THRUST VECTOR CONTROL STUDY FOR LARGE (260 IN.) ROCKET MOTOR APPLICATIONS

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ABSTRACT

A study program was completed which compared three types of thrust vector control (TVC) design concepts for application on a 260 in. diameter solid rocket motor (SRM). The three TVC design concepts were: (1) mechanical interference, (2) secondary liquid injection (LITVC), and (3) movable (flexible bearing) nozzle. The comparison was made on weight, cost, simplicity, and development risk. During the preliminary design phase, mechanical interference TVC design approaches were eliminated from further contention. Preliminary system designs were prepared for LITVC and flexible bearing nozzle TVC concepts; a tradeoff comparison was performed and a single detail design made for each of these two TVC concepts. The major components of the selected LITVC system consisted of a single toroidal tank, which contained the injectant (N_2O_4) and the pressurant (GN_2), and 16 equally spaced (about the nozzle circumference) electromechanical single-pintle injector valves. Two electrohydraulic servoactuators were used to vector the flexible bearing nozzle. The power source selected was a passive cold gas blowdown system using GN_2 as the pressurant.

INTRODUCTION

Reduced steering requirements for large solid propellant booster motors have dictated reexamination and consideration of TVC design variations which would reflect the lower steering angles and angular rates. A study was sponsored by NASA to define an optimum TVC system to fulfill the reduced requirements with reduced cost and complexity, improved reliability, and minimum development risk. Designs were prepared for mechanical interference, LITVC, and flexible bearing approaches. A tradeoff was performed within each of the approaches so that a single detail design of the most promising system could be made for each TVC concept. An optimum TVC system could be selected from direct comparison of the detail designs.

The basic vehicle employed to evaluate each system was a 260 in. diameter solid rocket motor (SRM) booster and S-IV-B upper stage similar to that described in Contract NAS8-21051 as the MLV-SAT-1B-5A. Identical fixed nozzles were used for both the LITVC and mechanical interference TVC systems, the only modifications being for attachment of the necessary TVC components. There was no modification required to the basic motor, except that it was assumed that access doors would be made available in the aft skirt. Design data used for the flexible bearing nozzle were supplied by NASA and were developed by Aerojet-General Corporation (Contract No. NAS 312049).

The study was divided into three phases: preliminary design, detail design, and cost analysis. This paper will deal primarily with the first two tasks, touching on the cost analysis only in the identification of major component cost. The following sections describe the nozzle and the three TVC systems. The tradeoffs conducted within each system are discussed, the selection of candidate systems, and the detail designs are presented. Since the maximum vector angle required is a function of the point of application of the side force, each TVC system has different design vector angle requirements. These requirements are presented in the appropriate section of the paper.

NOZZLE

Basic Fixed Nozzle--The same basic convergent-divergent nozzle (with appropriate modifications) is used for the LITVC and mechanical interference concepts. The nozzle has an initial throat diameter of 89.1 in., an initial expansion ratio of 8.515, exit cone half-angle of 17.5 deg, and exit diameter of 260 in. The aft closure mounting flange, whose upstream face is 55.10 in. forward of the nozzle throat, incorporates a 180 in. diameter mounting circle; the exit plane is 277.86 in. aft of the nozzle throat. The throat contour radius is equal to the throat radius. The above basic fixed nozzle dimensions, with the exception of the aft mounting flange location, also apply to the movable nozzle designs.

Low cost ablative materials are used to line the nozzle wherever possible: silica and asbestos filled Buna rubber from the aft closure mounting flange to a point 42 in. forward of the throat; and canvas, from 90 in. aft of the throat to the exit plane. Carbon cloth, backed with 0.42 in. thick silica cloth, lines the nozzle in the throat region.

Alloy steel (4130) is used as the structural support between the aft closure and exit cone mounting flanges. The exit cone structure is fiberglass, filament wound to a steel nozzle mounting flange.

Total nozzle weight is 47,900 lb; 38,120 lb of ablatives and insulation and 9,780 lb of structure.

Aerojet (AGC) Flexible Seal Nozzle--The AGC nozzle is a submerged movable design with a forward pivoted flexible seal joining the fixed and movable nozzle sections. The movable section structure begins at the exit cone mounting structure and extends forward beneath the throat, continues into the nozzle's submerged portion, extends down to the nose, and then loops back to the forward end ring.

The cylindrical cross section AGC seal has conical shims. The seal assembly consists of four 0.7 in. thick steel shims with an O.D. of 115.6 in. and five 0.3 in. thick elastomer layers. The pivot point is located 60.5 in. forward of the throat. This location reduces the missile turning moment and requires nozzle vectoring ± 1.61 deg to achieve the required thrust vectoring of ± 1.5 deg. The 180 in. diameter bolt circle is located 27.0 in. forward of the throat.

Total nozzle weight is 56,298 lb, 36,262 lb of ablative and insulation material and 20,036 lb structure. The fixed section weighs 8,899 lb and the movable section 47,398 lb.

MECHANICAL INTERFERENCE TVC SYSTEMS

A mechanical interference TVC system (MITVC) alters the resultant direction of axial thrust vector by mechanically interfering with the nozzle exhaust flow. The mechanical interface usually occurs in the form of a forward facing step in the supersonic section of the nozzle. Two TVC systems sometimes considered movable nozzle types, the flexible exit cone, and supersonic splitline were included in this section because both movable sections of the exit cone mechanically interfere with the exhaust flow.

LITERATURE SEARCH

Available published literature was reviewed for MITVC system data. Only those systems which are or have been operational, or are under development, were investigated. This restriction was imposed by considerations of reliability and cost which were the two most important criteria of this study. The six MITVC system candidates were: mechanical probes, jetevators, jet tabs, flexible exit cone, supersonic splitline and jet vanes.

Search results showed that development of MITVC systems had been concentrated on obtaining the maximum possible TVC angle with each system for a particular application. Few data were available on small vector angles (1 to 2 deg). There was also a general lack of scale-up data, resulting in many approximations for control element sizing. In all systems, choice of materials proved a considerable problem.

DESIGN REQUIREMENTS AND SELECTION CRITERIA

The requirements provided by NASA are as follows: Total side impulse requirement was 69.6 deg-sec. Maximum TVC angle was 1.03 deg acting at the nozzle exit plane. The magnitude of the side force requirement varied, depending upon its point of application in the nozzle; however, the maximum turning moment on the vehicle was fixed at 109.6×10^6 in.-lb. Maximum slew rate was 3 deg/sec. Motor burn time was 143 sec. Combustion gas temperature was assumed to be about 5,800° F.

Selection of the more promising MITVC system was based primarily upon reliability with respect to current technology and potential cost. Wherever two or more MITVC systems compared closely, weight, performance loss, development history and current development status provided secondary evaluation criteria.

MITVC SYSTEM SELECTION

Following the literature search the jet vane and flexible exit cone concepts were eliminated from further consideration on the grounds of a history of low reliability and high development cost and risk. The remaining four systems were studied in more detail.

Mechanical Probes--Analysis of available probe data indicates that the side force ratio Fs/Fa of an optimum probe system is directly proportional to the blockage area ratio Ap/Ai at the probe insertion point. Optimum probe location appears to lie at an X/L ratio of approximately 0.5. Probes were sized for X/L ratios of 0.4, 0.5, and 0.6, and for one, two, and three probes/quadrant. The results are summarized in Table I.

The projected area of a single probe located at X/L = 0.5 is 490 sq in. Probe loading is almost constant (141,000 lb) for all locations, since probe projected area requirements increase at about the same rate as the total pressures acting on the front face of the probe decrease. However, the bending moment acting on the probe may change, depending upon probe insertion depth. For a square probe (22.1 in. sides) at X/L = 0.5, the bending moment is very large (4.68×10^6 in.-lb). This may be reduced in two ways: (1) reduce insertion depth by increasing probe width; (2) adopt a multiple probe system.

Little work has been done on multiple probe systems. One test* indicated that distance between probes is critical if severe nozzle erosion is to be minimized.

At X/L = 0.5 a maximum thrust loss of 0.5 percent was estimated which would require more than 7,000 lb of additional propellant to achieve the total side impulse requirement for the 260 in. diameter launch vehicle.

Since the probe is completely immersed in the exhaust stream when providing TVC, its construction presents a problem. Some form of outer refractory shell with an inner graphite-type heat sink is the only form of construction for which survival has been reliably demonstrated in this type of environment. Uncooled probes tested to date have been small enough that solid tungsten could be used. Ablative probes can only be considered for short duration burn times; they are inefficient, since they must be overdesigned for the initial part of any duty cycle. The resultant larger probe carries higher bending moments and requires larger actuation forces because of an unavoidable component of the front face pressure force acting on the probe tip.

*Eastman, J. M. and Leining, R. B.: "Cooled Probe Thrust Vector Control," Bendix Aviation, South Bend, Indiana, January 1963 (Confidential).

Cooled probes offer the advantages of smaller dimensions, since the coolant contributes to the side force; nonrefractory construction may be employed as a result of lower probe operating temperatures. Investigation of various coolants revealed an excessive weight penalty in all instances (Table II). The values generated were based upon the lowest flow rate at which probes have survived (0.2 lb/sec of water per sq in. of probe projected area), determined by Bendix in their cooled probe program.

Development history of cooled and uncooled probes generally has not been impressive; both scaling data and mechanical solutions to probe construction are limited. Probes were eliminated from further consideration because of development risk.

Jetevators--Jetevators are one of the more highly developed forms of MITVC systems; however, operational jetevators (Bomarc and Polaris) withstood gas temperatures no greater than 4,800° F, and molybdenum was used extensively. The anticipated 5,800° F environment for the 260 in. diameter motor application reintroduces a materials problem.

The extremely large and heavy control elements are located immediately outside the 260 in. diameter nozzle exit plane, where small changes in control element geometry cause large changes in weight. Minimum jetevator deflection (and therefore jetevator size) is achieved with a spherical inner ring profile. This also offers the advantage of minimum torque requirements.

Total weight of the control elements for a jetevator TVC system meeting the requirements of the 260 in. diameter launch vehicle was estimated to exceed 11,000 lb. A 5 deg jetevator deflection, spherical inner ring profile, and a configuration employing a refractory face plate material backed up by a graphite-type heat sink were assumed, with an insulated steel ring providing the support structure.

Polaris and Bomarc test data indicate a thrust loss of approximately 0.55 percent, resulting in a requirement for 8,500 lb additional propellant. Actuation torque of approximately 300,000 in.-lb and a stroke of $\pm$12.1 in. would be required.

As in other systems, the motor actuation time precludes use of ablative control elements. Savings in weight would be partially offset by the larger jetevator ring required and greater structural bending loads. The nonuniform eroding surface would change the side force/jetevator deflection characteristics, thus introducing unpredictable control.

Fabrication of the spherical inner ring profile in a refractory material would be extremely costly and a large amount of material would be required. Any deviation from this contour would increase torque requirements by an order of magnitude.

The resultant increase in weight of nozzle support structure at such a large diameter would be prohibitive.

The jetevator TVC system for the 260 in. launch vehicle would weigh almost as much as a jet tab system and in view of the potentially higher cost and higher development risk, it was eliminated in favor of jet tabs.

Jet Tabs (Figure 1)--Jet tab design was based largely on data from the 156 in. diameter motor program*, mainly because: (1) the tabs developed during this program would produce almost 60 percent of the side force requirement of the 260 in. diameter launch vehicle; (2) a tab configuration had successfully demonstrated the capability to survive a 5,400°F exhaust environment. Much of this technology could thus be directly applied to 260 in. diameter motor jet tab design.

A multiple tab system, two tabs per quadrant, was selected in preference to a single tab system. With a projected area of 1,592 sq in., construction and handling of a single large tab would be exceedingly difficult. In any case, a single tab violates the aft skirt envelope of the 260 in. launch vehicle. The area of each tab in a dual tab system would be 850 sq in. Other advantages are reduced power requirements, more flexible geometry, and increased reliability of the launch vehicle as a result of the redundancy inherent in a multiple tab system.

A composite structure comprising a refractory face plate, graphite heat sink and insulated support structure, was assumed for the tab configuration. The weight of each tab was estimated to be 1,050 lb. Bending moments on each tab would be large, 3.4×10^6 in.-lb at full tab insertion. The bending moment contributes directly to the friction torque of the shaft. Total torque requirements were 107,800 in.-lb per tab, comprising 47,500 in.-lb aerodynamic torque, 15,300 in.-lb friction torque, and 45,000 in.-lb inertial torque.

A warm gas turbine system using eight linear hydraulic servoactuators was selected for the actuation system. Warm gas blowdown and rotary actuators were considered but both were rejected as being too heavy. The TVC duty cycle accounts for a total impulse loss of approximately 0.27 percent, resulting in an additional propellant penalty of 9,330 lb.

A weight breakdown, by component, of the selected jet tab TVC system is presented in Table III.

Supersonic Splitline (Figure 2)--A supersonic splitline nozzle employing a flexible bearing at the splitline location, appears particularly attractive in the light of current technology, and this concept was selected for the final tradeoff. The main advantages are force amplification and relatively low nozzle ejection loads.

*Lockheed Propulsion Co.: "156 In. Diameter Motor Jet Tab TVC Program," Final Report AFRPL-TR-64-167, January 1965 (Confidential).

Design and fabrication of large flexible bearings has reached a point at which the bearing required for the supersonic splitline could be designed and constructed with a high degree of confidence.

Cold flow test data indicate that an optimum joint location would lie at an expansion ratio of approximately 2:1. Pivot point location depends to a degree on joint design but ideally should be as near the splitline as possible.

Use of a gimbal ring as a means of vectoring the movable portion of the exit cone was investigated. This approach is unattractive from a weight standpoint, since the gimbal ring must be located at a relatively large diameter for the 260 in. motor application. Total nozzle weights would be 82,600 lb for the gimbal ring, 17,600 lb for the ring itself, and 58,900 lb for the flexible bearing.

Torque requirements are very high, 24×10^6 in.-lb for the gimbal ring and 27×10^6 in.-lb for the flexible bearing. A tradeoff of various actuation systems indicated the most promising means of producing the high torques is a warm gas turbine system driving a variable displacement pump. Peak power demands would be met using an accumulator incorporated in the delivery side of the pump.

A weight breakdown by component for the supersonic splitline nozzle is presented in Table IV.

The support structure necessary to contain the bearing would add a total of 10,990 lb to the baseline nozzle design.

The supersonic splitline and jet tab TVC systems were selected for the final tradeoff. Current technology can be applied with a high degree of confidence to the design of a supersonic splitline employing a flexible bearing. Development of the actuation system to meet the high torque requirements is considered a relatively low risk, low cost effort.

The major problem appears to lie in overcoming nozzle erosion immediately downstream of the splitline, where a particularly severe thermal environment exists and in the splitline survivability which has not been impressive to date. This latter consideration resulted in the elimination of the supersonic splitline in favor of jet tabs from a reliability standpoint.

The jet tab TVC system was recommended as the more promising system capable of meeting the requirements of the 260 in. diameter launch vehicle, based upon the criteria of cost and reliability. Jet tabs were chosen primarily because of their development status with respect to large motors. Despite the fairly large amount of tungsten involved for each tab, an extensive development effort is not anticipated to design a reliable tab meeting the requirements of the 260 in. motor TVC system.

Following the recommendation of the most promising TVC system in each category (mechanical interference, liquid injection, and movable nozzle) it became clear that MITVC was inferior to the other two systems from many aspects.

Development risk was significantly greater with the MITVC system, primarily because of the severe materials problem. More than 9,000 lb of additional propellant is necessary to overcome the performance loss of the jet tab system. Performance loss of the movable nozzle is negligible and LITVC actually provides thrust augmentation. The total weight of the jet tab TVC system, including the nozzle, would be 86,475 lb compared to 57,300 lb for the movable nozzle and 82,900 lb for LITVC. Accordingly, completion of a detailed design of the jet tab TVC system was considered unnecessary and work ceased on this concept.

LITVC SYSTEM STUDIES

The LITVC scheme implements vehicle guidance commands by injection of a secondary liquid into the nozzle supersonic exhaust stream. A side force to provide pitch and yaw control results primarily from the induced pressure unbalance acting over a portion of the internal nozzle surface area, and secondarily from the reaction force of the injected liquid.

The general objectives of the LITVC system design studies for application on a 260 in. solid rocket motor (SRM) of a MLV-SAT-1B-5A type vehicle were as follows.

1. Investigate liquid injection parameters and LITVC system components.

2. Compare potential LITVC system design approaches.

3. Select candidate LITVC system designs which would best meet the study goals.

4. In collaboration with NASA-Le RC, select the better design approach for a detailed design and analysis of a LITVC system for use on the 260 in. SRM.

The LITVC system design requirements utilized in this study are presented in Table V; the LITVC duty cycle is illustrated in Figure 3.

A summary of the design analyses, candidate systems evaluation tradeoff, selection of the better LITVC system design approach, and a description of the 260 in. SRM LITVC system design follow.

DESIGN ANALYSIS

Preliminary LITVC system weight tradeoff comparisons were made to screen and select the more promising liquid injectants for further detailed LITVC system design analyses. The tradeoff studies relied heavily on data from previous programs for prediction of injectant effectiveness. Performance curves for the candidate injectants were established by empirical correlations of existing LITVC data. Weight trade studies were performed per the design requirements of Table III to establish total LITVC system comparisons incorporating each candidate injectant.

To determine the amount of duty cycle injectant required, the side specific impulse (ISP_s) for each injectant corresponding to 0.42 deg thrust vector was utilized. The ISP_s for each injectant corresponding to 1.2 deg thrust vector was used to calculate the maximum injectant flow rate required per injector port.

Thiokol's "IBM Computer Program for Design of a LITVC System" was utilized to establish preliminary design data on the size and weight of LITVC systems using each of the candidate injectants for the established system requirements. The computer program calculated the amount of duty cycle injectant, total amount of on-board injectant required, and the maximum required injectant flow rate. The computer program also was used to calculate size and weight of actuation and pressurization subsystems, tankage, injector valves, power supply components, liquid and gas lines, plus the weights of hydraulic fluid, disconnects, filters, electrical cabling, brackets, and fittings.

For this weight study, a representative injectant tankage and pressurization system consisting of two toroidal tanks was selected. One tank contained the injectant; the other contained nitrogen gas initially charged at 3,000 psi and then regulated to maintain a constant injectant tank pressure of 600 psi. An electrohydraulic actuation system and 20 equally spaced single pintle-type injectors were also selected. For these weight tradeoff studies, it was felt that representative LITVC system weight comparisons could be made.

Utilizing the N_2O_4 LITVC system weight (35,180 lbm) as a baseline factor, the computer program results of the initial LITVC system launch weights (nozzle excluded) are compared in Table VI.

Each LITVC system was similar in all respects except for the type of liquid injectant used. As a result of the initial LITVC system weight tradeoffs, nitrogen tetroxide (N_2O_4) and aqueous strontium perchlorate [$Sr\ (ClO_4)_2 + H_2O$] injectants were selected for more detailed LITVC system design work.

An investigation was conducted to provide configuration data and tradeoff curves for use in the detailed design studies of N_2O_4 and $Sr\ (ClO_4)_2 + H_2O$ LITVC systems for the 260 in. motor. Several typical LITVC system techniques are illustrated in Figure 4. The most significant injection parameters, components, and subsystems investigated included the injector location, injection angle, number of injector orifices, injection pressure, type of injection valve, pressurization concepts, and tank configurations. The results of this investigation are shown in Table VII.

CANDIDATE SYSTEM EVALUATION TRADEOFF

The earlier tradeoff studies affected the choice of the candidate LITVC systems to be evaluated for 260 in. SRM application to a major degree. A cursory component breakdown of each of the candidate LITVC system configurations is presented in Table VIII. A comparison of the injectant and pressurant requirements, the estimated total launch and burnout weights (nozzle weight excluded), and estimated cost of each candidate LITVC system design are shown in Table IX.

Referring to the total (wet) launch weights in Table IX, the two aqueous Sr $(ClO_4)_2$ LITVC systems (No. 5A and 5B) exceeded the launch weights of their N_2O_4 counterpart designs (No. 4A and 4B) by 17 percent. The heavier aqueous Sr $(ClO_4)_2$ system launch weights resulted primarily from the increase in injectant weight (due to lower ISP_s capabilities than N_2O_4) and the requirement for a minimum of five injectors per quadrant (instead of four per quadrant with N_2O_4). Within the six N_2O_4 LITVC systems evaluated (systems No. 1 thru 4B), system No. 1, which utilized four cylindrical N_2O_4-GN_2 tanks, was estimated to be the most costly system, and also the heaviest at launch and burnout.

Table IX illustrates that of the eight candidate designs, LITVC system No. 3B was the second lightest N_2O_4 design at launch, had the lightest burnout weight, and was the least costly.

Thiokol Chemical Corporation-Wasatch Division and NASA-Lewis Research Center jointly determined that LITVC system No. 3B offered the greatest design potential and should be pursued further in the detailed design task. This decision was based primarily on system weight, cost, and simplicity.

DETAILED SYSTEM DESIGN

The LITVC system design developed for application on the 260 in. SRM is presented in Figure 5. The addition of an aft skirt access door was the only modification required to the basic vehicle design.

A single toroidal tank (nominal volume, 702 cu ft), which is shown in detail in Figure 6, contains both the GN_2 pressurant and the N_2O_4 injectant fluid. The GN_2-N_2O_4 tank is supported by a tubular system attached to the internal structural members of the vehicle aft flare. The tank support structure design has features to allow for misalignment, asymmetric loads from various sources, and possibilities for future support design structure modification and/or growth. Provisions are made for loading and unloading N_2O_4, filling and venting GN_2, emergency venting of N_2O_4 vapors, nonvortex distribution of the N_2O_4 from the tank to each of 16 injectors, and measurement of the unexpected N_2O_4.

At launch, the N_2O_4 liquid injectant occupies 473,470 cu in.; the GN_2 pressurant 739,800 cu in. The total N_2O_4 launch weight is 24,634 lbm including the required duty cycle fluid, plus allowances for expulsion efficiency, system errors, motor and LITVC performance tolerances, ullage, manifolds and injector leakage. The total minimum required GN_2 pressurant by weight is 1,690 lbm.

Initial GN_2 blowdown system pressure is 800 psia minimum with the vehicle fully loaded and ready for launching. The system blows down to 400 psia during the course of the flight. Experimental data indicate satisfactory N_2O_4 LITVC system performance at injectant pressures down to 400 psia within the thrust vector range and duty cycle requirements specified for this study.

The N_2O_4 is distributed from the toroidal tank to each of the 16 injectors through flexible expansion lines. Injection valve housings attached to the nozzle provide a support for each of the injector valve assemblies.

The injectors are electromechanically actuated pintle type valves which vary the flow rate by changing the effective flow area. These servocontrolled assemblies can modulate N_2O_4 flow from zero to 169 lbm/sec at 800 psi, and from zero to 120 lbm/sec at 400 psi. The injector valves use developed servocomponents to provide valve opening and closing time capable of achieving the required slew rate.

The TVC system has the capability of providing correction for all transient and steady-state perturbations in the pitch and yaw axes. The pitch and yaw controller subsystem provides: (1) servodrive amplifiers and coupling between the autopilot command signals and the injection valves; (2) a linearization of the side force-voltage relation; (3) controller integration of the liquid dump commands with the TVC requirements; and (4) compensation for quadrant interaction.

A component weight breakdown (nozzle excluded) of the 260 in. LITVC system is presented in Table X. The total initial weight is 38,801 lbm; the total burnout weight is 14,804 lbm.

A total LITVC system unit cost (nozzle cost excluded) of $165,940 was the result of individual component cost breakdowns based on a total of 10 development motors and 20 flight motors.

MOVABLE NOZZLE TVC SYSTEM

Previously designed TVC systems for large solid propellant rocket motor systems using movable nozzles were investigated. One common design feature of the systems studied was use of hydraulics to transmit power to the load. The most conventional nozzle actuation system was a warm gas generator powering a turbine-gear box hydraulic pump arrangement.

The systems investigated in this program included: (1) warm gas solid propellant-turbine-gear box-pump, (2) warm gas liquid propellant-turbine-gear box-pump, (3) warm gas blowdown, and (4) pneumatic actuators. Nine different configurations were considered under the first three items listed above.

A system utilizing pneumatic actuators (warm and cold gas) was studied briefly. However, flight type pneumatic servoactuators were expected to present problems in design and performance which could not be justified under stated objectives of low cost, complexity, and development risk, for a vehicle of this size.

A cold gas blowdown system would require either a 6,000 psi pressure source regulated down to 4,000 psi or an unregulated 4,000 psi system which would decay with discharge of hydraulic fluid. The unregulated system would require use of large actuators to meet the high torque requirement as system pressure decayed. Both of these configurations would require an excessively large system and were not considered further at this time.

For the preliminary design the following requirements were used.

Maximum nozzle vector angle	161 deg
Maximum nozzle slew rate	3.0 deg/sec
Maximum nozzle torque	17.726×10^6 in.-lb at 1.61 deg
Action time	143 sec*

For the detail design, these requirements were modified with the concurrence of NASA.

The primary task in the preliminary screening was selection of a power source to drive the actuators. The torque values used were obtained from the Thiokol Advanced TVC Digital Computer Program using the Aerojet bearing design. The same torque values, servoactuators, and plumbing were used for all configurations during the preliminary design. For comparison purposes the tradeoffs were made on the power sources only, since the actuators, plumbing, etc, would be common to all systems.

Actuator Design--Servoactuator design depends upon three parameters: force, stroke, and linear rate. The force is derived from the nozzle torque and actuator geometry. The stroke can be readily determined from the required nozzle vector angle and the lever arm. The rate, however, presents a more difficult and somewhat perplexing problem in spite of the fact that the nozzle angular rate is given as 3.0 deg/sec. The velocity of the actuator is directly related to the control

*The warm gas generators were sized to burn for 170 sec to permit the hydraulics to be pressurized for 27 sec prior to launch, allowing time for prelaunch checkout of the complete flight actuation system.

flow through the servovalve, which in turn is a function of the pressure drop across the valve. The drop across the valve is defined as:

$$P_v = P_s - P_1 - P_r$$

where P_v, P_s, P_1, and P_r are the pressure drop across the valve, supply pressure, drop across the load, and return pressure, respectively. Since P_1 varies with actuator position (because torque varies with position), P_v (control flow), and consequently the velocity attainable will be a function of actuator position, it becomes important to define the point or range of points where the rate of 3.0 deg/sec is required.

The preliminary duty cycle has a maximum excursion of 0.92 deg and a maximum rate of 1.84 deg/sec, both at approximately 20 sec. For the preliminary design a rate of 2.12 deg/sec at a vector angle of 0.825 deg in each plane was used. This will require 43.8 gpm flow to each actuator for a total flow rate of 87.6 gpm. A standard 50 gpm servovalve was selected for preliminary design. The actuator had a stroke of 2.7 in. and an effective area of 47 sq in. Operating pressure was set at 4,000 psi. For the detail design, slew rate was redefined and a different method was used to size the servoactuator.

SOLID PROPELLANT-TURBINE-PUMP SYSTEM

Six turbine pump configurations using a solid propellant gas generator, turbine gear box, and hydraulic pump were investigated; however, only two systems had pumps large enough to handle the total fluid requirements. The remaining systems employed either a nitrogen precharged accumulator or a warm gas generator pressurization source to supplement hydraulic flow during peak demands. A schematic of a typical turbine-pump system with a precharged accumulator is shown in Figure 7.

Of the systems without an accumulator, one employed a single large pump--the Vickers designed B70 type--and the other used dual pumps driven from the same gearbox. The single pump, delivering 100 gpm at 4,000 psi, was designed for more rigorous use than is required for this program and consequently is heavier, more complex and relatively more expensive than necessary. In keeping with the objective of simplicity and low cost, this system was not continued beyond the investigation stage. Dual pumps, each capable of flowing 48 gpm, were used to eliminate the need for an accumulator. The two positive displacement axial piston type pumps are lighter, less complex, and lower in cost than the single large pump. They are driven from the same turbine-gearbox arrangement. A single gas generator must supply sufficient power to drive both pumps.

Three designs with precharged accumulators utilized the same pump but rotated at different speeds. The degree of complexity remained the same; however, the power requirement is less at lower speeds, resulting in smaller gas generators. This advantage is partially offset by the necessity for a larger accumulator.

Major component weights for four of the solid propellant gas generator-turbine pump systems are compared in Table XI.

To overcome the pressure decay problem encountered with a precharged accumulator, a system was considered in which the accumulator was charged by the same warm gas generator that drives the turbine. System pressure can be maintained at 4,000 psi during accumulator fluid discharge. A switching arrangement (priority valve) is provided so that between cycles, the pump will refill the accumulator with hydraulic fluid, making it ready for the next demand. Of the several disadvantages of this system, the primary one is complexity due to the valving arrangement. To obtain system pressure, either a 4,000 psi generator or a differential area accumulator would be necessary. The added complexity of the valving plus the heavier gas generator and accumulator would not offset the advantage of maintaining system pressure for this application. For this reason this system was not considered further in the preliminary design phase.

Monopropellant-Turbine-Pump System--The turbine-gear box-pump arrangement used with this design is identical to that described in the solid propellant scheme. However the solid propellant gas generator is replaced by a liquid fuel (hydrazine) system (Figure 8). Hydrazine is pumped from the fuel tank by a centrifugal pump through a flow control valve to the decomposition chamber. Gas generated in the chamber is used to drive the turbine.

The output pressure of the turbine driven centrifugal pump is essentially independent of flow, but is a direct function of pump speed and consequently of turbine speed. The fuel valve senses pump output pressure and varies flow to the turbine as a function of pressure. Thus turbine speed is controlled and can be maintained almost constant over the entire hydraulic flow range.

The system is started by firing a cartridge propellant which drives the turbine to its operating speed and also raises the temperature of the catalyst bed to assist decomposition of the fuel during startup. System design follows that proposed by Sundstrand Aviation, Rockford, Ill., for a large SRM.

The system was sized for two hydraulic pump speeds with a precharged accumulator to supply additional flow for peak demands. As shown in Table XI, the weight difference between the liquid and solid propellant systems is insignificant. In reviewing current usage, complexity, and development status of the hydrazine system, it was decided that any weight difference would not overcome the lack of experience and development required for use of a liquid system of this size.

Warm Gas Blowdown--Warm gas blowdown is the least complex of the systems studied in the preliminary design. It utilizes a solid propellant warm gas generator to pressurize a reservoir which contains sufficient hydraulic fluid to meet the duty cycle requirements.

The most critical item in the design of a blowdown system is sizing of the reservoir. The required duty cycle presented nozzle angular velocity ($\dot{\delta}$) versus time. The integral of the absolute value of $\dot{\delta}$ gave 18.9 deg for the combined pitch and yaw signals. Using a safety factor of 1.2 a total of 2,455 cu in. of hydraulic fluid would be required, including servovalve leakage of 1.0 gpm, and a limit cycle oscillation of ± 0.1 deg amplitude and 0.2 Hz.

To permit use of the same actuators as on the previous systems, a 4,000 psi gas generator is used, which lowers the mass fraction, thus increasing the weight of the system. A schematic is shown in Figure 9, and major component weights are listed in Table XI. The system is heavier (by a factor of 2) than the turbine-pump system; however, it is less complex.

The major disadvantage with any blowdown system is that is is limited by the duty cycle. If nozzle vectoring exceeds the designed duty cycle during flight, the hydraulic fluid may be depleted prior to burnout. This possibility can be precluded by using a larger safety factor in sizing the reservoir, although this imposes a greater weight penalty on the system. Because of the duty cycle limitation and the weight factor, this system was made secondary to the turbine-pump system. Other approaches were considered, such as a closed cycle blowdown system in which the hydraulic fluid is recycled instead of dumped overboard. This closed cycle system is relatively insensitive to duty cycle perturbations. Two reservoirs (each smaller than the one used above) would be required as well as additional valving to recirculate hydraulic fluid. The added complexity and cost would not, however, be justified for this application.

PRELIMINARY DESIGN REVIEW

The system selected incorporates a solid propellant gas generator with a 60 gpm pump and a 600 cu in. accumulator.

As may be noted (Table XI) the weight differential among the turbine pump systems is insignificant. Final selection was based upon the simplicity and lower cost of the single pumping system. Dual pumps offer more flexibility in duty cycle perturbations, but the increase in cost and complexity offsets this advantage. The warm gas blowdown system was the least complex configuration investigated; however, because of the high loads throughout the duty cycle, this system was not considered competitive.

A redefinition of requirements was established in that gravity torque could be neglected if the motor were fired vertically during static test. Internal aerodynamic torque was computed as a function of expected chamber pressure and grain configuration as the motor burned (Figure 10). The duty cycle was multiplied by $\sqrt{2}$ and an additional 1.61 deg event was added at 60 sec (Figure 11). The system was designed to allow a maximum vector angle of 1.61 deg for the first 60 sec and 1.18 deg thereafter. The slew rate of 3.0 deg/sec was redefined as the average rate when stepped from hardover in one direction to 90 percent of hardover in the other direction.

If an average 3.0 deg/sec slew rate is to be attained the rate must exceed this value some of the time. To obtain maximum velocity without a detail study, it was assumed that the actuation system dynamics could be approximated by a second order system with a damping ratio of 0.8. Using a plot of amplitude vs nondimensional time ($\omega_0 t$) for second order system, the time to reach 90 percent of the final value is 2.95. For this application "t" is approximately 1.0 sec; hence, ω_0 is 2.95 rad/sec. The maximum velocity is then 4.02 deg/sec, requiring a peak flow of 74 gpm. Using this above new requirement, NASA requested that two systems be investigated (1) turbine pump and (2) passive cold gas blowdown.

The passive cold gas blowdown system (Figure 12) was designed for an initial pressure of 4,000 psi decaying to 3,000 psi at 60 sec. The effective area of the actuator was 30 sq in. Component weights for the two systems using the new criteria are compared in Table XII. Although the blowdown system is 30 percent heavier, it offers greater simplicity with a much smaller number of components. A detail design was then prepared using the simplified cold gas blowdown system.

DETAIL DESIGN

An analog computer simulation of the passive cold gas blowdown actuation system was conducted. A block diagram of the system is shown in Figure 13. Step inputs were applied to the system to insure stability and capability to meet the slew rate requirements. Response to a step input from hardover to hardover (Figure 14) shows that velocity averages 3.0 deg/sec while the maximum velocity is approximately 4.5 deg/sec in the positive direction. The response shown in Figure 14 was obtained at a time equivalent to 60 sec on the duty cycle. This time was considered to be the most critical since system pressure was expected to decay to approximately 3,000 psi and the maximum thrust vector angle was required. System pressure was held constant at 3,000 psi and the torque value was that corresponding to the torque curve at 60 sec. The effective area of the actuator, servovalve size, reservoir volume, nozzle compliance, and servoamplifier gain were varied to obtain an optimum response.

The duty cycle (Figure 11) was recorded on magnetic tape and used as the input to the problem. System parameters for these runs are shown in Table XIII.

A layout of the actuation system (Figure 15) illustrates the major components, two servoactuators and a large tank. The reservoir is made of 4340 steel and contains no barrier between the hydraulic fluid and the pressurant (the acceleration forces parallel to the longitudinal axis of the vehicle contain the fluid in the outlet of the tank). Total volume of the tank is 7,590 cu in., of which 2,530 cu in. is hydraulic fluid. The tank is mounted on the aft skirt with the centerline parallel to the longitudinal axis of the missile. The flight version expels hydraulic fluid from the aft end of the tank, while for static test the tank is reversed. A normally open solenoid valve is mounted on the expulsion end of the tank. Quick disconnects located near the nozzle bolt circle provide means of filling the hydraulic system and actuating the nozzle with ground power prior to launch. A check valve in the dump line is designed to open at a 50 psi differential pressure, creating a slight back pressure on the servovalves.

During firing the solenoid valve remains in the de-energized mode and the number of moving parts in the system is minimized. System complexity has been reduced to the point where only three major components are required, two of which are the servoactuators which would be necessary for any actuation system design. System simplicity is believed to compensate for any limitations which may be imposed by the duty cycle. Current space technology is adequate to define duty cycle requirements with sufficient accuracy that, with a moderate factor of safety, a blowdown system can be designed to perform satisfactorily.

System Cost and Weight--In computing the total weight of the passive cold gas blowdown system, weights of developed components were obtained from vendors. Other components such as the actuators, tank, bracket, etc, were calculated from drawings. The total weight of the system is 881.0 lb.

Cost figures obtained from vendor's quotes were based on a total of 20 flight motors, 10 development motors, and additional components required for bench tests and spares. The total cost for each actuation system is $38,925.

CONCLUSIONS

At the time this paper was written, no decision had been made as to the selection of the optimum TVC system. The weight and cost per system is summarized in the table below.

	Nozzle (lbm)	TVC (lbm)	Launch Weight (lbm)	Burnout Weight (lbm)	Average Cost/Unit (dollars)*
LITVC	47,900	38,801	86,700	57,700	1.806×10^6
Movable Nozzle	56,298	881	57,179	52,129	1.662×10^6

Current technology, development status, and risk indicates that a high degree of confidence in either TVC concept could be achieved.

*The cost/unit value generated was based on 20 flightweight motors and 10 development motors. The cost/unit value includes materials, tooling, bench tests, fabrication, and engineering. Detailed cost breakdowns will be published in the program final report.

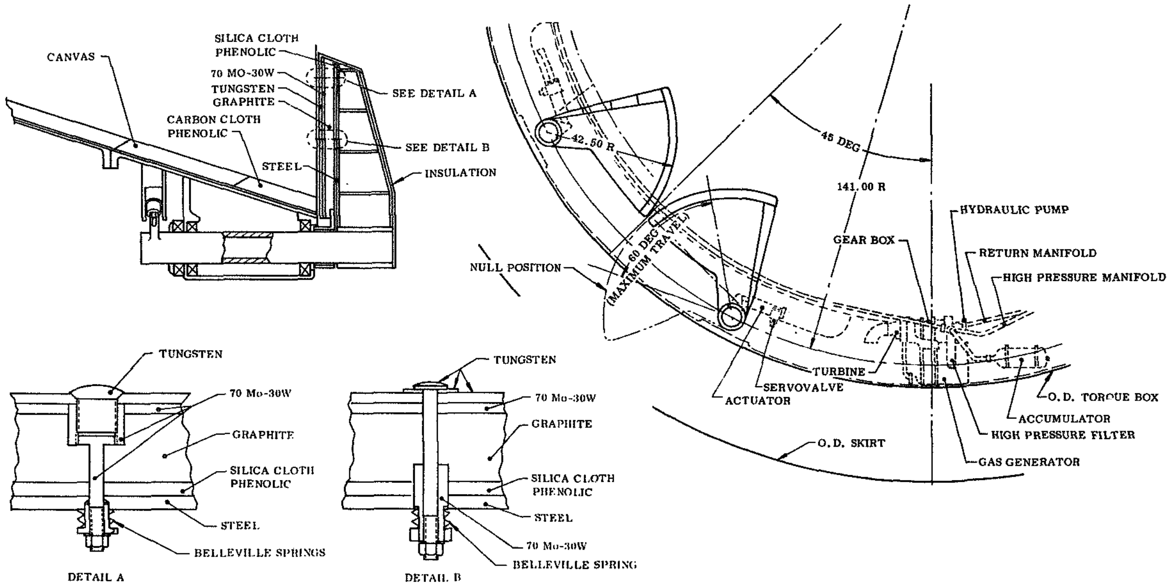

FIGURE 1. JET TABS

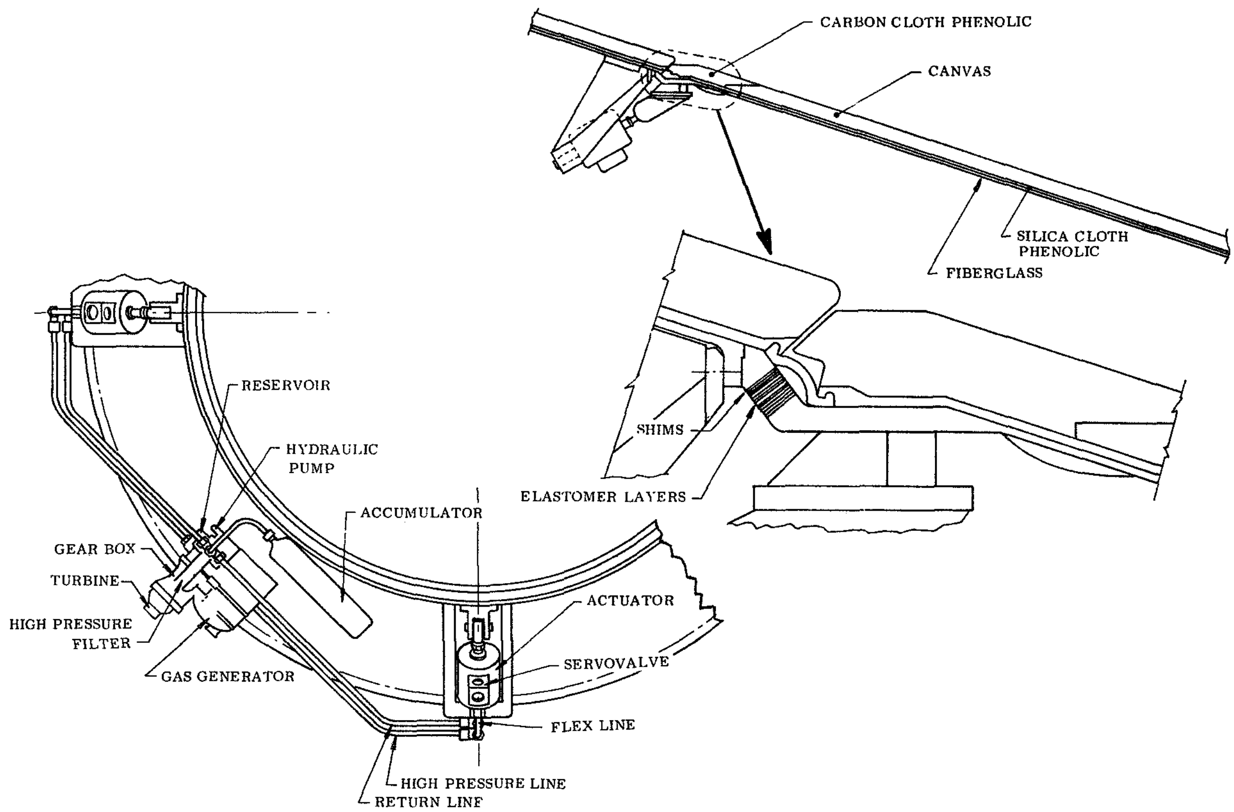

FIGURE 2. SUPERSONIC SPLITLINE

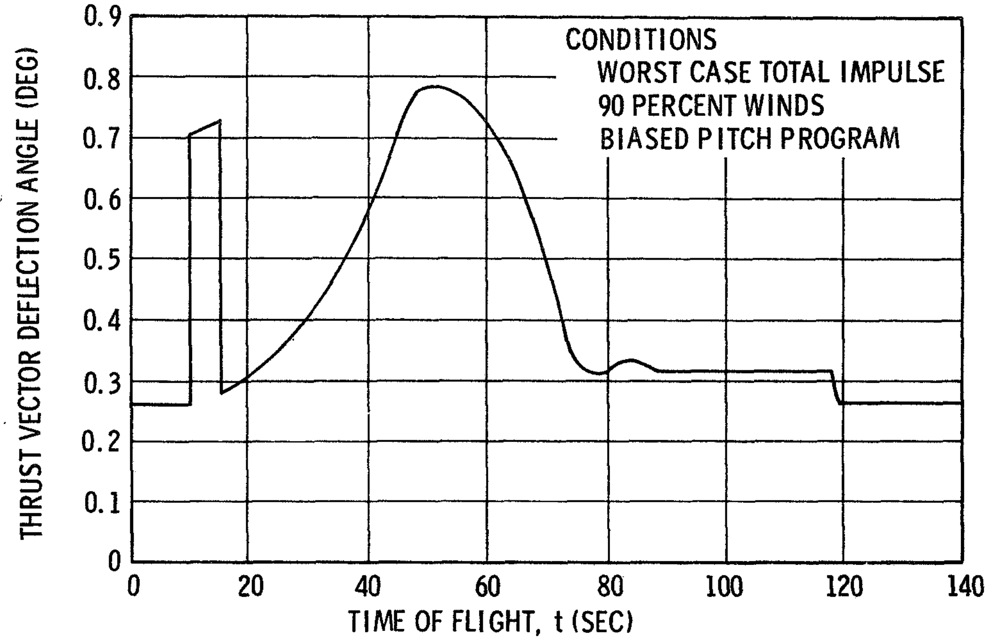

FIGURE 3. LITVC DUTY CYCLE

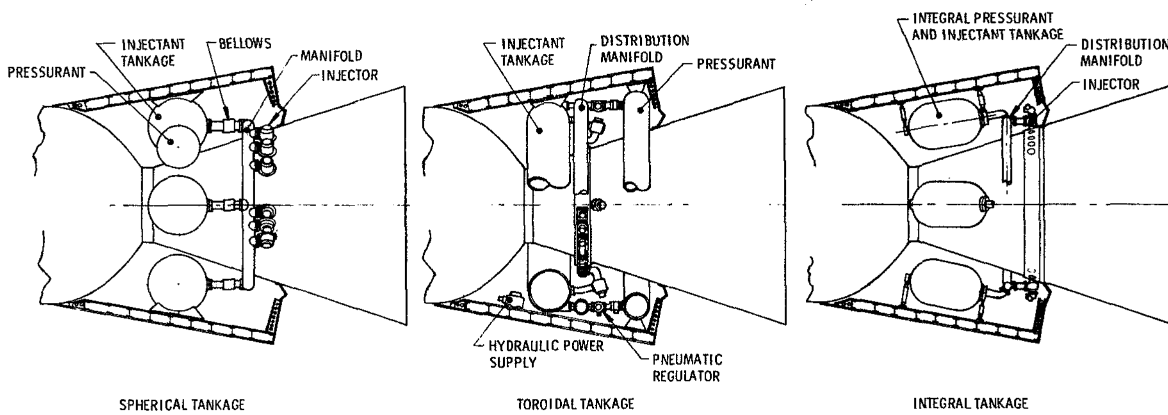

FIGURE 4. TYPICAL LITVC SYSTEM TECHNIQUES

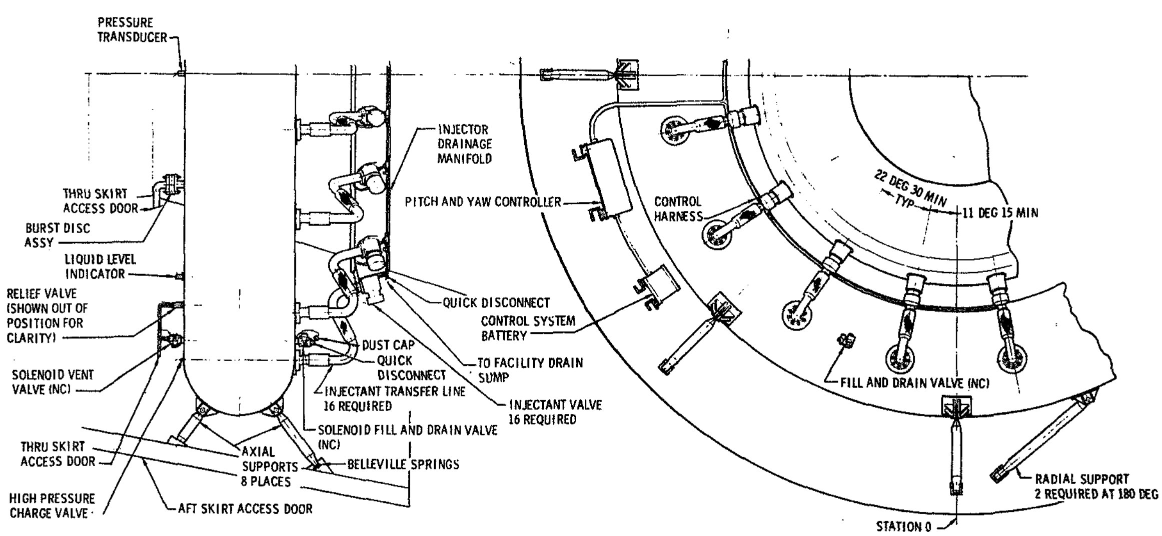

FIGURE 5. LITVC SYSTEM FOR 260 INCH SOLID ROCKET MOTOR

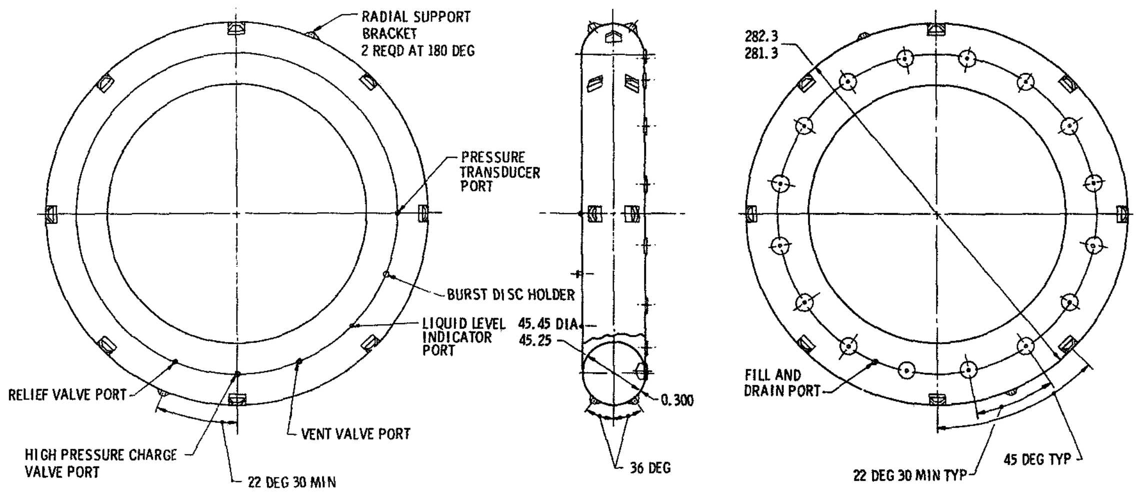

FIGURE 6. NASA 260 INCH SOLID ROCKET MOTOR LITVC SYSTEM GN_2 - N_2O_4 TANK

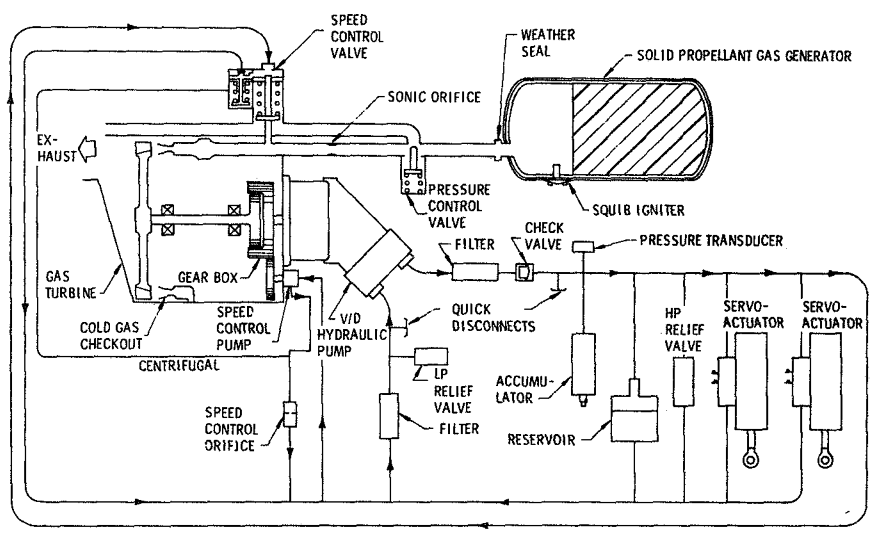

FIGURE 7. SCHEMATIC OF VARIABLE DISPLACEMENT PUMP APPROACH

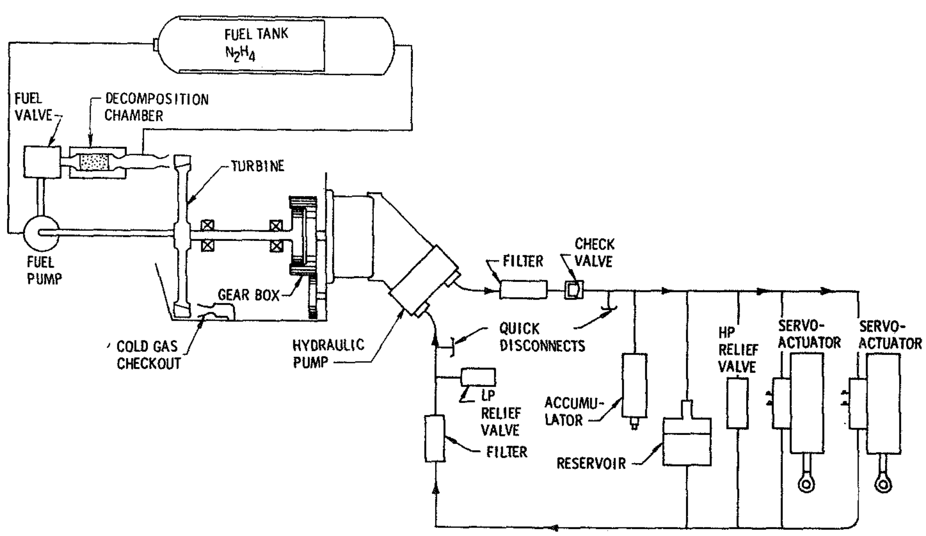

FIGURE 8. LIQUID FUEL SYSTEM (HYDRAZINE)

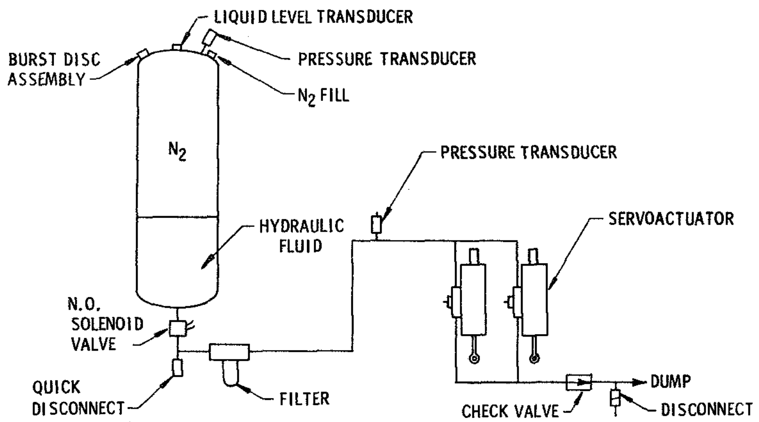

FIGURE 12. PASSIVE COLD GAS BLOWDOWN SYSTEM SCHEMATIC

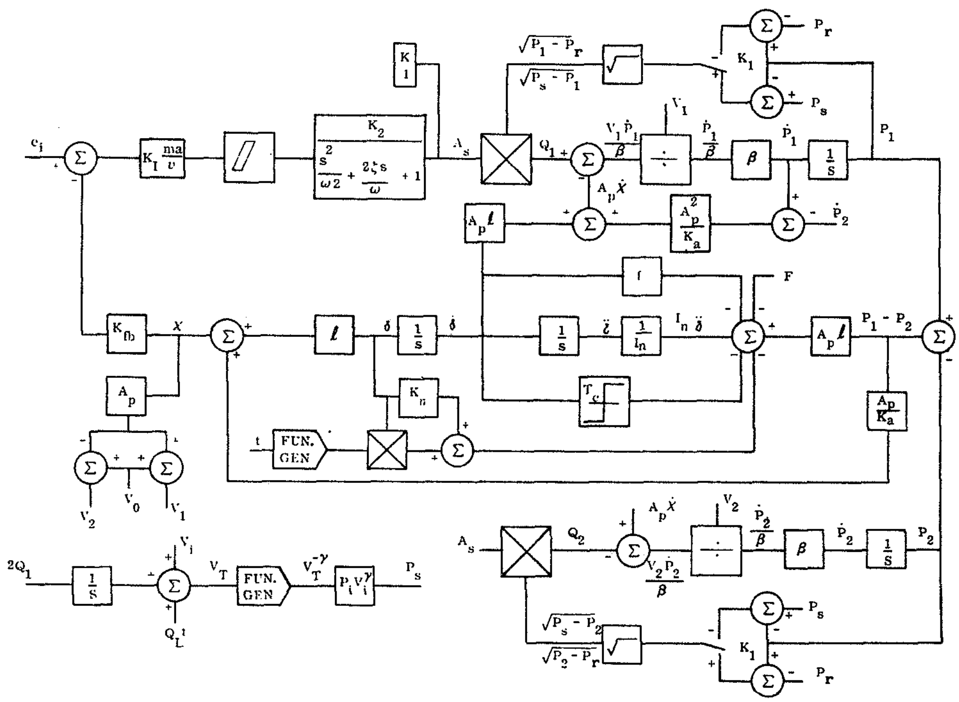

FIGURE 13. COMPUTER BLOCK DIAGRAM

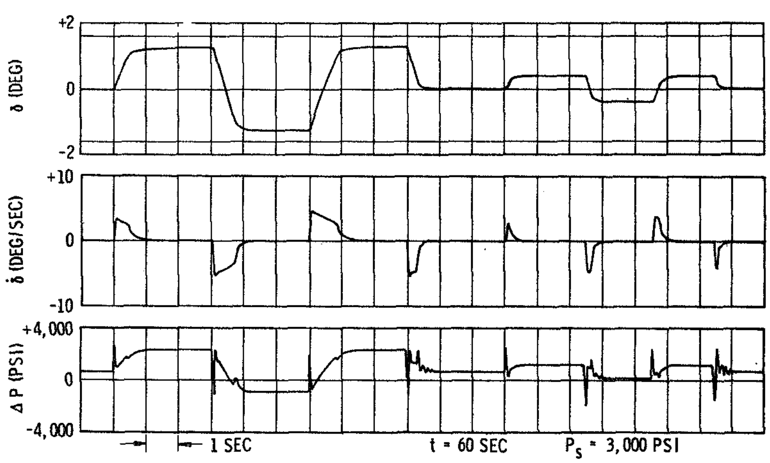

FIGURE 14. RESPONSE TO STEP INPUTS

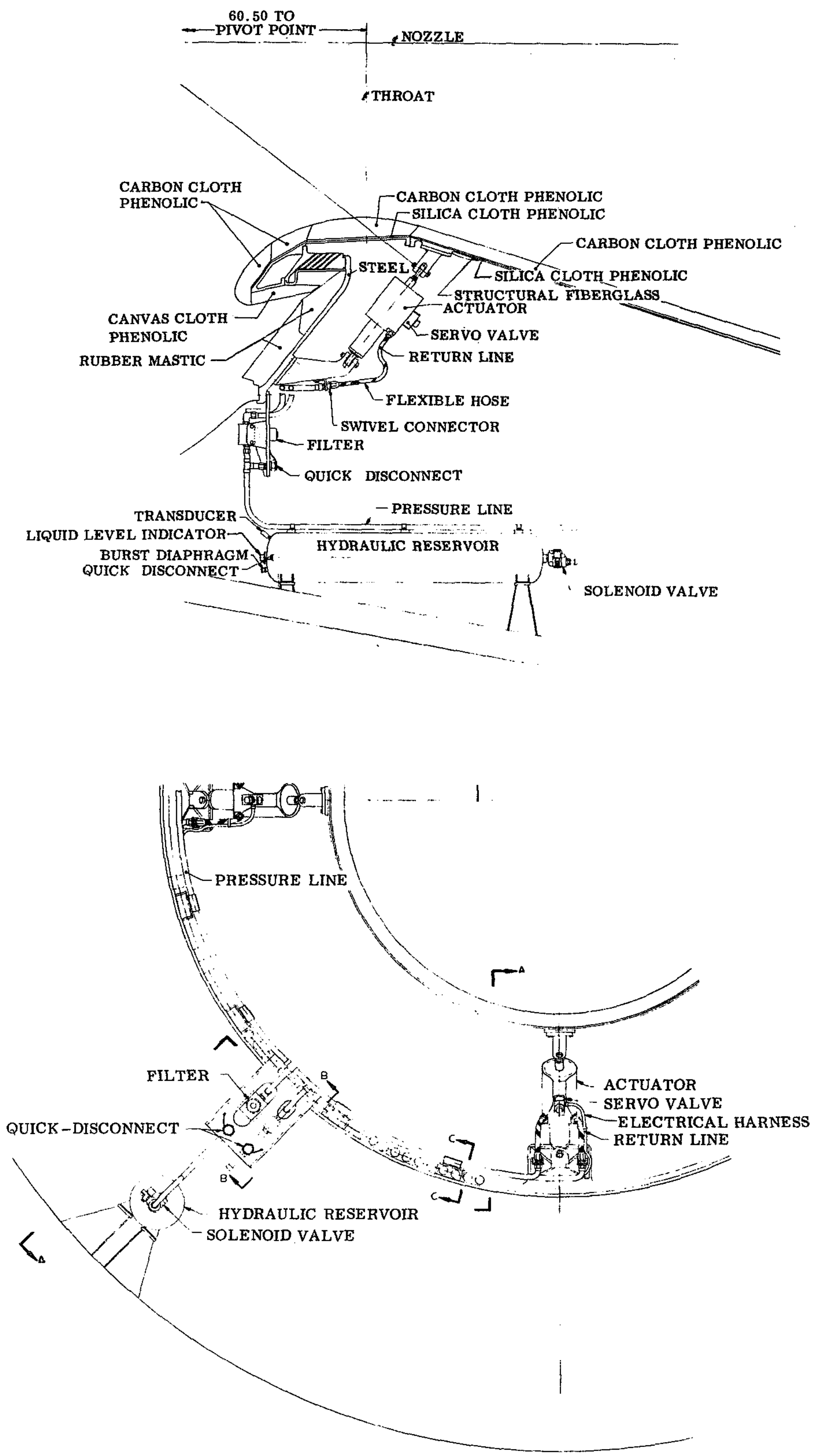

FIGURE 15. ACTUATION SYSTEM FOR MOVABLE NOZZLE

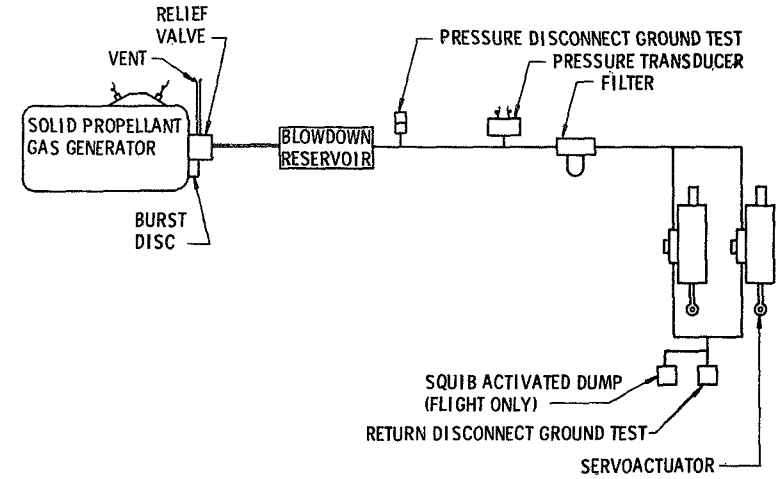

FIGURE 9. SCHEMATIC OF WARM GAS BLOWDOWN SYSTEM

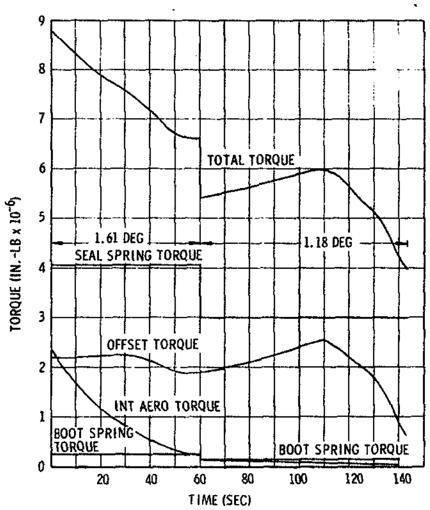

FIGURE 10. NOZZLE TORQUE VS TIME

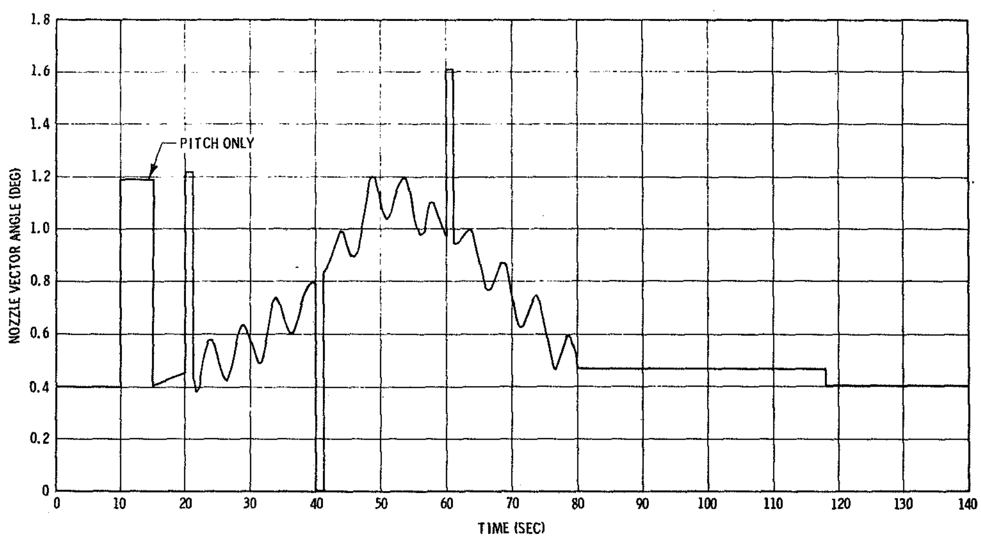

FIGURE 11. MOVABLE NOZZLE DUTY CYCLE

TABLE I

APPROXIMATE PROBE DIMENSIONS

$X/L = 0.4$ $A_p = 405$ sq in.			$X/L = 0.5$ $A_p = 490$ sq in.			$X/L = 0.6$ $A_p = 580$ sq in.		
No. Probes	D_p	H	No. Probes	D_p	H	No. Probes	D_p	H
1	17.8	22.7	1	5	98.0	1	5	116.0
1	22.3	18.2	1	10	49.0	1	10	58.0
1	26.7	15.2	1	20	24.5	1	20	29.0
1	31.2	13.0	1	30	16.3	1	30	19.3
1	35.6	11.4	1	40	12.3	1	40	14.5
2	5	40.5	2	5	49.0	2	5	58.0
2	10	20.3	2	10	24.5	2	10	29.0
2	15	13.5	2	15	16.4	2	15	19.3
2	20	10.2	2	20	12.3	2	20	14.5
2	30	6.8	2	30	8.2	2	30	9.7
3	5	27.0	3	5	32.6	3	5	38.6
3	10	13.5	3	10	16.3	3	10	19.3
3	15	9.0	3	15	10.9	3	15	12.9
3	20	6.8	3	20	8.2	3	20	9.7
3	30	4.5	3	30	5.5	3	30	6.45

A_p = Approximate Probe Projected Area (sq in.)

D_p = Approximate Probe Diameter (or width) (in.)

H = Approximate Probe Inserted Height (in.)

X = Axial Distance from Throat to Probe Insertion Point (in.)

L = Axial Distance from Throat to Exit Plane (in.)

TABLE II

MECHANICAL PROBE POTENTIAL COOLANT COMPARISON

	Uncooled	Water	N_2O_4	N_2H_4	Freon 114-B2	Freon 113	Aqueous Strontium Perchlorate
A_p (sq in.)	490	477	352	441	330	359	423
Weight Coolant (lb)	--	8,000	16,000	10,000	29,300	27,600	10,100
K_2	--	0.2	0.9	0.5	0.5	0.5	0.7
C_p (Btu/lb° F)	--	1.0	0.368	0.736	0.166	0.218	0.700

NOTES:
1. Weight of coolant calculated to meet NASA duty cycle
2. A_p = probe projected area (sq in.)

ASSUMPTIONS:
1. Probe location at $X/L = 0.5$
2. For probe, linearity factor $K_1 = 1$ $Fs/Fa = K_1 (A_p/A_I)$

   Fs = side force from probe A_p = probe projected area
   Fa = nominal axial thrust A_I = nozzle cross sectional area at probe location
3. K_2 = measure of injectant efficiency obtained from LITVC data.
4. Weight of coolant adjusted only for heat capacity of injectant. Heat of vaporization neglected.
5. Weight of coolant based upon satisfying total side impulse requirements of 260 in. diameter launch vehicle.

TABLE III

JET TAB TVC WEIGHT BREAKDOWN

Component		Weight (lb)
Modified Nozzle (excluding torque box)		65,360
Torque Box		12,000
Shafts (8)		2,128
Tabs (8)		6,264
Servoactuators (8)		280
Gas Generator		123
Pump		20
Turbine Gearbox		40
Hydraulic Fluid		35
Accumulator		25
Miscellaneous (lines, filter, disconnect, etc)		200
	Total	86,475

TABLE IV

SUPERSONIC SPLITLINE NOZZLE WEIGHT BREAKDOWN

Component		Weight (lb)
Nozzle Assembly		58,890
Servoactuators (2)		400
Gas Generator		280
Pump		28
Turbine Gearbox		42
Hydraulic Fluid		76
Accumulator		33
Miscellaneous (lines, filters, reservoir, etc)		226
	Total	59,975

TABLE V

LITVC SYSTEM DESIGN REQUIREMENTS

Total Injection Impulse (deg-sec)	60
Pitch and yaw; assumes a 0.25 deg thrust misalignment throughout the entire flight	
Total Injection Impulse (lbf-sec)	6.287×10^6
Maximum Required Equivalent Thrust Vector Angle Each - Pitch and Yaw (deg)	1.2
Maximum Required Equivalent Slew Rate (deg/sec)	3
Average Thrust Deflection Angle of Duty Cycle, 60 Deg-Sec ÷ 143 Sec (deg)	0.42
Average Side Force (lbf)	43,965
Ratio of Control Thrust Impulse to Total Vehicle Vacuum Thrust Impulse (percent)	0.727

The amount of injectant required for duty cycle operation must provide a minimum side impulse of 60 deg-sec (pitch and yaw total) between motor ignition signal and the end of motor action time (143 sec).

The total amount of injectant carried on board was to include the duty cycle injectant and allow for expulsion efficiency errors, motor and LITVC performance tolerances, injector valve leakage, and extra injectant to fill ducting and manifolds.

TABLE VI

LAUNCH WEIGHTS OF LITVC SYSTEMS

LITVC System	Weight Factor
Nitrogen Tetroxide, N_2O_4	1.00
Aqueous Strontium Perchlorate, Sr $(ClO_4)_2 \cdot H_2O$	1.35
Aqueous Lead Perchlorate, Pb $(ClO_4)_2 \cdot H_2O$	1.55
Freon 114B2	2.01
Freon 113	2.03
Hydrazine, N_2H_4	2.13

TABLE VII

SELECTED LITVC SYSTEM DESIGN CHARACTERISTICS

Type of Injectants	1. N_2O_4 2. Aqueous Sr $(ClO_4)_2$ solution
Injector Position	35 to 40 percent of nozzle length
Injection Angle	+15 deg upstream of a perpendicular to the nozzle centerline
Type of Injection Valve	Single pintle-type injectors
No. of Valves per Nozzle Quadrant	4 and 5
Type of Injector Actuation System	1. Electromechanical actuators/battery power source 2. Hydraulic actuators/electric motor pump power source 3. Hydraulic actuators/passive blowdown power source
Type of Injectant Pressurization	Nitrogen gas (GN_2) blowdown
Type of Tank Configuration	1. Single common toroidal tank* 2. Four common cylindrical tanks*
Injection Pressure	800 psia initially; blows down to 400 psia
LITVC Control System Scheme	Pitch-yaw controller

*No bladder required between pressurant and injectant because the vehicle acceleration forces parallel to the longitudinal axis of the vehicle maintain the liquid injectant in the aft end of the tank (location of injectant outlet ports).

TABLE VIII

COMPONENT BREAKDOWN OF CANDIDATE LITVC SYSTEM DESIGNS

LITVC System Component	LITVC System Designation							
	1	2	3A	3B	4A	4B	5A	5B
N_2O_4 Injectant	X	X	X	X	X	X	--	--
Sr $(ClO_4)_2$ Injectant	--	--	--	--	--	--	X	X
GN_2 Pressurant	X	X	X	X	X	X	X	X
Cylindrical Tanks (4)	X	--	--	--	--	--	--	--
Toroidal Tank (1)	--	X	X	X	X	X	X	X
Injectant Distribution Manifold	X	X	--	--	--	--	--	--
Injectant Ducts - Tank to Manifold	X	X	--	--	--	--	--	--
Injectant Ducts - Manifold to Injectors	X	X	--	--	--	--	--	--
Injectant Ducts - Tank to Injectors	--	--	X	X	X	X	X	X
Electrohydraulic Injector Valves	X	X	--	--	X	X	X	X
Electromechanical Injector Valves	--	--	X	X	--	--	--	--
5 Injectors/Quadrant	X	X	X	--	--	--	X	X
4 Injectors/Quadrant	--	--	--	X	X	X	--	--
Battery Assembly	X	X	X	X	X	--	X	--
Power Transfer Switch	X	X	X	X	X	--	X	--
Hydraulic Power Supply System (Electric Motor Pumps)	X	X	X	X	X	--	X	--
Passive Blowdown Hydraulic Power System	--	--	--	--	--	X	--	X

TABLE IX

WEIGHT AND COST COMPARISON OF CANDIDATE LITVC SYSTEM DESIGNS

	LITVC No. 1	LITVC No. 2	LITVC No. 3A	LITVC No. 3B	LITVC No. 4A	LITVC No. 4B	LITVC No. 5A	LITVC No. 5B
Injectant	N_2O_4	N_2O_4	N_2O_4	N_2O_4	N_2O_4	N_2O_4	$Sr(ClO_4)_2 + H_2O$	
Injectant Volume (total initial) (cu in.)	466,000	466,000	466,000	473,470	473,470	473,470	437,600	437,600
Injectant Weight (total initial) (lbm)	24,309	24,309	24,309	24,634	24,634	24,634	30,387	30,387
Pressurant	GN_2	GN_2	GN_2	GN_2	GN_2	GN_2	GN_2	GN_2
Pressurant Volume (total initial) (cu in.)	728,000	728,000	728,000	739,800	739,800	739,800	683,700	683,700
Pressurant Weight (total initial) (lbm)	1,650	1,650	1,650	1,690	1,690	1,690	1,560	1,560
LITVC System								
Estimated Total Launch Weight* (lbm)	37,105	33,758	32,938	33,104	33,353	33,340	39,006	39,174
Estimated Total Burnout Weight* (lbm)	13,424	10,077	9,257	9,107	9,356	9,252	9,405	9,454
Estimated LITVC System Unit Cost**	$452,950	$375,250	$268,950	$245,180	$311,380	$252,980	$327,820	$268,720

*Nozzle weight excluded.
**Nozzle cost excluded; unit cost based on thirty (30) 260 in. motors and LITVC systems.

TABLE X

NASA 260 IN. SRM LITVC SYSTEM COMPONENT WEIGHTS

Component	Weight (lbm)
Injectant-Pressurant Tank Assembly	10,470
Injectant: Nitrogen Tetroxide (N_2O_4)	24,634
Pressurant: Nitrogen Gas (GN_2)	1,690
Burst Disc Assembly, Operational Pressure Transducer, Liquid Level Indicator, Relief Valve, Solenoid Vent Valve, GN_2 Pressure Charge Valve, Solenoid Fill and Drain Valve, Quick Disconnect and Dust Cap	29
Injector Valves (16)	320
Injector Housings (16)	192
Tank to Injector N_2O_4 Transfer Liners (16)	240
Axial (16) and Radial Supports (2), and Aft Skirt Support Brackets (18)	950
Pitch and Yaw Controller	30
Control System Battery	40
Power Transfer Switch	8
Electrical Harness Assembly	160
Injector Valve Drain Manifold Assembly	18
Relief and Solenoid Vent Valve Tubing Assembly	16
Burst Disc Assembly Tubing Assembly	4
TOTAL INITIAL WEIGHT (LBM)	38,801
TOTAL BURNOUT WEIGHT (LBM)	14,804

TABLE XI

COMPARISON OF MAJOR COMPONENTS FOR MOVABLE NOZZLE TVC SYSTEM PRELIMINARY DESIGN

	Solid Propellant				Liquid Propellant		Warm Gas Blowdown
DESIGN PARAMETERS							
Pump Speed (rpm)	7,100	5,650	3,750	5,650	7,100	5,600	--
Pump Flow (gpm)	60	(2)96	(2)64	48	60	48	--
Power (hp)							
Pump Output	140	224	149	112	140	112	--
Pump Input	175	264	166	132	175	132	--
Gas	350	528	332	264	350	264	--
Gas Generator Flow Rate (lb/sec)	0.35	0.528	0.33	0.264	0.283	0.214	0.62
Fuel Weight (lb)	59.5	89.6	56.4	44.8	48	36.4	105
Accumulator Volume (cu in.)	600	--	440	1,050	600	1,050	2,450
COMPONENT WEIGHT (lb)							
Gas Generator	82.5	123	77	63.2	60	50	234
Accumulator	37	--	37	65	37	65	150
Reservoir	15.5	11	13.6	18	15.5	18	--
Turbine-Gearbox	42.5	47	43	40	42.5	40	--
Hydraulic Fluid	13.2	7	10.4	18	13.2	18	73.5
Pump	19.8	(2)39.6	(2)39.6	19.8	19.8	19.8	--
Accessories	--	--	--	--	10	8	6
Total Weight	210.5	227.6	220.6	224.0	198.0	218.8	463.5

TABLE XII

COMPARISON OF TURBINE-PUMP SYSTEM WITH THE PASSIVE COLD GAS BLOWDOWN SYSTEM PRELIMINARY DESIGN

	Turbine-Pump (lb)	Blowdown (lb)
Gas Generator	78.0	--
Pump	19.8	--
Reservoir	15.0	--
Turbine-Gearbox	40.0	--
Tubing-Fittings	50.0	50.0
Filters-Disconnects	20.0	20.0
Accumulator	20.0	250.0
Hydraulic Fluid	30.0	97.0
Actuator (2)	370.0	370.0
Servovalve (2)	38.0	38.0
N_2	1.7	58.0
Solenoid	--	10.0
	682.5	893.0
+10 Percent Contingencies	68.0	89.0
TOTAL	751.0	982.0
Major Component Cost per Motor	$66,044	$43,644
Nonrecurring Cost	$350,000	$150,000

TABLE XIII

SYSTEM DESIGN PARAMETERS

A_p	Actuator Area (sq in.)	30
l	Lever Arm (in.)	94.5
K_{fb}	Feedback Gain (v/in.)	5.0
I_n	Nozzle Inertia (in.-lb-sec^2)	5.3×10^6
K_a	Compliance (lb/in.)	2.0×10^6
K_I	Servoamplifier Gain (ma/v)	2.1
K_v	Servovalve Gain (No Load) (cu in./sec/ma)	23.1
X	Actuator Stroke (in.)	± 2.7
P_i	Initial Pressure (psi)	4,000