-pressurization, a simulated countdown, propellant draining, and test article purging and inerting.

After the tanking data has been analyzed to determine systems compatibility, the first series of seven static firings using flight nozzles on the engines will begin. The first two firings will be 10 and 60 seconds, respectively, followed by a 250-second firing. This slow buildup insures proper flame bucket water protection prior to a full-duration firing. These short firings have as major test objectives the verification of MPS prestart conditioning, MPS start and cutoff transients, propellant loading and draining at increased flow rates, anit-geysering, boiloff and replenish, thrust structure loads and compliance, cryo insulation performance, and limited thrust vector control (TVC) system operation.

The fourth through seventh firings are planned to last 490-500 seconds and to include such objectives as limited SSME throttling (minor due to nozzle configuration), stability, more extensive TVC system operation, and performance repeatability from prior tests.

Before the second series of eight tests is started, modification time is allocated to allow insertion of any required design changes to the MPS and related subsystems, permitting static firing verification of these changes before flight. The stub nozzles (35:1 expansion ratio) will be installed at this time.

The eighth firing (20 seconds) will allow a shakedown of stub nozzle and other modification effects on prestart conditions, MPS transients, flame bucket, ET pressurization, and thrust structure compliance. Test 9 will last 490 seconds and, for the first time, explore full throttling range performance. Tests 10 through 15 will be for full duration to obtain data on full range MPS performance, and MPS/TVC step input frequency response. Pogo pulsing will be done on three of the firings; pogo effect will be evaluated on all firings, determining the interaction of SSME valve and TVC operation, and pogo suppressor performance. ET feedout characteristics will be determined from the last six firings, with LO_2 liquid level sensor cutoffs planned for tests 10 through 14 and a LH_2 liquid level sensor cutoff planned for test No. 15.

Unique Shuttle Problems

The Space Shuttle has some unique design features that impose unusual requirements on the major propulsion systems test compared to previous programs. The first major difference that must be recognized in the main propulsion test article (MPTA) is the externally mounted tank. In past static firing programs, the vehicle was held in the stand in the aft region close to the main propulsion system being tested. The Shuttle MPTA will be held in the stand at the forward and aft SRB/ET attach points on the ET. This more closely duplicates the dynamic responses of flight and facilitates investigation of any pogo that might be present in the design. To evaluate tank feedout characteristics, the test article is canted 9 degrees in the test stand to approximate the angle of attack at propellant depletion in flight. Holding the test article so that all thrust loads go through the external tank, as in flight, and canting the test article at 9 degrees in the stand have created some unique analytical problems not encountered on prior programs.

Another unique design issue is the close spacing of the engines in the aft end of the orbiter. In flight, engine collision is prevented by the FCS logic. At MPTA a different approach will be used due to lack of redundant avionics. In flight, four separate computer systems protect against erroneous TVC commands, but on the MPTA only one command computer exists. Therefore, on MPTA each engine will have external mechanical stops on critical actuators to limit engine travel. In addition, the SATS computer will have software programs to check engine position versus TVC commands to prevent collision. When sufficient static firing operating confidence is gained, the mechanical stops will be removed, placing total control on SATS.

GROUND VIBRATION TESTS

A major program requirement is to determine the structural modal characteristics and transfer functions. The Shuttle ground vibration test (GVT) is a series of vibration tests at different sites. The GVT on the Shuttle is one of the first space vehicles that will utilize a four body system which departs from the standard/typical cylindrical booster spacecraft launch configuration

The basic objective of the GVT is to verify the math models used to determine the Shuttle analytical structural dynamic characteristics over the frequency range of interest (0 to 40 Hz). This is accomplished by obtaining verification data from modal surveys and transfer function measurements. As in all such programs, the difficulty is in duplicating the flight constraints and environment. Therefore, data from modal surveys of representative configurations will be obtained from ground tests to validate math models which are, in turn, used to calibrate flight modal characteristics.

The first major test is the orbiter horizontal GVT (HGVT). This comprises two subtests, soft and rigid mounted. This test will be conducted at Rockwell's Palmdale, Calif., manufacturing and test facility with the objective of acquiring orbiter modal characteristics for both free flight (using a low-frequency or soft suspension system) and mated test conditions (using rigid links). Aerodynamic flight control frequency response data also will be acquired during the soft GVT. The first full-scale flight orbiter (101) will be used for this test.

The second major test is the 1/4-scale model GVT, to be conducted at Rockwell's Downey, Calif., facility. This test will utilize replica models of the orbiter, external tank, and three sets of solid rocket boosters. Subtests consist of influence coefficient tests on each element, individual element tests, combined tests of the orbiter/ET for various tanking levels (to simulate several boost conditions), and combined tests of the orbiter/ET/SRB for liftoff, maximum dynamic pressure, flight conditions, and SRB burnout. SRB models filled with various quantities of inert propellant will be used. Objectives of the 1/4 scale program are to determine modal characteristics, compare math modeling and modal analysis results with actual test data.

The final major test is the mated vertical GVT (MVGVT) (Figure 31) which consists of five subtests utilizing full-scale hardware. This test will be conducted at the Marshall Space Flight Center (MSFC), Huntsville, Alabama, to determine modal characteristics and frequency response characteristics at the guidance and control sensor locations. One series of subtests will utilize orbiter 101 which was used for HGVT and an ET loaded to three different levels of water in the LOX tank to simulate different propellant loads. A set of empty SRB's will be added to simulate SRB burnout and a set of SRB's, full of inert propellant, will be utilized to simulate liftoff. All of these articles will be refurbished and later used in the flight program.

Prior to and during major tests, other supporting tests will be performed to evaluate areas of concern, either in more detail or to supplement the GVT in its ultimate goal of evaluating math models. Some of these tests and their objectives follow:

1. The 1/8-scale Shuttle model program which is being performed at Langley Research Center. It is being utilized to provide early data on behavior of Shuttle-configured structure, to isolate and study specific technical problems associated with each Shuttle element, and to develop test methods and data interpretation experience for application to 1/4-scale and full-scale GVT.

2. The Solid Rocket Motor (SRM) segment modal survey test which is to be performed at the Thiokol Wasatch (Utah) Facility will utilize a 160-inch long segment from an SRM filled with live propellant. Primary objectives are to acquire empirical verification of modeling techniques using complex shear modulus, verify assumptions on propellant linearity, and verify inherent damping characteristics of live viscoelastic propellant. It will also provide full-scale, live propellant response data not obtainable from MVGVT.

3. The external tank LO_2 tank modal survey test program is to be performed at MSFC. Test objectives are to determine experimentally structural dynamic and hydroelastic characteristics; to investigate the effects of hydroelasticity on modal frequencies, mode shapes, damping, generalized mass, and modal energy distribution of tank/fluid normal modes; to determine effects of tilt and propellant fill conditions on coupled fluid/structural dynamics and tank bottom pressure sensitivities; and to provide verification of the mathematical model to be used in Shuttle systems loads and pogo stability analysis.

4. SRM development and qualification firing tests will acquire GVT related test data. These GVT related objectives are to verify/determine thrust vector characteristics as a function of TVC motion profile and flexible bearing deflection, to determine nozzle structural interaction, TVC/nozzle system response characteristics, dynamic thrust vector alignment, characteristics of internal oscillating acoustic wave phenomena during SRM burn, near and far field acoustic environments, and vibro- acoustic transfer functions for SRM hardware.

5. The main propulsion test (MPT) to be performed at the NSTL, Miss., will gather GVT related data for the math modeling and modal response verification for pogo and dynamics loads analysis. This is also the key test for obtaining frequency response data for the orbiter engines.

The overall GVT and supporting tests have been scheduled to support the first manned orbital flight in early 1979. The HGVT is scheduled for summer 1976, 1/4-scale GVT for calendar year 1977, and MVGVT for last half of 1978.

SEPARATION SYSTEMS VERIFICATION

During boosted flight, two separation functions occur. The first takes place at solid rocket booster burnout when the boosters are jettisoned. This is accomplished by severing retention struts attached to the forward and aft ends of the external tank and imparting separation velocity with small solid-propellant rocket motors mounted in the forward and aft ends of the boosters.

The second separation takes place after main engine shutdown and just prior to orbital insertion and consists of jettisoning the external tank from the orbiter. The separation is in three parts: (1) the umbilical assembly between the orbiter and tank is hydraulically retracted, (2) explosive bolts in the forward and aft attach fittings are fired, then (3) separation velocity is imparted to the orbiter by firing the RCS engines.

Early in the Shuttle program an analysis of the scope of the separation test program was conducted to define the most cost-effective approach. As a result of this study, a decision was made not to conduct full-scale all-element separation tests but to conduct a series of tests at the subsystem or component level. These tests, in combination with wind tunnel and math model analysis and use of mockups, were presumed sufficient to assure proper in-flight separation verification.

External Tank/Solid Rocket Booster Separation System Verification

To define the required performance of all components in the system, wind tunnel tests and math models will be used. These two tools will help size all separation hardware and define separation rocket burn time to establish the required SRB trajectory to avoid recontact.

Component suppliers will conduct qualification tests defined by procurement specifications to assure performance of separation bolts, solid rocket motors, chamber pressure transducers, and other components.

At the MSFC, full-scale assemblies will be used to test structural capabilities of the forward and aft struts and to verify their separation characteristics. Final verification of the ET/SRB separation for operational use will be accomplished during development flight test. The separation dynamics math model also will be verified by the flight test program. Verification of this model will then allow investigation of off-nominal separations without the personnel hazard associated with this type of flight test.

Orbiter/External Tank Separation Systems Verification

As with the ET/SRB interface, extensive use of math model analysis and wind tunnel testing will be used to define separation forces and trajectories for the ET separation.

Component testing will be conducted by individual suppliers; however, because of the complicated umbilical system used for propellant, pressurization and electrical interfaces between the orbiter and the external tank, major umbilical testing will be conducted in-house at Rockwell's Space Division. These tests will verify the proper functioning of the LO_2 and LH_2 umbilicals and their retraction functions.

The total ET/orbiter aft interface consisting of the LH_2 and LO_2 umbilicals, structural attachments and mechanical separation systems, including the orbiter closeout doors, will be tested as an assembly to verify proper separation using a full scale mockup. The orbiter portion of this mockup will be mounted on a cam system which will reproduce the separation kinematics as predicted by the wind tunnel tests and math model.

The forward separation bolt will be tested in a manner similar to the ET/SRB bolt tests. All components will be exposed to temperature and vibration environments comparable to flight conditions.

FLIGHT READINESS FIRING (FRF)

The final step in the Space Shuttle system verification network prior to the first DDT&E flight will be a static firing of the Space Shuttle main engines using mated flight vehicle elements in as near as possible flight configuration, mated to the mobile launch platform (MLP) on the launch pad at KSC. The flight readiness firing (FRF) will be conducted as part of the countdown demonstration test (CDDT) for the first Shuttle manned vertical flight.

In previous space and missile programs, static firings and integrated flight control and propulsion tests were conducted at a test site prior to the vehicles arriving at the launch sites. However, due to the unique design and multi-elements of the Shuttle, all flight systems (propulsion, flight control and avionics) are not integrated until mated at the launch site. Although each element and subsystem of the Shuttle goes through development and verification test, including a main propulsion system test (MPT) utilizing a flight ET and orbiter aft fuselage with SSME's, the total integrated system is not available until the vehicle is mated at the launch pad. Two other important factors necessitated a flight readiness firing: (1) the Shuttle program has no unmanned flights scheduled and (2) no facility checkout vehicle. Specific system objectives realized from the FRF are:

1. First verification of the flight MPS and associated subsystem structural integrity and performance during SSME firing (exact launch conditions up to SRB ignition)
2. First verification of the adequacy of flame and heat protective shielding for SRB's and ET during SSME pre-liftoff firing and simulated launch abort shutdown
3. First integrated avionics/MPS test (SSME control and monitoring with orbiter avionics)
4. First integrated APU/hydraulics/SSME/flight control functional test
5. Additional verification of prelaunch servicing procedures and countdown timelines
6. Additional SSME cluster firing data to verify first flight vehicle MPS predicted performance.

The FRF will be conducted with an unmanned orbiter. Additional switch control functions have been provided in the orbiter for ground control through the launch processing system (LPS) that would not have been required for a manned FRF. These additional ground control functions, plus a modified flight software program, allows the Shuttle main propulsion system (MPS) to be tested at the launch pad with the vehicle configured for flight. The solid rocket boosters (SRB) flight control systems will not be activated for the FRF; however, SRB ignition commands and SRB holddown release signals will be verified. The orbiter T-O umbilicals and the external tank liftoff umbilicals will remain connected during the 20-second firing of the MPS. The orbiter orbital maneuvering systems (OMS) and the forward and aft reaction control systems (RCS) will not be activated during the FRF. OMS and RCS propellant will not be loaded. Orbiter flight control commands will be exercised during the 20-second firing. At the termination of the 20-second firing, the three SSME's will be sequencially shut down simulating a prelaunch shutdown. Following the post firing securing, a vehicle inspection and data analysis will be conducted and the vehicle reconfigured and prepared for the first vertical flight.

CONCLUSION

Selected technical challenges in integrating the Space Shuttle have been described above. Space did not permit including many others, such as avionics, software, flight-system-to-ground-system interfaces, acoustics, vibration, electromagnetic interference and payloads interfaces.

The results of the integrated analyses have led to the establishment of the requirements for the design of the Space Shuttle. The design of all the elements is nearing completion and fabrication is underway. Through an iterative process, analyses will be continued to incorporate refinements and to serve as an adjunct to major ground tests in the verification process.

The integration activities are planned to support ground testing in 1977 and 1978, leading to successful Shuttle flights in early 1979.

NOMENCLATURE

ACCUM	–	Accumulator
AEDC	–	Arnold Engineering Development Center
α	–	Angle of attack
ALT	–	Altitude
AOA	–	Abort-once-around
APT	–	Auxiliary propulsion test
APU	–	Auxiliary power unit
ATO	–	Abort-to-orbit
β	–	Angle of yaw
BSM	–	Booster separation motor
CDR	–	Critical design review
DDT&E	–	Design, development, test and evaluation
DOF	–	Degrees of freedom
ET	–	External tank
FCS	–	Flight control system
FPR	–	Flight performance reserve
FVF	–	First vertical flight
GN&C	–	Guidance, navigation, and control
GSE	–	Ground support equipment
HGVT	–	Horizontal ground vibration test
HPOTP	–	High pressure oxidizer turbopump
IMU	–	Inertial measurement unit
INCL	–	Inclination
JSC	–	Johnson Space Center
KSC	–	Kennedy Space Center
LARC	–	Langley Research Center
LPOTP	–	Low pressure oxidizer turbopump
MECO	–	Main engine cutoff
MMC	–	Martin Marietta Corporation
MPT	–	Main propulsion test
MPTA	–	Main propulsion test article
MSFC	–	Marshall Space Flight Center
MVGVT	–	Mated vertical ground vibration test
NOM	–	Nominal
NSTL	–	National Space Technology Laboratory
OFT	–	Orbital flight test
OMS	–	Orbital maneuvering system
$\dot{p}$	–	Roll acceleration
PDR	–	Preliminary design review
$\dot{q}$	–	Pitch acceleration, or maximum heating rate
$\bar{q}$	–	Dynamic pressure
$\dot{r}$	–	Yaw acceleration
RCS	–	Reaction control system
RECIRC	–	Recirculation
REPRESS	–	Repressurization
RI/RD	–	Rockwell International, Rocketdyne Division
RI/SD	–	Rockwell International, Space Division
RTLS	–	Return to launch site
SATS	–	Shuttle avionics test system
SOFI	–	Spray-on foam insulation
SRB	–	Solid rocket booster
SRM	–	Solid rocket motor
SSME	–	Space Shuttle main engine
SSV	–	Space Shuttle vehicle
TK	–	Tank
TP	–	Test point
TPS	–	Thermal protection system
TVC	–	Thrust vector control
T/W	–	Thrust-to-weight ratio
T_W	–	Wall temperature

ACKNOWLEDGEMENTS

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C.A. Scottoline – Aerothermodynamics

J.M. Hall – Flight Control

H.R. Wiener – Pogo

R.W. Westrup – Structural Dynamics & Loads

R.E. Field – Propulsion

B.E. Andrews – Main Propulsion Test

G.H. Russell – Ground Vibration Test

D.M. Dysart – Separation Tests & Flight Readiness Firing

ILLUSTRATIONS

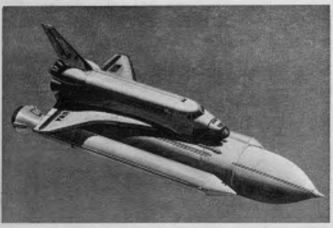
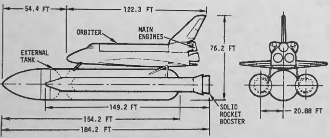


Figure 1. Space Shuttle Vehicle



Element	Length (ft)	Diameter (ft)	Inert Weight (lb)	Propellant Weight (lb)	Gross Weight (lb)	Sea Level Thrust (lb)
Orbiter	122.3	--	132,054	23,181	158,979*	-
SSME (3 each)	13.9	7.8	6,446	577	7,023	375,000
ET	154.2	27.6	72,490	1,564,704	1,637,279	-
SRB (2 each)	149	12.2	180,510	1,105,988	1,292,274	2,700,000

*Includes 3,744 pounds of non-propulsive consumables.

Total incl 2 SRB's & 3 SSME's	-	-	-	3,801,592	4,399,875	6,525,000

Figure 2. Space Shuttle Vehicle Characteristics

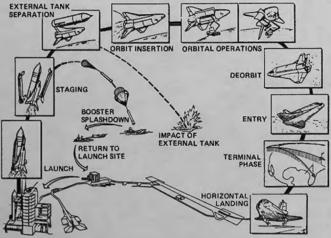


Figure 3. Shuttle Mission Profile

BASE-LINE REF. MISSION	LAUNCH SITE	OBJECTIVE	INCL (DEG)	ALT (NM)	DURATION	PAYLOAD (LB) ASCENT	PAYLOAD (LB) DESCENT	DELTA V REQMTS (FPS) OMS	DELTA V REQMTS (FPS) RCS
1	KSC	PAYLOAD DELIVERY	28.5	150	7 DAYS	65.0K	32.0K	650	100
2	KSC	COMBINATION REVISIT TO ORBITING ELEMENT & SPACELAB	55.0	270	7 DAYS	-	-	1250	120
3A	WTR	PAYLOAD DELIVERY	104.0	100	1 REV	32.0K	2.5K	250	100
3B	WTR	PAYLOAD RETRIEVAL	104.0	100	1 REV	2.5K	25.0K	425	190

Figure 4. Baseline Reference Missions

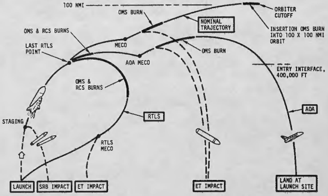


Figure 5. Space Shuttle Ascent Trajectory Flight Profile

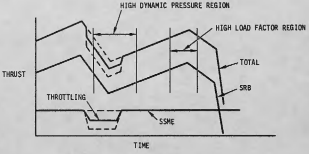


Figure 6. Thrust Profile Considerations

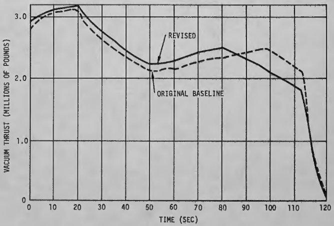


Figure 7. Comparison of Thrust-Time Curves

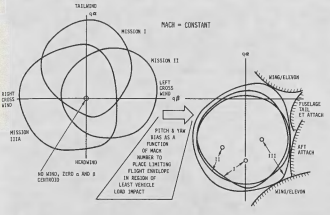


Figure 8. Trajectory Biasing

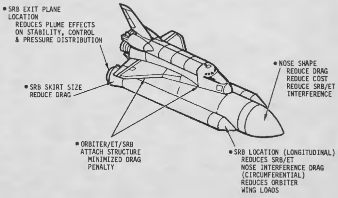


Figure 10. Shuttle Vehicle Aerodynamic Considerations

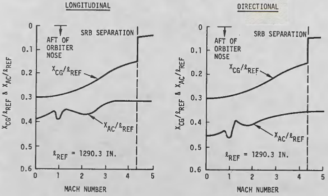


Figure 9. Launch Vehicle Stability Characteristics

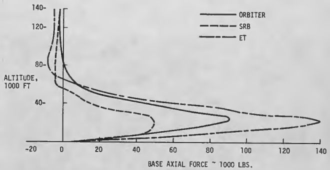


Figure 11. Element Power-On Base Axial Force

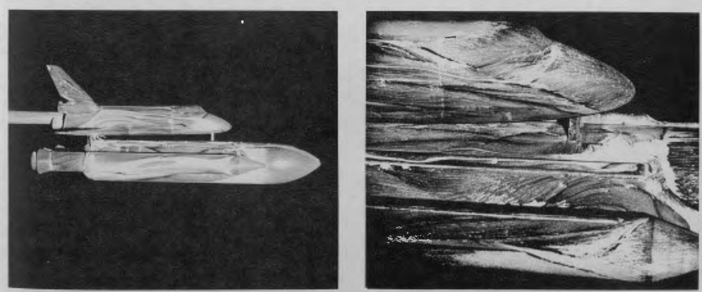


Figure 12. Surface Flow Patterns – Wind Tunnel Model

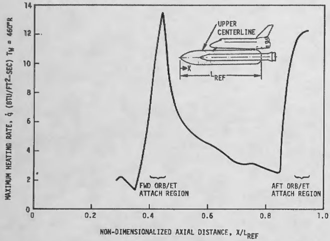


Figure 13. Maximum Heating Rate Distribution on ET Intertank and LH_2 Tank Upper Centerline

Figure 14. Scale Model for Predicting Plume Heating

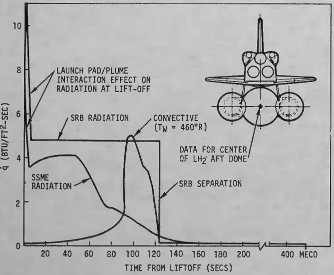


Figure 15. Plume Heating History at Center of ET LH_2 Aft Dome

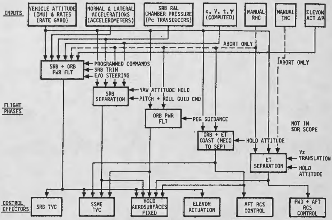


Figure 16. Ascent Flight Control Input/Output

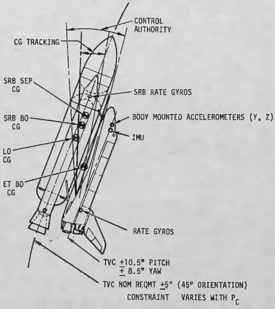


Figure 17. Configuration Considerations

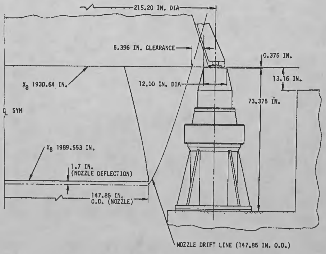


Figure 18. SRB Liftoff Clearance with Support Structure

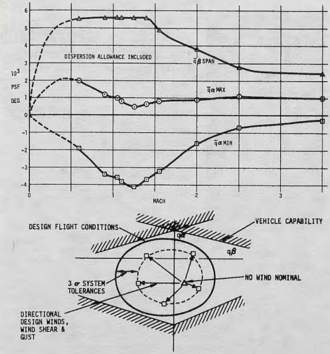


Figure 19. Load Relief Requirements

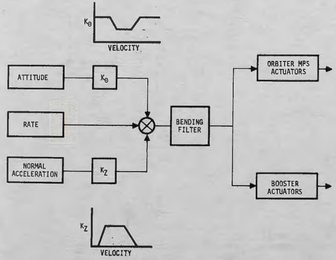


Figure 20. Blended Attitude/Load Relief Control

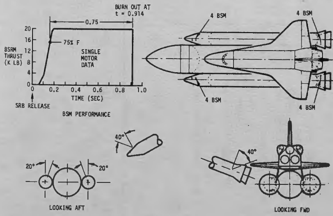


Figure 21. Booster Separation Motor Orientation

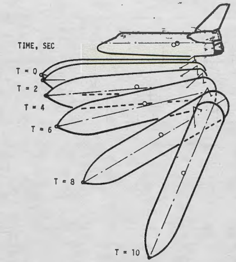


Figure 22. External Tank Separation From Orbiter

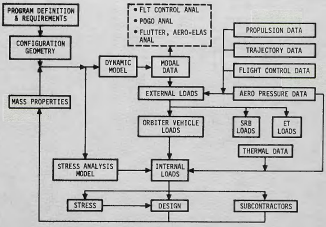


Figure 23. MMLS System Flow Diagram

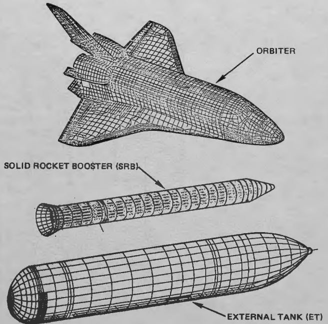


Figure 24. Finite Element Stress Models

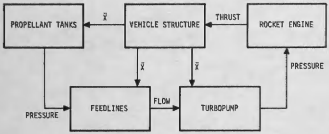


Figure 25. Pogo Stability Loop

•THRUST, POUNDS	
•SEA LEVEL	375,000 (RATED POWER LEVEL)
•VACUUM	470,000 (RATED POWER LEVEL)
•FPL, PERCENT	109% OF RATED POWER LEVEL
•CHAMBER PRESSURE, PSIA	2970 (RATED POWER LEVEL)
•AREA RATIO	77.5
•SPECIFIC IMPULSE (NOM), SECONDS	
•SEA LEVEL	363.2 (RATED POWER LEVEL)
•VACUUM	455.2 (RATED POWER LEVEL)
•MIXTURE RATIO (O/F)	6.0 ±0.06
•LENGTH, INCHES	167
•DIAMETER, INCHES	
•POWERHEAD	105 BY 94.5
•NOZZLE EXIT	94
•DRY WEIGHT, POUNDS	6339
•LIFE	
•HOURS	7.5
•STARTS	55

Figure 28. Space Shuttle Main Engine (SSME)

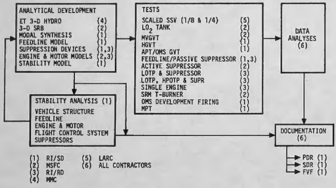


Figure 26. Pogo Analysis/Test Plan

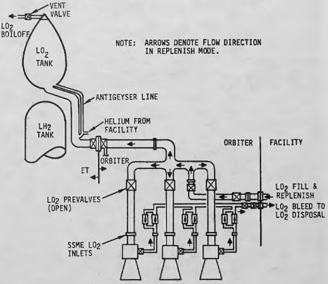


Figure 29. Shuttle LO_2 System

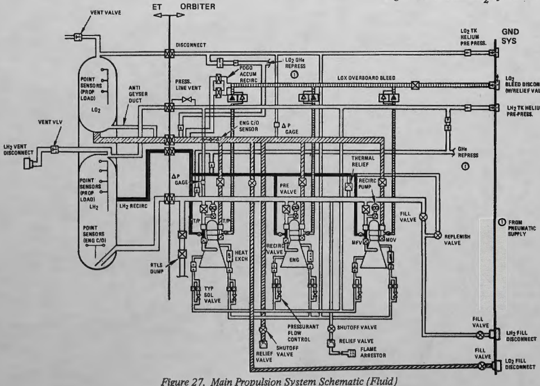


Figure 27. Main Propulsion System Schematic (Fluid)

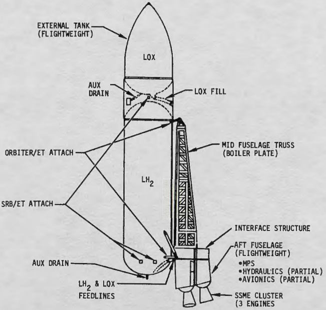


Figure 30. Main Propulsion Test Article (MPTA)

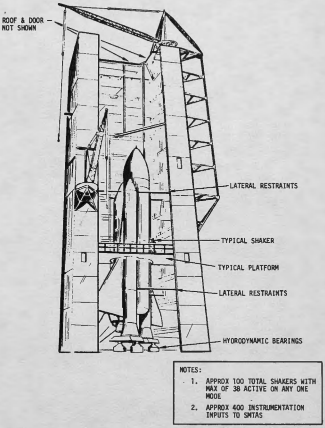


Figure 31. Representation of Mated Vertical Ground Vibration Test