[3] Thiruvengadam, A. and Preiser, H. S., "On testing materials for cavitation damage resistance," Hydronautics, Inc., TR233-3 (1964).

[4] Thiruvengadam, A., "A unified theory of cavitation damage," J. Basic Eng. **85**, 365–376 (September 1963).

[5] Thiruvengadam, A. and Waring, S., "Mechanical properties of metals and their cavitation damage resistance," Hydronautics, Inc., TR233-5 (August 1964).

[6] Thiruvengadam, A. Couchman, C., III, and Preiser, H. S., "Cavitation damage resistance of materials in liquid sodium," AIAA Preprint 64-363 (June 29–July 2, 1964).

Large Post-Saturn Launch Vehicles: Why? When? What?

WILLIAM G. HUBER*
NASA Marshall Space Flight Center, Huntsville, Ala.

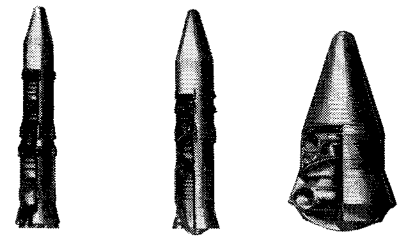
LAUNCH vehicles continue to be one of the major pacing items in the progress of space exploration. The largest launch vehicle now under development is the Saturn V; however, planning is underway for a post-Saturn vehicle that could greatly increase our capabilities for planetary exploration. This planning effort must first determine under what conditions a post-Saturn is justified. In other words, what is the size of space programs beyond which it is desirable to develop a launch vehicle larger than Saturn V? Then by having some indication of the future program, the questions of why we need a post-Saturn, when we need it, and what is the best concept can be answered.

Possible Vehicle Concepts

In order to proceed with the missions analysis work in parallel with the vehicle design effort, representative or baseline configurations were selected in each vehicle class (Fig. 1).† The selection of these baselines does not mean that they are best in their class, but only representative of the technology and availability offered by the class. It is then possible to make certain interclass comparisons with these baseline concepts.

Class I comprises expendable vehicles with propulsion systems currently under development. These include the M-1, the large solid, and the F-1 or an up-rated version of the F-1. The first-stage diameter is 65.5 ft; the second-stage diameter is 60 ft; and both stages use separate propellant tanks.

Class II represents advanced technology, principally in propulsion and recovery. New propulsion systems may use high chamber pressure, up to 3000 psia, and unconventional nozzles achieving some degree of altitude compensation. These advanced propulsion concepts are expected to improve the cost effectiveness of class II by 15%. The most desirable propulsion system is yet to be determined, since a great deal of experimental work is needed to verify the performance assumptions. Recovery and reuse of the first stage offers a 40% improvement in cost effectiveness for a launch rate of about 10/yr. Recovery of the second stage can offer further improvement of about 8%; however, the technical problems associated with recovery of items of this size from near-orbital velocities are serious, and at this time are not considered worth the potential gains for the class II.



CLASS	I	II	III	
HEIGHT (FT)	460	420	210	
			1 STAGE	2 STAGE
ENGINES	18 F-1A/3M-1	18/2 @ 1000K	18 @ 1000K	18/3 @ 1000K
THRUST (M-LB)	32.4	18	18	18
LAUNCH WT. (M-LB)	25.2	14.4	12-14.4	14.4
PAYLOAD (K-LB)	980	942	460-826	1250
PROP. MASS FRACTION	.920/.904	.897/.883	.928	.928/.922
GROSS WT./PAYLOAD	25.7	15.3	14.5	11.3

Fig. 1 Baseline post-Saturn vehicles.

Class III concepts utilize very advanced technology in the areas of propulsion, structures, and recovery from near orbital velocity. The ideal class III concept is a single-stage-to-orbit, fully recoverable vehicle. Several configurations have been studied. Analyses have shown the vehicle performance to be extremely sensitive to specific impulse and dry stage weight assumptions. Special features may include expendable tanks, solid and liquid JATO units, fluorine substitution, expendable second stages, and variable payload capability. A representative class III configuration was selected and is shown in Fig. 1. This is a basic single-stage-to-orbit, fully recoverable concept with 826,000-lb payload capability. For larger payload missions, an expendable second stage could give a 1,250,000-lb payload capability. The vehicle is about 120 ft in diameter and 220 ft high with second stage and payload.

Mission Analysis

In order to attempt to determine the role of a post-Saturn vehicle, four different mission models were developed, as indicated in Table 1. The missions assumed covered the orbital, lunar, and both the manned and unmanned planetary categories, and were constructed to represent possible follow-ons to existing programs for the 1970 to 1990 period. Table 1 indicates the magnitudes of the small and large programs without going into detail. Besides varying the number of missions, the size of each mission was varied in the manned planetary area. As an example, the initial manned Mars landing was assumed to be with a fleet of four ships for the large programs and only two ships for the small programs. Shown in Fig. 2 is the effect of schedule on the programs. For simplicity, the two extremes are shown. There is some variation of the selected missions with schedule. For example, a Mars capture mission was assumed for the small programs, but not for the large ones. For the small programs, the most ambitious mission selected was a manned Mars landing. Both the large programs included a Mars synodic base, and for the one on an optimistic schedule, another even more advanced mission was assumed.

Table 1 Assumed mission objectives

Program schedule	No. of orbital manned stations	Lunar base size[a]	No. of planetary missions	
			Unmanned	Manned
Large O^b	24	80	37	11
Large P^b	24	80	37	7
Small O	10	12	18	8
Small P	10	12	18	5

[a] Number of men.
[b] O = optmistic schedule, P = pessimistic schedule (see Fig. 2).

Presented as Preprint 64-279 at the 1st AIAA Annual Meeting, Washington, D. C., June 29–July 2, 1964; revision received August 28, 1964.

* Special Assistant to the Director, Future Projects Office. Member AIAA.

† A fourth class, not discussed herein, would comprise a chemical class I or II first stage with a nuclear second stage.

Table 2 Results

		Av. annual expenditures, $\times 10^9$		Average annual launch requirements				Equivalent orbital cost effectiveness, \$/lb			
				Schedule *O*		Schedule *P*		Schedule *O*		Schedule *P*	
Mix	Space program	Schedule *O*	Schedule *P*	S-V	P-S	S-V	P-S	DOC	TOC	DOC	TOC
1	Large	6.97	5.37	62.7	0	47.0	0	116.5	141.3	123.3	150.3
2		6.67	4.95	8.9	11.2	4.0	9.1	107.6	152.8	109.4	161.6
3		6.13	4.60	8.9	11.9	5.7	9.3	70.3	114.6	65.8	117.1
4		6.27	4.64	11.9	10.0	8.8	7.7	76.4	122.8	69.8	124.9
1	Small	3.33	3.40	20.4	0	15.0	0	100.2	129.4	156.6	195.6
2		3.88	3.62	4.4	3.9	3.0	2.9	129.1	208.5	133.3	238.3
3		3.78	3.51	4.6	4.0	3.2	3.0	89.0	168.7	91.6	190.8
4		3.76	3.55	6.1	3.3	4.9	2.3	95.1	182.9	107.0	217.5

Each set of assumed mission objectives was then accomplished with four different combinations of present and future launch vehicles. All vehicle mixes used the Saturn IB, Saturn V, and the 10-passenger re-usable orbital transport, which was assumed to be available in 1978 for the optimistic schedule programs and 1982 for the pessimistic. An improved capability of 300,000 lb to orbit was assumed for the Saturn V vehicle. The following are the vehicle mixes used: 1 = Saturn IB, Saturn V, 10-passenger; 2 = 1 plus post-Saturn class I; 3 = 1 plus post-Saturn class II; 4 = 1 plus post-Saturn class III.

With these combinations, it should be possible to determine the role of a post-Saturn launch vehicle in the over-all space program. For each mission objective, the spacecraft and the required space propulsion systems were identified, sized, and costed. A flight mode was then selected, either direct, as in the case of the lunar missions, or via orbital operations, as in the case of the most manned planetary missions. Orbital operations burden rates were assessed to those missions through orbit.[1,2] The sum of the direct flights, the mission flights to orbit, and the flights needed for orbital support gives the total launch requirements vs time. For expendable vehicles, the launch rate equals the number required; for re-usable vehicles, suitable re-use rates, turn-around time, and refurbishment cost were assumed. The program cost was determined by adding the spacecraft and space propulsion costs, the launch vehicle costs, and the orbital operations support costs. This was done on both a direct operating and total operating cost basis; the latter included development and facilities. Reliability assumptions then gave the mission yields (in pounds delivered, men, manyears, etc.). By combining these yields with the appropriate cost, the various indices of performance are determined.

Results

Results are shown in Table 2. The average yearly expenditure ranged from $3.3 × 10^9$ to $7.0 × 10^9$. Since the mission yield is relatively constant for each combination of

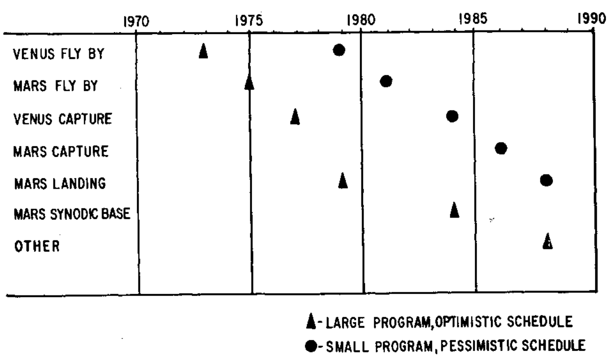


Fig. 2 Schedules for manned planetary missions.

launch vehicles, these costs represent a measure of effectiveness. For the large program on either the optimistic or pessimistic schedule, the combination of Saturn V and a class II post-Saturn (mix 3) appears to be the most efficient. The large program is too large for Saturn V only. The class III post-Saturn (mix 4) is closer to the minimum cost for the pessimistic schedule, since its late availability comes closer to matching the manned planetary mission schedule. The more the mission schedule is compressed, the more off-minimum the class III vehicle mix.

For the small programs, the Saturn V only (mix 1) represents minimum cost because of the reduced number of mission requirements. However, on the pessimistic schedule, the class II post-Saturn (mix 3) is not far from the minimum cost, since its availability is more compatible with the planetary missions. For the large programs, Saturn V only (mix 1) results in very high launch rates. The addition of any post-Saturn can greatly reduce these rates. In all cases the Saturn V requirements increase when the introduction of a post-Saturn capability is delayed from class I to class II to class III.

In an attempt to put all programs on an equal basis, an orbital equivalent cost effectiveness was calculated. This represents the cost effectiveness of the various transportation systems, if all launches were made to earth orbit. The results are shown in Table 2 in terms of both direct operating cost (DOC) and total operating cost (TOC), which includes facilities and development. The numbers shown represent the average of the combination of vehicles in each mix. For example, the effectiveness of mix 4 is an average of all the Saturn IB, Saturn V, and post-Saturn class III vehicles required to meet the assumed mission objectives. The 10-passenger re-usable orbital transport vehicle flights have been excluded in the calculation, since they are used to carry personnel and not cargo. For the large program, mix 3 (post-Saturn class II) is favored from both the direct- and total-cost effectiveness standpoint. The direct-cost effectiveness becomes greater for mixes 1 and 2 with a stretch-out in schedule, whereas it improves for mixes 3 and 4. This is because the stretched-out programs are more compatible with the later availability of the classes II and III post-Saturn. The Saturn V only is the most effective from a total-cost standpoint for the small program on an optimstic schedule, even though the post-Saturn class II mix is best on a direct-cost basis. If the small program schedule is stretched, then mix 3 becomes the most efficient.

Conclusions

The class II concept gave the best direct-cost effectiveness under all programs considered and the best total-cost effectiveness in all but the small program on an optimistic schedule. This class offers significant improvement over Saturn V and at a time that is compatible with the planetary missions. The class III vehicle did not produce favorable results because of its late availability. Programs with more pessimistic schedules would permit better utilization of class III, and would therefore produce better cost effectiveness.

It can be concluded that, with a relatively large space program, there is a role for post-Saturn. The exact size and composition of the program that economically justifies a large launch vehicle remain to be determined. The exact vehicle concept eventually selected will depend on the state of the art at the time of development initiation, but will probably include some degree of advanced propulsion and some form of recovery and re-use.

References

[1] Ehricke, K. A., "Economy of large launch vehicles including orbital labor cost," J. Spacecraft Rockets **1**, 611–619 (1964).

[2] Koelle, H. H., "Orbital burden rates for manned space missions," J. Spacecraft Rockets **1**, 620–626 (1964).

Roll-Rate Lag of Rockets Accelerating in the Upper Atmosphere

W. D. Parsons*

Space-General Corporation, El Monte, Calif.

Nomenclature

A = aerodynamic reference area, exposed fin area, ft^2
B = aerodynamic reference length, fin semispan, measured from vehicle centerline, ft
C_l = rolling moment coefficient, rolling moment$/qAB$
C_{l_δ} = derivative of rolling moment with fin cant δ, rad^{-1}
C_{l_p} = damping derivative of rolling moment with fin tip helix angle pB/V, rad^{-1}
D = dimensionless fin cant driving parameter, rad^{-1}
= $C_{l_\delta}\delta\rho_{bo}AB^2/2\beta I \cos\gamma_{bo}$ (const)
Ei = exponential integral, $Ei\,(-x) = \int_\infty^x \left(\frac{e^{-x}}{x}\right)(dx)$, Eq. (10)
I = roll moment of inertia, slug-ft^2
$-K$ = $dV/dln\rho$, fps
M = Mach number
p = roll rate, rad/sec
q = dynamic pressure, $(\rho/2)V^2$, psf
Q = dimensionless roll damping parameter, rad^{-1}
= $-C_{l_p}\rho_{bo}AB^2/2\beta I \cos\gamma_{bo}$ (const)
t = time, sec
V = trajectory velocity, fps
β = exponential atmosphere scale factor; bove h = 35,000 ft
$1/\beta$ = 21,000 ft
γ = flight path angle, measured from the vertical, deg
Δ = increment
δ = fin cant, average of all fins, rad
ξ = fin tip helix angle, pB/V, rad
ρ = air density, slug/ft^3

Subscripts

bo = at burnout
ss = steady state
$(\ )'$ = derivative with respect to ρ/ρ_{bo}
$(\dot{\ })$ = derivative with respect to time

Introduction

THE results of a 6-D machine program, reported in Ref. 1, indicated that the estimated burnout roll rate for the Aerobee 350, based on the steady-state assumption, i.e.,

$$pB/V = C_{l_\delta}\delta/-C_{l_p}$$

was too high by a factor of 1.24. This does not include the effects of induced rolling moment, a phenomenon that is not treated herein. The roll-rate lag could cause significant effects on certain payloads that demand close control of the postburnout coning angle. Therefore, it becomes important to understand the causes of the burnout roll-rate lag so that compensating increments of fin cant may be prescribed to restore the roll rate to its desired level. This additional roll-rate requirement will accelerate the occurrence of pitch-roll resonance so that it will occur at a lower altitude and Mach number where the vehicle loads may be greater.

Received June 29, 1964; revision received August 3, 1964.

*Senior Engineer, Aerodynamics Department. Member AIAA.

Method of Analysis

The rolling moment differential equation of motion is

$$I\dot{p} = qAB[C_{l_\delta}\delta + C_{l_p}(pB/V)] \qquad (1)$$

This equation may be transformed into an easily solved differential equation by merely changing the independent variable from t to ρ. This is accomplished with the rate of climb relation

$$dh/dt = V\cos\gamma \qquad (2)$$

and the exponential atmosphere equation

$$\rho/\rho_{\text{sea level}} = e^{-\beta h} \qquad (3)$$

Equation (3) is differentiated and substituted into (2), yielding

$$d(p)/dt = -\beta V(\rho/\rho_{bo})\cos\gamma d(p)/d\rho/\rho_{bo} \qquad (4)$$

Applying Eq. (4) to (1) gives

$$p'(B/V) = [-AB^2\rho_{bo}/2\beta I\cos\gamma][C_{l_\delta}\delta + C_{l_p}(pB/V)] \qquad (5)$$

Differentiating pB/V gives

$$(pB/V)' = p'(B/V) - (pB/V)(V/'V) \qquad (6)$$

Substitution of Eq. (6) in (5) produces the transformed differential equation of motion

$$\xi' + (V'/V - Q)\xi = -D \qquad (7)$$

which, assuming only that Q and D are constant, has the immediate integral solution

$$\xi = \left[\int_{\rho_{ss}/\rho_{bo}}^{\rho/\rho_{bo}} \left(\frac{V}{V}\right)_{bo} e - Q\rho/\rho_{bo}(-Q)d\left(\frac{\rho}{\rho_{bo}}\right)\right]\xi_{ss}\frac{eQ\rho/\rho_{bo}}{V/V_{bo}} \qquad (8)$$

where, by definition, $\xi_{ss} \equiv C_{l_\delta}\delta/(-C_{l_p}) = D/Q$.

Integration requires that the velocity be described as a function of the density. For sounding rockets accelerating swiftly in the upper atmosphere, a close description is given by $dV/dln\rho = -K$ (a constant). It can be shown that many types of rockets do indeed exhibit this linear, semilogarithmic variation, which exists for several thousands of feet below the burnout altitude, the region where nearly all of the dynamic roll lag occurs. At the lower altitudes, where this simple relation begins to fail, the steady-state analysis will apply so that the dynamic solution is not required. Although other analytic forms could be constructed, this one has an additional advantage in that it leads to a simple integration when it is employed in Eq. (8). Upon integrating it and satisfying the important burnout condition, the result is

$$V/V_{bo} = 1 - (K/V_{bo})\ln(\rho/\rho_{bo}) \qquad (9)$$

Use of Eq. (9) in (8) permits integration in terms of the exponential integral. The integration is begun at the steady-state condition so that the initial steady-state conditions are satisfied:

$$\xi/\xi_{ss} = 1 + \frac{eQ\rho/\rho_{bo}}{(V_{bo}/K) - \ln(\rho/\rho_{bo})}\int_{Q\rho_{ss}/\rho_{bo}}^{Q\rho/\rho_{bo}} \frac{e^{-x}}{x}\,dx \qquad (10)$$

The values of this integral are tabulated, for example, in Ref. 3, as $Ei(-x)$. It is noted that $Ei(-x) < 0.01$ for values