JOURNAL OF SPACECRAFT AND ROCKETS
Vol. 34, No. 2, March–April 1997

Fundamentals and Issues in Launch Vehicle Design

Robert S. Ryan* and John S. Townsend†
NASA Marshall Space Flight Center, Huntsville, Alabama 35812

The fundamentals of launch vehicle design are examined using simplified single-stage, two-stage, and Space Shuttle performance equations. The single-stage-to-orbit launch vehicle is very sensitive to the performance-critical parameters of mass efficiency, propulsion efficiency, and loss management. Cost and operations coupled in the performance equation further complicates the design process. Launch vehicle design is optimized when the performance and programmatic drivers are balanced. Programmatic drivers include affordability, reusability, operability, abort/safety, and reliability. The issues, disciplines, and potential problems that characterize the building of a future launch system are presented. The history of the Space Shuttle is used as the benchmark example. Robustness is the key to uncoupling the design factors so that optimization can occur, but typically robust designs define low-performance systems. Future space launch vehicles must develop new technologies to reshape the design parameter sensitivities of the robustness and performance functions.

Nomenclature

DMF_n	= dry-mass fraction, stage n [$n = 1, 2$ (one-usable propellant stage mass/total stage mass)]
g	= gravitational acceleration
I_{sp_n}	= specific impulse, stage n ($n = 1, 2$)
ℓn	= natural log
m_n	= stage n total mass ($n = 1, 2$)
m_o	= total vehicle mass; two-stage, $m_o = m_1 + m_2$
m_{p1}	= usable solid rocket booster (SRB) propellant mass
m^*	= Space Shuttle main engine (SSME) propellant mass used during SRB stage
n	= number of stages; $n = 0$ references the single-stage-to-orbit vehicle
α	= $1/\{1 + \beta[(1 - \mathrm{DMF}_2)/(1 - \mathrm{DMF}_1)](m_2/m_1)\}$ or $m_{p1}/(m_{p1} + m^*)$
β	= ratio of (SSME propellant used from liftoff to SRB separation)/(total SSME propellant used from liftoff to main engine cutoff)
ΔV	= velocity change from liftoff to orbit
ΔV_{loss}	= gravity losses, drag losses, etc.
λ	= $(1 - \beta + \beta \cdot \mathrm{DMF}_2)$ or $[(m_2 - m^*)/m_2]$

I. Introduction

A PRIMARY mission goal for NASA and the aerospace community is to design, develop, and maintain higher-quality space launch vehicle systems for less money. Because laws of physics determine launch vehicles to be high-energy systems, designs tend to be extremely complex when man/payload, safety, performance, and reliability are considered. Manned spaceflight is, therefore, an expensive business. NASA's budget has been steadily declining since 1993 and is projected to continue falling through the year 2000. At the same time, the Shuttle fleet is aging and operational costs are increasing. Today's cost per flight is estimated at $360 million (yearly Shuttle budget/number of flights per year). These dollars do not reflect the original development and inventory costs, but occur primarily from refurbishment, operations, upgrades, and life-cycle costs. There is approximately an equal cost on each launch for the development and operations of the payload that flies. If spaceflight is to continue into the 21st century with the same benefits and enthusiasm as mankind has witnessed in the past, then cost—the real-dollar cost per mission—must be reduced by an order of magnitude.

The challenge that confronts the aerospace community is to design a manned space launch vehicle that meets the basic mission requirements of performance, reuse, abort/safety, and reliability while reducing program operations and cost. Many engineers and managers believe that the operations cost factor will drive future vehicle designs to the single-stage-to-orbit (SSTO) concept. This paper does not address the feasibility of the SSTO design, but rather examines the fundamentals and issues in launch vehicle design. The design problem is a multiparameter optimization task and requires balancing of performance and programmatic drivers. This set of trades/sensitivities augmented by a series of imposed constraints always results in a suboptimized system. The degree of suboptimization coupled with the nonideal effects (management of losses) determines the success or failure of the program. New technology development is one key to the success of future aerospace programs. Technological areas that must be advanced include structures/materials, propulsion, manufacturing, thermal protection system (TPS), avionics, and health monitoring.

The paper begins with the development of the idealized rocket performance equation to determine key design drivers. It then examines the incorporation of nonideal effects into the vehicle design optimization tasks, followed by a look at the Saturn and Shuttle experiences for trends and lessons learned. The basic problem of balancing performance and programmatics, along with some approach for a solution, also is presented. The paper concludes with a review of the technology thrusts required if the low-cost space vehicle launch systems are realized.

II. Launch Vehicle Basics/Fundamentals

The fundamentals in launch vehicle design are well known as are most of the basic design issues. However, today's increased emphasis on cost and operational efficiencies merits that both be re-examined. The tight coupling between performance, cost, and operations dictates that all basic principles and historical experience be completely defined and understood if future launch vehicle design is to be successful.[1,2] First, the performance of a launch vehicle is examined from an idealized viewpoint.

A. Single-Stage-Rocket Idealized Performance

The single-stage-rocket idealized-performance equation can be represented in terms of dry-mass fraction as

$$\Delta V = -g \cdot I_{\mathrm{sp}} \cdot \ell n[\mathrm{DMF}] - \Delta V_{\mathrm{loss}} \tag{1}$$

Equation (1) identifies three key overall performance factors: 1) propulsion system efficiency, I_{sp}; 2) structural efficiency or total

Presented as Paper 96-1194 at the AIAA Dynamics Specialist Conference, Salt Lake City, UT, April 18–19, 1996; received Oct. 7, 1996; revision received Jan. 8, 1997; accepted for publication Jan. 8, 1997.

*Deputy Director, Structures and Dynamics Laboratory, currently retired. Associate Fellow AIAA.

†Aerospace Engineer, Structures and Dynamics Laboratory. Member AIAA.

dry-mass effects vs propellant mass, DMF; and 3) losses, ΔV_{loss}. In real life, these design parameters are interrelated in very complex ways. For example, propulsion system inefficiencies increase the propellant requirements, thus adding structural weight and/or tank volume. This weight increase requires more propulsion system output, such as first-stage thrust to compensate. Also, additional propulsion is needed to maintain liftoff acceleration required for pad clearance in high winds. Nonideal effects and the natural losses in a complex system create the same closed-loop escalation, thus compromising design simplicity, robustness, operations, and cost, as well as performance. Minimization of losses is also very critical, for losses are equivalent to payload-to-orbit. Figure 1 shows a graph of these effects for SSTO vehicle dry weight and dry-weight margin. The different curves represent a comparison of the state-of-the-art structural and propulsion system technologies to predicted technology improvements in each discipline, as well as improvements in loss management (risk mitigation). Because these curves are nonlinear, an SSTO vehicle design must be on the flat portion of a curve at the chosen weight-margin percentage, otherwise the total dry-mass sensitivity is nonlinear and becomes prohibitive at realistic margin assumptions. It is clear that for an SSTO design, if a key parameter is missed slightly, the performance is not obtained. The same observations can be applied to multistage vehicles but with less sensitivity.

B. Two-Stage-Rocket Idealized Performance

A two-stage vehicle or some partial version, such as the Space Shuttle, is less sensitive to design uncertainties. An idealized two-stage-rocket performance equation is as follows:

$$\Delta V = -g \cdot I_{\text{sp}_1} \cdot \ln[m_2/m_0 + \text{DMF}_1 \cdot (m_1/m_0)] - g \cdot I_{\text{sp}_2} \cdot \ln[\text{DMF}_2] - \Delta V_{\text{loss}} \qquad (2)$$

Equation (2) models the same characterization as for the SSTO Eq. (1) but with some major shifts. There are two separate propulsion and structural systems, with a modifying term in the first-stage terms that includes the first-stage-mass-to-total-vehicle-mass ratio and the second-stage-mass-to-total-vehicle-mass ratio. The ΔV_{loss} term defines losses that occur for both first- and second-stage burn and must be managed as discussed earlier. Performance characteristics of the single-stage, two-stage, and Space Shuttle idealized rocket equations are compared next.

C. Idealized Rocket Performance Characteristics

Assuming a special design case of equivalent stage masses, specific impulses, and dry-mass fractions, Fig. 2 plots the characteristic velocity ratio of a two-stage/single-stage vehicle vs the dry-mass fraction. Note that two-stage velocity is considerably higher than SSTO velocity for a given dry-mass fraction and engine I_{sp}. In this example, velocity increase translates into a 20% decrease in I_{sp} for the two-stage design or a 60% increase in its dry-mass fraction. Thus, an equivalent two-stage rocket design has a built-in margin and, as a result, is less sensitive to design uncertainties than a single-stage rocket.

For several reasons, including the desirability of having all liquid propulsion engines healthy and verified at liftoff, the Space Shuttle uses parallel-burn dual propulsion systems, dropping off one system (solid rocket motors) after 120 s of flight time. Equation (3) is an idealized equation of the Shuttle system using two propulsion systems with separate masses for each system:

$$\Delta V = -g[\alpha I_{\text{sp}_1} + (1-\alpha) I_{\text{sp}_2}] \cdot \ln[\lambda(m_2/m_0) + \text{DMF}_1(m_1/m_0)] - g \cdot I_{\text{sp}_2} \cdot \ln[\text{DMF}_2/\lambda] - \Delta V_{\text{loss}} \qquad (3)$$

Notice in Eq. (3) that as λ and α approach one (i.e., β approaches zero), this idealized Space Shuttle equation approaches the two-stage idealized rocket equation (2).

Using Eq. (3) and the data from the Space Shuttle (see Table 1), a parametric evaluation is accomplished. Figure 3 is a plot of the results, showing the effects of the percentage of Space Shuttle main engine (SSME)-usable propellant consumed during the parallel-burn portion of flight on delta escape velocity. The operation point of the Shuttle is shown at approximately 22%. Notice that there is some performance loss due to the parallel burn vs an idealized two-stage rocket, where β equals zero. These idealized equations do not contain losses. In general, loss sources include path deviation, load increases, propellant reserves, environment uncertainty, and technology readiness misses. To this point, using idealized equations and not considering the impact of cost, operations, and other programmatic issues, the characteristics of staging vs performance enhancement efficiencies have been determined. It is also clear that an SSTO rocket requires highly efficient structure and propulsion

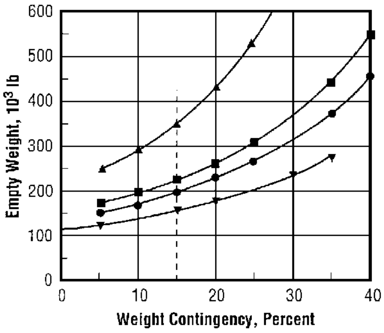

Fig. 1 Vehicle dry weight vs dry-weight margin.

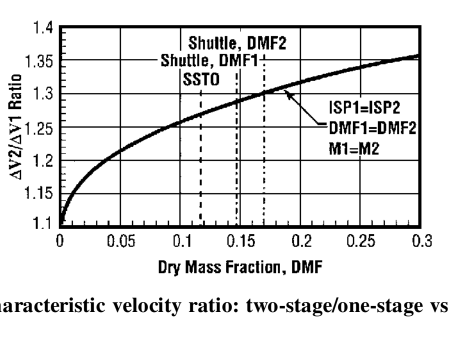

Fig. 2 Characteristic velocity ratio: two-stage/one-stage vs dry-mass fraction.

Table 1 Shuttle system mission weight and performance summary (typical/lb)

Vehicle wt at liftoff	4,511,795	m_0
1st stage wt	2,600,318	m_1
SRB × 2-inert wt	375,800	m_{s1}
SRB × 2-propellant wt	2,224,518	m_{p1}
2nd stage wt	1,911,477	m_2
Shuttle system at main engine cutoff	322,807	m_{s2}
SSME propellant expended	1,588,670	m_{p2}
SRB dry-mass fraction (1st stage)	0.144	DMF_1
Shuttle system 2nd stage dry-mass fraction	0.169	DMF_2
SRB I_{sp} (Av, LBF-SEC/LBM)	265	I_{sp_1}
SSME I_{sp} (vacuum, LBF-SEC/LBM)	452	I_{sp_2}
SSME propellant used from liftoff to SRB separation, lb	356,081	m^*
SSME propellant SRB stage ratio	0.224	β
2nd stage true mass ratio: $[(m_2 - m^*)/m_2]$	0.814	λ
1st stage true propellant mass ratio: $[m_{p1}/(m_{p1} + m^*)]$	0.862	α
Ratio of 1st stage inert mass to 2nd stage mass: $\text{DMF}_1 \cdot \{m_1/m_2\}$	0.196	η

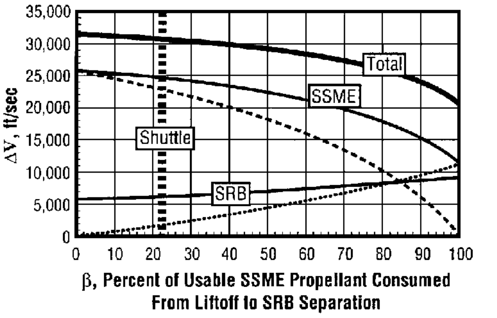

Fig. 3 Idealized Space Shuttle velocity.

systems, as well as an intense program of loss management (risk mitigation). Taking the next step requires that the programmatics be added to the design equation.

III. Top-Level Issues

In today's culture, a launch system cannot be designed, built, and sold on performance alone. Economy must be a fundamental part of the design equation, as well as other programmatic issues such as operations, reusability, reliability, and safety/abort. David Pye, in *The Nature of Design*, said, "When economy is brought into the design equation the product is strongly impacted."[3] The same is true of all programmatic factors. As shown in Fig. 4, the top-level issues in the technical (performance) are mass efficiency, propulsion efficiency, and managing losses, whereas for the programmatics they are operability, reusability, affordability, safety/abort, and reliability. The driving fact, in practical design, is that the performance and the programmatics are strongly coupled, many times in a highly nonlinear characterization. This strong coupling between technical and programmatics therefore requires a balancing act (optimization) between the coupled drivers (see Fig. 4).

This relationship of design for performance and cost is compounded by a series of constraints that must be applied to any design to meet the requirements of the program. When these constraints are applied, a suboptimal system always occurs. The degree of suboptimization, the sensitivities, and trades drive the concept selection and finally the program success. The first driver is always weight and how it is managed, paralleled with propulsion system efficiencies and loss management. Constraints that are fundamental to vehicle concept selection and design suboptimization are discussed in the following paragraphs.

A. Constraints

1. Thrust-to-Weight at Liftoff

The acceleration at which the vehicle leaves the pad has two important effects on the vehicle design/performance. Ground winds, aerodynamics, and thrust misalignments cause the vehicle to drift from the desired path. This drift also can create potential impacts of the vehicle with the launch pad. The faster the acceleration, the less relative drift that occurs near the tower. The angle of attack also is reduced as the vehicle leaves the pad. The second effect of higher acceleration is the initial kick that reduces gravity losses. As a result, design has a constraint of a minimum liftoff thrust to weight, which obviously influences the weight/propulsion thrust trades. Typically, the thrust-to-weight ratio is no lower than 1.1. Preferably, it should be around 1.2.

2. Manned/Unmanned

Manned flight requires that the vehicle acceleration not exceed the human endurance limit. Trained astronauts can handle up to 5 g acceleration, whereas a normal citizen can handle 3 g. This limit makes the vehicle suboptimal in terms of performance. In addition, the propulsion system must be throttled to meet these constraints, which is a complication to the propulsion system. A closed-loop control system is preferred. In addition, the required life support further drives the configuration, adding complexity to the design.

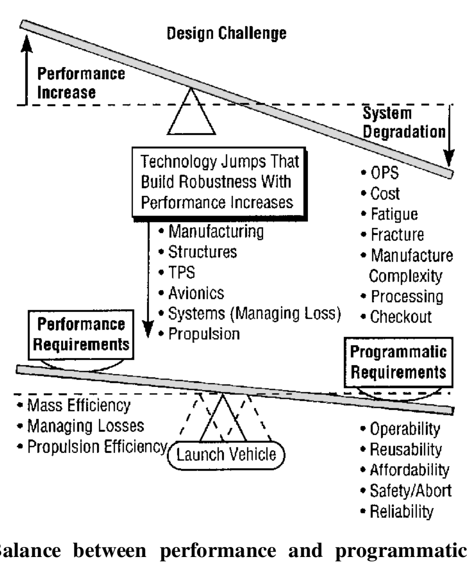

Fig. 4 Balance between performance and programmatic requirements.

3. Contingencies

A constraint usually is placed on the contingencies, plus an aggressive program for management of these contingencies is put in place as a major part of the design process. Contingencies must deal with all potential losses and their management. Part of the management of a losses program is constraints. For example, dynamic pressure q must be controlled by constraints. This constraint is placed to control flutter boundaries and loads, adding complexity on the propulsion system (throttling), or on the trajectory by lofting. Lofting creates a much higher performance loss than throttling. For example, on the Space Shuttle, reducing q by 1 psf costs 25 lb of payload throttling, whereas for lofting, it costs 250 lb of payload. Usually, a constraint is placed on angle of attack to reduce aerodynamically induced loads and, thus, keep structural weight bracketed. This constraint introduces complexity in the trajectory shaping (wind biasing) and load relief control, as well as causes performance losses. Design is a trade between performance loss from drift vs performance loss resulting from increased structural weights due to larger angles of attack and loads.

Other constraints can be placed to bracket events such as stage separation conditions. These constraints also impact performance, further suboptimizing the system. Ascent rates are another source of constraints because they drive the thermal protection system design.

An assumption on constraint is usually placed on the design for weight margins, normally stated as dry-weight margins. Current history of launch vehicles places this number around 30% (growth of weight during design and development). It is desirable in today's environment to reduce this constraint number to 15%. This level of reduction in the weight growth constraint dictates an aggressive weight management program. The program must not only deal with structures, but also the losses that drive the weight. One alleviation approach is an optimization approach that reduces propellant tankage, such as burning high-density propellants earlier and high-energy propellants later in flight. Utilization of the atmospheric oxygen also helps this constraint.

4. Payload Accommodations

Payload constraints/requirements come in several forms, such as weight, length, diameter, and attachment approach. These constraints can drive the aerodynamic configuration, such as fairings/shrouds for nose compartments, to payload doors for other configurations. Attachment mechanisms must be a part of the vehicle design with means of accommodating payload deployment or astronaut egression.

5. Cost/Reliability

Cost constraints/goals have a dominant effect on the design. There are many design considerations from cost viewpoints. Reduction of the number of piece parts is a key driver. Modularization is, in general, a cost alleviator. Weight has been equated with cost, introducing weight constraints. This can be a misnomer because weight can be added to obtain simplicity and robustness, which reduces cost. This trade can be extended to trading SSTO vs multistage vs technology advancement.

6. Reusability

The requirement for reusability is a major design driver. The first impact is how to bring the vehicle safely back to Earth, bleeding off the energy that was added to the system to get the performance. Initially, parachutes were used. The Space Shuttle uses wings (structure) to fly the vehicle back. The propulsion system (engines) can be restarted and used to kill off the energy. Re-entry also requires a complicated thermal protection system to absorb or dissipate the heat generated when re-entering the Earth's atmosphere. Because all the energy is never eliminated, Earth landings in various forms are required either as landing gear or water impact and recovery.

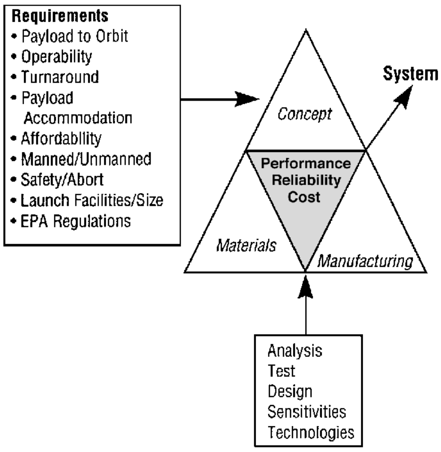

Fig. 5 Launch vehicle design process.

Reusability also implies design for fatigue, damage, wear, refurbishment, etc. This means a detailed understanding of dynamic and thermal environments, parts, wear, etc., as well as an intensive program of inspections, health monitoring, part refurbishment and retirement, and obsolescence. Reusability implies several design impacts, not the least of which is the design for inspections, disassemblage, etc., to ensure safe and successful launches.

Future launch vehicles must bring two other constraints into the design equation: total cost and operations efficiencies. As discussed previously, the first task is to get the required performance regardless of the concept chosen. This is a balancing act between the factors listed earlier. Without performance, cost and operations have no meaning. Bringing cost and operations into the performance equation greatly complicates the design process.

B. Design Process

The design process must incorporate into the design equation a consideration of these constraints. The process must have a system focus that considers the launch vehicle, the launch facilities, the payloads, and the payload accommodation as a total system or program. System focus is necessary to ensure compatibility and proper trades. With this focus, the concept selection and design process becomes an iterative sequence considering all of the requirements and constraints. Several concepts are selected along with materials and manufacturing approaches and analyzed through sensitivity studies, test, technologies, etc., to determine the best concept solution (Fig. 5). The design follows in the same manner. In all cases, as stated previously, solutions are now suboptimal. The best design solutions flow out from the metrics of performance and programmatics, where cost is a large driver.

In summary, the design process must concern itself with optimization between a constrained set of systems that are strongly coupled between the performance and the programmatics (see Fig. 4). Weight is always a critical concern and, thus, mass efficiency is a major technical issue. Propulsion efficiency is the balancing side to the weight issue. Playing these together as a vehicle with the constraints and programmatic drivers quickly uncovers the technical requirement: to efficiently manage the losses. These losses occur from unexpected environments, unexpected phenomena, unrealized technologies, requirements growth, etc. If not managed properly, losses can quickly eat into the performance margin and greatly increase launch vehicle cost and operations.

IV. Heritage Lessons

NASA's rich heritage in transportation systems (performance-wise, very successful) provides the illustration of what to consider for the future.[1] The Saturn family and the Space Shuttle are rich in this heritage, with the Space Shuttle having the greatest database source because of its reusability and numerous flights.

A. Saturn Family

The Saturn family consisted of three basic vehicles, the Saturn I, the Saturn IB, and the Saturn V. The Saturn I was a state-of-the-art vehicle built using Redstone, Jupiter, tooling, materials, and propulsion systems, plus the Pratt and Whitney RL-10 engine. The Saturn IB was an evolution of the Saturn I, but contained upgraded H-l engines and a new liquid hydrogen high-thrust engine for the second stage. Because Saturn IB was originally an evolutionary test bed for Saturn V (the moon rocket), its second stage (S-IVB) was the third stage of Saturn V as well as the instrument unit. The characteristics of Saturn V are shown in Fig. 6. Saturn V was designed, in general, as a robust vehicle. The initial performance studies indicated a requirement of only four F-1s for the first stage and four J-2s for the second stage. It was decided, however, after weight growth due to unexpected phenomena, to use five engines each on the first and second stages. The extra engine per stage, along with added propellant, provided increased performance and improved base heating. The design was fortuitous in that this robustness allowed the addition of the Lunar Rover on later moon flights and the successful launching of Skylab Space Station.

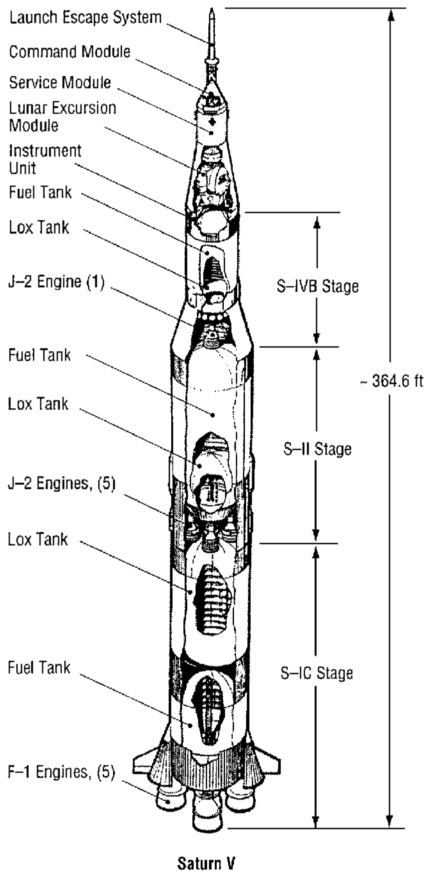

Fig. 6 Characteristics of the Saturn V launch vehicle (liftoff weight = 6,400,000 lb and payload weight = 285,000 lb).

Saturn taught many valuable lessons, not the least of which is the requirement for systems interdisciplinary focus. For example, it was standard practice then to use rigid-body, control-load relief using accelerometers to determine angle of attack. Saturn V design started with this concept, previously developed and verified on Saturn IB. This approach worked well for an assumed rigid vehicle. However, when the elastic lateral bending modes, in conjunction with realistic wind gust, were included, the loads increased on the forward third of the vehicle. After many aeroelastic and elastic body control studies were run, it was determined more optimum not to use rigid-body load relief control. Vehicle loads were reduced by staying with pure rigid-body attitude control, monthly mean wind biasing, and rate damping the first lateral bending mode.

Pogo was another system problem experienced on both the first and the second stages of the vehicle. Pogo is an unstable longitudinal oscillation that occurs from the coupling of the propulsion system, the structural dynamic system, and the acoustic system. Figure 7 shows the structural gain and frequency differences between the first Saturn launch (AS-501), which was pogo free,

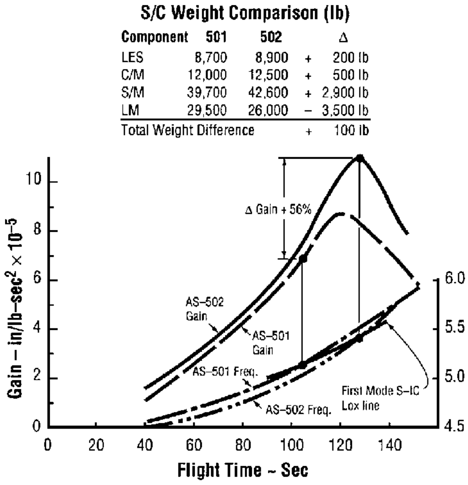

Fig. 7 AS-501/AS-502 first mode longitudinal structural dynamic characteristics.

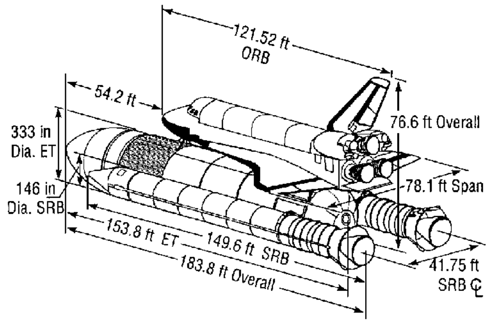

Fig. 8 Space Shuttle configuration and characteristics.

and the second Saturn launch (AS-502), which had an unstable pogo oscillation. The gain and frequency shift was caused by shifting in the Apollo lunar mass distribution (only 100 lb total change) as shown at the top of the figure. The message was very clear—system sensitivities must be clearly understood. The S-II pogo problem, as well as several other problems, taught the same lesson. Saturn also taught the prudence of using an evolutionary building-block approach for new and expanding technology systems.

B. Space Shuttle

The Space Shuttle has been a very successful program. It is the only reusable transportation system to space in use today; however, it has not been without problems and is costly to operate. The Space Shuttle history is probably the greatest source of lessons for the development of future transportation systems. Figure 8 is a sketch of the Shuttle configuration, giving the basic characteristics. It is reusable except for the external propellant tank.

The evolution of the Space Shuttle has followed a very interesting path. Originally, NASA Headquarters placed a series of constraints or design goals on each element. The total vehicle was limited in terms of gross liftoff weight and dry weight per element. In the early design phases, these goals were not being met and performance was degrading. For example, initially the orbiter was between 20,000 and 30,000 lb overweight. The original lifetime or mission model was given as 100 missions. The result of this weight creep was the institution of a weight savings program. To save weight, the SSME used basically an all-welded approach and no fracture control. The other elements reduced weight also. The orbiter took weight out of the wing. The external tank (ET) reduced its design safety factor from 1.4 to 1.25 for all well-known loads, saving approximately 10,000 lb. In addition, the design was modified to reduce the mission model to 55 missions and increase the SSME thrust to 109% of the nominal design. It was believed that the reduction of the mission model to 55 missions would cover the added fatigue damage of the 109%. However, this was not the case and many fatigue problems resulted. Criteria for winds were reduced and monthly mean wind biasing was instituted to save weight by reducing loads.

Several unexpected occurrences further compounded the problem. The SSME's I_{sp} was 2.5 s short. The solid rocket motor (SRM) had an I_{sp} shortage of 1.5 s. On the Space Transportation System (STS)-1, two unexpected environment deviations occurred. STS-1 lofted significantly because of a missed aerodynamic plume interaction effect that changed the pitch overturning moment and shifted the aerodynamic distribution on the orbiter wing. The result was a load increase of 30–40%. The solution to this problem cost the program approximately 5000 lb of payload and greatly reduced launch probability for the winter months. STS-1 also had a large SRM-induced overpressure wave that induced large dynamic loads on the vehicle. The solution required modification to the launch platform to suppress this acoustic wave.

As a result of these and other problems of performance and the fatigue and wear problems of the SSME, many design changes have evolved to increase performance and reduce maintenance problems. These include but are not limited to the high-performance SRM, new alternate high-pressure turbopumps, the super lightweight ET, plus many optimization approaches.

In summary, the Space Shuttle as it evolved experienced many performance degradations due not only to lack of technology readiness, but also the losses that occurred due to unexpected environments.[1,2] A brief discussion of several major problems experienced in the Shuttle program along with their solutions and the major lessons/focuses that resulted is given below.

1. *Liftoff Response*

Liftoff of the Space Shuttle is a very dynamic event that is very complex in nature. The event starts with propellant fill, which, because of the propellant temperatures, shrinks the tank structure and stores energy that is released later. During the liftoff sequence, the first step is to ignite the SSMEs and power them up to full thrust to ensure engine health. Because of the vehicle's asymmetry in the pitch plane, the vehicle bends from its cantilevered SRB attach points to the mobile launch platform, and squeezes the tank between the SRBs, storing energy. The preliftoff phase introduces a large dynamic load into the aft SRB skirt. The SRMs then are ignited at 95% SSME thrust, releasing the vehicle from the pad and subsequently releasing all of the stored energy as structural dynamic transients. Dynamic response designed much of the vehicle structure and was a major design issue. To help alleviate the effect of the stored energy on the Space Shuttle dynamic response, two changes were made: 1) the SSME engine starts and on-pad abort shutdowns were staggered and 2) the SRM ignition was delayed to occur at the minimum of the engine-induced stored bending moment. Even with these changes, impacts occurred. First, there was some SSME propellant performance loss due to the SRB ignition delays. Second, the liftoff dynamics were still large, creating design changes due to load sensitivities, also increasing structural weight and decreasing performance. Third, to adequately understand and predict these loads, a massive structural dynamics model had to be generated and verified. This model contained approximately 300 modes below 30 Hz.

2. *Maximum Dynamic Pressure Response*

Although the Space Shuttle is aerodynamically stable, the large wing surface introduces large forces and creates large dynamic (rigid-body) response to wind gust and shears. These aerodynamic induced loads were a design problem; therefore, the Shuttle has several design features for load alleviation. First, the trajectory is wind biased to the monthly mean wind. Second, there is rigid-body load relief control in pitch, yaw, and roll, as well as elevon load relief. Load relief introduces path errors, which translates into performance loss. When the trajectory is corrected after the high dynamic pressure regime to get back near an optimum path, a large angle of attack results. Large aerothermal loads then are created, impacting the thermal protection system design, and another performance loss occurs. These losses were about 1000–2000 lb.

a. STS-1 aerodynamic response. On the first Space Shuttle launch, the vehicle lofted substantially more than predicted. Also, the orbiter wing strain gauges read 30–40% higher than expected. The cause turned out to be a missed prediction of the aerodynamic distribution on the orbiter wing, creating a larger total vehicle overturning moment, and higher-than-expected wing loads. The cause was the plume effects of the propulsion systems coupled with the tunneling effects of the SRMs and ET compared to the orbiter wings. The solution was either beef up the orbiter wing or fly a less optimum trajectory. The final approach was a combination of the two options. The orbiter wing leading edge was beefed up and the trajectory was changed from a −2-deg angle of attack to a −6-deg angle of attack. Figure 9 shows a plot of q-alpha vs Mach number for the original design trajectory and the new design trajectory. Included are all of the parameter variations and their effect compared to the nominal. The net result was a 5000-lb payload loss plus a reduction in launch probability due to winds-aloft constraints. As a result, a day-of-launch wind response simulation based on wind measured every 2 h before launch was baselined. If the wind-induced loads were too high, the launch was held until better wind conditions prevailed. Originally, the launch probability for the most windy months (January, February, and March) was approximately 65%. As a way of increasing this launch probability, a day-of-launch I-load update was developed and implemented that biases the trajectory to the wind measurement taken 2 h prior to launch. Before implementing this approach, four Shuttle launches were delayed because of winds aloft.

3. SSME Wear and Fatigue Examples

One fatigue problem will be given as an example for the SSME, as well as a liquid oxygen (lox) pump-bearing wear problem.

a. 4000 Hz. To achieve the high performance, the SSMEs lox flows at a high velocity and pressure. The lox flows into the powerhead dome, passing through a splitter at the inlet tee. Figure 10 is a sketch of the hardware and cracks. Approximately 20% of the SSMEs buzzed at full power at 4000 Hz, creating large accelerations on the gimbal block and cracking the splitter. The failure cause turned out to be small differences in the splitter trailing edge, all within specifications, which created a vortex-shedding-type oscillation. The fix was simple: Taper and smooth the splitter

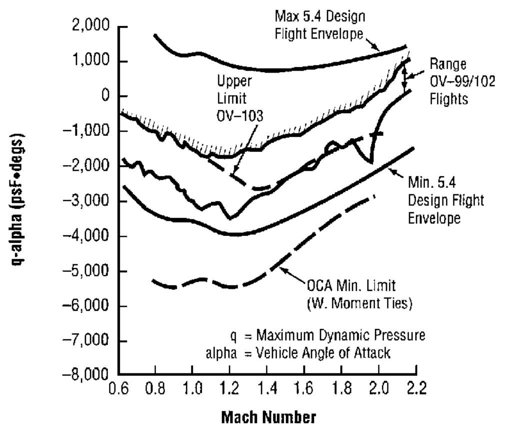

Fig. 9 Space Shuttle ascent flight envelope limits.

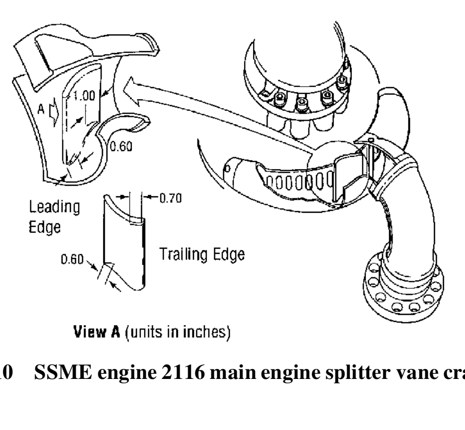

Fig. 10 SSME engine 2116 main engine splitter vane cracks.

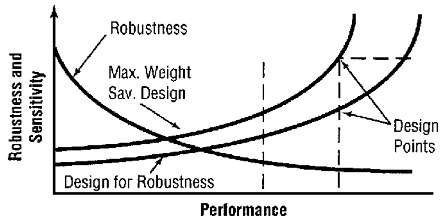

Fig. 11 Robustness curve, sensitivity vs performance.

trailing edge and open up the inlet gap between the splitters. The design change has eliminated the 4000-Hz oscillations. It took approximately 2 years of intense effort to understand and solve this problem.

b. Lox pump bearing wear. The upgraded replacement lox pump experienced excessive ball-bearing wear during development. The bearings were steel, but unlubricated because they ran in lox. Many changes were attempted, such as better bearing coolant flow, bearing clearances, guided cage changes, and bearing tilt, all to no avail. The bearing ball material was then changed to silicon nitride; the bearing race remained steel. No wear has occurred since changing the bearing ball material. In fact, pumps have operated up to 20,000 s with no problems evidenced. The bearing wear problem solution took approximately 1 year of dedicated effort of a combined government/contractor tiger team to solve the problem.

4. Summary of Space Shuttle Lessons

Several lessons and areas of focus resulted from the Space Shuttle program.

a. Performance requirements. Launch vehicle sensitivities to environments, manufacturing errors, design changes, etc., increase as the performance requirement increases. Inversely, the vehicle robustness decreases as performance increases. Figure 11 is a plot of this principle. The robustness curve is essentially the S-N fatigue curve of a material, whereas the sensitivity curve is its inverse. The SSTO dry weight vs dry-weight margin (shown previously on Fig. 1) also follows this sensitivity trend. The only way to relieve this curve for a given system is through technology jumps that increase performance and robustness simultaneously or through trades on weight for robustness. This is a design principle that must be understood and practiced if future projects are to be successful.

b. System focus. The basic conclusion is that the problems experienced have not, in general, been from poor discipline penetration technology, but lack of a strong systems engineering focus. The history of problems that have occurred supports this finding. Therefore, systems engineering focus is necessary for successful design. Each specialist must have the indepth technical knowledge of his or her specialty and an understanding of interfacing disciplines to maintain a system focus so that all of the interactions are robust and accounted for.

c. Loss management. It is inevitable that performance losses will occur; however, it is mandatory that a loss management program (risk mitigation) be established at the beginning of the program. The Space Shuttle had many unexpected losses that required many innovative and expensive recovery approaches. Losses occur because of unrealized technologies, unexpected environments, and systems integration/interactions. Sensitivity studies are at the heart of any loss management program.

d. Operations and cost. The last major lesson that is evident from the Shuttle is that cost and operations must be a fundamental part of the concept selection and the design equation, otherwise operations become very complex and cost escalates. Proper consideration of cost in design requires the establishment of metrics to judge the concepts and designs. It is also desirable to have determined all key parameters that drive cost and operations.

V. Solutions

Until this point, the discussion has focused on the idealized fundamentals of a launch vehicle, the nonideal effects experienced in Saturn and Shuttle programs, and some generalized lessons learned. Key issues in the design of future launch systems evolve from these lessons. This section deals with the key issues, how the design process relates to them, and a solution approach.

A. Design Process

The design concept selection/design process must start with the key issues by drivers. Figure 4 identifies the drivers as performance and programmatic with the appropriate issues under each. With this set of drivers and issues in the design process, Fig. 5 takes the basic requirements derived from the drivers and issues and attempts to find a solution. The solution, for example, can be a concept with given materials and manufacturing processes. Perturbations then are made in the materials and manufacturing blocks, and this system is evaluated. This process is continued until options run out, then a new concept is assumed and the cycle is repeated until an acceptable solution is found. It is emphasized that the design process cannot just consider the vehicle but also must include payloads, payload accommodation, and launch facilities. Design is not just a launch vehicle but a launch system. The design process, therefore, must focus on the issues that require solutions through the sensitivity and trade studies.

B. Basic Solution Approach

John London III in his dissertation says, "The high cost of space transportation can be attacked on several broad fronts, including launch vehicle design and manufacturing, launch operations, procurement streamlining, and program management."[4] London further states, "Air Force Command specifies four basic characteristics of any launch system: capability, reliability, affordability, and responsiveness. However, improving each of these characteristics starts with the vehicle design, and the concepts—to drastically reduce launch costs will have a positive benefit to all four." His study reinforces the premise stated above regarding the inherent coupling between the programmatic and the technical.

The solution to the design problem that considers cost and operations, as well as performance, must have a three-pronged approach. The first prong is to design in robustness that achieves the desired performance.[5] Robustness of this type then decouples the programmatic and performance drivers to a large degree, providing a satisfactory solution (see Fig. 4). If robustness cannot be achieved with performance, then the design has a much more difficult problem—shifting the emphasis to the other two prongs. Multistage vs single stage can be a part of this concept selection trade. Another approach to robustness is through technology jumps that produce the high performance with robustness. This design approach invariably drives up the developmental cost. Simplicity is another approach to robustness. Load path is one way of achieving simplicity.[6] The challenge to the designer is to keep open and pursue all the options.

The second prong emphasizes process improvement from operation, to manufacturing, to health monitoring, etc. Although this emphasis can result in improvements, it cannot by itself make big jumps in efficiency. Coupling process improvement with robustness can lead to major jumps in optimization.

The third prong deals with relaxing criteria and requirements. Criteria and standards for launch vehicles have evolved to be very conservative and restrictive, yet successful. Scrubbing these criteria and dealing with true reliability/probability can result in some improvement but must be pursued carefully so that safety is not compromised.[7]

If this three-pronged approach is to be achieved for future systems then, in general, it will be a balancing act between the three prongs. Far-out thinking is required to accomplish this task. For example, is it necessary to split the launch vehicle function into a cargo carrier and a manned-only carrier vs the current Shuttle concept of having both in the same vehicle? Is there a way of utilizing the Earth's atmosphere during that portion of the flight to save propellant tankage? Coupled with far-out thinking, major technology thrusts must occur.

VI. Technology Thrust

Propulsion, structures, TPS, avionics, health monitoring, and manufacturing are the major top-level thrusts for future space launch vehicles from a technical standpoint that must be driven if low-cost systems are to result. Specific examples include tripropellant or other advanced engines (aerospike); advanced materials and manufacturing (aluminum, lithium, and lightweight composite structures and tanks); and low-maintenance, reusable thermal insulation. The focus of new technology development must include not only performance issues but these other factors as well. The technical and programmatic issues are mass efficiency, propulsion efficiency, managing losses, operability, reusability, affordability, and safety/abort. Tables identifying the drivers of each major issue can be found elsewhere.[1]

VII. Conclusions

In summary, if future launch systems are to meet the current envisioned goals, a multiprogram design process must be followed. Design must optimize between programmatic and performance drivers. Design must emphasize technologies that build performance with robustness. A strong risk mitigation program must be followed with focus on the system and loss management. Finally, the goals can be met only if a robust low-cost manufacturing program is pursued.

The design of an STS is a very complex, system-focused, integrated activity. Many complex trades are required to determine the final suboptimal systems. Several fundamentals are key to this success and must be emphasized.

1) Weight and its management are a prime focus. Every extra pound that is sent to orbit costs energy. The paradox is that weight efficiencies can create other problems such as more complex dynamics, higher technology development cost. Weight management is necessary and must include materials weight growth, environment management, components management, etc.

2) As the performance requirements increase, the sensitivities to limit unknowns, uncertainties (environments), and manufacturing increase. Operations at these high-performance levels increase the amount of analyses and tests required for verification and safe operations, as well as more inspections and care in operations. Many times these high-performance levels force development of more exotic technologies.

3) Reusability drives the system hard. It requires first a mechanism to re-enter the Earth's atmosphere and land, including the thermal propulsion system for re-entry heating. The additional drivers are fatigue, wear, refurbishment, etc.

4) Engineering design starts with decomposition or compartmentalization to ensure efficiency in the design task. Components and subsystems then are reintegrated for the system integrity/verification. Most problems occur because of the breakdown of this process. Success in space transportation design requires a concerted system focus.

References

[1] Ryan, R. S., and Townsend, J. S., "Fundamentals and Issues in Launch Vehicles Design," AIAA Paper 96-1194, April 1996.

[2] Ryan, R. S., "A History of Aerospace Problems, Their Solutions, Their Lessons," NASA TP 3653, Sept. 1996.

[3] Pye, D., *The Nature of Design*, Reinhold, New York, 1969.

[4] London, J., III, "LEO on the Cheap," Air Univ. Press, Maxwell AFB, AL, 1994.

[5] Ryan, R. S., "The Role of Structural Dynamics in the Design and Operations of Space Systems: The History, the Lessons, the Technical Challenges of the Future," AIAA Paper 94-1726, April 1994.

[6] Pugh, S., "Total Design, Integrated Methods for Successful Product Engineering," Addison–Wesley, Reading, MA, 1991.

[7] Blair, J. C., and Ryan, R. S., "Role of Criteria in Design and Management of Space Systems," *Journal of Spacecraft and Rockets*, Vol. 31, No. 2, 1992, pp. 328, 329.

J. A. Martin
Associate Editor