VOYAGER CAPSULE ENGINE SYSTEM DESIGN CONSIDERATIONS

SINGLE VERSUS MULTIPLE ENGINE CONFIGURATION

Power Systems Division

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1. INTRODUCTION

One of the key propulsion design decisions to be made for Voyager Capsule Bus is the selection of the number and type of rocket engines for the deorbit and landing phases of the flight. Several of the more attractive configurations are shown in Figure 1. A solid rocket motor or a liquid rocket can be used for the deorbit engines, while either one, three or six variable thrust liquid rocket engines are needed for the landing maneuver. The six-engine configuration is of interest because it provides good reliability through engine shutdown capability, while the three-engine configuration provides ease of vehicle control through differential throttling of the engines.

The purpose of this paper is to discuss some of the advantages of the six vs. three vs. single engine configurations for Capsule Bus propulsion. A four-engine configuration, similar to the three-engine configuration, is discussed with the three-engine arrangement. This document does not discuss all facets of the problem with equal thoroughness because the comments and inputs are limited to the comments of a propulsion-oriented organization.

2. ENGINE SYSTEM DESIGN

The configuration discussion is based on one feed system design concept. Figure 2 illustrates the schematic required for each engine/feed system combination. This configuration is based upon using positive seal squib valves to enclose the propellants within tankage until they are required for landing or landing and deorbit, if liquid engines are

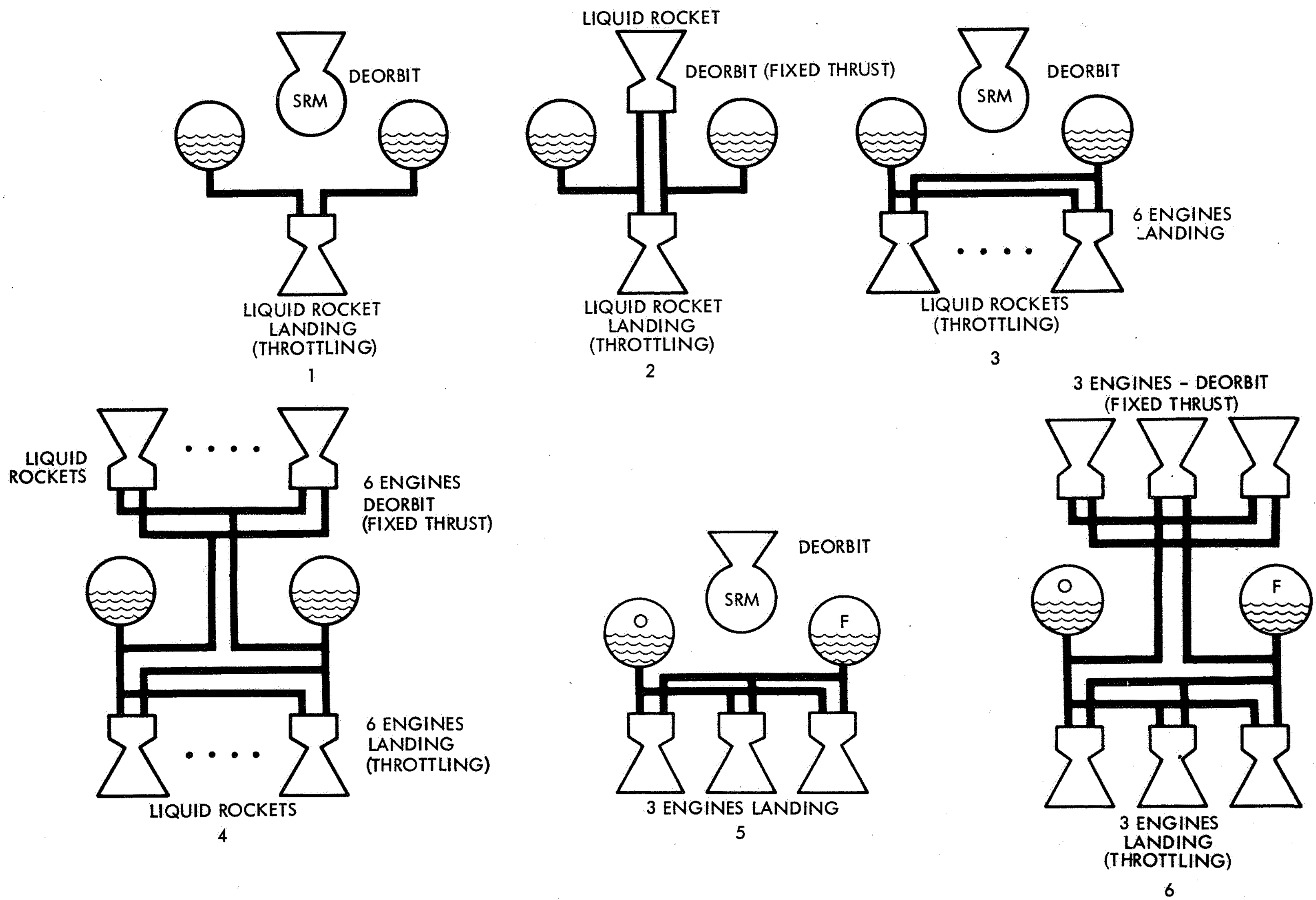

Figure 1. Alternate Propulsion Configurations

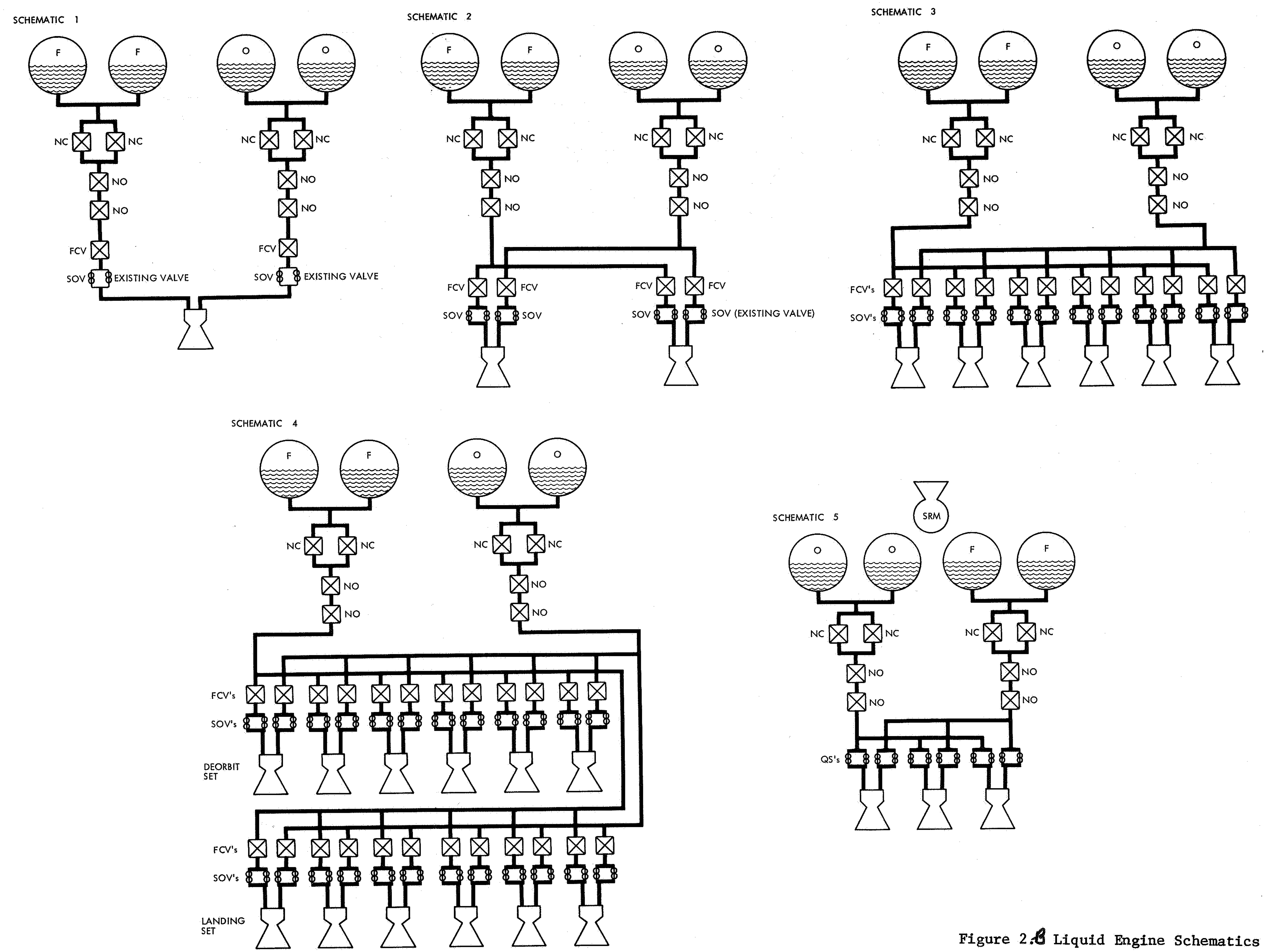

Figure 2.A Liquid Engine Schematics

Fig. 2-A

3-A

3 A

used for the latter. Only one set of squib valves is proposed for the system whether or not both the deorbit and landing engines are to be fed by the same system. This design requires that the engine valves hold the propellants during deorbit and entry if the same liquid feed system is used for both maneuvers. Thus, a quad-redundant engine valve (parallel and series redundancy) should be used for the feed systems using both deorbit and landing because these valves represent the only seal to propellant leakage. If the feed system is to be used for landing only, then only parallel redundancy is needed in the engine valves to guarantee engine starting, while upstream squib valves can guarantee engine sealing after engine shutdown.

One feature to note about all of the liquid engine schematics is the method to be used for venting all lines. Venting can be accomplished by opening the engine valves while the vehicle is in space; however, in the case of Schematics (3) and (4), if one of the engine valves opens and fails to close, it can cause a major leakage path upon introducing propellant to the engine or a deflagration when an engine start is attempted with a leaking valve. Thus, since a single failure can abort the mission, quad-redundant valves were introduced in this schematic as well.

For discussion purposes, the schematics illustrated and discussed above will be used in comparing the advantages of the six-, three- and one-engine systems. It is assumed that the liquid rockets used for deorbit are fixed-thrust versions of the throttleable engines used for the lander vehicle.

The following aspects of the three systems are discussed in this report:

1) Mission reliability
2) Failure sensing equipment
3) Checkout and countdown reliability
4) Interacting failure modes
5) System performance
6) System weight
7) Vehicle configuration
8) Flight control
9) Leakage, plumbing and contamination
10) Feed system dynamics
11) Electrical, EMI and power requirements
12) Telemetry requirements
13) Base heating
14) Spacecraft heat load
15) Engine developmental effort
16) Propulsion system developmental effort
17) Growth potential

2.1 MISSION RELIABILITY

Exact determinations of reliabilities of single engines and multiple engine clusters of various possible configurations were made but are not included in this report. Rather, failure probability equations were developed for determination of the ratio of probability of failure of a multi engine cluster and probability of failure of a single engine. Figure 3 is a plot of this relation for various values of failure probability of single engine (P_e), fraction of engine failures which would be catastrophic (λ), and the ratio of failure probability of engine failure sensing and shutdown system to P_e (β). A supporting plot (presented in Figure 4) of P_e, β and λ is made for unity failure probability ratio.

The charts were plotted for destructive (catastrophic) failure percentage ranging from 0 to 60 percent in Figure 3, and 0 to 20 percent in Figure 4, to indicate trends; although this demonstrated percentage is normally much less. Depending upon the exact engine studied, the catastrophic failure percent may average near 10 percent of all failures. Test and use data generated within TRW on LMDE indicates a catastrophic failure rate of much less than 5 percent of all major failures.

It will be noted from the charts that the failure probability ratio of multi engine versus single engine configurations increases with an increasing fraction of destructive failures. This ratio also increases with the relative complexity and, thus, unreliability of the failure sensing and shutdown systems. Basic analyses of control systems indicate that a three-engine cluster control system should range near a β ratio of 0.6 and a six-engine cluster near 0.8.

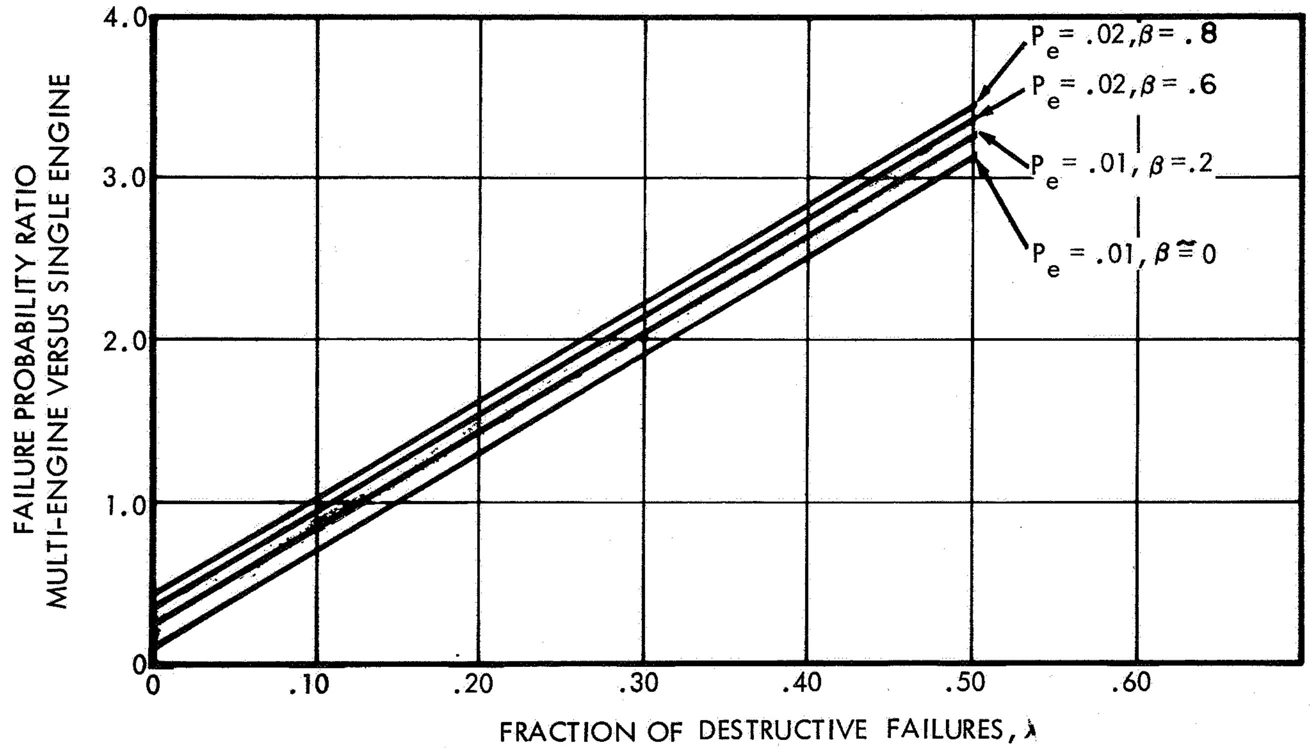

Figure 3. Failure Probability Ratio

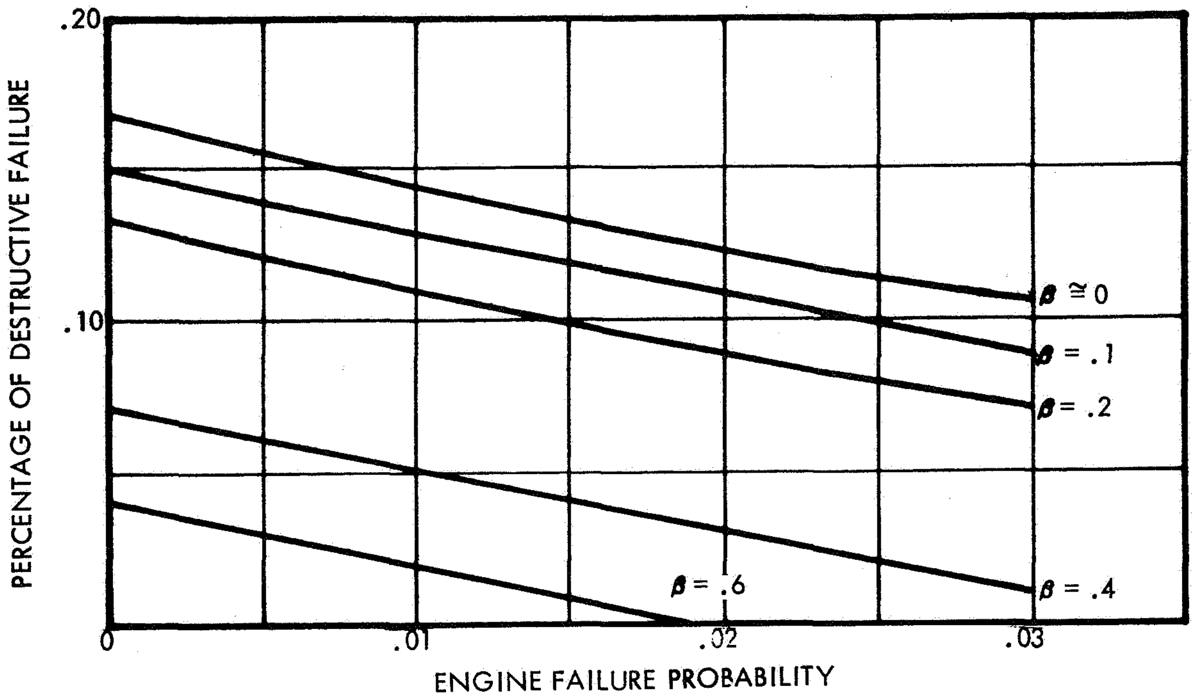

Figure 4. Engine Failure Probability Versus Destructive Failure

Generally speaking, the more simple the mechanism or system, the more reliable it can be expected to be. In the case of determination of flight, countdown, or functional reliability of a single engine versus an engine cluster, it is obvious that the addition of ducting, joints, interface structure, sensing equipment and shutdown controls to multiple engine configurations increases unreliability of the system. For the single or three-engine system, the necessity for a gimbal may add incremental unreliability. More important, for an engine suspected to be unreliable, a cluster will provide back-up redundancy in case of non-critical failure. However, the probability of catastrophic failure is increased along with added engines, and this added probability may well decrease the effectiveness of the redundancy afforded by additional engines.

The results of this study show an operational reliability advantage of multiple engine clusters over single engine in the areas of zero to unity failure probability ratio with extremely high reliability sensing and shutdown control systems and less than 10 percent destructive failures. Utilizing sensing and shutdown control reliability values more nearly representing a state of the art expected in the near future, the advantage of a single engine configuration becomes quickly apparent within the same parameters. There should be no doubt that the reliability of the engine interconnect sensing and shutdown control system is extremely important. It should also be apparent that, for a successful mission, catastrophic failure probability must be extremely low (hopefully non-existant) for any engine configuration.

2.2 FAILURE SENSING EQUIPMENT

It is very likely that some type of failure sensing system will be required for the six-landing-engine case. A failure sensing system detracts significantly from the overall system reliability as discussed in Section 2.1, but it is required if the vehicle flight would be uncontrollable as a result of one engine failing to throttle. A review of the failure modes and needs for failure sensing pointed out this need for the six-engine configurations used to land (see Table I), but not deorbit. The need for this system can be minimized by making most failure modes drive the engine closed, but there are several modes that require a failure sensing and engine shutdown system. A chamber pressure sensing system using the engine P_c transducers is a most plausible method. A majority voting system would be used wherein all the P_c signals could be compared and if one were significantly out of line, that engine which it is part of would be shutdown. Eliminating failure sensing is a desirable feature of the single or three-engine arrangement because: (1) mission and countdown reliability are increased if it is deleted, (2) a significant development effort will be required to qualify the failure sensing equipment, and (3) a tie-in between the telemetry systems and engine controls should be avoided.

2.3 CHECKOUT AND COUNTDOWN RELIABILITY

In prelaunch ground checkout of multiengine clusters it is necessary to test all functional components in the system for any mode of failure. To do otherwise would jeopardize the flight since it would not be known

TABLE I

MALFUNCTION DETECTION REQUIREMENTS (FAILURE MODE ANALYSIS)

A. DEORBIT (FIXED THRUST)

Failure Mode	Consequence	Failure Sensing Required
Single Valve Leakage	None	No - Redundant Valves
Jammed Actuator	None	No - Fixed Thrust
Jammed Injector	None	No - Fixed Thrust
Engine Hot Spot	None - heat shield	No - Too Much Instrumentation
Chamber Failure	Destructive	No - Too Rapid
Inoperative Actuator	None	No - Fixed Thrust
Open Command	None	No - Fixed Thrust
Close Command	None	No - Five Out of Six and Use Up Propellants

B. LANDING (VARIABLE THRUST)

Failure Mode	Consequence	Failure Sensing Required
Single Valve Leakage	None	No - Redundant Valves
Jammed Actuator	Abort	Yes - Shutdown Engine
Jammed Injector	Abort	Yes - Shutdown Engine
Engine Hot Spot	None - heat shield	No - Too Much Instrumentation
Chamber Failure	Destructive	No - Too Rapid
Inoperative Actuator	Abort	No - If Fails Shut
Open Command	Abort	Yes - Shutdown Engine
Close Command	None	No - 5/6 Operates

whether built-in redundancy features were present or not. Redundancy benefits contributing to flight performance, therefore, do not exist in the prelaunch checkout phase. This means, then, for checkout conditions a six-engine cluster has all its components effectively in series (zero redundancy) and the cluster's failure probability can be expected to be several times greater than that for a single engine. Such is the case as is borne out by the comparison of single and multiple engine systems given below:

Definitions:

P_e = Probability of checkout failure (Q_s) of a single engine in any mode

P_s = Probability of checkout failure in a six-engine cluster's failure sensing and shutdown system

$\beta = P_s/P_e$

Then:

$(1 - P_e)$ = Probability of no failure in a single engine system

$(1 - P_e)^6$ = Probability of no failures in a six-engine cluster

$(1 - P_s)$ = Probability of no failure in sensing and shutdown

$(1 - P_e)^6(1 - P_s)$ = Probability of no failures in six-engine cluster combined with its sensing-shutdown system

$Q_c = 1-(1 - P_e)^6(1 - P_s)$ = Total probability of failure in prelaunch checkout of six-engine assembly

The ratio of six-engine to single-engine failure probabilities then is:

$$\frac{Q_c}{Q_s} = \frac{1 - (1 - P_e)^6(1 - P_s)}{P_e}$$

but since $P_s = \beta P_e$ $\qquad \dfrac{Q_c}{Q_s} = \dfrac{1 - (1 - P_e)^6(1 - \beta P_e)}{P_e}$

Results obtained by insertion of various values for P_e and β are tabulated below:

	$\beta = 0.1$	$\beta = 0.2$	
$P_e = 0.01$	6.1/1	6.2/1	Q_c/Q_s
$P_e = 0.02$	6.1/1	6.2/1	

As expected, a six-engine assembly probably would exhibit about six times more failures during checkout than a single-engine assembly and this factor changes very little for variations in P_e and β . This same 6-to-1 ratio also holds for launch conditions involving two single-engine versus two six-engine cluster assemblies, all of which must operate.

The excess of the ratio Q_c/Q_s over a 6.0 figure can be attributed directly to the failure sensing and shutdown system necessary for the multiple engine configuration.

It should be noted the fraction of total engine failures expected to be destructive (λ) does not appear in the equation for Q_c/Q_s. This is because in ground prelaunch checkout, all failures, destructive or not, must be detected and repaired.

The 6-to-1 advantage of the single-engine system during checkout actually is conservative. The greatly increased complexity and cost of ground support equipment required to test adequately the redundant components within the six-engine assembly have not been considered.

During countdown, a total of 192* shutoff valves per landing system must be functioning properly. A failure of any of these valves may require removal and replacement of the component thus compromising the sterilization integrity of the engine system.

2.4 INTERACTING FAILURE MODES

Any multiengine arrangement can be subject to interacting failure modes that lower the system reliability below a simple product of individual failure probabilities. Even though each engine, by itself,

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* Considering two Voyager spacecraft, six engines per spacecraft landing system and six engines per spacecraft deorbit system; a total of twenty-four series-quad (8 valves per unit) shutoff valves or 192 valves are required. This compares to thirty-two valves for a single engine system.

has a certain reliability and a certain percentage destructive failure, a multiple engine system will have additional failure modes that can lead to overall mission failure. Several examples of this type of failure mode are given below:

1) Leakage in one engine could be ignited by another engine starting, causing an engine compartment fire.
2) A power drain by one engine can drain a common power supply for the propulsion system.
3) Pressure transients and oscillations can be induced within the feed system due to the interlocked manifold system. These interactions may lead to destructive loads on the system.
4) Electrical signals or interference from one engine can cause malfunction within other engines.

A number of multiple engines (clusters) have been developed and each has required a significant development effort to work out system interaction effects to minimize interacting failure modes.

2.5 SYSTEM PERFORMANCE

Unlike the single- or three-engine configurations, small system performance penalty accrues in a six-engine configuration due to increased gravitational trajectory losses if only five out of six engines are operating. (It is assumed that the six-engine configuration has the same total maximum thrust as the three- or single-engine arrangement.) If five out of six redundancy is claimed, additional propulsion system weight must be added to account for the increased propellant, tankage and pressurization system weight required if one engine fails. An estimate of these weight increases were made for two cases - engine failure during deorbit and engine failure during landing. These estimates were based upon 350-pound propellant weights for both deorbit and landing. The propulsion system weight increases are approximately 5 pounds for deorbit and 21 pounds for landing, for a total of 26 pounds. This increase effectively reduces the total capsule payload weight by 26 pounds.

2.6 WEIGHT SYSTEM

In general, for a given vehicle thrust level, the weight of the system will increase as the number of engines are increased. Table II shows the weights for three alternative engine arrays for the Voyager Capsule Bus. A single engine is of 6500-pound thrust, the three-engine array uses 2500-pound thrust engines, and the six-engine array uses 1900-pound thrust engines. All engines are of the throttling Lunar Module type. For the purposes of this comparison, the weight of the gimbal actuators for the single engine should be included since these are not chargeable to the other two configurations. This weight is approximate. All engines are assumed to operate at the same maximum chamber pressure and expansion ratio. The weight is also shown for the ablative wall thickness required to limit the maximum wall temperature to 350^{o}F. Additional weight would also be required for the multiengine configurations if suitable shielding to limit the plume base flow would have to be provided.

TABLE II

LANDER CONFIGURATIONS DATA FOR PRELIMINARY DESIGN

P_c = 325 psia ϵ = 50 N_2O_4/50-50

100 second burn time

Ablative chamber

Thrust Rating, lb	No. of Engines	Total Thrust, lb	Weight of Each Basic Engine, lb	Total Weight of Engines for Basic Configuration, lb	Total Weight of Engines for Max OD Temperature of 350°F, lb
1200	6	7200	39.5	237	326
2500	3	7500	62.0	188	240
7500	1	7500	160.0	160	204

Note: An optimized engine design which takes into account nozzle expansion ratio and duty cycle should permit an improvement of the values presented.

Another consideration in the selection of the optimum engine configuration for the Voyager Capsule Bus is the effect of thrust malignment on system weight. The degree of this penalty for the multi-engine system depends on whether the error in thrust alignment for these engines will be assumed to be worst case errors (in which case all thrust vectors are misaligned in a single vector direction), or if the thrust vectors are assumed to be RMS for a nominal total thrust vector misalignment. In either case, the attitude control propellant requirements must be increased to compensate for this effect. This penalty is chargeable to both the six- and the three-engine configurations with fixed engines, and is also reflected in the accuracy with which the single-engine gimbal system can align the thrust vector. However, it appears that, in all cases, the multiengine configuration will suffer a greater ACS propellant penalty than will the single engine. Similar in effect to the thrust vector malignment is the engine-to-engine variation in thrust magnitude and performance, including the dead band and other control inaccuracies on thrust magnitude control. These variables will have to be accounted for as an increase in ACS propellant requirements, and, again, are chargeable to the multiengine configurations. In this case, these errors are not chargeable to the single engine system.

2.7 VEHICLE CONFIGURATION

One of the reasons often mentioned for favoring the multiple engine configuration is that the vehicle configuration is more desirable with it. Three points are made against the single engine arrangement: first, the engine is located in the center where it is desirable to locate the

the experimental packages that that may change in weight and need the center location to properly perform experiments; second, the nozzle exit cone protrudes, serving as an obstacle to landing on a nonuniform Mars surface; and, third, the natural gimbal plane for the engine is at the throat, but the throat is located near the vehicle c.g., which minimizes the gimbal effectiveness. Suggestions are presented to resolve these points in favor of the single engine.

2.7.1 Payload Location

Investigation of the types of the experimental equipment to be carried on the Capsule Bus indicate that off-center locations can be used in their current planning.

2.7.2 Nozzle Extension Crushing

A columbium nozzle extension is designed to the critical condition of resistance to buckling at maximum thrust. Within this constraint, the extension is made as thin as possible to minimize crushing load requirements. In the case of the LMDE, the extension is step-tapered to a nominal wall thickness of 0.010 inch in the aft 30 percent of the cone.

The actual crushing forces and energy were determined experimentally at the temperature environment of the 25 percent thrust level. The results of this test are shown in Figure 5. The extension was tested under conditions of impact with a flat surface and with a velocity equivalent to lunar impact. Thus, the force to initiate buckling is relatively high. If impact occurs on one edge as on a sharp surface, the initial force would

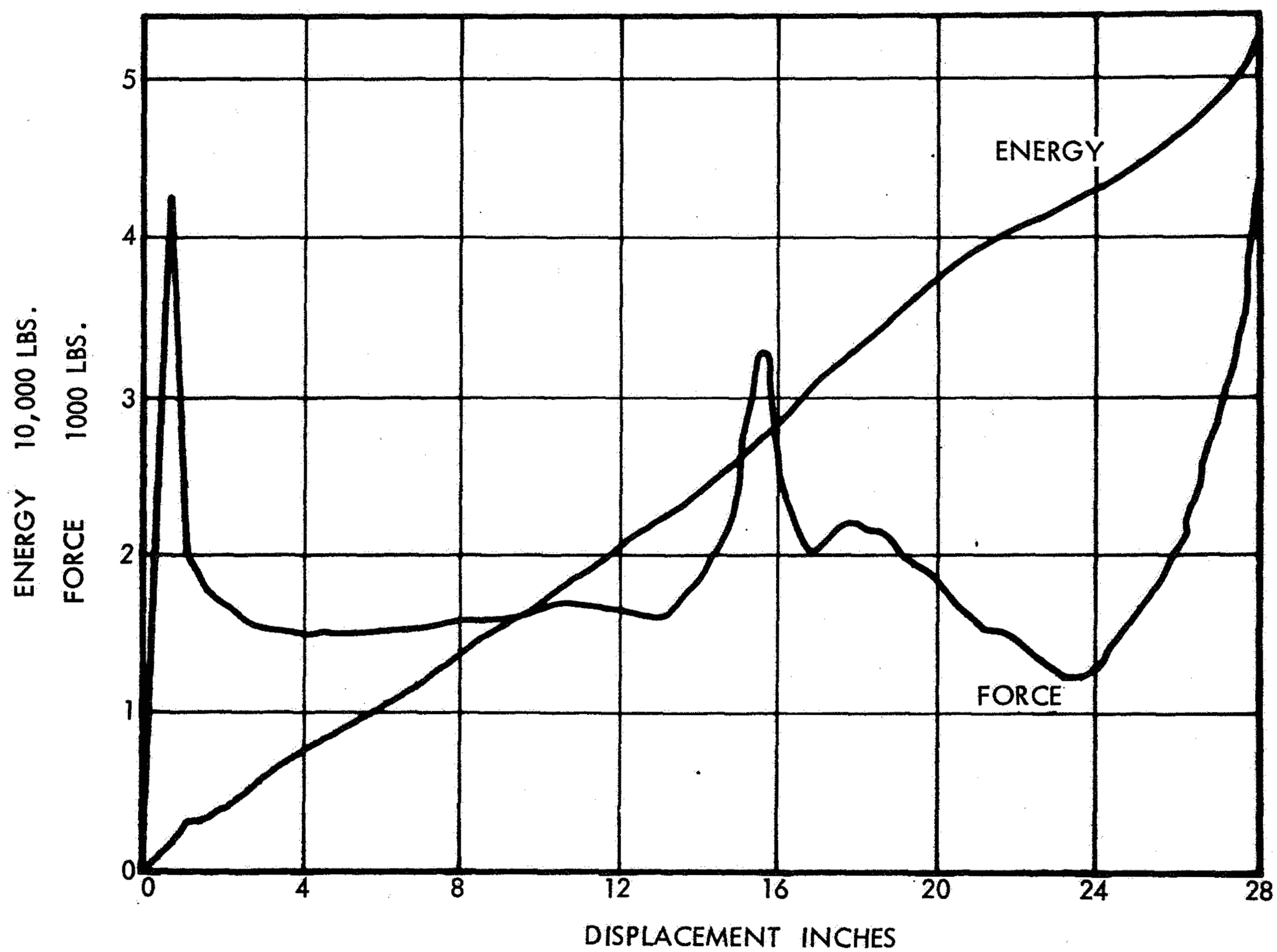


Figure 5. LMDE Nozzle Extension Crush Test Results

be reduced; however, if the cone is crushed to the same degree, the overall energy requirements would be the same.

It can be concluded from the test data available that a columbium extension with the wall thickness tapered to a minimum thickness within the basic constraints can be crushed to approximately 40 percent of its original length.

The length of a columbium extension that may be used on any given engine is limited only by the temperature limitations of the oxidation-resistant coating. This is currently 2400°F. A specific individual engine thermal analysis is required to establish this point.

2.7.3 Gimbaling

Gimbaling of a modified LM descent engine system around a point forward of the engine by mounting a gimbal above the oxidizer inlet tube can be accomplished by a universal type device with two spherical bearing axis (as shown in Figure 6) or with a single partially enclosed spherical ball joint (as shown in Figure 7). There are many possible alternates to these two concepts and a design study would be required to attain the optimum design. In both cases it will be necessary to reinforce the basic injector structure to carry the thrust and vibration load as of this type of mounting. Attachment lugs for mounting actuation devices can be readily provided at any point on the head end assembly or on the chamber. Gimbaling should be the preferred method of attitude control for a single engine arrangement if no other system considerations are imposed.

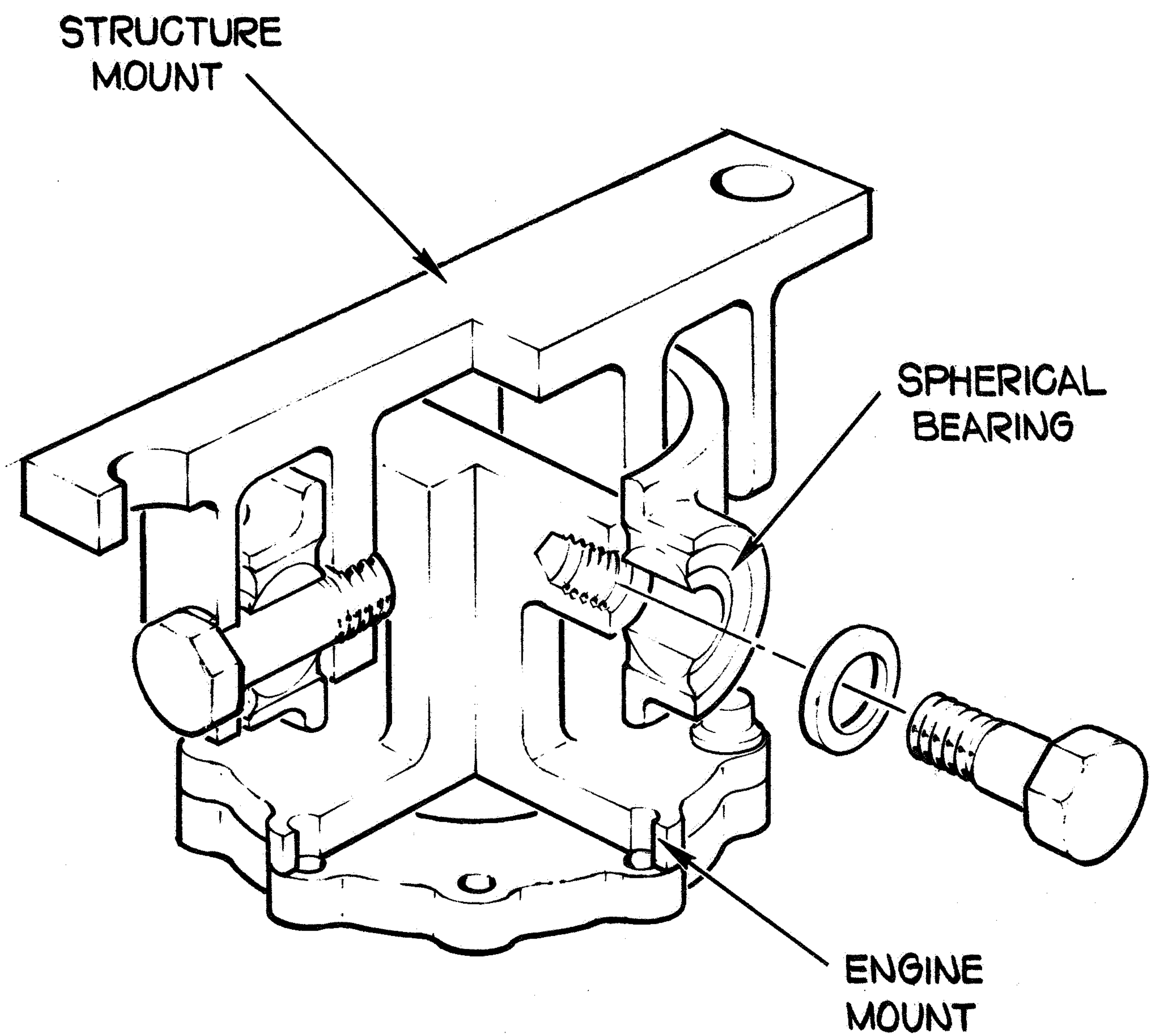


Figure 6. Universal Type Forward End Gimbal

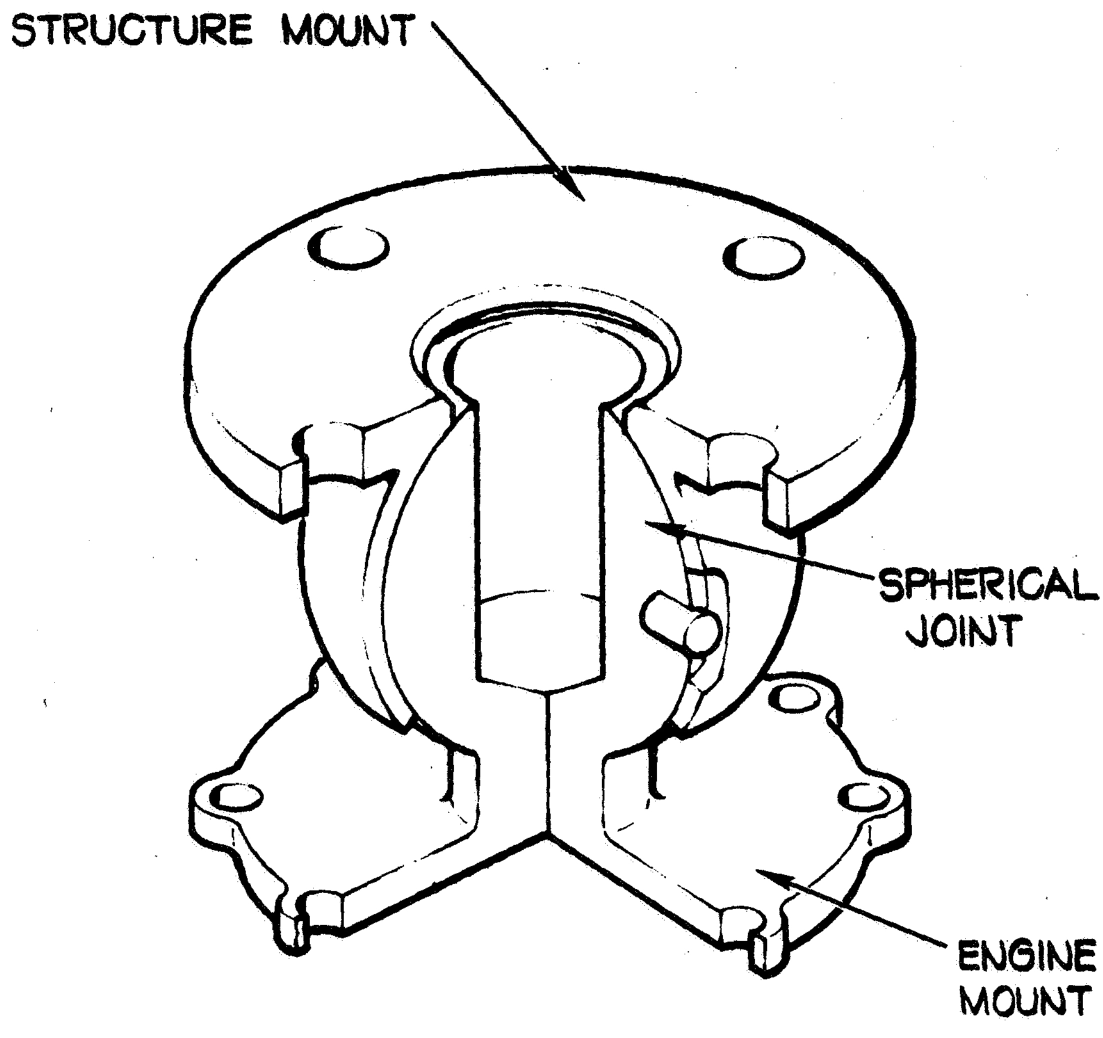


Figure 7. Spherical Forward End Gimbal

SPHERICAL
FORWARD END GIMBAL

FIGURE 7

2.8 FLIGHT CONTROL

The decision of whether to use a single high thrust engine or multiple low thrust engines on the Voyager Lander has significant effects on the flight control system design. While it would be presumptuous to attempt a total comparison of the respective flight dynamics at this time, there are certain factors which should be considered from the standpoint of engine control. The following paragraphs present a discussion of these factors as they affect single-, three- and six-engine concepts.

2.8.1 Open Loop Errors

A prime consideration in the selection of the number of engines should be the three sigma thrust vector errors which can be expected and the effect on vehicle control. While a total error evaluation must include the effects of C.G. shifts and installation variabilities, this study is limited to engine thrust vector errors. Thrust magnitude errors have the greatest influence and, thus, thrust direction errors resulting from such things as throat erosion were not considered.

a) Thrust Magnitude Errors

The relative error in total thrust magnitude can be evaluated assuming that the percentage errors for the small and large engines are equal. Engine thrust is given by Equation (1) and the percentage thrust variability by Equation (2).

$$F \propto (\dot{w}_o + \dot{w}_f)\ C_F\ C^* \tag{1}$$

where:

$\dot{w}_o$ = oxidizer flow rate

$\dot{w}_f$ = fuel flow rate

C_F = thrust coefficient

C^* = characteristic velocity

$$\frac{\partial F}{F} = \left(\frac{\dot{w}_o}{\dot{w}_o + \dot{w}_f}\right) \frac{\partial \dot{w}_o}{\dot{w}_o} + \left(\frac{\dot{w}_f}{\dot{w}_o + \dot{w}_f}\right) \frac{\partial \dot{w}_f}{\dot{w}_f} + \frac{\partial C_F}{C_F} + \frac{\partial C^*}{C^*} \tag{2}$$

Combining these errors gives the 3δ thrust error, P, per Equation (3):

$$\left(\frac{\partial F}{F}\right)_{3\sigma} = P \tag{3}$$

If N engines are used to deliver the total thrust as given by Equation (4), then the total thrust variability is given by Equation (5).

$$F_t = F_1 + F_2 - - - F_N \tag{4}$$

$$\frac{\partial F_t}{F_t} = \frac{\partial F_1 + \partial F_2 --- \partial F_N}{F_1 + F_2 --- F_N} \tag{5}$$

$$= \frac{\partial F_1}{NF_i} + \frac{\partial F_2}{NF_i} + --- \frac{\partial F_N}{NF_i}$$

where:

F_i = thrust level per engine

Combining these errors by Equation (6) gives the total 3δ thrust magnitude error.

$$\left(\frac{\partial F_t}{F_t}\right)_{3\sigma} = \left[\sum_{i=1}^{N} \left(\frac{\partial F_i}{NF_i}\right)^2\right]^{\frac{1}{2}} \tag{6}$$

$$= \frac{1}{\sqrt{N}} \left(\frac{\partial F_i}{F_i}\right)$$

$$= \frac{P}{\sqrt{N}}$$

where:

$$\left(\frac{\partial F_i}{F_i}\right) = \frac{\partial F}{F} = P$$

It can therefore be concluded that the greater the number of engines, the lower the total thrust variability. However, the significance of this variability relationship is minimized if the thrust magnitude errors are accounted for in the guidance and control system. This is discussed in further detail in Section 2.8.3.

b) Disturbing Moment Errors

Of greater significance to the flight control system are the disturbing moments introduced by thrust magnitude errors. While the total thrust variability requires only a thrust level correction, the displacement of these vector errors from the vehicle C.G. results in destabilizing moments. The following moment variability equations describe this relationship case for the one-, three- and six-engine systems, where principal axis have been assumed for the latter two.

Single Engine

$$M = F\ell \tag{7}$$

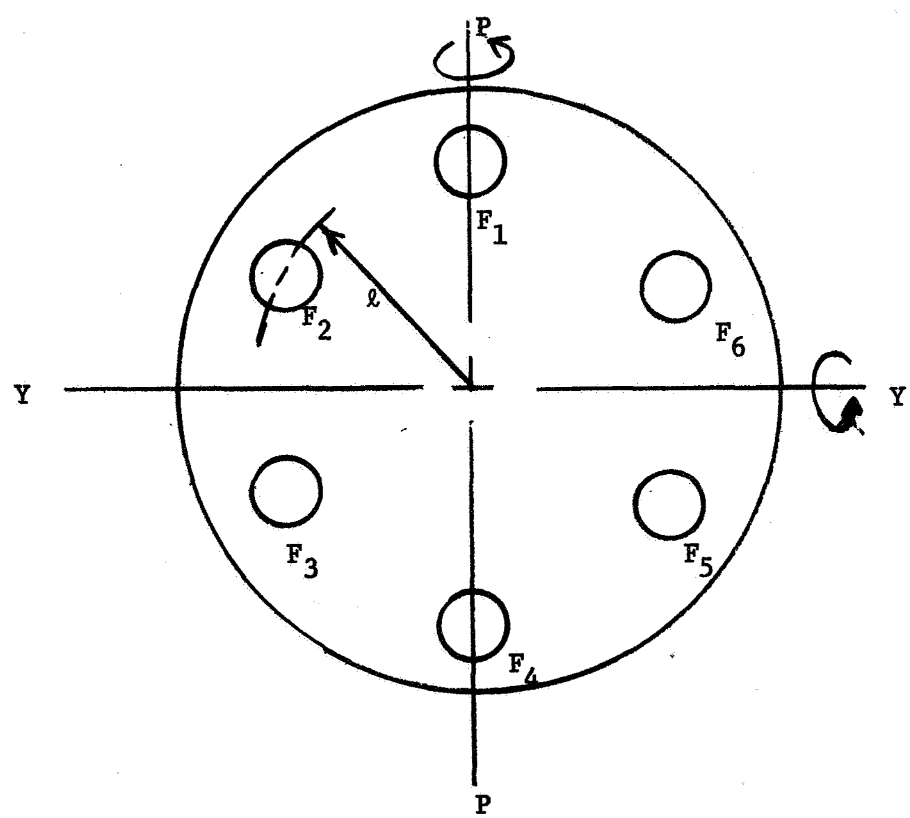
where:

ℓ = moment arm = 0

$$\partial M = \cancelto{0}{\ell}\,\partial F + F\,\partial\cancelto{0}{\ell} \tag{8}$$

$$= 0$$

Six Engines



The moments about the pitch axis are:

$$M_P = F_6 \ell \text{ Cos } \Pi/6 + F_5 \ell \text{ Cos } \Pi/6 - F_2 \ell \text{ Cos } \Pi/6 - F_3 \ell \text{ Cos } \Pi/6 \qquad (9)$$

The moment variation is given by:

$$\partial M_P = \ell \text{Cos } \Pi/6 \ (\partial F_6 + \partial F_5 - \partial F_2 - \partial F_3) + F_i \text{ Cos } \Pi/6 \partial \cancel{\ell}^{\,0} \qquad (10)$$

Combining these errors gives the following 3 moment error based on the equivalent percentage thrust magnitude error where F_i is the thrust per engine and is variable over a 10:1 range:

$$(\partial M_P)_{3\sigma} = 2F_i \ \ell \text{Cos } \Pi/6 \ (\partial F_i/F_i) \qquad (11)$$

$$= 2F_i \ \ell P \text{ Cos } \Pi/6$$

The yaw axis moment can be determined similarly:

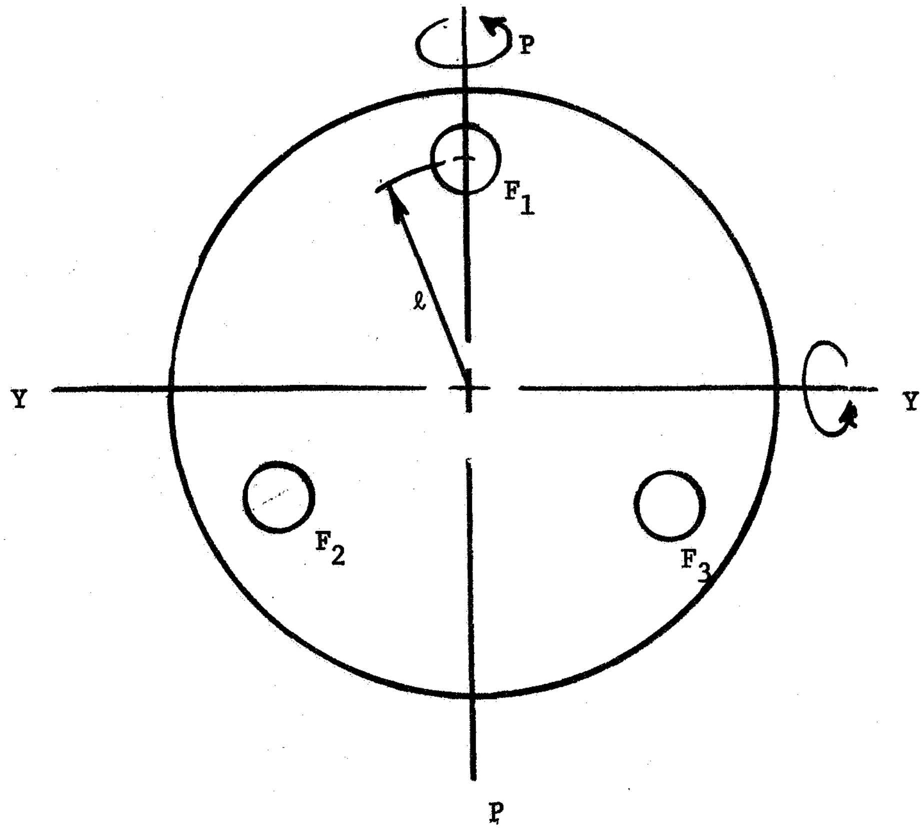
$$M_Y = F_3 \ \ell \text{Sin } \Pi/6 + F_5 \ell \text{Sin } \Pi/6 - F_2 \ell \text{Sin } \Pi/6 - F_6 \ell \text{Sin } \Pi/6 \qquad (12)$$

and,

$$(2M_Y)_{3\sigma} = \sqrt{2} \ F_i \ \ell \ \left(\frac{\partial F_i}{F_i}\right)_{3\sigma} \ (2 \text{ Sin}^2 \ \Pi/6 + 1)^{1/2} \qquad (13)$$

$$= \sqrt{2} \ F_i \ell \ P \ (2 \text{ Sin}^2 \ \Pi/6 + 1)^{1/2}$$

Three Engines



The moments about the pitch and yaw axis are given by:

$$M_P = F_3 \; \ell \mathrm{Cos}\; \Pi/6 - F_2 \; \ell \mathrm{Cos}\; \Pi/6 \tag{14}$$

$$\partial M_Y = F_2 \; \ell \mathrm{Sin}\; \Pi/6 + F_3 \; \ell \mathrm{Sin}\; \Pi/6 - F_1 \ell \tag{15}$$

Combining the moment variabilities in the same manner as for the six-engine case, the following 3σ errors can be defined:

$$(\partial M_P)_{3\sigma} = \sqrt{2} \quad F_1 \; \ell P \quad \mathrm{Cos}\; \Pi/6 \tag{16}$$

$$(\partial M_Y)_{3\sigma} = F_i \, \ell P \quad (2 \sin^2 \Pi/6 + 1)^{1/2} \tag{17}$$

It can be seen from these relationships that the use of multiple engines introduces a mean disturbing moment. This relationship can be used in conjunction with the anticipated thrust schedule to evaluate the additional propellant (impulse) and control which must be utilized to maintain stability. It is felt that this effect is considerably more significant than the thrust magnitude correction requirement.

2.8.2 Attitude Control

One of the potential advantages of using multiple throttleable engines is the use of differential thrust for attitude control. However, as shown in Paragraph 2.8.1 (a), multiple engines also contribute significantly to the required level of control. By controlling all engines in parallel infinite thrust magnitude resolution can be achieved from 10 percent to 100 percent of maximum thrust. Therefore, differential throttling can be used effectively for attitude control with good magnitude (moment) and direction control capability. Similarly, a two axis gimbaled single engine can give equal resolution characteristics. Therefore, the question resolves into one of the relative actuator control requirements. While the six-engine case requires six throttle actuators, the gimbaled engine requires two gimbal actuators. The requirement for mixing thrust level and attitude control signals is discussed in Paragraph 2.8.3.

2.8.3 Control System

a) Flow Control. One of the main advantages to the single-engine approach is that only a single throttle actuator is required. Therefore, there is no requirement for transient thrust matching as in the case of multiple engines. If this cannot be accomplished in the latter case, moment errors in addition to those discussed in Paragraph 2.8.1 (b) are introduced. These transient moment errors will actually dictatate the stability (response) requirements of the attitude control system while the errors derived in Paragraph 2.8.1 (b) will establish the additional propellant (impulse) requirements. The weight penalty for multiple actuators, whether electromechanical or electrohydraulic, and the additional electrical circuits must also be considered. On the other hand, the response of the smaller engines will be better assuming that a sufficient power supply is available.

b) Guidance and Control Interface. The type of guidance and control system selected will affect the relative performance of the potential engine concepts. For example, if the vehicle is controlled to a specific altitude, velocity and acceleration schedule using appropriate feedback loops, then the engine is throttled based on an error signal rather than by actually commanding a thrust level. If, however, the engine is controlled in an open-loop manner, such that the controller follows a predetermined thrust schedule, it is imperative that tight thrust repeatability be maintained. If the latter approach is used, then the thrust magnitude error established in Paragraph 2.8.1 (a) is extremely important while it is not important in the former guidance scheme. The exception to this is performance variations caused by off-design mixture ratio operation which is duscussed in the following section.

A second consideration is the necessity for mixing thrust (throttling) and differential thrust (attitude control) signals. In the multiple engine case, it is necessary to add the two signals for each engine while in the single-engine case, the gimbal gain must be modified based on the nominal thrust level. Therefore, it appears that regardless of the guidance system selected, it will be necessary to determine the nominal thrust level either through actuator position or chamber pressure measurements.

c) Mixture Ratio (Performance) Control. The main factor contributing to the C^* variation of Equation (2) is mixture ratio which can vary within limits as a result of temperature, pressure and supply system resistance variations. If it is necessary from the standpoint of performance and propellant utilization to consider mixture ratio control, it is more easily adapted to the single engine case than to multiple engines. In order to be effective, it would be necessary to control the mixture ratio of each engine independently, thus, requiring an equal number of controllers. The exact mixture ratio control design would depend upon whether an open-loop (implied mixture ratio) or closed-loop (measured mixture ratio) system could be used.

d) Conclusions. The effect of off-design engine operation on propellant consumption and control requirements is significant. In addition, the number of actuators and electrical circuits is a disadvantage to the multiple engine concept. The question of attitude control or stabilization should be answered based on a detailed weight, response and power consumption analysis. The response characteristics of a multiple engine system must be better than those of a single-engine although these engines must be dynamically matched to avoid introducing further disturbing moments. A single-engine approach is a simpler system from the standpoint of flight control.

2.9 LEAKAGE, PLUMBING AND CONTAMINATION

An examination of some of the hardware problems associated with the use of multiple (six or three) throttling engines versus one large engine has been conducted. The schematic in Figure 2 represents the systems which were studied.

The problems identified are discussed briefly below:

(a) Residual Propellants. The cross-sectional flow area of lines for each engine will be about 1/6 of that for a single engine if six engines are used and 1/3 if three engines are used. Line lengths, however, will not scale down. In fact, almost the same length of line will be required for each engine as for a single unit. The trapped propellant between the squib SOV and the engine SOV can, therefore, be expected to be six or more times greater for a six-engine configuration than for a single engine and 3 times greater for a three-engine configuration.

(b) Leakage and Mars Surface Contamination. Because of the larger number of lines, joints, flanges, etc. (varies directly with number of engines), the possibility of leakage at shutdown is greater by a factor of the number of engines than with a one-engine configuration.

(c) Engine Manifolding. Since it will be almost impossible to design a manifold that will supply the same inlet pressure to all engines, provisions have been made to orifice each engine individually. This requires prior knowledge of each engine's location. Without this capability, it is possible to have unbalanced thrust from the various engines. This effect can be significant at the maximum thrust level. It will be almost impossible to assure that all engines will give the same thrust for a particular command signal. Except for maximum thrust, the unbalanced forces resulting from this condition may require either a complicated thrust sensing system so that each engine can be throttled individually, or a matching of engines. The sources of nonuniform thrust of the engines are given below:

Positioning Accuracy. The ability to position each engine at a particular thrust level is about 1 percent of maximum thrust.

Engine Variability. Engine-to-engine variability is probably no better than 1 percent for any single position. This error is referenced to a standard configuration.

Run-to-Run Repeatability. This includes the effect of positioning accuracy and hysteresis within an engine. Combined effects are probably no better than 0.5 percent.

Erosion. If ablative chambers are used, throat erosion will alter the thrust of an engine at a thrust position. It is unlikely that a uniform erosion rate can be predicted for all six engines.

Engine Alignment. Location of the multiple engines in a ring will require that the thrust center line be determined within close limits.

(d) Contamination. The number of smaller flow passages, filters, and control orifices associated with a multiengine configuration will significantly increase the possibility of clogging, or blocking of some key flow components.

It can be deduced that the use of multiple engines will require a more stringent control of tolerances for the various components, a more involved development program, and a more complex system to achieve the same results as a single engine.

2.10 FEED SYSTEM DYNAMICS

For the Voyager Lander, the use of a single engine versus six engines must be evaluated on the basis of system dynamic considerations. The primary performance parameters are impulse degradation and engine variability. Engine variability is of concern because of the large flight control disturbing moments which can be developed during startup, shutdown, and throttling. The primary system parameters which can introduce off-performance operations are variations in power supply voltage, propellant temperatures, head-end assembly temperatures, hot and cold engine operation, and duty cycle. Since a six-engine configuration will tend to introduce significant temperature gradients between engines as a result of the engine hot gas recirculation, and since these recirculation patterns are not predictable, this system characteristic must be investigated. The following discussion briefly outlines some of the major points from both engine operation and configuration considerations:

(a) The engine feed line configuration with six engines will tend to be more complex and, therefore, tend to be more susceptible to trapping vapor gas (due to zero g operation) upstream of the shut-off valves. This trapped gas can introduce large variations in the initial start transient from engine-to-engine which can produce large disturbing flight control moments.

(b) The multiple feed lines associated with six engines can introduce: (1) lower feed line hydraulic frequencies which tend to increase the engine performance (impulse) degradation for minimum impulse bit and associated command frequency duty cycle, and (2) the starving of engines (as a result of the feed line location and cross-coupling of engines) and, therefore, a reduction in engine performance.

(c) The differential temperatures of the engine head-end assemblies (as a result of engine hot gas recirculation and the location of the engines) can introduce significant variations in engine performance as a function of engine position. The performance variation is a result of pressure drop changes in the injector and valves, changes in the response times of the flow control valves and shut-off valves, and the variation of dribble volume fill time. The effects of these temperature variations are most severe during engine startup and shutdown. As a result of these thrust transient variations between engines, large disturbing moments can be experienced. As an example, the firing of one engine prior to the remaining engines will cause a large moment which cannot be balanced by differential throttling, since it is the only operating engine. This transient thrust unbalance will dictate the stability (response) requirements of the attitude control system.

(d) The differential temperatures of the engine thrust chambers (hot and cold engine operation) can introduce significant variations on engine performance. These engine temperatures can be further varied by the location of the engines with relationship to the engine hot gas recirculation flow patterns and the design of the thermal shielding. These flow patterns are not predictable, which introduces statistical errors associated with this phenomena.

(e) The effects of engine transient response due to changes in the engine power supply can vary significantly. The power supply is a primary factor in determining the response of the throttle and shut-off valve actuation times. The power supply can change as a result of the number of power supply units (there is a trade-off between the number of power supply units, and the variability in these separate units, and reliability) and the regulation between these separate units and temperature changes in the electrical networks. The temperature variability can be introduced by engine operation (hot and cold engines) and the effectiveness of shielding the engine hot gases.

2.11 ELECTRICAL SYSTEM

The multiple versus single engine electrical systems are compared for weight and power requirements in Table III. There are also several more subtle considerations worthy of attention. Perhaps the most significant problem of the spacecraft design is that of preventing undesirable electrical interactions between the multiple engine installation, both in the reliability and in the electromagnetic compatibility aspects. Control functions to each of the multiple engines would have to be isolated, but synchronized, such that short circuit or open curcuit failures in any one engine or engine circuit would not degrade performance of the remaining engines.

Another significant problem in the use of multiple engines, especially where solenoid valves are concerned, is that of electromagnetic compatibility. The filtering and cable shielding necessary to suppress up to 24 solenoid circuits operating simultaneously impose a significant weight penalty upon valve drivers and spacecraft cabling beyond those given in Table III.

The problems associated with protecting inherently susceptible components, such as explosive valves, are compounded by the multiple installation.

TABLE III

WEIGHT AND POWER REQUIREMENTS

FOR ONE, THREE AND SIX ENGINE CONFIGURATIONS

Component	Power (watts) @ 28vdc LMDE	3L	6L	Weight (lb) LMDE	3L	6L
Throttle Actuator		NA	NA	(1)	NA	NA
Peak	225(2)					
SS	41			NA		
J-Box				5.75		
TAR & CCB	1.6			(3)		
TIR	2			(3)		
TVSA Valves	NA	270(2)	540(2)	NA	(1)	(1)
ECA		9	18		4.5	9.0
Shutoff Valves	65	204	408	(1)	(1)	(1)
Pressure Transducer	.35	1.05	2.1	0.25	0.75	1.5
Injector Position Indicator	NA	(4)	(4)	NA	0.3	0.6
Interface Control Panel		NA	NA		1.5	4.5
J-Box Bracket				1.5	NA	NA
Interconnecting Equipment				7.25	11.7	23.4
Peak Power (2)	294	484	968			
Total Weight				14.75	18.75	37.5

Notes:
(1) Weight included elsewhere.
(2) Intermittent -- during throttling only.
(3) Weight included in J-Box weight.
(4) Power included in ECA power.

Spacecraft signal and power distribution to multiple engines greatly increases the problems of cable routing and the concomitant envelope definition. Development and fabrication costs of the spacecraft and engine cabling would be proportional to the number of engines selected. Costs of engine throttle actuators are also directly proportional to the number of engines.

The application of six or three engines imposes peak power demands of 969 watts or 484 watts, respectively, on the spacecraft, as opposed to the single-engine configuration demand of 294 watts. Development of fail-safe power distribution and control functions represents difficult requirements to be imposed upon the spacecraft. Electromagnetic compatibility problems are compounded by the multiple installation configuration. Weight and procurement costs of both the spacecraft and engine electrical systems are nearly directly proportional to the number of engines in the configuration.

2.12 TELEMETRY

In addition to the engine instrumentation weight and power requirements of the multiple installations (summarized in Table III), the cost of telemetry and facilities equipment required to verify flight readiness and mission performance must be considered.

The nature of the measurement requirements is such that no economies can be effected either in cost or weight of instrumentation between the single high-thrust engine and one of the smaller multiple engines. Total telemetry system costs therefore are directly proportional to the number of engines required.

A typical engine installation includes 15 to 20 flight-readiness test points, including all valve and actuator positions and interface pressures and temperatures. The GSE equipment required to support two spacecraft with up to 12 engines and 240 channels of information would cost hundreds of thousands of dollars more than that for 40 channels.

Development and procurement costs of instrumentation, telemetry equipment, facilities, and GSE to support flight readiness tests and flight data transmission are nearly directly proportional to the number of engines in the vehicle configuration. Two spacecraft with six engines each could require up to 200 more channels of information than single engine installations. Data retrieval from Mars on the propulsion system is greatly simplified by the use of a single engine configuration.

2.13 BASE HEATING

The firing of multiple engines has, inherent in its operation, the problem of base heating. The base heating is a result of radiation from the exhaust plumes as well as the recirculation or reversed flow of combustion gases caused by expanding multiple plumes. In addition, improper spacing of the engines could cause choking of the external flow field with even higher heating rates. Finally, in an environment containing a form of oxidizer, an additional after-burning effect is possible, especially with engines running fuel-rich along the nozzle walls.

Although the choking problem associated with the engine spacing can be eliminated through judicious arrangement of the engines, the heating caused by radiation and recirculation is unavoidable. Solution of this problem may necessitate the use of a protective heat shield across the entire base of the spacecraft to prevent the hot gases from flowing over critical areas upstream of the engines.

In addition to the added weight penalty of the base heat shield, difficulty arises in designing the shield since present state-of-the-art analytical techniques are far from adequate in predicting base heating phenomena. Thus, a semiempirical approach must be used. As stated in a recent article in the J. Spacecraft*, "Design data for a base heat shield are generated through model tests and empirical relations developed through experience. Hot model testing is expensive and may be almost as complex as testing of the prototype." In addition, data are not available for the case where the external flow opposes the nozzle flow. This case may cause increased recirculation. Also, in order to define the severity of the after-burning problem, testing would have to be performed in an environment which, in addition to simulating the Martian ambient pressure, would also have to simulate the composition with respect to the amount of oxidizer present and the relative flow field around the spacecraft. This, of course, would involve a costly testing program.

2.14 SPACECRAFT HEAT LOAD

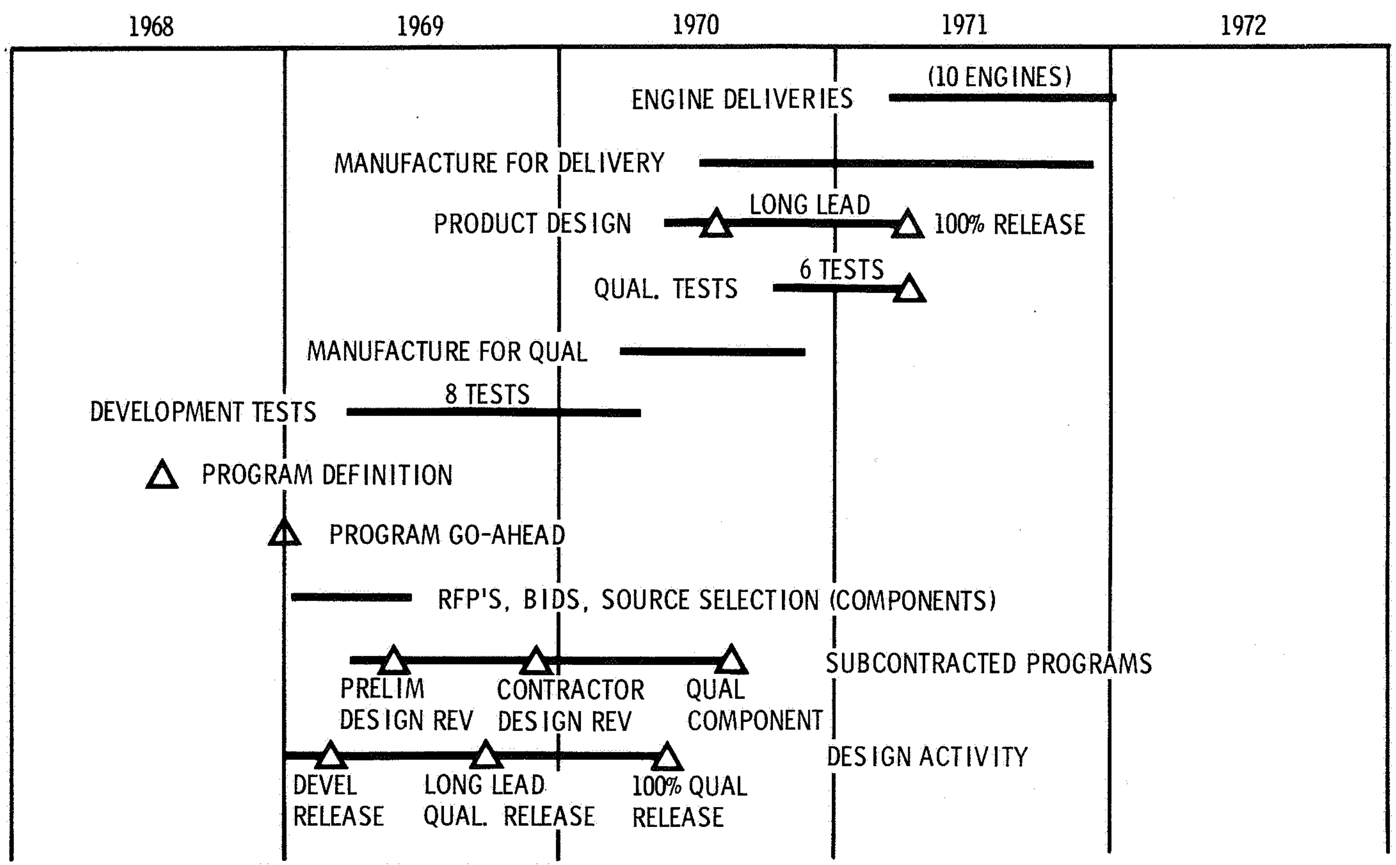
An obvious and probable multiple engine configuration is one in which the central (spacecraft centerline) portion is utilized for

* H. B. Wilson, "Results from Short - Duration Altitude - Chamber Techniques for Simulating Rocket Base Heating Problems," J. Spacecraft, May, 1967

instrumentation or payload. Under this scheme the central package would be exposed to a surrounding radiative heat source both during the firing and, to a greater extent, during the soakback period following the firing. Cooling of the central package by radiation would be hampered by the reduced view of space caused by the surrounding multiengine propulsion system. On the other hand, a single-engine arrangement would be a more concentrated heat source, much easier to shield, and leaves the spacecraft open for radiation and convective cooling.

2.15 ENGINE DEVELOPMENTAL EFFORT

One of the most important advantages of the single-engine configuration is the availability of a flight qualified engine in the proper thrust range (6000 to 9000 pounds) for Voyager Capsule Bus. The LMDE engine will complete its flight qualification in August 1967. Modifying the LMDE for the Voyager Capsule Bus would require a total program of 2-1/2 years through qualification after Phase D go ahead. New engines would require 3-1/2 to 4 years for qualification. Figures 8 and 9 present development schedules for LMDE and a new 1900-pound-thrust engine for Voyager Capsule Bus. The LMDE program would cost from $15 to $25 million including deliveries; a new engine would cost from $35 to $45 million including deliveries. Detailed cost breakdowns for these two cases are shown in Tables IV and V.



REFERENCES FROM QUAL "B" LMDE:

1. 6300-750 LB THRUSTS
2. RECONTOUR NOZZLE AND REMOVE ABLATIVE MATERIAL
3. NEW VALVES AND PLUMBING
4. USE MMH (MAYBE)
5. REDEVELOP ACTUATOR AND OTHER ELECTRONICS (THRUST RANGE AND STERILIZATION)

Figure 8. LMDE Modified for Voyager Capsule Bus

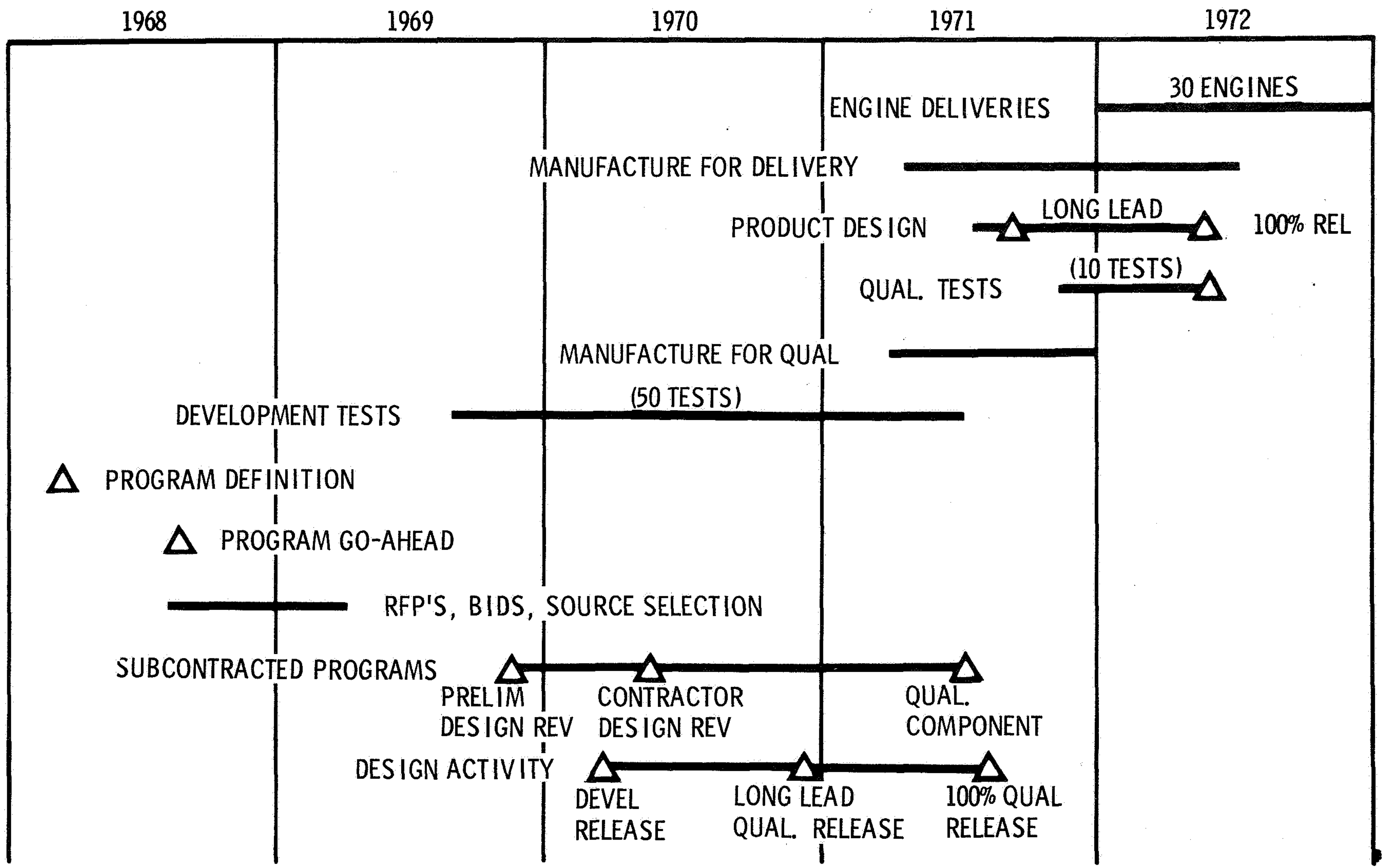


Figure 9. 1900 lb Thrust Engine for Voyager Capsule Bus

TABLE IV

VOYAGER FLIGHT CAPSULE LANDING ENGINE

PRICING DATA - 7000-LB THRUST LMDE APPLICATION

Assumptions:

1. Thrust - 600 to 7000 lb

2. Quantities - Development ....2 Engines
   4 Thrust Chambers

   Qualification ....2 Engines
   4 Thrust Chambers

   Delivery ....12 Engines

3. Schedule - Definition .... 6 months

   Development ....24 months

   Delivery ....18 months

   Total Program ....36 months

4. B & P Price - Development $13,310K

   Delivery ($300K x 12).. 3,600K

   Contract Total $16,910K

TABLE V

VOYAGER FLIGHT CAPSULE LANDING ENGINE

PRICING DATA - 1900-LB THRUST CLUSTER ENGINE

Assumptions:

1. Thrust - 160 to 1900 lb

2. Quantities - Development .... 8 Engines
14 Thrust Chambers

Qualification .... 4 Engines
6 Thrust Chambers

Delivery ....25 Engines

3. Schedule - Definition .... 6 months

Development ....36 months

Delivery ....18 months

Total Program ....48 months

4. B & P Price - Development $30,250K

Delivery ($170K x 25)... 4,250K

Contract Total $34,500K

2.16 PROPULSION SYSTEM DEVELOPMENTAL EFFORT

A six-engine configuration will require the greatest amount of system development testing because of the many additional modes of operation and the many more items of equipment that require integration into a working system. A six-engine system development test series would include:

a) Failure mode tests on the engine feed system

b) Interacting failure tests on the engine and feed systems

c) Failure sensing and corrective action tests

d) All up system tests with five and six engines operating

e) EMI and electrical system tests

A three-engine configuration would require less system and failure mode testing, but there would be a significant amount of system interaction testing required. EMI should be less of a problem as well.

The single-engine configuration should require the least amount of propulsion system development effort. Experience on the LM vehicle has demonstrated that with a minimum of feed system/engine tests that the vehicle has been readied for flight operations. LM has required approximately one year of system tests with a single-engine configuration. The other configurations should require significantly more.

2.17 GROWTH POTENTIAL

The current 7000-pound Voyager Capsule Bus requires approximately 6500 to 7500 pounds thrust for two-stage aerodynamic deceleration. If the capsule weight grows in the future, additional decelerating thrust will be required for near optimum performance. Additional thrust would require additional engine development or built-in thrust growth potential in the original engines. If growth potential is built-in, it generally is reflected in additional weight for the propulsion system. If, however, an existing engine is used that is derated for the Capsule Bus application and this engine is competitive weightwise with other engine configurations, growth potential can be achieved with no significant sacrifice in weight. This point can be best illustrated by an example. Table VI compares the weights of six, three and one engine arrangements with a total thrust of 7200 pounds. It then shows the weight increment required to increase the thrust level to 8230 pounds to match an increase in capsule weight of 1000 pounds. The weight increase due to growth potential is approximately 12 pounds for a three-engine arrangement and 18 pounds to provide the same growth potential for a six-engine arrangement. Since a single-engine configuration, using a derated engine, has no penalty for growth, it is most desirable from a growth potential viewpoint.

3. CONCLUSIONS

Considering all of the factors presented, the single-engine arrangement is the most desirable of the three arrangements even if a new engine were required. With an engine available for the single-engine arrangement, it becomes the most logical configuration for Voyager Capsule Bus.

TABLE VI

WEIGHT PENALTIES DUE TO GROWTH POTENTIAL

Weight of Capsule (lb)	Total Thrust (lb)	Single Derated Engine Weight (lb)	Three Engine Weight* (lb)	Six Engine Weight* (lb)
7000	7200	180	186	234
8000	8230	180	198	252
Increment due to growth		0	12	18

* Based on ablative engine