valves then would close and helium injection for anti-geysering protection would be terminated. It is proposed that a proper degree of redundancy be provided in the ground system to assure a fail safe arrangement. A test plan approach has been developed to support the LOX anti-geysering program. The test plan itself is still in work along with the type of hardware and test facility to accomplish the objectives of the program. Test schedule is:

Hardware on-site and installation start	February 1, 1976
Test start	September 1976
Test completion	March 1977

Proposed test configuration is shown in Figure 32 .

3.3.1.3 <u>Electrical Subsystem</u>

System Design

For the design development test and engineering phase of the Space Shuttle program, the external tank electrical subsystem includes: (1) operational instrumentation, (2) electrical distribution, (3) lightning protection, and (4) development flight instrumentation as appropriate.

Operational instrumentation includes those external tank intruments required to monitor and control tank-related functions from the start of propellant loading through tank separation. Each instrument is supposed to be individually hardwired through the tank electrical distribution cable assemblies to the ET/Orbiter umbilical connectors. Amperage limiting protection is provided by the Orbiter for those circuits penetrating the ET tanks to preclude the generation of ignition sources. Since this instrumentation consists only of sensors and cabling from them to the interface, no circuit grounds are made to the tank structure. All sensor leads are individually returned to the Orbiter for single point grounding. Cable shields are, however, grounded to tank structure to satisfy electromagnetic compatibility requirements.

Development flight instrumentation is, by definition, non-critical for external tank operation and will be installed on the main propulsion test article at NSTL and on the first six external tanks. The principal requirement from a safety standpoint is that this instrumentation shall not cause the failure of any critical external tank function. The general design and construction of the development instrumentation is the same as previously described for the operational instrumentation. Electrical power for the instrumentation assemblies is supplied through the Orbiter umbilical interface. There are two operational instrumentation cable harnesses inside the LOX and liquid hydrogen tanks. The cables are made of teflon (FEP) insulated wire, and the sensors are attached with fixed splices, insulated and sealed with heat-shrinkable TFE teflon tubing and meltable FEP teflon. Each cable is routed through a separate cryogenic feed-through connector mounted in the noseplate of the LOX tank and the forward dome of the liquid hydrogen tank. The wire bundles inside the tanks are spot tied with lacing tape and supported by corrosion resistant steel bands with teflon cushions. The use of teflon (FEP) insulated wire in contact with LOX has been identified as a potential hazard since it includes both a fuel (teflon) and a potential ignition source (electrical energy) interfacing with LOX.

The philosophy expressed in NASA's NHB 8060.1A, "Flammability, Odor, and Offgassing Requirements and Test Procedures for Materials in Environments That Support Combustion," is that the design of LOX systems should preclude any ignition sources interfacing with the media. If this goal cannot be met, any material used in the proximity of a source of electrical energy shall be evaluated in the proposed configuration. Evaluation should be made using the worst-case electrical and environmental conditions and by applying the techniques of NHB 8060.1A, Test No. 4, "Electrical Wire Insulation and Accessory Flammability Test." Results of the Apollo 13 incident and subsequent testing have shown that teflon will not pass such a test in a cryogenic high pressure oxygen environment. See Figure 33 . MSFC has stated that Saturn Launch Vehicle test experience with teflon (TFE) coated wire shows that: (1) teflon coated wire insulation cannot be ignited under LOX by any electrical over-load, (2) teflon coated wire insulation can be ignited in gaseous oxygen by approximately 800% electrical overload and will propagate, and (3) in the unlikely event of ignition under operational conditions, fire will not propagate through the feed-through-connector at the tank wall. Shuttle sensors are similar to those used on the Saturn second stage, S-II. Analysis and testing (similar to that which will be accomplished for Shuttle) were conducted subsequent to the Apollo 13 incident for the S-II sensors and demonstrated that no safety problem existed. It was stated that the temperature on the cable will be sufficiently below the sublimation point of teflon to maintain a safe condition in the cabling. The Panel pointed out that while the size of the wires was small and the potential of applying in excess of the 800% overload appeared minimal there still could be some chance of a problem, and suggested further consideration.

Current Status

MSFC will conduct worst-case current overload testing and analysis in the LOX environment using actual ET hardware and all circuit protection devices (in their worst-case credible consequences of their failures). Testing would include sensor shorts, opens, normal operation and electronic failure modes. This issue will be considered resolved if the above testing is successful. It was also suggested by the Panel that all other similar non-metallic materials' applications be reviewed and appropriate disposition made.

The External Tank design incorporates features to protect the structure and subsystems from the direct and indirect effects of triggered atmospheric discharges during transportation, prelaunch, launch and flight operations. Methods employed to provide lightning protection are intended to assure that low resistance paths are provided on the External Tank surfaces to distribute lightning currents through the structure and to guide currents around or over nonmetallic areas. At this time lightning protection on the nose cap consists of a short nose rod and conductive aluminium strips cemented onto the vehicle and electrically bonded to the structure. The LOX hydrogen and inter-tanks incorporate thin aluminum strips, adhesive-bonded to the external insulation surface and electrically bonded to the LOX tank skin. Further protection measures include the use of twisted wires on all internal circuits and twisted shielded cables in exterior cable tunnels.

The only significant problem noted by MSFC was the possibility that the diverter strips could debond or melt in flight and the resultant debris could possibly damage the Orbiter in some manner. This problem is currently under study to determine alternate designs and to further understand the impact of strips melting or debonding.

3.3.1.4 Separation and Disposition

The External Tank interfaces with the Orbiter and the Solid Rocket Boosters. In the mission events time-line, the Solid Rocket Boosters are separated from the External Tank/Orbiter combination and then the External Tank is separated from the Orbiter. The ET/SRB attach configuration is shown in Figure 34, and the aft and forward attach configurations between the External Tank and the Orbiter are shown in Figures 35 and 36. The separation hardware in both the Orbiter and Solid Rocket Booster case are designed by their respective contractors (Rockwell International and Thiokol) and not by the tank contractor since the External Tank portions of separation interfaces are passive. Martin-Marietta Corporation does support the Rockwell International and MSFC (SRB) efforts in defining, designing and testing the separation hardware. Aspects of the ET/Orbiter separation have been discussed under the Orbiter Section 3.1 and the same will apply to the Solid Rocket Booster Section 3.4. Only those Orbiter and SRB actions that can affect the External Tank's ability to separate safely and be disposed of during its return to earth are discussed here.

The Solid Rocket Booster (SRB) separation from the External Tank (ET) follows this sequence: (1) Orbiter receives separation cue from the Solid Rocket Booster, (2) Orbiter arms' separation system pyrotechnic initiator controllers on both of the SRB's 0.8 seconds after the Orbiter cue is given, (3) Orbiter issues fire commands to separation system "A" on both SRB's simultaneously 2.5 seconds after the Orbiter cue, and (4) Orbiter issues fire commands to separation system "B" on both SRB's simultaneously 40 milliseconds after the system "A" fire commands.

Actions to be taken if for some reason this separation does not take place are to be examined further by the Panel. All the prime contractors and the NASA Centers are involved since this is an interface problem.

The External Tank separation from the Orbiter follows this sequence: (1) forward Orbiter reaction control system deployment, (2) fluid and electrical umbilical separation, (3) forward and aft structural attachment release, and (4) Orbiter maneuver away from the External Tank. Sequencing of all separation operations and commands are initiated and controlled by the Orbiter. As a result of new loads analyses for the ascent portion of the mission, the External Tank/Orbiter aft attach loads have increased, requiring hardware modifications which do not appear to unduly affect the separation events mentioned above. There are some safety concerns that result from the separation process which have been discussed with the Panel: (1) LH_2 and LOX trapped between the feed-line closure valves and released as the External Tank and Orbiter separate pose a potential fire/explosion hazard and, (2) External Tank recontact with the Orbiter vehicle primarily due to Orbiter hardware problems.

External Tank entry and disposal after release from the Orbiter has been of great interest to the Panel. Ground rules, constraints, and guidelines applicable to the External Tank disposal problem have been stated as:

(a) No External Tank impact below 60° South parallel, based on State Department international agreement.

(b) External Tank impact locations shall be in ocean areas with minimum ship traffic densities.

(c) External Tank impact locations shall be no closer than 200 nautical miles from land masses.

(d) External Tank impact location and dispersions are predictable.

(e) External Tank rupture for nominal missions shall not occur above 240,000 feet altitude.

(f) External tank distruct from any cause shall not occur within four (4) nautical miles of the Orbiter.

On normal missions the External Tank separates from the Orbiter at almost orbital velocity. The impact site is therefore sensitive to variations in the tank velocity and other conditions at separation. The question then is whether the selected design can ensure that the tank or the debris will always land in an acceptable ocean area. Aborts and catastrophic situations during launch and ascent also must be considered, and the added hazard of having large quantities of propellant and oxidizer under such situations must be taken into account.

A major consideration in the proper disposal of the tank is the point in the ascent at which time the Orbiter main engines are cutoff. The definition of the MECO (Main Engine Cut Off) is currently baselined as occurring at an altitude of 60 n. mi. for nominal mission and at 55 n. mi. for an "abort-once-around" mission. Based on these altitudes, the MECO conditions for a launch from KSC are as follows:

(a) For a nominal mission, the altitude of 60 n. mi. with a velocity of 25,383 feet per second and an angle of attack of 0.5 degrees.

(b) For an abort mission (AOA), with an altitude of 55 n. mi. with a velocity of 25,317 feet per second and an angle of attack of 0.75 degrees.

(c) For the return-to-the-landing-site (RTLS) abort mode, the MECO target is at 230,000 feet (37.8 n. mi.) with a velocity on the order of 6,500 FPS.

These MECO conditions for a launch from KSC are valid for a wide variety of launch inclinations and payload weights. Figure 37 is typical of the tank disposal landing footprint for nominal and AOA conditions.

There are two major challenges associated with the safe reentry of the External Tank. The first is the premature breakup due to LOX and hydrogen tank ruptures as well as determination of actual breakup altitude and uncertainty of the dispersion of the resultant debris. The second is the inability to assure tank impact predictability without the use of system that causes the tank to tumble. The tumbling condition must be achieved before the tank has any chance of "skipping" due to aerodynamic lift, as well as having a tumble rate that prevents the occurrence of the "frisbee" effect, which occurs at too high a tumble rate. Typical effects of three different nominal entry conditions are shown in Figure 38 . These assume a tumble rate of 30 degrees per second maximum and $\pm$ 1.3 degrees per second as minimums. The frisbee effect shown in Figure 39 becomes noticeable at tumble rates in excess of 30 degrees per second. Premature tumbling might also result in contact of the External Tank and the Orbiter. As a result of current studies, the following two ground rules have been established for an acceptable tumble system: (1) no tumble action to be initiated prior to 60 seconds after separation from the Orbiter, and (2) acceptable tumble rates are between 10 and 50 degrees per second. Martin Marietta Corporation currently is conducting studies to refine and define an "optimum" system to satisfy the ground rules noted above. The systems being considered are:

(a) Blow down, using LOX vent valves

(b) Solid rocket thrusters

(c) LOX and hydrogen tank "blow holes."

3.3.1.5 Thermal Protection Subsystem

In November 1974 the Thermal Protection Subsystem baseline was changed due to a significant increase in expected thermal heating environment and to a requirement to minimize ice formation and its impact on the Orbiter. This new baseline data affected the insulation material used on the three major sections of the tank: LOX tank including the nose cone, the inter-tank, and the hydrogen tank.

Current design thermal inputs to the External Tank segments based on analyses through December 1974 are:

(a) For the LOX tank forward ogive section the induced thermal environment can be as high as 10.5 btu/ft^2-sec, but new hypersonic wind tunnel data indicates a value that could be as high as 16 btu/ft^2-sec. The LOX tank, inter-tank and hydrogen tank thus are considered to be subject to heating values in excess of that normally acceptable for the proposed new insulation material (Upjohn CPR-421 spray-on foam insulation (SOFI). The CPR-421 is considered appropriate for heating values up to about 6 btu/ft^2-sec but are unacceptable at valued around 10-11 btu/ft^2-sec. The material used on structure subjected to very high heat rates is an ablator material called SLA-561 with a silicone sealant coat. These areas include the Orbiter aft attach strut, forward attach strut, liquid hydrogen feedline and crossbeam, and the LOX tank conduit.

In addition to preventing ice formation and heat input to cryo fluids, one of the major reasons for the insulation is to preclude the air liquification because liquid air is high in oxygen content when boiling off, and compatibility problems exist when it contacts hydrocarbon materials.

NASA and the prime contractor are currently conducting studies and tests to establish an insulation configuration that will satisfy known induced and natural environments with a capability for future possible heating rate increases. They feel that neither trajectory shaping or external tank configuration changes are practical methods of alleviating this problem.

3.3.1.6 Ground Support and Logistics

The mode of transportation for the External Tanks to the launch site has been settled. Barges will be used in a manner similar to that for the Saturn launch vehicle stage movement (S-IC and the S-II). The use of any carrier aircraft has been ruled out at this time because of the modifications required, cost and safety implications.

To assure propellant and oxidizer cleanliness, the following requirements have been levied on the External Tank system:

(a) The LOX and hydrogen tanks will be cleaned per MSFC - Spec - 164A, with no particle larger than 1000 microns.

(b) At the exit of each tank, propellant screens will be installed. For the hydrogen tank this will be a 400 micron "glass bead rated" screen, and for the LOX tank an 800 micron "glass bead rated" screen.

(c) All lines and components downstream of the filters shall be cleaned to a maximum particle size of 400 microns for the liquid hydrogen and 800 microns for the LOX.

It was noted that the External Tank design common fill and delivery lines insure that any contamination introduced into the system during propellant loading will be delivered to the main engines. Therefore, the ground systems and the Orbiter lines have to be cleaned to at least the same levels as the External Tank lines which interface with the Orbiter.

3.4 Solid Rocket Booster

Prior to liftoff the Orbiter Main Engines are ignited and brought to full thrust and both Solid Rocket Motors are armed and ignited from simultaneous ignition commands. At approximately 150,000 foot altitude, the thrust of both Solid Rocket Motors will have decayed to less than 25% of nominal. At this time separation of both Solid Rocket Boosters is initiated and the Orbiter and External Tank continue toward orbit. Upon successful separation of the Solid Rocket Boosters, a sequence is initiated for individual recovery of the two booster units. Parachutes are deployed along the trajectory of each unit to provide for soft impact within a predefined recovery zone. Each booster is to be floated by entrapped air until the arrival of a recovery ship or ships. The flight time, launch to splashdown, takes about 7 minutes and 15 seconds.

The Solid Rocket Booster element of the Space Shuttle system is made up of seven subsystems: (1) the solid rocket motor, (2) the thrust vector controls, (3) separation subsystem containing mechanical and ordnance equipment, (4) the recovery subsystem containing mechanical and parachute equipment, (5) avionics, (6) structure, and (8) a destruct or range safety subsystem.

The Thiokol Corporation in Wasatch, Utah was selected as the Solid Rocket Motor contractor. They have completed the design of most of the tooling for the fabrication of the motor cases and procurement is underway. The contractural awards for the structures, separation motors, recovery system, thrust vector control, and avionics had not been completed at the time of the Panel's review. However, since the Solid Rocket Booster Preliminary Design Review was completed in November 1974, the Panel was able to review the detailed design of the booster components. As mentioned in an earlier section on management, the overall integration of the booster is being performed by the Marshall Space Flight Center in Alabama. NASA plans to select a booster assembly contractor in fiscal year 1977.

3.4.1 Solid Rocket Motor

System Design

The solid rocket motor includes the case, propellant, igniter and nozzle as shown in Figure 40. Flexibility in fabrication and ease of transportation and handling are made possible by a segmented case design. The propellant grain is shaped to reduce thrust approximately one-third some 55 seconds after liftoff to prevent overstressing the vehicle during the period of maximum dynamic pressure. The grain is of conventional design, with a star-shaped perforation in the forward casting segment and a truncated cone perforation in each of the segments and the aft closure. The contoured nozzle expansion ratio is 7.16:1. The rocket motor case is made up of ten separate segments with specific joints to meet the structural requirements and weight needs as shown in Figure 41. The following is a performance summary of the rocket motors under nominal conditions at 60^{o}F.

(a)	Vacuum delivered impulse, lb-sec	290.6×10^{6} (T=1 sec.)
(b)	Burn Time, seconds	122
(c)	Propellant burning rate, in/sec	0.411 (at 1000 psi)
(d)	Specific Impulse, average, lb-sec	262.2×10^{6}

The Solid Rocket Motor ignition hardware consists of an igniter and dual redundant standard man-rated initiators. These initiators are separated by an independent electrically dual redundant (2 motors and 1 shaft) electro-mechanical safe and arm device. Each initiator is fired by an independent Pyrotechnic Initiator Controller (PIC) upon command. The safe and arm device is maintained in the safe position by a mechanical safety pin until a given point in the countdown at which time it is removed. The device remains in the safe position until the arm-command is given immediately prior to the motor ignition.

The items associated with weight and weight control are:

(a)	Motor Mass Fraction	0.884
(b)	Total Solid Rocket Motor, lbs.	1,254,210
(c)	Solid Rocket Motor, lbs.	1,227,250

Current Status

There have been studies on alternate propellants to minimize HCl release above 65,000 feet (ozone layer) during the ascent portion of the mission. To date the studies indicate that it is technically feasible to minimize (less than 3% by weight) or eliminate the release of HCl above 65,000 feet. However, there would be a probably payload loss of 2,000 to 7,000 pounds. These studies will continue as one of NASA's efforts to reduce the atmospheric impact from the Space Shuttle operations.

NASA has noted that the Solid Rocket Motor and booster components fabrication requirements are considered to be the current state-of-the-art technology which has been demonstrated in systems such as the Titan III rocket now in use.

Thrust mismatch of the two rocket motors is of great concern to the designers and the operation of the Shuttle system. As a result of this concern, NASA and its contractors, continue to pay a great deal of attention to having both the rocket motors ignite and essentially tail-off simultaneously and an acceptable thrust mismatch during normal ascent. The reproducibility limits, based on the latest analysis, are shown in Figure 42. Thus there will most likely be a need to match pairs of rockets. The specification requires that there not be a mismatch greater than 710,000 pounds during the tail-off thrust period at around 115 seconds after ignition.

The POGO phenomenon is not expected to manifest itself in the burning characteristics of the rocket motor. However, the potential for this motor to contribute to POGO will be explored fully by the program offices as a part of the overall POGO effort.

3.4.2 Thrust Vector Control

System Design

The Thrust Vector Control subsystem controls the angle of the nozzle of the rocket motor, in order to obtain the proper flight trajectory. Each Solid Rocket Booster contains a Thrust Vector Control assembly consisting of redundant hydraulic power units and two actuators. If one of the hydraulic power units fails, a valve in the actuators isolates the failed unit and this prevents any loss of thrust vector capability. The servovalves for each actuator are hardwired across the SRB/ET interface and accept steering commands from the Orbiter guidance and control system to provide motor deflection. The basic requirements for this control system are:

(a)	Torque, inch-pounds	4,200,000
(b)	Rate, degrees per second	5
(c)	Acceleration, radians per sec^2	2
(d)	Gimbal Angle, degrees	5
(e)	Redundancy	Fail safe as minimum.

Current Status

The current design is a fail operational/fail safe design. The Thrust

Vector Control has a maximum gimbal capability of 7.1 degrees and provides torques in excess of those required for known loadings. Since the loads effort is a continuing activity the loads may change upward but appear not to be a major problem at this time.

3.4.3 Separation Subsystem

System Design

The Solid Rocket Booster separation subsystem consists of the forward and aft separation motor assemblies, the forward attachment unit and the aft attachment and umbilical pull-away unit, Figure 43.

The separation sequence for the booster is:

(a) Orbiter receives separation cue from both boosters,

(b) Orbiter arms two sepration system pyrotechnic initiator controls on both the A and B units in both boosters 0.8 seconds after the cue is given to the Orbiter,

(c) The Orbiter issues fire commands simultaneously to the "A" unit on both the boosters at 2.5 seconds after the cue,

(d) Orbiter issues the fire command simultaneously to "B" unit separation assemblies on both boosters some 40 milliseconds after system "A" has been given the fire command.

The cue received by the Orbiter is in the form of a pressure signal when the Solid Rocket Motor chamber pressure has reached 50 ± 15 psia on any two pressure sensors used for this purpose. The separation system avionics is shown in Figure 44.

Current Status

The forward and aft separation motor asseblies each consist of four separation motors and ignition ordnance which are fired to impart side thrust to the expended booster. There has been a recent change in the motors to reduce, if not eliminate, the impingement of the motor plumes on the Orbiter Thermal Protection Subsystem. These changes are noted here:

	Before	Current
Thrust Level, lbs.	12,000	20,000
Burn Time, seconds	2	0.75
Propellant Restrictions	none	max. metal or stabilizing additives - 2% burn rate additives - 1%
Igniter Case Material	glass phenolic	non-debris generating
Igniter Propellant	no restriction	same restrictions as main propellant
Thrust Tail-Off Rate	no restriction	Tail-off to 50% chamber pressure limited to 100 milliseconds
Motor Location	SRB forward back of frustum and aft skirt	Nose frustum and aft skirt

The forward attachment unit consists of an SRB fitting, called a thrust post, supported by the SRB forward attachment structure which mates with an External Tank fitting. This forward attachment provides longitudinal SRB/ET restraint and transmits thrust from the SRB to the ET/Orbiter. The SRB and ET mating surfaces are held together by a double-ended separation bolt which is internally redundant for the separation function. A standard manned spacecraft initiator pressure cartridge is mounted on both ends of the double-ended separation bolt. At separation, both of the separation cartridges are fired and the resultant pressure buildup drives an internal piston at each end of the separation bolt toward the separation plane to effect bolt fracture. Operation of either piston will fracture the bolt.

The aft SRB/ET attachments include a lower, upper, and diagonal strut assembly which provide lateral and rotational restraint between the SRB/ET. Each strut assembly consists of a SRB and ET fitting held together by a double ended separation bolt similar in design and operation to the forward attachment separation bolt. The "pull-away" connectors used at each SRB/ET interface carry the electrical circuits as follows:

(a) Forward Attachment	1
(b) Aft Strut (Diagonal)	1
(c) Aft Strut (Upper)	5
(d) Aft Strut (Lower)	3

As a result of the latest Shuttle system loads analysis, December 1974, there is an effort underway to redesign the forward thrust fittings and aft attachment struts. This will result, most likely, in some weight increases. There is no expected change to the basic concept of the separation assembly described here.

3.4.4 Recovery Subsystem

System Design

The booster recovery subsystem provides the necessary hardware to control the descent (velocity and attitude) after separation from the External Tank. The recovery subsystem includes those items used to separate, deploy, disconnect, control attitude, float, and provide for location of the expended booster. Figure 45 shows the booster recovery (separation to splashdown) events and associated parameters of performance at each stage. The booster recovery main chutes, drogue and frustum, and booster itself are buoyant. The recovery system is redundant except for the beacon and flashing light.

Briefly the sequence of events is as follows. A command is sent from the Orbiter to the Solid Rocket Booster just before separation to apply battery power to the recovery logic network and at the same time to arm the nose cap thruster for deploying the drogue, the frustum ring detonator for main deploy, and the main chutes disconnect. Two barometric switches are set to close at high altitude (below 19,000 feet) and at low altitude (below 10,000 feet). At high altitude the nose cap thruster fires, pushes the nose cap away from the booster, and deploys the drogue chutes. At low altitude the frustum ring detonator fires, the drogue chute pulls the frustum away from the booster, and deploys the main chutes. After a time delay the nozzle extension is jettisoned and the impact switches are armed. A third barometric switch will close at a very low altitude to turn on the impact recorder just prior to water impact. At impact the impact switches close and after a time delay the main chutes are disconnected and the beacon and light are turned on. The nose section of the booster, containing the majority of the recovery hardware, is shown in Figure 46.

The maximum vertical velocity for the booster at water impact has been set at 100 feet per second.

Current Status

The Panel's major interest was directed toward questions concerning the inherent safety of a reusable Solid Rocket Booster. The solid rocket case, the parachutes and the hardware for the separation of the booster from External Tank were of the greatest interest. In this section the parachutes and separation hardware are discussed, while the motor case is discussed under the "Structures" paragraph which follows. The separation hardware includes the forward and after separation motor assemblies, forward and aft strut attachment units and the umbilical pull-away connector units. The separation motors are burned out after use and require replacement, as does the ignition ordnance. As noted in the reviews conducted at MSFC the electrical connectors and wiring are the major items requiring retest and rehabilitation for reuse in the booster. The attachment struts and fittings are a part of the structure and are covered in that section. The replacement of used pyrotechnic cartridges and retest of the connectors and wiring is the important task.

Refurbishment of the parachutes (drogue and main) is new to NASA experience in that NASA's current approach is to not reuse space recovery parachutes. However, there is a great deal of DOD experience available with regard to reusing parachutes, e.g., aircraft braking chutes, cargo parachutes and personnel parachutes.

The material in Table X is indicative of the approach used in defining the ability to reuse a drogue or main chute. More specifically, the following data have been developed for commonly used materials such as nylon and dacron:

(a) Prolonged ultraviolet explosure produces strength loss of 50% within seven days.

(b) High temperatures result in severe strength loss after only 10 hours of exposure at 350° F.

(c) Since these materials are hygroscopic (absorb water), they show only a slight strength loss when subjected to high humidity.

(d) Radiation other than ultraviolet is very harmful and thus chutes require shielding.

(e) Vacuum conditions do not appear to materially affect the cute properties.

3.4.5. Avionics

Systems Design

The Booster Avionics consists of the following assemblies: electrical, instrumentation, control rate gyro, recovery, range safety, and failure detection.

A significant portion of the electrical and instrumentation assemblies are included in two line replaceable units, the forward and aft integrated electronics assemblies. Both contain the logic and networks distributior, multiplexer-demultiplexer, signal conditioner and the forward two data buss couplers.

The electrical system consists of a 28 VDC battery supplying power for separation, deployment and recovery functions through the logic and network distributors. These distributors, one forward and one aft, also provide the 28 vdc power from the Orbiter to signal conditioners and associated measuring devices during the ground and flight period when the boosters still are a part of the total Space Shuttle vehicle.

The avionics associated with the recovery activities consists of the following components: (1) Altitude/impact switch assembly, (2) X-band radar transponder (beacon system), (3) X-band radar antenna (beacon system), and (4) two flashing lights.

Range safety subsystem, which is not yet defined, is to provide the destruct capability for the boosters in case of early termination of the flight. This system has been defined in the Level II "Space Shuttle Program Flight and Ground System Specification", JSC-07700 Vol. X, updated to May 1975, as "an add-on destruct system --- which does not require any action by the crew prior to initiation of an abort. The system function shall be dependent on real-time range safety down-linked paramaters and/or tracking data for the period after liftoff up to SRB/ET separation."

Current Status

Based on the material provided to the Panel, the following is the status of the range safety system:

The design concept and selection of system components are complete except for conical shaped charge to be placed in the solid rocket booster element. Currently the program is involved in an effort to fully integrate the system design from the standpoint of ground-to-flight vehicle and between the flight vehicle elements. Acceptance of basic design concept by the Air Force Eastern Test Range is still under discussion. Working interfaces have been established between all organizations affected by the range safety system design, development and utilization. Discussions between these groups, reviews and planning sessions are being established.

The failure detection setup for the booster provides the failure detection capability during boost phase of the flight. This setup had not been defined sufficiently for presentation to the Panel during its early Spring review at MSFC.

3.4.6 Structures and Reusability

The reusability aspects of the Solid Rocket Booster are so closely tied to the structural design capabilities that these two aspects of the booster program are discussed together in this report. Basically the only non-structural hardware built for reuse are the electrical and instrumentation equipment, thrust vector control assembly and such recovery items as the parachutes. The Solid Rocket Motor case and attendant structure are all considered as a part of the structural assembly.

The current baseline for reuse of the Booster components is:

Structures reuse	40 times
Solid Rocket Motor Case and Nozzle	20 times
Thrust Vector Control assembly	20 times
Electrical and Instrumentation reuses	20 times
Recovery assemblies	10 times
Batteries	1 time

Structural design features to support the booster reusability program include such things as: (1) external protective coatings, (2) weld-free solid rocket motor case, (3) water-tight compartments using welded aluminum skins, (4) bulkheads for protection of the avionics (electrical and instrumentation items in the forward portion of the booster, (5) stiffening rings along the aft quarter of the booster structure to help take the water impact loads, and (6) the use of a smooth surface for the application of thermal protection material around the aft skirt which covers the nozzle. The Solid Rocket Motor case is designed with 0.009 metal thickness beyond that required for flight loads, fracture mechanics and water impact. To allow for wear due to "grit" cleaning during refurbishment for additional refillings. The Solid Rocket Motor case joints are described in Figure 41.

Current Status

An integral part of the structural design procedure includes a "Fracture Control Plan" for the Solid Rocket Booster and motor. This plan establishes the requirements for reporting, non-destructive testing (inspection), failure documentation, traceability, service life recording, proof testing, and environmental control of all portions of the structures defined as susceptible to structural failure due to flaws and cracks. In line with this plan, materials are selected and characterized for specific Solid Rocket Booster and motor environments and fabrication processes and refurbishment requirements. One of the problems in designing the booster/motor structures is to account for fracture under other than plane-strain conditions and to provide a practical means for predicting life under the complex time-stress histories occurring during pad operations, boost phase of the mission and recovery of the booster.

Other questions open at the time of the Panel's review deal mainly with the structural aspects of the booster element.

The specified reuse requirements and the designs to meet them are dependent upon the definitions of service life, safety factors and their derivation. Some thoughts relative to reuse which are pertinent to assuring a safe and cost effective booster are: (1) what will wear out or be rendered unserviceable after the specified number of reuses that will not wear out or be unserviceable after a greater or lesser number of reuses or cycles, and (2) what would be designed differently if the design were required to be made to meet a higher number of reuses.

Noise (vibroacoustic effects) generated by the Solid Rocket Motors and the Main Engines on the pad and soon after liftoff may impose severe requirements. The determination of these effects and the design constraints are still under study.

The booster design and expected attrition rates are highly dependent upon the extent of damage due to water impact loads. These stresses are dependent upon booster velocity, angle of impact, temperature of the structural material and surface conditions such as winds and sea state. Computer analysis programs have been developed to analyze (1) initial impact, (2) cavity formation and collapse of the water volume, (3) maximum booster penetration into the water and at the same time water penetration into the throat of the rocket motor, and (4) rebound and slapdown on the water surface.

There are also those events associated with the time when the booster is in the water and the ships and men begin to retrieve the boosters from the water. The degree that these operations impact the design of the booster has not been fully explored by the Panel at this time.

From the time a solid propellant rocket grain is cast until it has burned away in the performance of its mission, it is subjected to an array of stress-inducing environments including gravity, propellant curing loads, handling shocks and vibrations, and the pressurizations and accelerations that accompany ignition, launch, and flight. The possibility of safety related problems resulting from any one or combination of these environments will be examined in later reviews by the Panel.

Lightning protection requirements for the Solid Rocket Booster are similar to those for the Orbiter. Equipment requiring protection include pyrotechnics, thrust vector control sensors and switching circuits, all exposed electrical cables, and the integrated electronic assembly (data buss couplers, signal conditioner, multiplexer-deplexer, logic and network distributor.

Current lightning protection design measures include the follow ing: (1) single point ground on power circuits, (2) use of twisted wire pairs, (3) delays of 2 ± 1 millisecond in the many switching functions, and (4) use of metallic cable tunnel to protect cable runs forward and aft and the use of multi-grounded overall shields on all ordnance cabling.

Electrical interfaces between the Orbiter, External Tank, and the Solid Rocket Booster do not fully satisfy the lightning design criteria. Interface design is being studied at this time to obtain a reasonable solution to this problem. On the SRB program several tests are being planned to validate the lightning protection arrangement: (1) cable core test on SRB equipment as required, (2) full scale lightning test on the External Tank/Booster attach struts with ordnance installed, and (3) cable tunnel attenuation tests.

3.5 Launch and Landing Element

The launch and landing aspects of the Shuttle program are considered an element in the same manner as the Orbiter element, External Tank element, SSME element and the Solid Rocket Booster element. The Launch and Landing element is under the jurisdiction of NASA's Kennedy Space Center. There are other prime and secondary sites, but the discussion here centers on the requirements, design, development, validation, launch, and landing preparation plans at KSC.

The design and operation of the launch/landing site is as much a key to achieving a low cost Shuttle system with rapid turnaround after a flight as any other element of the program. KSC's past roles on the manned and unmanned programs, in which facilities and know-how have been developed for the receipt inspection assembly, checkout and launch, plays a large part in their ability to meet their current and projected role in the Space Shuttle program. More specifically the Launch and Landing Project conducted at KSC covers the following activities:

(a) Shuttle vehicle element receiving (including all that goes with such activities, e.g., inspections), assembly of the Shuttle vehicle including buildup from the elements to the total ready-to-fly vehicle, checkout and launch.

(b) Recovery/retrieval operations for the Orbiter and Solid Rocket Booster.

(c) Ground Operation taking into account the necessary sustaining engineering, logistics, maintainability and the turnaround operations.

(d) Facilities and Ground Support Equipment, such as the Runway, Orbiter Processing Facility, Launch Control Center, Flight Test Control. A major innovation will be the Launch Processing System to satisfy the requirements for an automated launch checkout.

With regard to payloads, KSC will prepare and install the Spacelab delivered by the European consortium, the automated payloads, the Air Force Interim Upper Stage Vehicle and the TUG vehicle and all other payloads.

The KSC interface with the NASA Flight Research Center at Edwards, California, includes a major role in the Approach and Landing Test program.

At Vandenberg Air Force Base, California, KSC will assist the Air Force in planning and will provide expert help in the area of turnaround operations, facilities, launch support equipment and payloads operations.

Recognizing that the Panel has not had the opportunity to examine the Shuttle program from the KSC viewpoint in any detail, the focus was on a small number of areas of particular interest to the Panel at this time: Solid Rocket Booster retrieval, landing facilities and landing controls, Orbiter Thermal Protection Subsystems maintenance, turnaround operations, and Launch Processing Subsystem. The Panel did, however, receive an orientation briefing on the total KSC role, responsibilities and plans to carry them out.

3.5.1 Solid Rocket Booster Retrieval

Systems Design

So we have noted, the Marshall Space Flight Center has responsibility for the development of the Solid Rocket Booster, including the intact reentry of the booster into the ocean. KSC, however, is responsible for developing the retrieval system for returning the boosters to dry land for refurbishment and preparation for reuse.

Retrieval of the boosters, parachutes, and other recoverable objects will be accomplished using surface vessels. The retrieval vessels will tow the boosters to KSC; other objects recovered will be brought onboard the vessels themselves. Shuttle developmental launches will, of course, be used to test and refine vehicle recovery/retrieval systems. The boosters are expected to impact at a point some 130 to 150 nautical miles downrange in an impact footprint defined as a 10 x 33 nautical mile ellipse. Once the boosters are located and the vessels are near enough, divers are sent to plug the nozzle.

Then the booster is dewatered and it attains what is called a "log" mode. Parachutes are coiled on reels and the nose cone frustum is lifted on board the vessel and the boosters towed home.

Current Status

The retrieval system definition is in its early stages and will be examined in more detail as the necessary design, interface and operational details are worked out. Among the questions yet to be answered are the number of tracks to have on the SRB impact recorder, and the baseline for the "station set" used in the SRB retrieval and disassembly

3.5.2 Landing Facilities and Landing Control

Systems Design

These facilities and controls can be divided into the following specific items: (1) Primary landing sites, KSC and VAFB used for test and operational flights, (2) secondary landing sites with particular emphasis on Flight Research Center/Edwards AFB used for the Approach and Landing Test program using the carrier aircraft, and (3) the Mission Control Center at Johnson Space Center, Houston, Texas.

The Orbiter Landing Facility at KSC is located approximately 1.5 miles north and west of the Vehicle Assembly Building (VAB) and extends 15,000 feet to the northwest. It is composed of the following:

(a) Airfield pavements of 15,000 ft x 300 ft with 1000 ft. overruns on each end, a two-way that is 10,600 ft. long and 50 ft. wide leading to the Orbiter Processing Facility, and a parking apron just off the main runway and coincidental with the two-way 490 ft. x 550 ft.

(b) Airfield lighting along the standard approach, runway touchdown and centerline, and the runway edge.

(c) A landing aids control building at the southeastern end of the runway containing hardware for flight and ground control including the Orbiter landing instrumentation system with S-band/UHF communications, TACAN, Microwave Scanning Beam Landing System (MSBLS) and related installations.

Current Status

The current status of the Orbiter landing facility at KSC is as follows:

(a) Construction awards have been made for Phase I and II and the requirements for Phase III are in the planning stage.

(b) Phase I construction on the runway, two-way, parking apron, arifield lighting, electrical power and water mains is to be completed in August 1976.

(c) Phase II construction on the landing aids control building, instrumentation facility, utilities support and cabling systems is expected to be completed in September 1976.

(d) Phase III, TACAN, Communication systems (MSBLS, Comsec, etc.), propellant and gases systems, high energy aim point, cinetheodolite system, Orbiter mating device, and other landing support equipment are all in planning and requirements review stages.

(e) Test planning includes the utilization of the Shuttle Training Aircraft to validate the ground landing aids and control systems.

(f) Significant issues at the time of the Panel review (March 1975) were: (1) Additional facilities required for cinetheodolites and the high energy aim point, (2) Runway grooving spacing which is to be between 1" and 2", and (3) While the microwave Scanning Beam Landing System has been selected to support the Orbiter landing, its location at the end of the runway is under discussion (i.e., on the centerline or off the center line).

The current program specifications call for the Johnson Space Center's Mission Control Center to retain control of the Shuttle elements (vehicle and, particularly, the Orbiter) throughout the mission including entry, landing and rollout to a stop on the runway. There is still some discussion as to the best location for control of the Orbiter during the Terminal Area Energy Management portion of the mission (from about 70,000 ft. altitude to roll-out on the runway). The Panel will follow this question until its resolution to assure that crew safety and successful vehicle return receive appropriate attention.

During the last half of 1974 the question "Is there a need for an overrun barrier at KSC, Edwards AFB or Western Test Range?" was asked in earnest. As presented to the Panel, a thorough analyses was made to determine the need for such barriers. The factors influencing the requirements were: (1) touchdown point on the runway, (2) velocity of the Orbiter at touchdown, (3) Orbiter characteristics, e.g., drag, stability, etc., (4) coefficient of friction (wheels to runway), and (5) the brake system capabilities. The "worst cast" roll-out performance used in the analysis assumed: hot day, wet runway (ungrooved), landing weight of 230,000 pounds, maximum landing velocity and landing long, and with a single tire blow-out at landing.

Analysis indicates that the Orbiter would require a total runway of 15,530 ft. Since the runway is 16,000 ft., the runway barrier requirements were deleted.

3.5.3 Ground Turnaround

Systems Design

Turnaround operations include:

(a) Landing (a portion of which is covered in previous paragraph)

(b) Orbiter safing, maintenance and checkout (this includes the Thermal Protection Subsystem maintenance)

(c) External Tank and Solid Rocket Booster preparation

(d) Shuttle Vehicle Assembly

(e) Pre-launch checkout and launch

Current Status

During the early Panel reviews it was evident that the 160 hour requirement is a major design driver. Therefore, the Panel is interested in assuring that this requirement will not adversely affect ground or crew safety. KSC is trying to meet this turnaround requirement and assume a safe vehicle through the use of the computerized Launch Processing System (LPS). In addition, ground operations are being designed to use proved techniques and optimize the level of inspection while reducing subsystem level checkout time as performance confidence is achieved. Evolution of the 160 hour turnaround is shown in Figure 47.

Two of many management aids in respect to turnaround are mentioned here because of their significance. The Shuttle Turnaround Analysis Group (STAG) chaired by KSC, has been established as the Government-contractor team responsible for Shuttle System integrated program turnaround allocations and assessments. The system integration contractor (Rockwell International, Space Division) assists KSC in the evaluation of the element-level reports and analysis reports.

The Shuttle Turnaround Analysis Report (STAR) is prepared by KSC and is submitted to the JSC Space Shuttle Program Office to depict the current status of the operational turnaround functions.

KSC considers the following four basic areas in developing the operational team concept: (1) Definition of functions in detail, (2) degree of autonomy to be provided, (3) depth of management oversight required, and (4) the varied personnel skills necessary to achieve the turnaround objectives.

The handling of the Orbiter TPS is one of the more difficult assignments during the turnaround period. Inspection and refurbishment will require constant attention to assure the adequacy of the TPS for the next mission. The TPS tiles are fragile in comparison to most other items on the Orbiter and must be handled accordingly. A major element of the post landing operations at KSC is the performance of preliminary checks of the TPS surface to determine in a gross manner the quantity of damage sustained during the mission and particularly during entry and landing. Once the vehicle has beem taken to the Orbiter Processing Facility a detailed examination of the tiled surface is made. The methods by which this will be done have not been fully defined, but will be examined in the future reviews.

The Launch Processing System makes use of modular, or building block, structure which will allow the hardware and software to be configured to accommodate differing requirements in the checkout, maintenance, and launch functions. In the launch support configuration, test engineers, manning LPS consoles in the Launch Control Center, perform testing and prepare for launch. The LPS in the maintenance and checkout configurations has LPS consoles located in areas such as the Orbiter Processing Facility, Vehicle Assembly Building High Bays and Hypergolic Maintenance Facility. The following points were made to the Panel regarding the requirements for the LPS in the checkout configuration:

(a) Tese automation - faster, repeatable, better discipline, realtime test results;

(b) Standardization of hardware and software - computers, displays, data transmission, hardware interfaces documentation, training and maintenance;

(c) General purpose/high density consoles - fewer operations per system, more burden on the machines and th multiple use of equipment;

(d) Test engineer oriented language to eliminate middleman-programmer, make engineer responsible for the entire system;

(e) Rapid access to engineering data and work control system.

Open issues at the time of the Panel's review included the continuing review of requirements for the system, preliminary design review planning and development flight instrumentation data processing and LPS requirements for the Payloads.

The station set is defined as an accumulation of units of GSE required to support a specific activity or phase of vehicle assembly, test launch or pre-launch. There are three types of GSE units or models in order to affect the greatest degree of cost effectiveness. These are:

Type I - Critical to 160-hr timeline or final system verification or hazardous operations.

Type II - Functional interface with the vehicle.

Type III - No vehicle interface or interfaces with vehicle but requires minimum design control.

The Panel asked about the requirements with respect to reliability and safety. The following requirements and philosophy apply: (a) The Launch essential and safety critical ground support items are identified and that particular list is updated and provided to management for their understanding and control, (b) Failure Modeand Effects Analyses (FMEA's) and hazard analyses are required for all launch essential and safety critical GSE, (c) All launch essential and safety critical GSE require that for the certification program, acceptance shall consist of one or any combination of analysis, similarity or actual testing.

One of the open questions to be resolved is the timelines of documentation data from the element contractors (Orbiter, External Tank, Solid Rocket Booster, Space Shuttle Main Engine) which affects

KSC's ability to plan for and define spares and maintenance requirements and affects the facility design activity as well.

One of the challenges during turnaround will be the assembly of the total Shuttle vehicle, since Shuttle elements require very tight stacking tolerances, well designed equipment, and well trained personnel to assure proper control of stacking procedures.

Factors being considered now in the design of the mobile launcher and launch pad are:

(a) Engine exhaust rebound back up into the space vehicle creating a vibroacoustic problem, as well as thermal problems.

(b) The engine quench system (water system).

(c) The hole-sizing in the platform to accommodate the Solid Rocket Booster exhaust.

(d) The requirements for payload unbilicals.

(e) Facilities to minimize External Tank ice formation and affects of ice shedding.

(f) Orbiter Thermal Protection Subsystem tile protection.

(g) Payload handling requirements and their implementation, e.g., the payload cleanroom facility.

4.0 SAFETY, RELIABILITY, QUALITY

4.1 System Design

For our purposes reliability (probability of failure), quality (excellence in producing hardware/software), safety (freedom from injury or loss) are all a part of the so-called "Risk Management System" or "Space Shuttle Assurance Program." These are obviously interrelated activities and as such are not covered separately in this document.

The Space Shuttle risk management system is built on prior manned flight program experience and modified to meet Shuttle requirements. Safety analysis process is shown schematically in Figure 48. Each of the element contractors and each of the participating NASA Centers conduct its own safety, reliability and quality programs. In addition, the Rockwell International Space Division in Downey, California, as the system contractor, conducts an integrated safety analysis operation. The total Shuttle program requirements including reliability, safety and quality are delineated in the Level II program requirements' documents JSC 07700, Volumes I-XVIII. Compliance with these requirements is further addressed in numerous documents. For instance, the approach to reliability is addressed in Volume I, "Master Verification Plan." Volumes II through V have the requirements for the element verification plans. The element verification plans describe the way the requirements are to be met, e.g., test, analysis, and inspection. The specific plans covering reliability, quality and safety are submitted by the element contractors to the appropriate project elements in NASA for review and approval.

4.2 Major Reviews

The major risks and uncertainties determined by various assessment teams and permanent organizations are reviewed by management as a part of their review system. The Preliminary Design Review for Orbiter No. 102 and the Shuttle System Preliminary Design Review are examples of such events. Figure 49 shows that at the time of the Orbiter 102 Preliminary Design Review twenty (20) subsystem failure modes and effects analysis documents have been issued. These documents covered 947 components in terms of possible failure modes and their impact on the crew and mission.

The Safety Analysis Report indicated 200 Orbiter hazards and the corrective actions being taken. This analysis covers such situations as: (1) illness/injury/loss of personnel, (2) collision/impact/erosion, (3) fire/explosion/implosion, (4) loss of or unsafe environment, (5) crash landing/ditching, and (6) loss of flight control.

Hazard analysis is performed at the subsystem level and, in cases where Failure Mode Effect Analysis have identified critical items for the Critical Items List, the analysis is performed to a lower level of detail.

The Critical Items List contains the single failure points and criticality 3_I items identified by the FMEA. Criticality 3_I are all those items not having a potential effect on loss of life or vehicle or loss of mission. They also meet one or more of the following criteria: (1) redundant elements are not capable of checkout during normal ground turnaround, (2) loss of a redundant element is not readily detectable in flight, or (3) all redundant elements can be lost by a single credible event or cause.

4.3 Safety Analysis Process

The safety analysis process for the Shuttle program is being implemented in the following basic steps: (1) identification of safety concerns, (2) analysis of safety concerns for credibility and criticality, (3) initiation of Shuttle hazard analyses, and (4) tracking and closing out Shuttle hazard analyses. Each of these steps is described below.

4.3.1 Identification of Safety Concerns

A system safety concern is any design or operational issue that has a potential impact on personnel or hardware. The concern may be identified by any person or organization on the program and must be dispositioned. For instance, the system contractor's safety office reviews the element contractor's hazard analyses and FMEA's to determine if a possible safety problem may propagate across elements of the Shuttle from an identified hazard or failure on any one element.

The system contractor's safety office also reviews the planned operations of the Shuttle for potential safety problems. This is to be done for each mission phase. In addition there is a continuing effort by Rockwell International's Space Division engineering and other groups to identify other issues which have a safety implication.

4.3.2 Analysis and Resolution of Safety Concerns

Every safety concern identified to the system contractor's safety office will be analyzed for credibility and criticality. Credibility means that there is a real possibility that the event may happen. Criticality means that, if the concern occurs, there would be personnel injury, loss of the vehicle, or major damage to ground facilities. If the concern is both credible and critical, then action has to be taken to preclude undesirable consequences or minimize possibility of occurrence. If the concern cannot be resolved, management must review and decide upon the risk to be accepted. Experience has shown that the great majority of the safety concerns identified can be shown to be not credible or critical.

4.4 Shuttle System Safety Concerns

Safety concerns as presented to the Panel during its May inspection trip to the Space Division of Rockwell International are shown in Table XI.

The hazards resulting from fluids used throughout the Shuttle mission, with particular reference to the fire and toxicity problems, are outlined in Table XII. Only two phases of the mission would appear to be essentially clear of problems, the ascent and orbit periods. A partial resolution of this problem was to separate incompatible materials and environments by compartmentizing or sealing off of the Orbiter where practical so there were no hazardous fluids in the pressurized crew compartment. In addition to sealing off compartments, an active purge, such as dry nitrogen gas, is used to dilute the concentration of hazardous gases. Warning devices have been developed to alert the crew and ground control. Contingency procedures at launch pad and during mission will be formalized. Figure 50 depicts this approach schematically.

The Orbiter flight vent and purge system described in Section 3.1 "Orbiter Element" to minimize the hazardous gas problem is augmented by the ground hazardous gas detection system designed and developed by the KSC organization. This ground system has been defined and the remaining major development items are the sensors for the cryogenic and hypergolic portions of the system. For the cryogenic subsystem, these are mass spectrometer, electrochemical sensors, and portable hydrogen sensor. For the hypergolic subsystem these are the portable hypergolic sensor and the air oxidation chemistry analyzer hardware. The flight system operation depends upon defining what is a hazardous fluid condition. For example, dissassociation of leaked fluids must be known for detection and hazard assessment (N_2O_4 in humid atmosphere forms nitric acid) as well as autogenous ignition temperature at altitude (low pressures) for Orbiter fluids. These data will be obtained in the coming months through a series of inhouse and contract activities.

Current Status

The Panel requested that the following safety concerns be discussed during their visits to both NASA Centers and Contractors. Each of these concerns is presented below along with the current status at the time of our review.

Solid Rocket Booster Ignition Overpressure - Large over-pressures on Orbiter and External Tank structures and surfaces may be imposed by the booster exhaust shock-wave at ignition. The over-pressure wave is assumed to reflect asymmetrically from the pad flame deflector and travel up the vehicle, applying pitch plane loads. Tests are to be conducted on a Shuttle model at MSFC to acquire valid pressure distributions and intensities. Resolution has been targeted for November 1975.

Unscheduled SSME Shutdown During Boost - SSME design provides internal, automatic shutdown mechanisms to achieve safe engine shutdown when critical performance parameters are not within tolerance requirements. Investigation has shown that the remaining two engines are necessary to achieve intact abort, and that a two-engine-out condition may well result in vehicle loss. One approach being studied to resolve this concern is to have a single engine shutdown inhibit or disable the internal shutdown mechanisms for the two remaining main engines.

This inhibit capability would be accomplished by automatic electrical "lockup" of the engine control valves in their last position, and by incorporating an inhibit coil on the emergency shutdown solonoid.

Crew Rescue From Orbit - If for any reason the vehicle is unable to return to earth from orbit, no rescue capability exists during the early flight test program, but a "rescue orbiter" would be available during the operational periods. Various ideas are being explored to achieve a rescue capability during the early flight test portion of the program.

Solid Rocket Booster Thrust Mismatch - Booster thrust mismatch can occur at any time during the burning period. The periods of greatest concern are at liftoff, maximum dynamic pressure and at the end of burning period (tailoff). During liftoff, the specification for the Shuttle system calls for a maximum mismatch of 300,000 pounds. This value appears conservative based on results of Titan IIIC statistical analysis of ignition transient. Ignition transient is still being evaluated by MSFC/Thiokol/Rockwell for better definition of the time mismatch action. The impact of a mismatch at the maximum "q" condition is to add an additional load on the flight control system elements in the yaw direction. The Shuttle structure and flight control system has the capability to adequately account for such additional loads. The Booster tailoff thrust differential indicates that a 710,000 pound mismatch is controllable with normal control capability. The

710K value has been established as a requirement which occurs about 115 seconds after ignition. However, when Booster nozzle actuator or SSME engines fail the separation of the Booster from the External Tank is delayed for up to 4 seconds to reduce the mismatch thrust and provide acceptable separation conditions. The extent of the control capability that can be exerted during tailoff continues to be studied to assure adequate flight control and separation ability.

4.5 Orbiter Safety Concerns

Orbiter Structural Elements

Structural deformation may prevent emergency egress from crash landings. Orbiter 102, to be used for first orbital flights, has added overhead escape panels which are used in conjunction with ejection seats, but the panels will remain after ejection seats are removed. There is a current study to ascertain the value of using the overhead hatches on all Orbiters. The ability to compartmentize or isolate hazardous fluids is discussed in the fire/toxicity section above. There must be continuous control to assure that hardware assigned to the "structures" category does not include items similar to the Skylab meteroid shield.

Doors

The major point is that during entry all doors must be closed. If the payload doors do not close then the crew must use EVA and secure them. There are continuing studies on elimination of doors and methods of assuring their proper positioning throughout the mission.

Payload Retention

Payloads must be adequately constrained during normal or abort landings to avoid damage to the crew.

Thermal Protection System

This has been covered in detail in Section 3.

Hydraulics

Loss of flight control due to failure of single actuators which are used for elevon control was studied by Rockwell International and NASA. They accepted the risk of being involved in relying upon a single actuator.

Ejection Seats

The possibility of collision between the ejection seats following ejection is under evaluation at this time.

Orbital Maneuvering Subsystem

Large quantities of OMS propellant requires that it be managed to assure proper center of gravity conditions during nominal and abort trajectories. Orbiter aerodynamics analysis and mass properties analysis are being performed to determine allowable residual propellant quantities and the quantities to be dumped. This work is expected to continue through the next fiscal year with resolution at the end of that time.

Data Processing System (Software)

Generic software errors may not be detected in the software verification program based on prior experience in this area. A study is under way to determine the degree of degradation due to expected errors and possible work-arounds to maintain operational control.

Hydrogen Fire During RTLS Abort

During the return to landing site abort a hydrogen concentration is expected to exist in the wake of the Orbiter. The location of the exhaust, vent, and dump locations are a safety concern.

Landing/Deceleration Subsystem

The Panel has questioned the ability of the landing gear gravity deployment system to support the Orbiter Landing trajectory (altitude, time, distance). What is the basis for confidence in the reliability of the free-fall system that landing gear will be in the down and locked position? When working properly is there sufficient time to achieve the down and locked position prior to touchdown? What contingency plans are available if the landing gear system does not operate properly?

Because of the Panel's interest in this area a brief description of the gear units and doors and their operation during landing provided here for a better understanding of the above three questions. Figures 51 and 52 show the nose gear and main gear installation. The nose gear retracts forward and up in the forward fuselage, and the main gear retracts forward and up into the wing. The weight of the nose gear system is about 1300 pounds and the individual main gear about 2500 pounds. Crew selection of landing gear "down," after the arm switch has been selected, accomplishes two functions for the nose gear. It energizes the landing gear selector valve, porting pressure to he down-side of the nose gear strut accuator and the down-side of the uplock release actuator. In addition, a redundant pyrotechnic backup system is sequenced to release the uplock, if the primary hydraulic system fails to operate in a "short" period of time.

There is only one primary hydraulic power system configuration for the nose gear operation. The gear then "free falls" from the wheel well, there by driving the mechanically linked doors open. Aided by weight and aerodynamic effects, the gear should reach the full down position and be locked in position by the action of a spring loaded bungee. The motion of the gear before locking down will be damped by an oil snubber to prevent any damage to the locking linkage. Down pressure to the strut actuator aids in the extension cycle, but in the event hydraulic power should be lost, it is not required to extend or lock the gear down. Gear downlock and gear/door uplock switches provide cock-pit indication of gear position. The extension cycle is designed to be accomplished at all velocities up to and including 300 knots within a time limit not to exceed 10 seconds.

The main landing gear extension cycle is identical to the nose gear with the following exception. In place of the backup pyrotechnic release system, two additional secondary hydraulic systems are provided for the uplock release actuator. Therefore, crew selection of landing gear "down" ports down pressure from the primary hydraulic system to the strut actuator and to the uplock release actuator. It should be noted that any one of the three systems is sufficient to release the main gear and door uplocks and initiate gear extension. Primary pressure to the strut actuator aids extension but is not required, as the weight and aerodynamic effects on "free fall" gear are sufficient for gear extension and locking via a spring bungee.

There is an Autoland System interface with the landing gear system which has not been fully defined as yet. Operation of the gear, during the landing operation, is actuated as late as 14 seconds before touchdown. Manual gear extension is achieved by the pilot throwing a gear extension switch after he sees a light on the display panel. It is expected in the Autoland system that the autoland hardware would accomplish the same action at about the same time. The problem then is obvious. With a maximum of ten seconds allowed for the gear to go into the down and locked position and the action initiated some 14 seconds before touchdown, there is little if any leeway for problems in response or deployment. Therefore, the reliability of the system must be very close to 100 percent during that 14 second to 4 second period prior to touchdown or some alternate action capability must be supplied along with a longer period to achieve down and locked gears.

4.6 Range Safety

Current requirements have established the range safety system as an add-on unit only for the design, development, test and engineering flights. The baseline system is shown in Figures 53 and 54. This system is still under discussion between NASA and the Air Force. Basically, the range safety system is required to provide for: (1) safety of lives and property, both on the ground and in flight, (2) External Tank propellant dispersion, and (3) protection against overt/covert destruction of the vehicle and against "false alarms" due to electromagnetic interference or spurious signals.

Issues under study at this time include the following:

(a) External Tank propellant dispersion and their impact on Orbiter (MSFC).

(b) Crew ejection seat inhibit which inhibits range safety system operation. Adequate procedural safeguards and time delays appear necessary to maximize astronaut survival if destruct action is required.

(c) Shutting down of the Orbiter's main engines upon receipt of the range safety destruct system arm signal.

(d) Inflight safing of the "safe and arm" device by the Orbiter software.

(e) Monitoring of the safe and arm device to prevent inadvertant safing of the range safety destruct device.

4.7 Materials Usage and Control

One of management's major controls to assure the design and construction of safe and efficient hardware is in the materials' usage area. This includes not only the compatibility of materials with their environment from the standpoint of flammability and toxicity but also with regard to their stress corrosion/fracture mechanics susceptability. The Shuttle program, using the experience gained from prior manned programs and military and commercial activities, has developed materials' programs for each element as well as for the integrated Shuttle system. Requirements set by the program and affecting all elements within the program are set forth in Paragraph 3.6.2.1, JSC 07700, Volume X, "Space Shuttle Flight and Ground Systems Specification," and the JSC document SE-R-0006A, dated April 1973, "Requirements for Materials and Processes."

These requirements include the following:

(a) Each element must have a controlling document on materials and processes stating the specifications and standards to be used. There is a drawing review and sign-off by a materials' engineer.

Materials testing and "allowables" are covered by:

(a) Flammability, odor, outgassing in NASA NHB 8060.1A.

(b) Thermal-vacuum stability in NASA SP-R-0022.

(c) Special tests as approved by JSC where it is felt