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PARAMETRIC MISSION ANALYSIS

by

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1. INTRODUCTION

This memorandum represents a preliminary publication of the parametric mission analysis work performed simultaneously in partial fulfillment of study obligations for Post-Nova III, EMPIRE Follow-On and the Planetary Transportation Model Studies. A more detailed report follows.

2. METHOD OF APPROACH (Fig. 1, 2)

The method of approach is outlined in the following steps:

(1) Matrix of Principal Lunar Mission Profiles (Definition of Mission Profiles by the principal maneuvers involved) (Tab. 1)

(2) Numerical Definition of Principal Lunar Mission Maneuvers (Tab. 2a)

(3) Mission Definitions in Terms of total velocities (Tab. 2a)

(4) Definition of Principal Mission Objectives and Correlation of Principal Mission Objectives with Mission Profiles for Lunar Activities (Fig. 3)

(5) Vehicle Data for Lunar Missions (Tab. 2b, 2c)

(6) Payload Data for Lunar Missions (Tab. 2d, 2e)

(7) Maneuver Reliability (Tab. 2f)

(8) Mission Reliability (Tab. 2f)

(9) Variation of Net Payload Weight with Orbital Departure Weight and Ideal Mission Velocity (Fig. 4)

(10) Variation of Net Payload Weight to Lunar Surface with Orbit Departure Weight for Missions L-5 and L-6 (Fig. 5)

(11) Survey of Planetary Mission Objectives (Tab. 3)

(12) Planetary Mission Groups (Tab. 4)

(13) Matrix of Principal Planetary Mission Profiles (Tab. 5)

(14) Hyperbolic Excess Velocities Used for Definition of Principal Planetary Mission Maneuvers (Tab. 6)

(15) Definition of Principal Planetary Mission Maneuvers (Tab. 7)

(16) Correlation of Principal Planetary Mission Objectives with Mission Profiles (Tab. 8)

(17) Vehicle Data

(18) Payload Fractions for Given Interplanetary Missions and Interplanetary Vehicles (Tab. 9)

(19) Determination of Life Support System Weights

(20) Determination of the Weight of Spare Parts for the Planetary Missions

(21) Determination of the Weight of the Destination Payload

(22) Correlation between Missions and Earth Launch Vehicles (Tab. 10)

(23) Launch Vehicle Evaluation (Fig. 6)

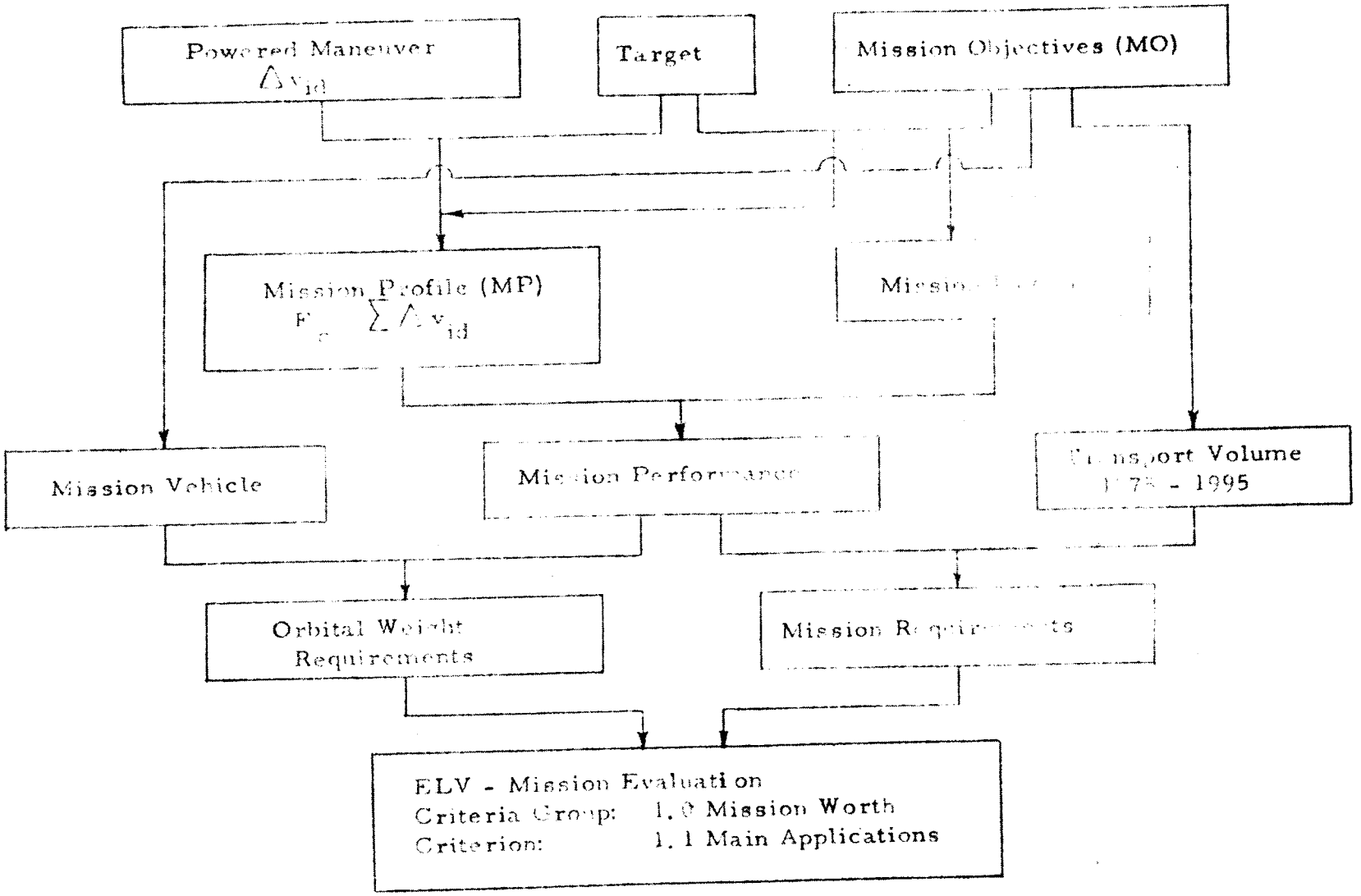


Fig. 1 Methodology of ELV - Mission Requirement Evaluation.

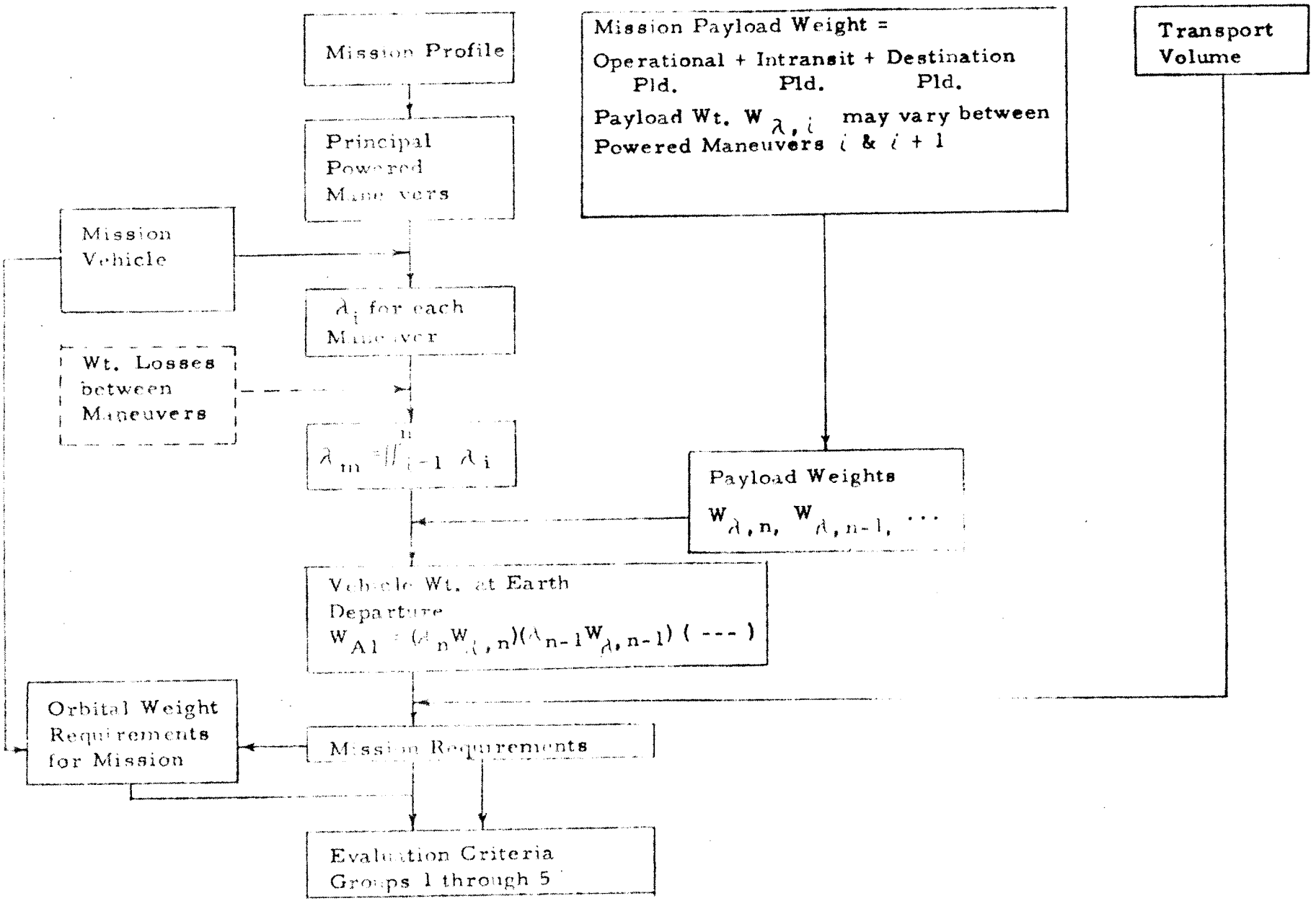


Fig. 2 The Role of Mission Payload Weight Fraction in Mission Requirement Evaluation.

Tab. 1 Matrix of Principal Lunar Mission Profiles

Definition of Mission Profiles / Principal Maneuvers		1 Earth to orbit	2 Inject. into cislunar transf. orbit	3 Direct Lunar landing	4 Establishment of circular Lunar orbit	5 De-orbit & Lunar landing	6 Ascent into Lunar orb.	7 Inject. into transf. orb. to Earth	8 Establishment of circular Earth orbit
No.	Mission Profile Designation								
L-1	Direct Lunar delivery	(*)[1]	*	*					
L-2	Lunar delivery via Lunar parking orbit	(*)	*		*	*			
L-3	As L-2 and return via parabolic entry	(*)	*		*	*	*	*	
L-4	As L-2 and return to Earth parking orbit	(*)	*		*	*	*	*	*
L-5	Interorbital shuttle & delivery from Lunar parking orbit		*		*	*		*	
L-6	Interorbital shuttle & Lunar surface to orbit shuttle		*		*	*	*	*	*

1) (*) Alternate mission profile

Tab. 2a **LUNAR MISSION DATA**

I. Principal Maneuvers		Δv (ft/sec)
1.	Earth-to-orbit (~ 200 n. mi. altitude)	30,000
2.	Injection into 60-hr transfer orbit	11,000
3.	Direct Lunar landing	9,000
4.	Establish circular Lunar orbit (~ 20 n. mi. altitude)	3,200
5.	De-orbit and Lunar landing (continuous thrust descent)	6,000
6.	Ascent into 20 n. mi. Lunar orbit ($F/W_o = 1\ g_{☾}$)	5,950
7.	Injection into 60-hr transfer orbit to Earth	3,200
8.	Establish circular Earth orbit (~ 200 n. mi.)	11,000

II. Mission Definitions		Δv_{tot} (ft/sec)
L-1	= 2 + 3 = direct Lunar delivery	20,000
L-2	= 2 + 4 + 5 = Lunar delivery via Lunar parking orbit	20,200
L-3	= 2 + 4 + 5 + 6 + 7 = Lunar delivery as L-2 and Earth return via parabolic entry	29,320
L-4	= 2 + 4 + 5 + 6 + 7 + 8 = Lunar delivery as L-2 and return to Earth parking orbit	40,350
L-5	= (2 + 4 + 7 + 8) + (5) = Interorbital shuttle and delivery from Lunar parking orbit	(28,400) (6,000)
L-6	= (2 + 4 + 7 + 8) + (5 + 6) = Interorbital shuttle and Lunar surface-to-orbit shuttle	(28,400) (11,950)

Tab. 2b LUNAR MISSION DATA (Continued)

III Vehicle Data (Subscripts refer to the maneuvers listed sub. I.)

Mission	L-1			L-2			L-3			L-4		
Orbital Launch Wt. (10^3 lb)	250	500	750	250	500	750	250	500	750	250	500	750
I_{sp}(sec)	450	450	450	450	450	450	450	450	450	450	450	450
Mass ratio, μ_1	2.14	2.14	2.14	2.14	2.14	2.14	2.14	2.14	2.14	2.14	2.14	2.14
Propellant fraction, Λ_1	.532	.532	.532	.532	.532	.532	.532	.532	.532	.532	.532	.532
Mass fraction, x_1	.86	.88	.90	.86	.88	.90	.86	.88	.90	.86	.88	.90
Load fraction, λ_1	.38	.394	.408	.38	.394	.408	.38	.394	.408	.38	.394	.408
Mass ratio, μ_1	1.83	1.83	1.83									
Propellant fraction, Λ_1	.453	.453	.453									
Mass fraction, x_1	.85	.87	.89									
Load fraction, λ_1	.467	.479	.49									
Mass ratio, $\mu_{4,1}$				1.89	1.89	1.89	1.89	1.89	1.89	1.89	1.89	1.89
Propellant fraction, $\Lambda_{4,1}$				.47	.47	.47	.47	.47	.47	.47	.47	.47
Mass fraction, $x_{4,1}$				.85	.87	.89	.85	.87	.89	.85	.87	.89
Load fraction, $\lambda_{4,1}$				.446	.46	.471	.446	.46	.471	.446	.46	.471
Mass ratio, $\mu_{6,1}$							1.88	1.88	1.88	1.88	1.88	1.88
Propellant fraction, $\Lambda_{6,1}$							.468	.468	.468	.468	.468	.468
Mass fraction, $x_{6,1}$							.85	.87	.89	.85	.87	.89
Load fraction, $\lambda_{6,1}$							.45	.462	.474	.45	.462	.474
Mass ratio, μ_6										2.14	2.14	2.14
Propellant fraction, Λ_6										.532	.532	.532
Mass fraction, x_6										.86	.88	[illegible]
Load fraction, λ_6										.38	.394	.408
I_{sp}(sec)	765	765	765	765	765	765	765	765	765	765	765	765
Mass ratio, $\mu_{1,1}$	2.33	2.33	2.33									
Propellant fraction, $\Lambda_{1,1}$	.57	.57	.57									
Mass fraction, $x_{1,1}$	.82	.84	.86									
Load fraction, $\lambda_{1,1}$	.305	.323	.347									
Mass ratio, $\mu_{1,4,1}$				2.35	2.35	2.35	2.35	2.35	2.35	2.35	2.35	[illegible]
Propellant fraction, $\Lambda_{1,4,1}$				.575	.575	.575	.575	.575	.575	.575	.575	.575
Mass fraction, $x_{1,4,1}$				.82	.84	.86	.82	.84	.86	.82	.84	.86
Load fraction, $\lambda_{1,4,1}$				.30	.316	.332	.30	.316	.332	.30	.316	.332
I_{sp}(sec)							450	450	450			
Mass ratio, $\mu_{6,1}$							1.88	1.88	1.88			
Propellant fraction, $\Lambda_{6,1}$							.468	.468	.468			
Mass fraction, $x_{6,1}$							.85	.87	.89			
Load fraction, $\lambda_{6,1}$							.45	.462	.474			
Mass ratio, $\mu_{6,1,6}$										2.35	2.35	[illegible]
Propellant fraction, $\Lambda_{6,1,6}$										.575	.575	[illegible]
Mass fraction, $x_{6,1,6}$										.82	.84	[illegible]
Load fraction, $\lambda_{6,1,6}$										.30	.316	[illegible]

Tab. 2c LUNAR MISSION DATA (Continued)

III. Vehicle Data (Concluded)

Mission	L-5			L-5			L-5		
Orbital launch wt. (10^3 lb)	250	500	750	250	500	750	250	500	750
I_{sp}(sec)	450	450	450	765	765	765	825	825	825
Mass ratio, $\mu_{2,1,2,1}$	7.14	7.14	7.14	3.34	3.34	3.34	2.918	2.918	2.918
Propellant fraction, $\Lambda_{2,1,2,1}$	.86	.86	.86	.70	.70	.70	.656	.656	.656
Mass fraction, $x_{2,1,2,1}$	.88	.90	.92	.83	.85	.87	.84	.86	.88
Load fraction, $\lambda_{2,1,2,1}$	.023	.044	.065	.156	.176	.195	.218	.236	.254
I_{sp}(sec)	400	400	400	400	400	400	400	400	400
Mass ratio, μ_2	1.6	1.6	1.6	1.6	1.6	1.6	1.6	1.6	1.6
Propellant fraction, Λ_2	.375	.375	.375	.375	.375	.375	.375	.375	.375
Mass fraction, x_2	.89	.90	.91	.90	.91	.92	.91	.92	.93
Load fraction, λ_2	.578	.583	.588	.583	.588	.593	.588	.593	.597

Mission	L-6			L-6			L-6		
Orbital launch wt. (10^3 lb)	250	500	750	250	500	750	250	500	750
I_{sp}(sec)	450	450	450	765	765	765	825	825	825
Load fraction $\lambda_{2,1,2,1}$	.023	.044	.065	.156	.176	.195	.218	.236	.254
I_{sp}(sec)	450	450	450	450	450	450	450	450	450
Mass ratio, $\mu_{3,1}$	2.29	2.29	2.29	2.29	2.29	2.29	2.29	2.29	2.29
Propellant fraction, $\Lambda_{3,1}$	.563	.563	.563	.563	.563	.563	.563	.563	.563
Mass fraction, $x_{3,1}$	.84	.86	.88	.85	.87	.89	.86	.88	.90
Load fraction, $\lambda_{3,1}$	.33	.345	.36	.337	.352	.367	.345	.36	.374

Tab. 2d LUNAR MISSION DATA (Continued)

IV. Payload Data

Mission	L-1			L-2			L-3			L-4		
Orbital launch wt. (10^3 lb)	250	500	750	250	500	750	250	500	750	250	500	750
Mission load fraction: (All maneuvers: I_{sp} = 450 sec)												
$\lambda = \lambda_2 \; \lambda_3$	.178	.188	.20									
$\lambda = \lambda_2 \; \lambda_{4,5}$				.169	.181	.192						
$\lambda = \lambda_2 \; \lambda_{4,5} \lambda_{6,7}$							.0764	.0839	.091			
$\lambda = \lambda_2 \; \lambda_{4,5} \lambda_{6,7} \lambda_8$										.028	.033	.0372
Net payload weight (10^3 lb)	37.8	79	127.5	37	77	122	14.9	32.6	53.2	5.3	11.9	20
Mission load fraction: (Maneuvers (2, 3, 4, 5): I_{sp} = 765 sec Maneuvers (6, 7): I_{sp} = 450 sec)												
$\lambda = \lambda_{2,3}$	.305	.323	.347									
$\lambda = \lambda_{2,4,5}$				.30	.316	.332						
$\lambda = \lambda_{2,4,5} \lambda_{6,7}$							.15	.146	.157			
$\lambda = \lambda_{2,4,5} \lambda_{6,7,8}$										.09	.10	.11
Net payload weight (10^3 lb)	68.5	145	234	64	135	212.5	24.5	53	85.6	16.4	36.5	60.2

Net pld. wt. ~85 to 90% of gross pld. wt. λ • Orb. Launch Wt.

Tab. 2e LUNAR MISSION DATA (Continued)

IV. Payload Data (Concluded)

Mission	L-5			L-5			L-5		
Orbital Launch wt. (10^3 lb)	250	500	750	250	500	750	250	500	750
I_{sp} = 450/400 sec:									
$\lambda = \lambda_{2,4,5,6}\lambda_5$	.0133	.0256	.0382						
Net payload wt. (10^3 lb)	2.2	7.5	19						
I_{sp} = 765/400 sec:									
$\lambda = \lambda_{2,4,5,6}\lambda_5$				.0916	.1035	.1155			
Net payload wt. (10^3 lb)				17.1	38.5	64.6			
I_{sp} = 825/400 sec:									
$\lambda = \lambda_{2,4,5,6}\lambda_5$							.1281	.14	.152
Net payload wt. (10^3 lb)							[illegible]	78.2	113

Mission	L-6			L-6			L-6		
Orbital Launch wt. (10^3 lb)	250	500	750	250	500	750	250	500	750
I_{sp} = 450/400 sec:									
$\lambda = \lambda_{2,4,5,6}\lambda_{5,6}$	.0076	.0152	.0234						
Net payload wt. (10^3 lb)	1.2	4.7	10.8						
I_{sp} = 765/400 sec:									
$\lambda = \lambda_{2,4,5,6}\lambda_{5,6}$				.0526	.062	.0733			
Net payload wt. (10^3 lb)				9.05	21.4	37.9			
I_{sp} = 825/400 sec:									
$\lambda = \lambda_{2,4,5,6}\lambda_{5,6}$							.0752	.085	.095
Net payload wt. (10^3 lb)							12.9	29.3	49.6

Tab 7 Definition of Principal Planetary Mission Maneuvers*)

	1 Earth to orbit	2 Earth escape ($r^* = 1.1$)	3 Planet capture	4 Hyperbol orb to surface	5 Capture orb to surface	6 Capt orb to moon & land'g	7 Surface (planet or moon) to orbit	8 Planet escape	9 Earth capt	10 Hyperb orbit to Earth surface	11 Capt orb to Earth surface	12 Moon surf to orbit	13 Transf from capt orbit to Moon	14 Direct Moon land'g	15 Moon capt	16 Moon capt orb to Moon surface
	30,000			Depends on hyp approach velocity $\Delta v = 1.7\sqrt{v_\infty^2 + 2K/r}$	♀ 11,300 ♂ 14,000 ♃ 13,800		♀ 11,300 ♂ 11,400 ♃ 13,800		Depends on capt maneuv Given below is $(v_E)_{100\,km}$ (EMOS)	Depends upon max all hyp entry vel $v = (v_E^*)_{100\,km} - (v_E^*)_{all}$	1000 (de-orb maneuver)	5950 (ascent into circ orbit @ 20 n mi altitude)	2000	9000	4000 (@ 20 n mi capt orb altitude)	6000 (cont thrust descent)
Escape & Capture Velocities (km/sec)																
[illegible]		11.22[2)]														
[illegible]		23.4 25.5	29.2 48.3					23.8 38.4	0.50 0.512							
[illegible]		12.8														
[illegible]		12.5 15.2	10.4 13													
[illegible]		15.2	14.7					23.2	0.48							
[illegible]		13.9														
[illegible]		13.9	10.7													
[illegible]		14.4[1)] 14.8[2)]	9.3[1)] 14.8[2)]					20.4[1)] 21.8[2)]	0.669[1)] 0.749[2)]							
[illegible]		12.8	7.7					10.2	0.392							
[illegible]		8.3 14 46 22.2														
[illegible]		33.2 46	41 56.8													
[illegible]		23.4[1)] 23.6[2)] 31.6[1)] 34[2)] 50.4[1)] 57.6[2)]	41[1)] 25[2)] 42.7[1)] 56[2)] 84.4[1)] 94.3[2)]					35.9[1)] 36.7[2)] 31.6 31.6 48.4 53.3	1.15[1)] 1.645[2)] 1.255 1.10 1.10 1.255							
[illegible]		81 55 37.2 [illegible]														
[illegible]		81 55 37.5 30	133 102 54.9 30.4													

*) Values of Maneuvers 5 & 7 taken as 1.17 times circular velocity at surface to account for gravitational losses, except for Venus, where the dense atmosphere leads to a lower value for landing and [illegible] for ascent. Escape and capture velocities are excess over circular velocities at the distance r_2^* indicated, multiplied by 1.055 to account for gravitational losses. They are based on the hyperbolic excess velocities of Tab [illegible]

1) favorable mission years. 2) unfavorable mission years

Tab. 2f LUNAR MISSION DATA (Concluded)

V. Maneuver Reliability

Maneuver	2			3		4	5			6		7	8	Rendez-vous (L-6)
							De-orbit	Lunar Landg.						
I_{sp} (sec)	450	765	825	450	765			400/450	765	400/450	765			
1970	.90	.85	-	.85	.80	As 2	As 2	.85	.80	.90	.85	As 2	As 2	.90
1975	.95	.90	.85	.90	.85			.90	.85	.96	.92			.97
1980	.99	.98	.95	.97	.95			.95	.95	.99	.97			.98

VI. Mission Reliability

Mission	L-1			L-2			L-3			L-4		
Year	'70	'75	'80	'70	'75	'80	'70	'75	'80	'70	'75	'80
I_{sp} - 450 sec	.765	.855	.96	.62	.77	.94	.50	.70	.92	.45	.66	.91
765 sec	.680	.765	.93	.49	.62	.89				.35	.51	.85
765/450 sec							.45	.63	.88			

Mission	L-5			L-6		
Year	'70	'75	'80	'70	'75	'80
I_{sp} - 450/400 sec	.45	.70	.90	.36	.65	.87
765/400 sec	.40	.56	.87	.32	.52	.84
825/400 sec	-	.40	.77	-	.37	.75

Lunar Mission Objectives	Mission Profiles					
	L-1	L-2	L-3	L-4	L-5	L-6
1.0 Exploration from Landers			*			
2.0 Earth-Dependent Lunar Bases						
2.1 Small Base ($\leq$ 10 persons)	*	*				
2.2 Medium Base (10 - 100 persons)	*	*		*	*	*
3.0 Partly Self-Sufficient Lunar Bases						
3.1 Medium Base (10 - 100 persons)	*	*		*	*	*
3.2 Large Base ($>$ 100 persons)	*	*		*	*	*

Fig. 3 Correlation between Principal Mission Objectives and Mission Profiles for Lunar Activities.

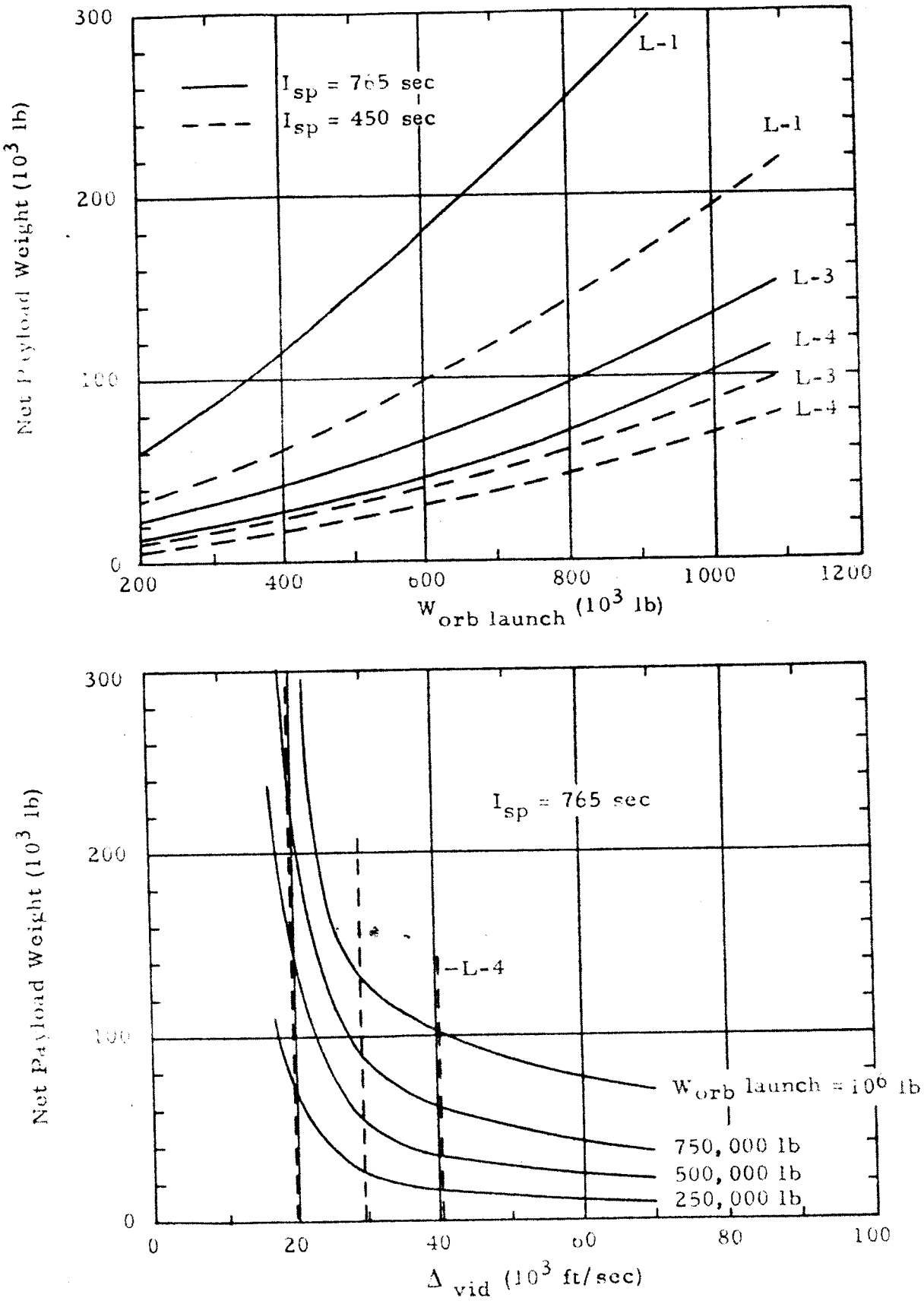


Fig. 4 Variation of Net Payload Weight with Orbital Departure Weight and Ideal Mission Velocity.

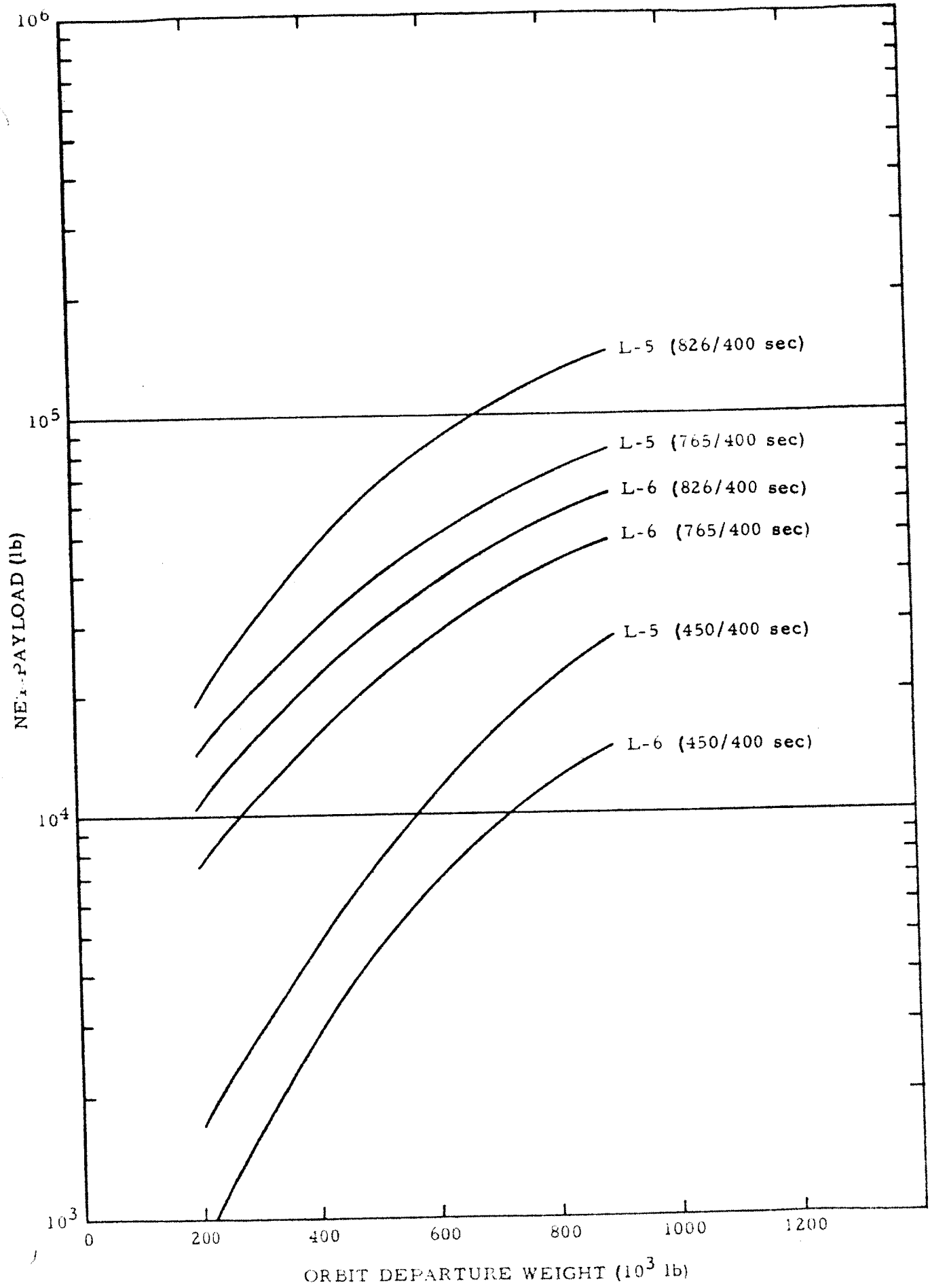


Fig. 5 Variation of Net Payload to Lunar Surface with Orbit Departure Weight for Missions L-5 and L-6.

Tab. 3 Survey of Mission Objectives

Objective								
1. Exploration								
1.1 Orbital	*	*	*	*	*	*	*	*
1.2 Atmospheric entry without surface landing		(*)[1]		*	*	*	*	
1.3 Surface landing	*	(*)[2]	*					*
1.4 Landing on satellite(s)			*	*	*	(*)	*	
1.5 Landing on asteroids (examples for scientifically interesting bodies: Icarus (1566), Hidalgo (944), Adonis)								
2. Temporary Base for Exploration and Research								
2.1 Planet surface, single base	*		*					(*)
2.2 Planet surface, multiple base net			*					
2.3 Suitable satellite(s)			*	*	*	(*)	*	
2.4 Asteroid (e.g. Icarus, Adonis, Geographos)								
3. Permanent Base								
3.1 Planetary surface, extended extraterrestrial life research and planet conditioning			*					
3.2 Planetary surface, raw material exploitation	?		?					
3.3 Suitable satellite(s)			*	*	*	(*)	*	
4. Refueling Base								
4.1 Planet surface			*					*
4.2 Suitable satellite(s)				*	*	(*)	*	

[1] Worthwhileness questionable; [2] Further data from instrumented Landers may eliminate justifications for this effort.

Tab. 4 Planetary Mission Groups

Mission Group	1-Way	Round-trip
1.0 Fly-By	*	*
2.0 Capture	*	*
3.0 Landing	*	*
4.0 Planetary moon landing	*	*

TAB 5 Matrix of Principal Planetary Mission Profiles

No.	Mission Profile Designation / Maneuvers	1 Earth to orb	2 Inject into hyp Earth dep orbit	3 Planet capture	4 Hyperbolic orbit to place	5 Capture orb to surface	6 Capture orb to Moon & landing	7 Surface (planet or moon) to orbit	8 Inject into hyp planet dep orb	9 Earth Capture	10 Hyp orb to Earth surface	11 Capt orb to Earth surface	12 Moon to orbit	13 Capture orb to Moon	14 Direct Moon landing	15 Moon capture	16 Moon capture orbit to Moon surface
1	[illegible]								*		*						
2	[illegible]								*								
3	[illegible]								*	*							
4	[illegible]								*	*							
5	[illegible]																
6	[illegible]																
7	[illegible]			*													
8	[illegible]																
9	[illegible]								*		*						
10	[illegible]																
11	[illegible]			*					*								
12	[illegible]								*		*						
13	[illegible]								*	*							
14	[illegible]																
15	[illegible]								*	*				*	*	(*)[1]	(*)
16	[illegible]								*	*				*	*	(*)	(*)
17	[illegible]								*	*			*	*	*	(*)	(*)
18	[illegible]			*													
19	[illegible]			(*)	*	(*)											
20	[illegible]			(*)	*	(*)											
21	[illegible]			(*)	*	(*)							*				
22	[illegible]			(*)	*	(*)		*	*		*						
23	[illegible]			(*)	*	(*)		*	*		*						
24	[illegible]	*		(*)	*	(*)		*	*	*							
25	[illegible]			(*)	*	(*)		*	*	*					*	(*)	(*)
26	[illegible]			(*)	*	(*)		*	*	*					*	(*)	(*)
27	[illegible]			(*)	*	(*)		*	*	*					*	(*)	(*)
28	[illegible]			(*)	*	(*)		*	*	*			*		*	(*)	(*)
29	[illegible]			*			*										
30	[illegible]			*			*										
31	[illegible]	*		*			*										
32	[illegible]			*			*	*	*								
33	[illegible]			*			*	*	*								
34	[illegible]			*			*	*									
35	[illegible]			*			*			*							
36	[illegible]			*			*			*							
37	[illegible]			*			*			*					*	(*)	(*)
38	[illegible]			*			*			*					*	(*)	(*)
39	[illegible]	*		*			*			*			*	*	*	(*)	(*)
40	[illegible]			*			*			*							

[1] (*) alternate mission profile

Tab. 6 Hyperbolic Excess Velocities Used for Definition of Principal Planetary Mission Maneuvers

All velocities are given so as to permit 20 to 40 days departure window at the transfer periods given

Mission	Earth Dep. (EMOS)	Planet Arr. (EMOS)	Planet Dep. (EMOS)	Earth Arr. (EMOS)
Me-1.0 (1-way) ($70 \leq T_1 \leq 100$ d)	.25[1] - .30[2]			
Me-2.0 (round-trip) ($200 \leq T \leq 300$ d)	.32[1]	.35	.30	.33
	.35[2]	.55	.45	.35
V-1.0 (1-way) ($108 \leq T_1 \leq 132$ d)	.12			
V-2.0 (1-way) ($96 \leq T_1 \leq 102$ d)	.11[1]	.10		
	.18[2]	.145		
V-2.0 (round-trip) ($360 \leq T \leq 400$ d)	.18	.18	.32	.30
Ma-1.0 (1-way) ($140 \leq T_1 \leq 180$ d)	.15			
Ma-2.0 (1-way) ($160 \leq T_1 \leq 180$ d)	.15	.14		
Ma-2.0 (round-trip) ($440 \leq T \leq 460$ d)	.16[1]	.13	.26	.55
	.17[2]	.20	.28	.65
Ma-2.0 (round-trip) ($800 \leq T \leq 1000$ d)	.12	.10	.14	.12
J-1.0 (1-way) ($190^{1)} \leq T_1 \leq 220^{2)}$ d)	1.0			
220 250	.80			
280 310	.60			
430 460	.40			
J-2.0 (1-way) ($400 \leq T_1 \leq 460$ d)	.45	.55		
300 340	.60	.80		
J-2.0 (round-trip) ($1100 \leq T \leq 1200$ d)	.32[1]	.29	.47	1.04
	.33[2]	.34	.48	1.16
($850 \leq T \leq 950$ d)	.43[1]	.62	.42	1.17
	.46[2]	.69	.42	1.03
($550 \leq T \leq 600$ d)	.65[1]	.98	.62	1.03
	.73[2]	1.08	.66	1.17
S-1.0 (1-way) ($360^{1)} \leq T_1 \leq 400^{2)}$)	.96			
(480 520)	.70			
(720 760)	.50			
(1020 1060)	.41			
S-2.0 (1-way) ($360 \leq T_1 \leq 400$)	.96	1.41		
(480 520)	.70	1.1		
(720 760)	.49	.63		
(1020 1060)	.41	.40		

[1] Favorable mission years

[2] Unfavorable mission years

Tab. 8 Correlation of Principal Planetary Mi

Principal Mission Obj.	Target Mission Profile MP-	Mercury 1.0	Mercury 2.0	Mercury 3.0	Mercury 4.0	Venus 1.0	Venus 2.0	Venus 3.0	Venus 4.0	Mars 1.0	Mars 2.0	Mars 3.0
1.0 Exploration												
1.1 Planet from Orbit		+	+	+		+	+*			+	+*	
1.2 Atmospheric Entry without Surface Landing							*					
1.3 Surface								+*				+*
1.4 Satellite(s)												
2.0 Temporary Base(s)												
2.1 Planet				*								*
2.3 Satellite(s)												
3.0 Permanent Base(s)												
3.1 Planet												
3.1.1 Establishment												*
3.1.2 Maintenance										X*	X*	*
3.2 Suitable Satellite												
3.2.1 Establishment												
3.2.2 Maintenance												

\+ Instrumented Probable (One-way Mission)

X One-Way Mission, Manned or Automatically Controlled Non-Manned Tran

* Round-Trip, Manned

ion Objectives with Mission Profiles

	Jupiter				Saturn				Uranus				Neptune				Pluto			
.0	1.0	2.0	3.0	4.0	1.0	2.0	3.0	4.0	1.0	2.0	3.0	4.0	1.0	2.0	3.0	4.0	1.0	2.0	3.0	4.0
	+*		+*		+*	+*			+*	+*			+				+	*		
			+*				+*				+*				+*					
			+				+				+				+				*	
*				*				*				*				*				
	X*	X*		*	X*	X*		*				*			*	*			*	

ort Vehicle

Tab. 9 **Payload Fractions for Several Interplanetary Missions and Interplanetary Vehicles**

Principal Maneuvers and Correction Maneuvers		M-0 Earth Ascent	M-1 Earth Escape	Corr. Maneuver	M-2 Planet Capture	M-3 Planet Escape	Corr. Maneuver	M-4 Earth Capture to Circ. Orbit	Summations (λ-values for correction maneuvers included)
			Venus Round-Trip, V-2.0 (360-400 d)						
$\Delta v (10^3$ ft/sec)		30	15.2	1.0	14.7	23.2	1.5	(23)	55.6 (78.6)
O_2/H_2	I_{sp}		450	450	450	450	450	450	
	x		0.92	0.92		0.90		0.90	$\lambda_{123} = 0.00735$
	λ		0.292	0.28		0.09		0.115	
O_2/H_2 with Nucl. M-1	I_{sp}		765	450	450	450	450	450	
	x		0.88	0.92		0.90		0.90	$\lambda_{123} = 0.0116$
	λ		0.46	0.28		0.09		0.115	
Nucl. with O_2/H_2 M-4	I_{sp}		825	450	825	825	450	450	
	x		0.9	0.92	0.88	0.92	0.92	0.92	$\lambda_{123} = 0.0746$
	λ		0.515	0.93	0.52	0.335	0.895	0.145	
Nuclear (S.C.R.)	I_{sp}		845	900	900	900	900	900	
	x		0.90	0.88		0.88	0.86		$\lambda_{123} = 0.105$
	λ		0.525	0.535		0.375	0.35		$\lambda_{1234} = 0.0369$
Nuclear (G.C.R. w/ S.C.R. M-4)	I_{sp}		2000	450	2000	2000	450	900	
	x		0.86	0.92	0.84	0.82	0.92	0.86	$\lambda_{123} = 0.294$
	λ		0.755	0.93	0.755	0.62	0.895	0.36	$\lambda_{1234} = 0.106$
Nuclear (All G.C.R.)	I_{sp}		2000	450	2000	2000	450	2000	
	x		0.88	0.92	0.86	0.84	0.92	0.82	$\lambda_{123} = 0.307$
	λ		0.761	0.93	0.761	0.635	0.895	0.625	$\lambda_{1234} = 0.192$
				Mars Round-Trip Ma-2.0 (440-460 d)					
$\Delta v (10^3$ ft/sec)		30	14.8	1.0	14.8	21.8	1.5	(50)	53.3 (103.3)
Nuclear with O_2/H_2 M-4	I_{sp}		825	450	825	825	450	450	
	x		0.92	0.92	0.90	0.88	0.92		

		Mars Round-Trip [illegible] (800-1000 d)						
Δv (10^3 ft/sec)		12.8	2.0	7.7	10.2	2.0	(14.2)	
O_2/H_2	I_{sp}	450	450	450	450	450	450	$\lambda_{123} = 0.0665$
	x	0.92	0.92		0.90	0.92	0.9	$\lambda_{1234} = 0.0298$
	λ	0.364	0.48		0.44	0.865	0.49	
		Jupiter J-2.0 (1-Way), Ell Capt., 350-400 d						
Δv (10^3 f/tsec)		40	1.0	40				82
Nuclear	I_{sp}	900	900	900				
(S.C.R.)	x	0.92	0.88					$\lambda_{12} = 0.0268$
	λ	0.185	0.145					
Δv (10^3 ft/sec)		40	1.0	30				
Nuclear	I_{sp}	900	900	900				
(S.C.R.)	x	0.92	0.88					$\lambda_{12} = 0.0462$
	λ	0.185	0.25					
		Saturn S-1.0 (1-Way), 350-400 d						
Δv (10^3 ft/sec)		31 + 40	1.0					72
Nuclear	I_{sp}	900	450					
(S.C.R.)	x	0.92 0.92	0.92					$\lambda_{1a1b} = 0.049$
	λ	0.285 0.185	0.93					
		$t = \lambda_{1a}$ $t = \lambda_{1b}$						
		Mars Synodic Base						
Δv (10^3 ft/sec)	30	14.8	1.0	14.8	21.8	1.5	50	133.9
Nuclear	I_{sp}	5000	5000	450	5000	5000	450	5000	$\lambda_{1234} = 0.4$
Pulse	x	0.84	0.84	0.92	0.84	0.84	0.92	0.84	$\lambda_{01234} = 0.323$
	λ	0.8	0.895	0.93	0.895	0.85	0.895	0.68	
		Jupiter. Capture. J-2.0 (Round-Trip) 850-950 d						
Δv (10^3 ft/sec)	23	34	2.0	56	31.6	2.0	85	233.6
Nuclear	I_{sp}	5000	5000	450	5000	5000	450	5000	$\lambda_{123} = 0.296$
Pulse	x	0.84	0.84	0.92	0.84	0.84	0.92	0.84	$\lambda_{0123} = 0.25$
	λ	0.841	0.775	0.815	0.65	0.787	0.865	0.51	$\lambda_{1234} = 0.151$
								$\lambda_{01234} = 0.127$
Nuclear	I_{sp}	2000	2000	450	2000	2000	450	2000	$\lambda_{123} = 0.074$
(G.C.R.)	x	0.88	0.88	0.92	0.88	0.86	0.92	0.84	$\lambda_{0123} = 0.048$
	λ	0.65	0.53	0.865	0.338	0.55	0.865	0.126	$\lambda_{1234} = 0.0093$
								$\lambda_{01234} = 0.006$

Tab. 10 **Applications**

	Potentially Attractive ELV																								
	1						2					3						4				5		6		
	1.1	1.2	1.3	1.4	1.5	1.6	2.1	2.2	2.3	2.4	2.5	3.1	3.2	3.3	3.4	3.5	3.6	4.1	4.2	4.3	4.4	5.1	5.2	6.1	6.2	6.3
Earth Orbit Missions																										
ED-1 Establishment of Small Installations (≤100t)	*						*	*																		
ED-2 Establishment of Intermediate Install. (100-450t)	*	*	*									*	*	*	*	*	*					*	*	*	*	
ED-3 Establishment of Large Installations (450t)			*									*	*	*	*	*	*									
ED-4 Supply & Maintenance of Installations	*	*					*	*	*	*	*															
ED-5 Passenger Transport	*	*					*	*	*	*	*															
Lunar Missions																										
Lunar Exploration from Landers (L-3)	*	*		*	*																					
Establishment of a Small Base (10 persons) (L-1, L-2, L-3)	*	*		*	*							*	*	*	*		*	*	*	*	*			*	*	
Establishment of Medium Base (10-100 persons) (L-1, L-2, L-3)		*		*	*							*	*	*	*		*								*	*
Establishment of Large Base (100 persons) (L-1, L-2, L-3)				*	*							*	*	*	*		*			*	*				*	*
Supply & Maintenance of Small & Medium Base	*	*		*	*							*	*	*	*		*	*	*	*	*	*	*	*	*	
Supply & Maintenance of Large Base		*		*	*							*	*	*	*		*			*	*		*	*	*	
Lunar Passenger Transport				*	*	*						*	*	*	*		*	*	*	*	*			*	*	
Venus																										
Capture Mission (Ve-2, 4) (8-12 persons) (Orbital Recov. from Circ. or Ell. Orbit)	(*)	(*)		*	*	*						*	*	*	*		*							*	*	
Mars																										
Capture Mission (6-10 persons)		*		*1)	*1)	*1)							*	*											*	*
Exploratory Landing Mission (8-12 persons)		(*)		(*)	(*)	(*)							*	*											*	*
Extended Landing Mission (12-30 persons)		(*)		( )	(*)	(*)							*	*											*	*
Synodic Landing Mission (30-40 persons)		(*)		(*)	(*)	(*)							*	*											*	*
Long-Term Base, Small (40-100 persons)				(*)2)	(*)2)	(*)2)							*	*											*	*
Long-Term Base, Medium (100-200 persons)				(*)2)	(*)2)	(*)2)							*	*											*	*
Long-Term Base, Single Large, or Network of Med. & Small Bases (200-1000 persons)													*	*											*	*
Supply & Maintenance of Small & Medium Base		*3)		*3)	*3)	*3)							*	*											*	*
Supply & Maintenance of Large Base																										
Passenter Transport																										

Earth Launch Vehicle (ELV) Classification

Evaluation Criteria

Column groups: Chemical ELV; Nuclear ELV; Air-Breathing / Nuclear ELV [illegible]

Evaluation Criteria
1. Mission Worth
1.1 Mission Applications
1.2 Advancement Factors
1.3 Performance Sensitivity
1.4 Cost [illegible]
1.5 Mission [illegible]
1.6 Relative [illegible]
1.7 [illegible] Effectiveness
1.7.1 Direct
1.7.2 Indirect
1.8 Rate of Obsolescence
1.9 Growth Potential
2. Service Reliability
2.1 Number of Events Required for Mission Success
2.2 Launch Vehicle Mission Reliability
2.3 Launch Malfunction Sensitivity
2.4 Flight Malfunction Sensitivity
2.5 Launch Rate Flexibility
2.6 Service Duty Cycle
3. Operational Availability
3.1 [illegible] Mi[illegible]
3.2 State of the Art [illegible]
3.3 Growth Potential Availability
4. Operational Characteristics
4.1 [illegible]
4.2 [illegible] Characteristics
4.3 [illegible] Radiation [illegible]
4.4 Thrust Vector Control [illegible]
4.5 Engine-out Sensitivity
4.6 Launch Sites & Facilities
4.7 Launch Operations
4.8 [illegible]
4.9 [illegible]
4.10 Operations during Descent
4.11 Recovery Operations
4.12 Refurbishing Operations
5. [illegible] Characteristics
5.1 Critical Development Items
5.2 State of the Art [illegible]
5.3 Value Analysis
5.4 Special Facility Requirements
5.5 Special [illegible] Requirements
5.6 [illegible]
5.7 [illegible]
5.8 Special [illegible] Problems

Figure 6. Earth Launch Vehicle - Evaluation Criteria Matrix

3. DETERMINATION OF PAYLOAD FRACTION λ_i FOR EACH MANEUVER

The payload fraction, defined as ratio of gross payload weight $W_{\lambda i}$ to ignition weight W_{Ai} at the beginning of the i th powered maneuver, is given by

$$\lambda_i \equiv \frac{W_{\lambda i}}{W_{Ai}} = 1 + \frac{1}{x}\left(e^{-\tau/I_{sp}} - 1\right) \tag{3-1}$$

where

x = mass fraction $W_p/(W_p + W_b)$

W_p = propellant weight

W_b = wet inert weight

τ = v_{id}/g

The relation can usefully be plotted as λ versus τ/I_{sp} with x as parameter(Fig. 7).

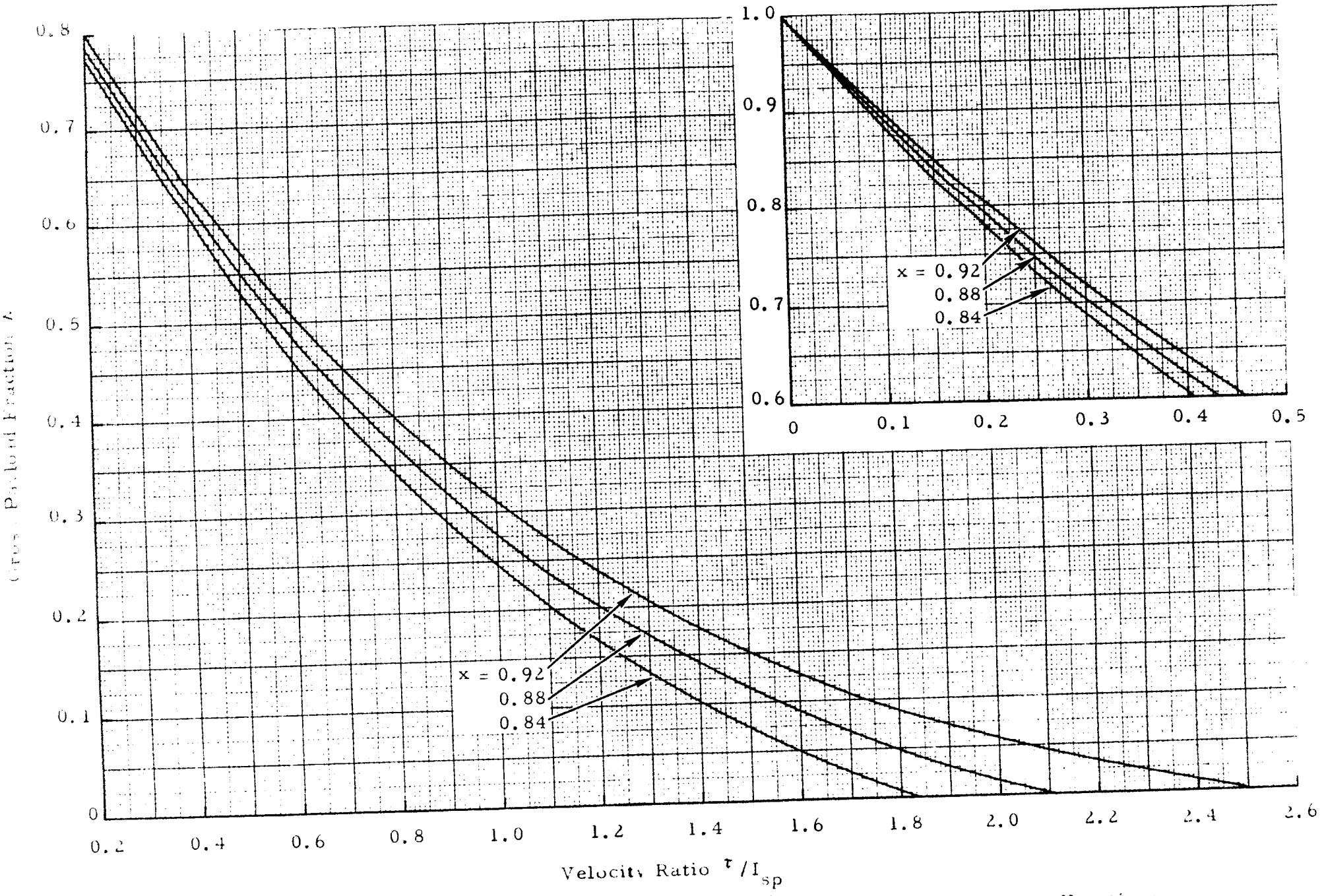


Fig. Gross Payload Fraction as Function of Velocity Ratio for Several Mass Fractions x

4. DETERMINATION OF THE INITIAL VEHICLE MASS

In a ballistic-type multi-stage vehicle, the overall payload fraction of, say, a 3-stage missile is

$$\lambda_{123} = \lambda_1 \lambda_2 \lambda_3 \tag{4-1}$$

For determination of the initial mass of an interplanetary (or lunar vehicle) this approach is inadequate, because of losses and planned weight reductions en route. These are designated as W_j in the systematic weight break-down/reproduced from the EMPIRE final report and attached here, "Principal Events in Capture Mission" (Tab. 11) They consist of two weight groups, namely weight W_{ji} jettisoned in a "house cleaning process prior to powered maneuver M-i; and weight $W_{j(i,\,i+1)}$ eliminated during the period between two major maneuvers M-i and M-(i+1). Generally, $W_{j(i,\,i+1)}$ and $W_{j\,(i+1)}$ belong to the same stage or module and can be lumped together as

$$\Delta W_{(i+1)} = W_{j(i,\,i+1)} + W_{j(i+1)} \tag{4-2}$$

or, written in a different form,

$$\Delta W_{(i+1)} = p_{(i+1)}\, W_{A(i+1)} \tag{4-3}$$

The method assumes that the terminal gross payload, designated as $W_{\lambda\,4}$ in a 4-maneuver round-trip capture mission is known or definable. This, then, is the payload of the vehicle during its Earth approach maneuver M-4, and the vehicle's corresponding ignition weight is

$$W_{A4} = \frac{1}{\lambda_4}\, W_{\lambda\,4} \tag{4-4}$$

During transfer from planet to Earth the weight W_{j34} was eliminated. Prior to the Earth arrival maneuver M-4 the weight W_{j4} was eliminated. Thus

$$\Delta W_4 = W_{j34} + W_{j4} \tag{4-5}$$

whence the "gross payload weight" of the vehicle during the departure maneuver

M-3 from the target planet is

$$W_{\lambda 3} = W_{A4} + \Delta W_4 \tag{4-6}$$

Therewith, the M-3 ignition weight of the space vehicle becomes

$$W_{A3} = \frac{1}{\lambda_3}(W_{A4} + \Delta W_4) = \frac{1}{\lambda_3 \lambda_4} W_{\lambda 4} - \frac{1}{\lambda_3} \Delta W_4 \tag{4-7}$$

Correspondingly, for M-2 (target planet capture),

$$W_{A2} = \frac{1}{\lambda_2 \lambda_3 \lambda_4} W_{\lambda 4} + \frac{1}{\lambda_2 \lambda_3} \Delta W_4 - \frac{1}{\lambda_2} \Delta W_3 \tag{4-8}$$

and for M-1 (Earth departure),

$$W_{A1} = \frac{1}{\lambda_1 \lambda_2 \lambda_3 \lambda_4} W_{\lambda 4} + \frac{1}{\lambda_1 \lambda_2 \lambda_3} \Delta W_4 - \frac{1}{\lambda_1 \lambda_2} \Delta W_3$$

$$+ \frac{1}{\lambda_1} \Delta W_2 \tag{4-9}$$

Now, let

$$\Delta W_4 = p_4 W_{A4} = \frac{p_4}{\lambda_4} W_{\lambda 4}$$

$$\Delta W_3 = p_3 W_{A3} = \frac{p_3}{\lambda_3 \lambda_4} W_{\lambda 4} + \frac{p_3 p_4}{\lambda_3 \lambda_4} W_{\lambda 4} \tag{4-10}$$

$$\Delta W_2 = p_2 W_{A2} = \frac{p_2}{\lambda_2 \lambda_3 \lambda_4} W_{\lambda 4} + \frac{p_2 p_4}{\lambda_2 \lambda_3 \lambda_4} W_{\lambda 4}$$

$$+ \frac{p_2 p_3}{\lambda_2 \lambda_3 \lambda_4} W_{\lambda 4} + \frac{p_2 p_3 p_4}{\lambda_2 \lambda_3 \lambda_4} W_{\lambda 4}$$

or, simply,

$$\Delta W_4 = \frac{p_4}{\lambda_4} W_{\lambda 4}$$

$$\Delta W_3 = \frac{p_3}{\lambda_{34}} W_{\lambda 4}(1 + p_4) \tag{4-11}$$

$$\Delta W_2 = \frac{p_2}{\lambda_{234}} W_{\lambda 4}(1 + p_3 + p_4 + p_3 p_4)$$

where

$$\lambda_{34} = \lambda_3 \lambda_4$$

$$\lambda_{234} = \lambda_2 \lambda_3 \lambda_4 \tag{4-12}$$

From these equations, the following relations are obtained for the ignition weight

(A) <u>If an M-4 is involved</u>

$W_{\lambda 4}$ given

$$W_{A4} = \frac{1}{\lambda_4} W_{\lambda 4} \tag{4-13}$$

$$W_{A3} = \frac{1}{\lambda_{34}} W_{\lambda 4} (1 + p_4) \tag{4-14}$$

$$W_{A2} = \frac{1}{\lambda_{234}} W_{\lambda 4} (1 + p_4 + p_3 + p_3 p_4) \tag{4-15}$$

$$W_{A1} = \frac{1}{\lambda_{1234}} W_{\lambda 4} (1 + p_2 + p_3 + p_4 + p_2 p_3 + p_2 p_4 + p_3 p_4 + p_2 p_3 p_4) \tag{4-16}$$

(B) <u>If an M-4 is not involved</u>

$W_{\lambda 4}$ given

$$W_{A4} \equiv W_{\lambda 4} \tag{4-17}$$

$$W_{A3} = \frac{1}{\lambda_3} W_{\lambda 4} \tag{4-18}$$

$$W_{A2} = \frac{1}{\lambda_{23}} W_{\lambda 4} (1 + p_3) \tag{4-19}$$

$$W_{A1} = \frac{1}{\lambda_{123}} W_{\lambda 4} (1 + p_2 + p_3 + p_2 p_3) \tag{4-20}$$

The p-values represent the weight expenditures between the major powered maneuvers in fractions of ignition weight of their stage. They are useful parameters in determining the orbital departure weight as well as the ignition weight for all major maneuvers during a planetary mission as function solely of $W_{\lambda 4}$ (predetermined), λ_i (I_{sp}, Δv_{id}, i.e. f(mission profile)) and p_4 (which can be varied parametrically or

determined in accordance with a suitable pattern).

In Tab. 9, payload fractions are listed for several interplanetary vehicles. From these, the various ignition weights and the orbital departure weight can be determined for given values of $W_{\lambda 4}$ and p. Mor M-4, in Tab. 9, it was assumed that it would serve only the reusability of the interplanetary vehicle. Therefore, capture into a circular orbit is assumed. Where reusability is not important or unfeasible, it is assumed that no M-4 is needed, because either the terminal payload (a fraction of $W_{\lambda 4}$) enters hyperbolically, or the crew is picked up by hyperbolic rendezvous.

Tab. 11 Principal Events in Capture Mission

Weight of vehicle fully assembled in satellite orbit (initial weight)	W_{S1}
Weight consumed prior to launch	W_{j1}
Weight, following mission readiness test, at ignition (launch weight)	W_{A1}
Burnout weight following M-1	W_{B1}
Propellant weight consumed during M-1	$W_{p1} = W_{A1} - W_{B1}$
Wet inert weight of vehicle section (escape booster) staged following M-1	W_{b1}
Weight at the beginning of coast (initial outbound coast weight)	W_{C1}
Weight consumed or jettisoned during planetocentric coast and heliocentric coast, including weight consumed for correction or spin and de-spin maneuvers, boiloff losses etc.	W_{j12}
Weight at termination of transfer coast, at the beginning of preparations for the M-2 maneuver (terminal outbound coast weight	W_{C2}
Weight eliminated in preparation of capture maneuver M-2; "house cleaning", i.e., jettisoning of all items no longer needed at the end of the outbound coast. This measure is taken to ensure that no unnecessary weight is accelerated	W_{j2}
M-2 ignition weight	W_{A2}
M-2 burnout weight	W_{B2}
M-2 useful propellant weight	$W_{p2} = W_{A2} - W_{B2}$
Wet inert weight (M-2 propulsion unit) staged following M-2	W_{b2}
Weight at beginning of capture period in target planet satellite orbit (initial capture weight)	W_{S2}
Weight consumed or jettisoned during capture period, including propellant consumed for whatever adjustment maneuvers are necessary	W_{j23}
Weight at termination of capture period (terminal capture weight)	W_{S3}
House cleaning in preparation of M-3	W_{j3}

Tab. 11 Principal Events in Capture Mission
(Concluded)

M-3 ignition weight	W_{A3}
M-3 burnout weight	W_{B3}
M-3 useful propellant weight	$W_{p3} = W_{A3} - W_{B3}$
Wet inert weight (M-3 propulsion unit) staged following M-3	W_{b3}
Weight at beginning of coast (initial return coast weight)	W_{C3}
Weight consumed or jettisoned during planetocentric and heliocentric coast, including weight consumed for correction, or spin and de-spin maneuvers, boiloff losses, etc.	W_{j34}
Weight at termination of transfer coast, at the beginning of preparations for the M-4 maneuver (terminal return coast weight)	W_{C4}
House cleaning in preparation of M-4, including jettisoning of the entire LSS except the EEM	W_{j4}
M-4 ignition weight	W_{A4}
M-4 burnout weight	W_{B4}
M-4 useful propellant weight	$W_{p4} = W_{A4} - W_{B4}$
Wet inert weight (M-4 propulsion unit) staged following M-4	W_{b4}
Ignition weight for correction maneuver M-5 prior to Earth entry	W_{A5}
M-5 burnout weight	W_{B5}
Wet inert weight jettisoned following M-5 and prior to Earth entry	W_{b5}
Initial Earth Entry Module (EEM) weight	W_{E1}
Terminal EEM weight (terminal mission weight)	W_{E2}

5. LIFE SUPPORT SYSTEM

Four major weight groups are to be considered:

5.1 Basic life support system dry weight (net dry weight without abort system and radiation shelter)

5.2 Radiation Shelter

5.3 Ecological System and Emergency Power Supply

5.4 Abort System

5.1 Basic LSS Dry Weight

Reference System: L-42

Basic LSS Dry weight (w/o ecological system and rad. shelter): 31,590 lb

Total Volume: 6500 ft^3

Vol. specific weight = 4.85 lb/ft^3

Crew size: 8 persons

Total vol. per crew member: 810 ft^3/p

Basic dry weight per crew member: 3950 lb/p

Now we will make the following assumptions:

(a) Total volume per crew member decreases with increasing crew size; (Fig. 8)

$$v_{tot} = 800 \left(\frac{n}{8}\right)^a \qquad n = \text{number of crew members} \quad a = -0.1 \tag{5-1}$$

(b) Total volume per crew member increases with increasing mission time (Fig. 9)

$$v_{tot} = 800 \left(\frac{T}{400}\right)^b \qquad T = \text{mission time (d)} \quad b = 0.6 \tag{5-2}$$

(c) Therefore, the total volume per person as function of crew size and mission time can in the first order be represented by the relation

$$v_{tot} = 800 \left(\frac{n}{8}\right)^a \left(\frac{T}{400}\right)^b \qquad (ft^3/p) \tag{5-3}$$

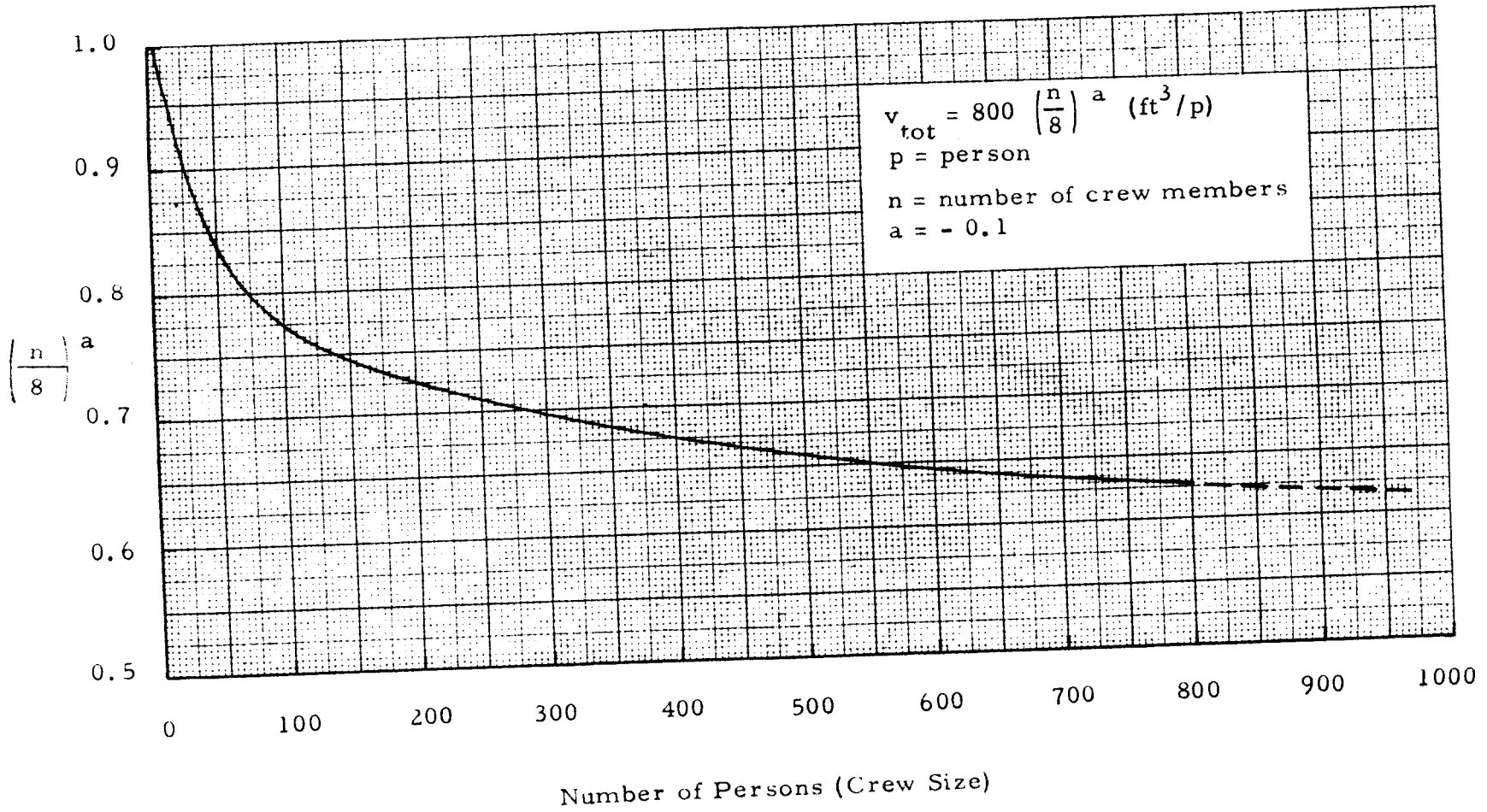


Fig. 3 Variation of LSS Specific Total Volume per Person as Function of Crew Size

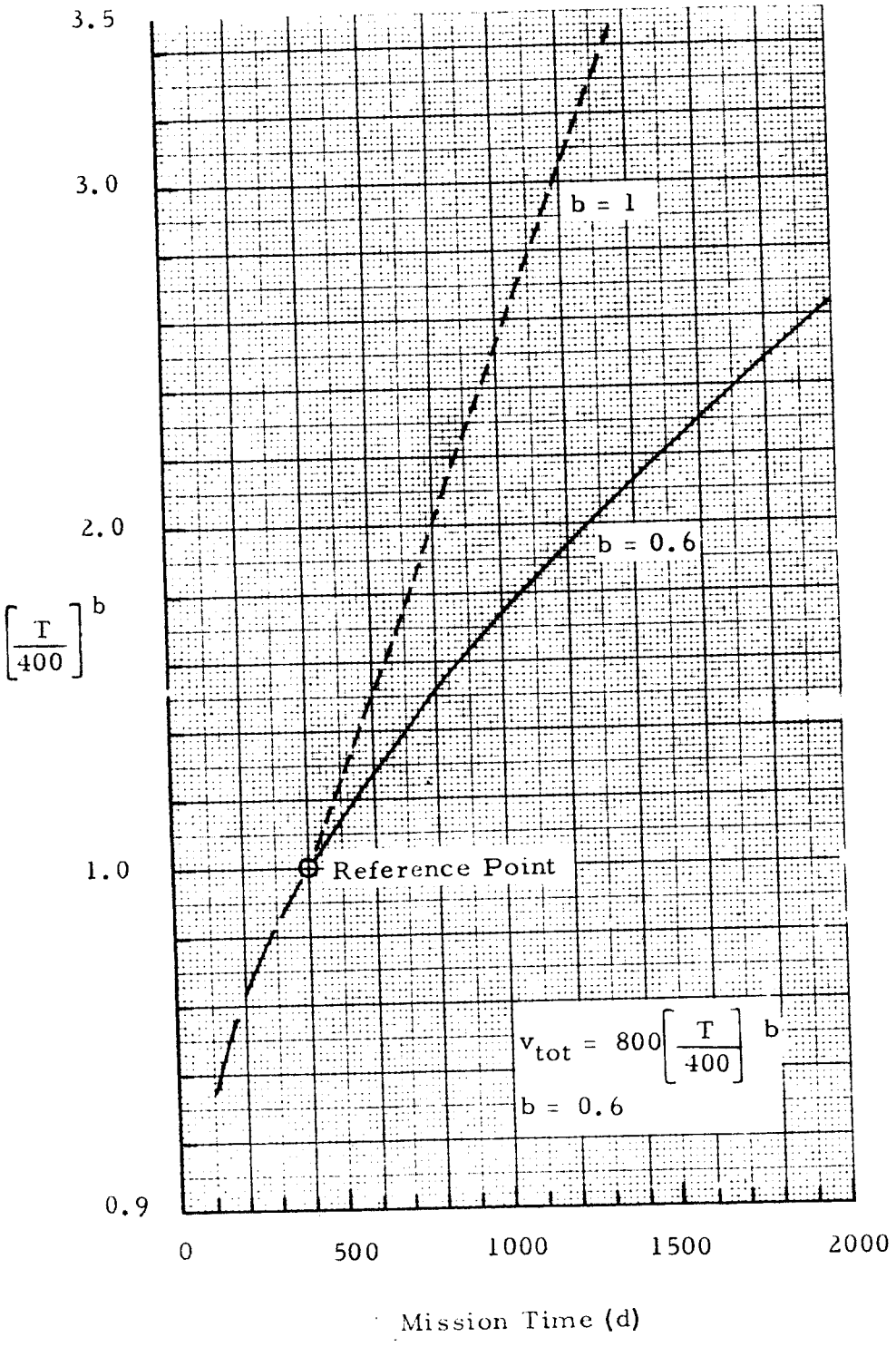


Fig. 9 Variation of LSS Specific Total Volume per Person as Function of Mission Time

(d) For a given type of mission, e.g. capture mission, the crew size increases with travel time (Fig. 10)

$$n = n_o \left(\frac{T}{400}\right)^{c\left(\frac{T}{400}\right)}$$

n_o = basic crew size for 400-day capture mission; n to be taken as the next higher full number; c = 0.3 (5-4)

(e) Therewith Eq. (5-3) becomes

$$v_{tot} = 800 \; \frac{n_o}{8} \left(\frac{T}{400}\right)^{c\left(\frac{T}{400}\right)} a \left(\frac{T}{400}\right)^{b} \qquad (ft^3/p) \qquad (5\text{-}5)$$

which makes the specific volume a function of mission time as independent variable and of the basic crew size as parameter. The basic crew size is defined as the size of the crew required to run the ship and take care of the vehicle oriented and the mission oriented functions. Not included are crew members for special tasks, such as surface excursion, visit to a moon etc., since these excursions may not vary as to duration and objectives even though the mission time varies. They must be evaluated separately as function of their objectives and duration. The increase in basic crew size aims to take into account the following factors which tend to cause an increase:

Higher probability of crew members getting incapacitated.

Reduced overall average efficiency due to monotony, fatigue effects, etc.

More mission tasks, because more observations and measurements will be made, which would be less relevant or not practical during faster flights (especially on-board biological experiments involving many generations of animals)

More vehicle-oriented tasks, because a greater variety of failures is likely to occur and a greater variety of skills will be required to keep the vehicles operational

If Eq (5-5) is applied to an n_o = 8 person crew, the following picture results:

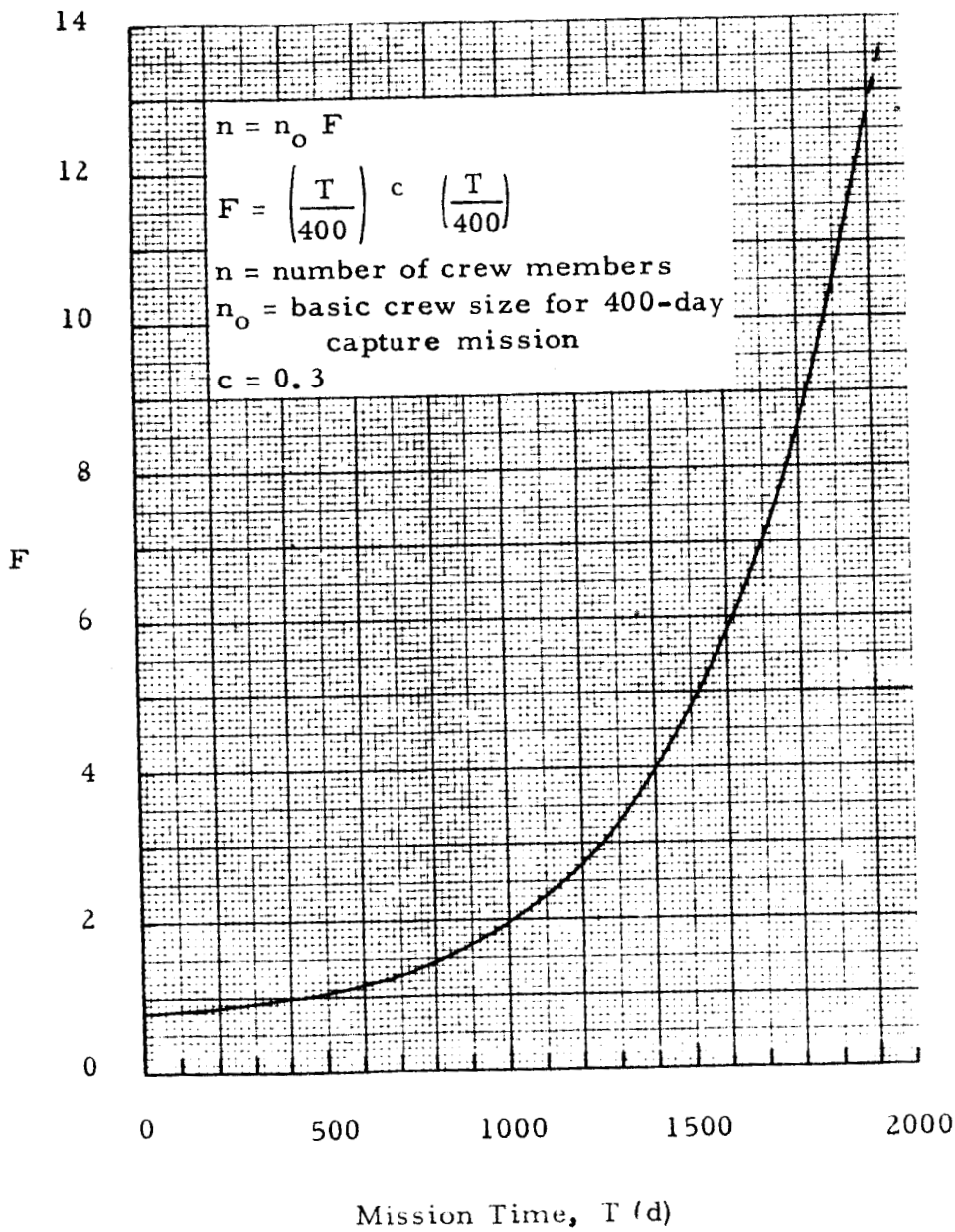


Fig. Variation of Crew Size as Function of Mission Time

$n_o = 8,\ a = -0.1,\ b = 0.6,\ c = 0.3$

T(d)	200	400	600	800	1000	2000
n	8	8	10	13	16	102
$A = 800 \left[\text{Eq. (5-5)}\right]^a$	800	800	780	750	740	610
$B = (T/400)^b$	1.0	1.0	1.261	1.515	1.734	2.624
$v_{tot} = A \cdot B\ (ft^3/p)$	812.5	812.5	1000	1155	1307	1629
$v_{tot} \cdot n = V_{tot}\ (ft^3)$	6500	6500	10.000	15.000	20,900	166,000

The principal functions are presented in Fig. 11.

(f) The volume for L-42 costs about 4.85 lb of basic dry weight per ft^3.

This specific weight is likely to decrease with increasing LSS volume. Therefore

$$w_{sp} = 4.85 \left(\frac{V_{tot}}{6500}\right)^d \ (lb/ft^3)$$

$$V_{tot} = \text{total volume of LSS } (ft^3);\ d = -0.05 \qquad (5\text{-}6)$$

(f) Therewith, the dry weight of the basic LSS becomes (Figs. 12, 13, 14),

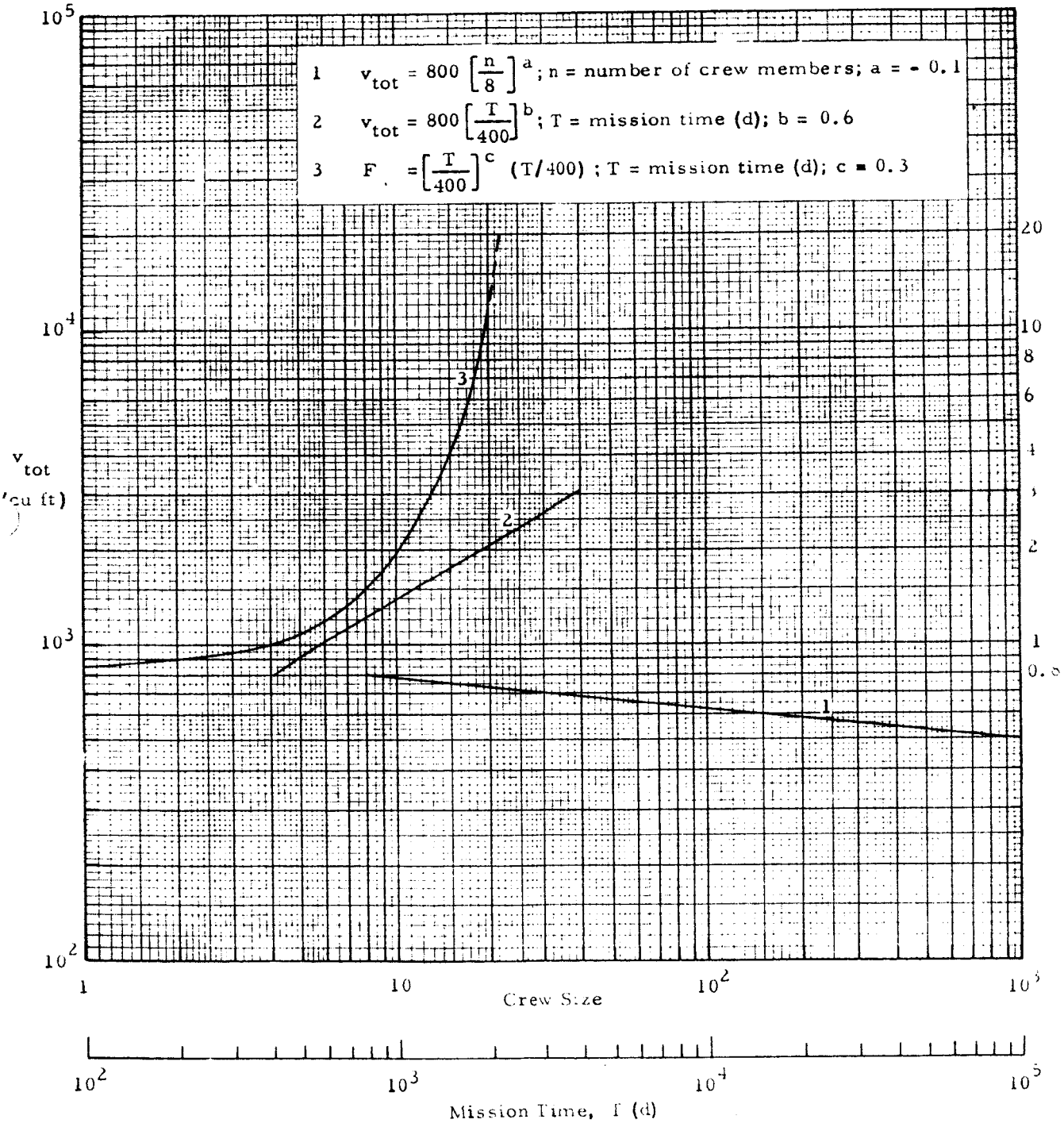
$$W_d = V_{tot}\, w_{sp} = n\, v_{tot}\, w_{sp} = n\, v_{tot} \cdot 4.85 \left(\frac{n\, v_{tot}}{6500}\right)^d \ (lb)$$

$$= 800 \cdot 4.85\, n \left[\frac{n_o}{8}\left(\frac{T}{400}\right)^{c\left(\frac{T}{400}\right)}\right]^a \left(\frac{T}{400}\right)^b \left\{\frac{800\, n \left[\frac{n_o}{8}\left(\frac{T}{400}\right)^{c\left(\frac{T}{400}\right)}\right]^a \left(\frac{T}{400}\right)^b}{6500}\right\}^d$$

$$= 3880\, n \left[\frac{n_o}{8}\left(\frac{T}{400}\right)^{c\left(\frac{T}{400}\right)}\right]^a \left(\frac{T}{400}\right)^b \left\{\frac{800\, n \left[\frac{n_o}{8}\left(\frac{T}{400}\right)^{c\left(\frac{T}{400}\right)}\right]^a \left(\frac{T}{400}\right)^b}{6500}\right\}^d \ (lb) \qquad (5\text{-}7)$$

For the case of $n_o = 8,\ a = -0.1,\ b = 0.6,\ c = 0.3$ and $d = -0.05$ one obtains

T(d)	2000	400	600	800	1000	2000
n	8	8	10	13	16	102
$V_{tot}\ (ft^3)$	6500	6500	10.000	15,000	20,900	166.000
$w_{sp}\ (lb/ft^3)$	4.85	4.85	4.75	4.655	4.57	4.126
$W_d (lb)$	31.590	31.590	47.500	70.000	95.600	693.000
$W_d/n\ (lb/p)$	3950	3950	4750	5390	5980	6790


1 $v_{tot} = 800 \left[\frac{n}{8}\right]^a$; n = number of crew members; a = - 0.1
2 $v_{tot} = 800 \left[\frac{T}{400}\right]^b$; T = mission time (d); b = 0.6
3 $F = \left[\frac{T}{400}\right]^c$ (T/400) ; T = mission time (d); c = 0.3
10^5
10^4
10^3
10^2
v_{tot}
(cu ft)
20
10
8
6
4
2
1
0.8
F
1
2
3
1
10
10^2
10^3
Crew Size
10^2
10^3
10^4
10^5
Mission Time, T (d)

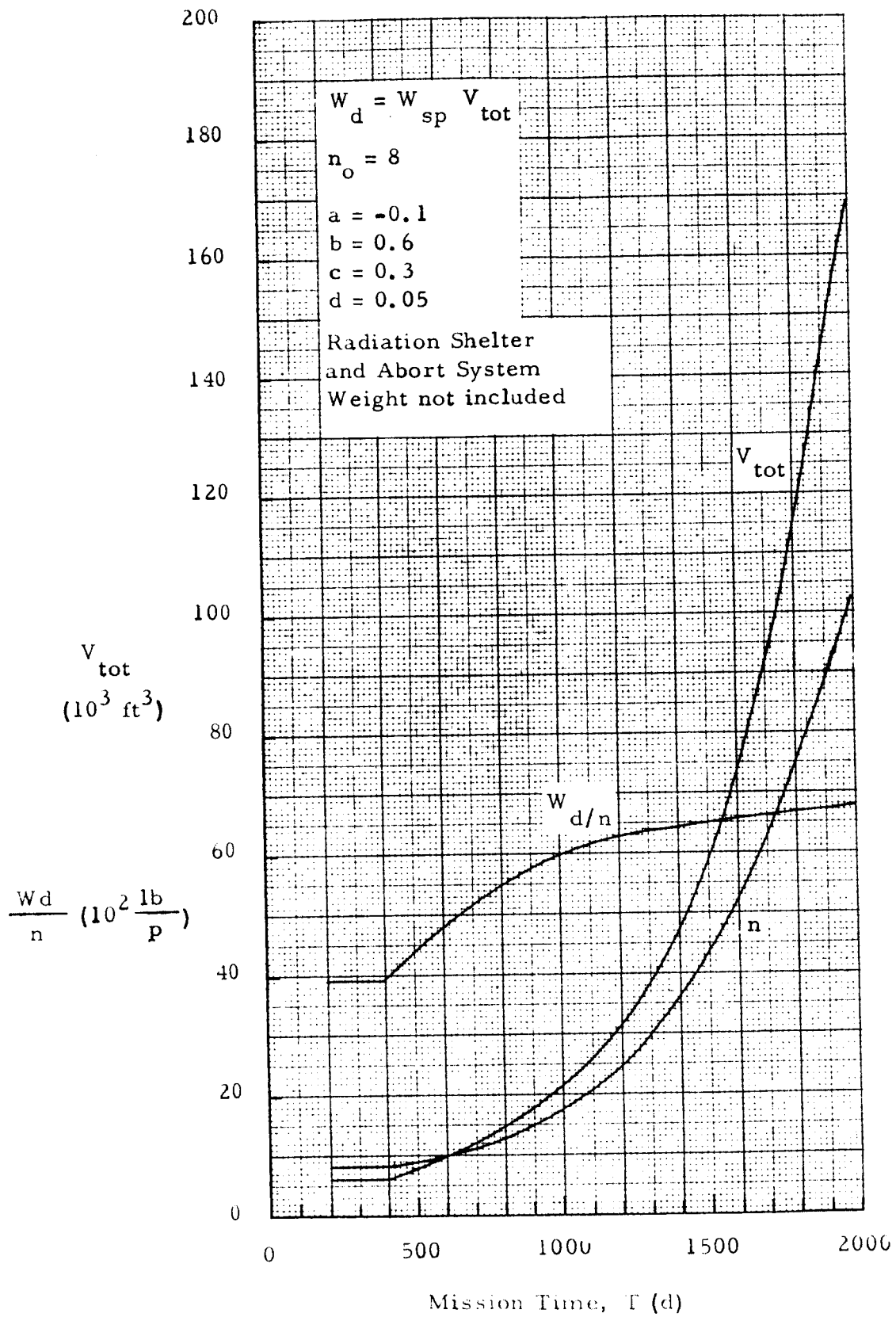


Fig. Variation of Total Volume of LSS, Crew Size and LSS Dry Weight per Person as Function of Mission Time

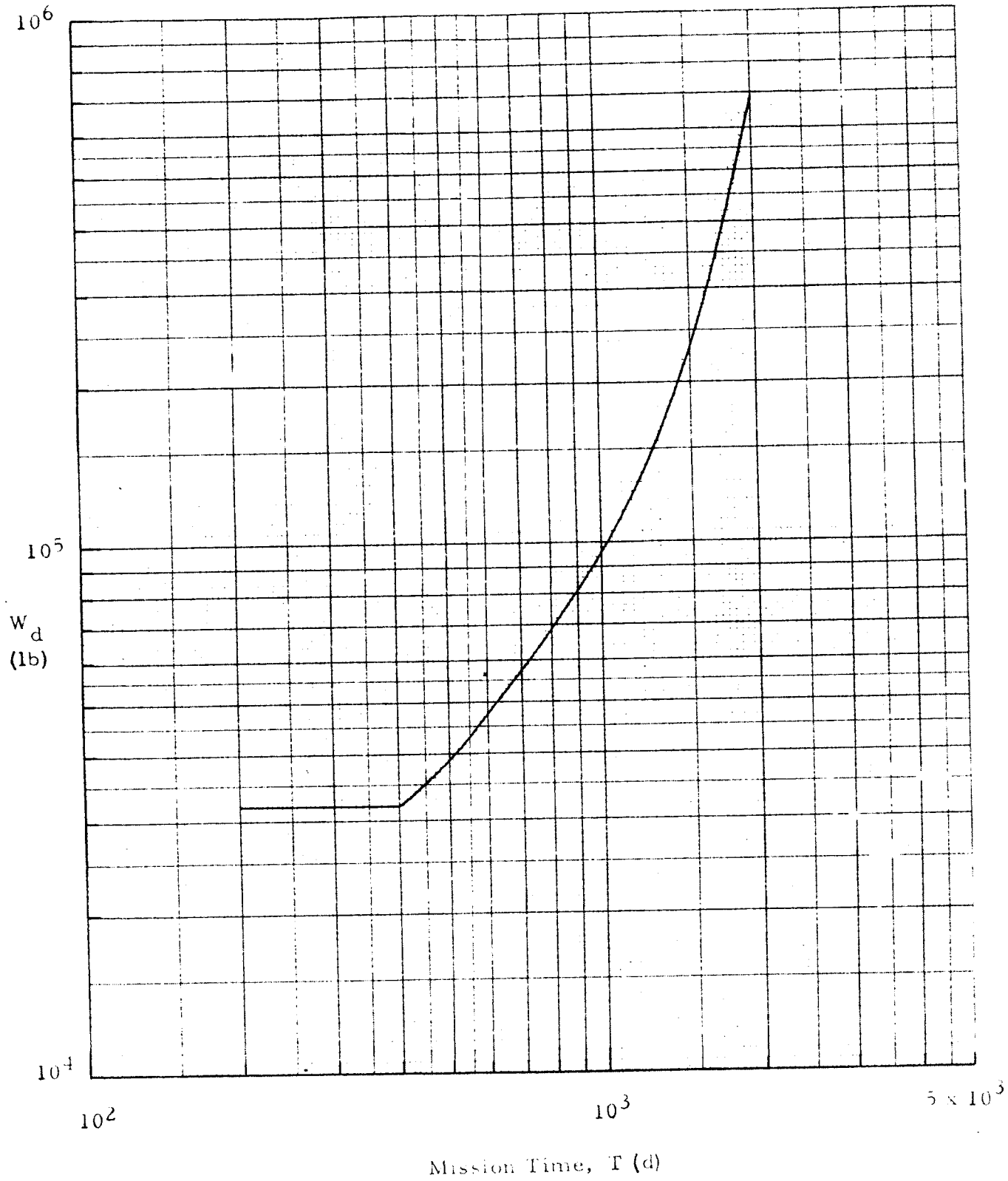


Fig. 3 Variation of Dry Weight of LSS with Mission Time (Radiation Shelter and Abort System not included)

5.2 Radiation Shelter

Reference system: L-42

8-person radiation shelter for mission year 1975/76 to Mars.

Volume: 400 ft^3; Weight = 11,800 lb

Mission time: 400 d

Average radiation dose = 1 rad/day

Volume specific weight: 29.6 lb/ft^3

CH_2 shielding area weight for above mission year: 11 gm/cm^2

= 22.5 lb/ft^2

(a) **Shielding requirements vary with solar activity during mission year. Presently, for the period 1975-2000 an average value of 13 gm/cm^2 = 26.6 lb/ft^2 is selected for a 400 day mission to Mars and 1 rad/day.**

(b) The shielding weight is also a function of the heliocentric distances covered during mission. Thus, for a mission to a more distant planet, the required shielding weight should decrease. However, the usually longer mission period has a compensating effect since it requires a reduction of the acceptable daily radiation dose. Moreover, even missions to the outer planets may lead close to the Sun (usually during the return flight); so do missions to Mars. Therefore, pending a more accurate treatment, the following assumptions are made for applying a correction factor f_c:

400-day Mars mission: $f_c = 1.0$

600-day Mars mission: $f_c = 1.15$

800-day Mars mission: $f_c = 1.25$ (5-8)

1000-day Mars mission: $f_c = 1.3$

400-day Venus mission: fc = 1.15

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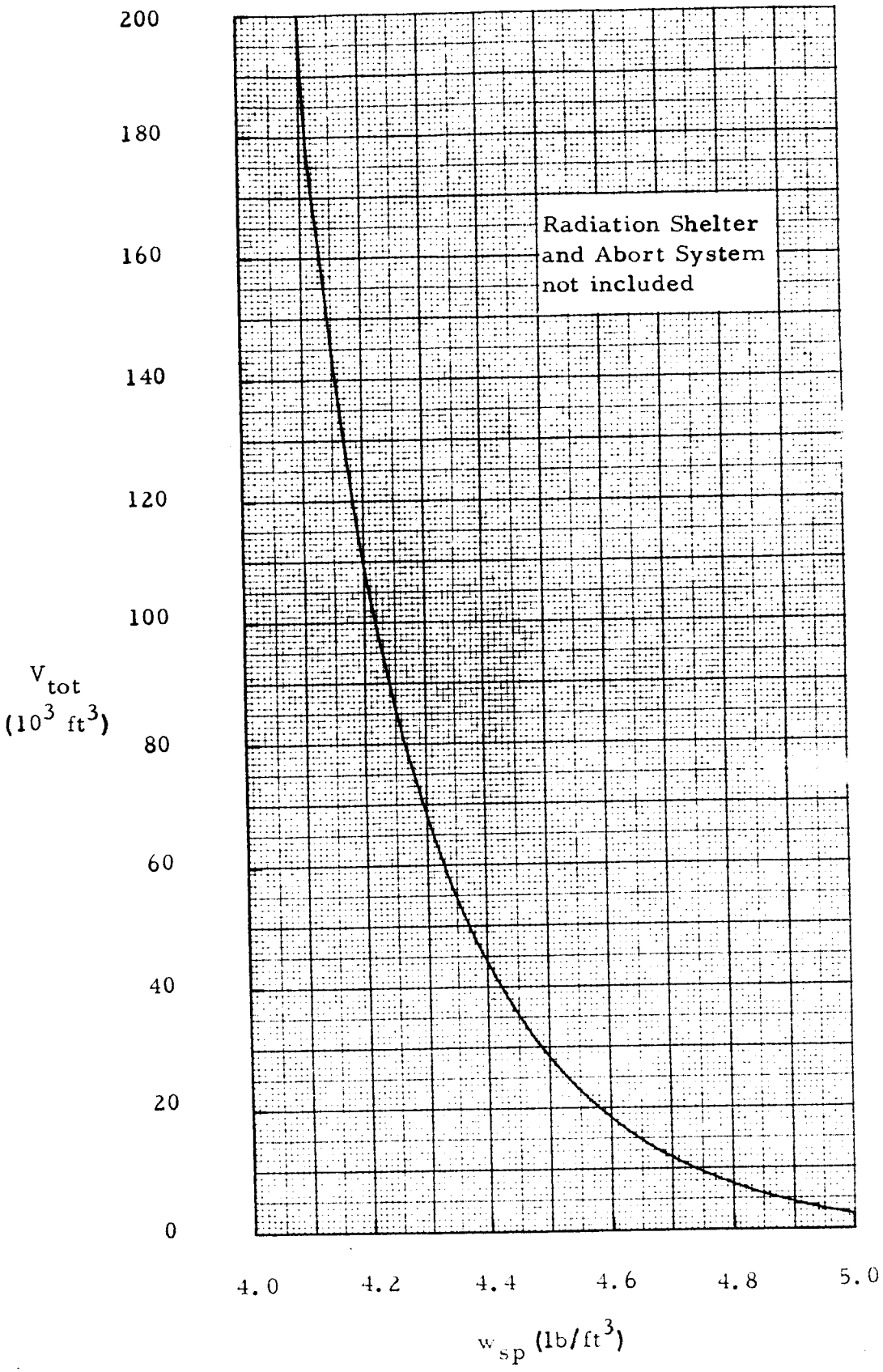

Fig. [illegible] Variation of Volume Specific Weight of Dry LSS as Function of LSS Total Volume

300-day Mercury mission: $f_c = 1.6$

800-day Jupiter mission: $f_c = 0.9$

1000-day Saturn mission: $f_c = 0.7$ (5-8)

It is realized that these figures are subject to variations depending upon mission time, mission profile and permissible daily radiation dose. They will be refined to these effects. Presently they merely represent a trend.

(c) Presently, it is assumed for the sake of simplicity, that the radiation shelter is always a section of a cylinder. Therefore, the following relations hold (d = diameter, h = height or length of section):

$$V = \frac{\pi}{4} d^2 h = \text{volume of shelter (inner volume)} \tag{5-9}$$

$$S = \pi d h + \frac{\pi}{2} d^2 = \pi d \left(h + \frac{d}{2}\right) = \text{total surface area of shelter} \tag{5-10}$$

$$\frac{S}{V} = \frac{4\left(h + \frac{d}{2}\right)}{d h} \tag{5-11}$$

$$S = \frac{4\left(h + \frac{d}{2}\right)}{d h} V \tag{5-12}$$

(d) The required shelter volume is assumed to be about $400\,(T/400)^{1/3}$ ft^3 for 8 persons or $50\,(T/400)^{1/3}$ ft^3 per person (for $T > 400$ d only). Therefore, if n is the crew size, the surface area of the shelter becomes

$$S = \frac{200\left(h + \frac{d}{2}\right)}{d h} n \left(\frac{T}{400}\right)^{\frac{1}{3}} \quad (\text{ft}^2) \tag{5-13}$$

(e) Based on the inner volume of the reference shelter, the weight per unit surface area is

$$w_S = \frac{11.800}{S} = \frac{11.800}{\pi \cdot 9.5(5 + 4.75)} = \frac{11.800}{291} = 40.5 \text{ lb/ft}^2$$

Therefore, for the reference mission of 400 d to Mars at 1 rad/day the area specific weight of the radiation shelter will be taken as $w_S = 40$ lb/ft^3. Since the radiation shield weight amounts to about 22.5 lb/ft^3, there are about 18 lb/ft^2 or about 45% of

the total weight absorbed in interior equipment and furnishings of the radiation shelter (e.g. emergency ecological provisions (incl. water), medical provisions, emergency sanitary equipment, "hospital beds", etc.). This weight thus amounts to about 5200 lb or 650 lb/p average for each crew member. This weight allowance is tentatively varied with $(T/400)^{1/3}$ for $T > 400$ d and kept constant for $T < 400$ d.

(f) Thus, the radiation shelter weight consists of two parts: radiation shield weight,

$$W_S = f_c \, w_s \, S \tag{5-14}$$

and the interior weight, W_{int},

$$W_{int} = 650 \, n \left(\frac{T}{400}\right)^{\frac{1}{3}} \tag{5-15}$$

With the use of Eq. (5-13) and consideration of (5-8), the radiation shelter weight becomes then (Fig. 15),

$$\begin{aligned} W_{RS} &= W_S + W_{int} \\ &= f_c \, w_S \frac{200\,(h + \frac{d}{2})}{d\,h} \; n \left(\frac{T}{400}\right)^{\frac{1}{3}} + 650 \, n \left(\frac{T}{400}\right)^{\frac{1}{3}} \text{ (lb)} \\ &= n \left(\frac{T}{400}\right)^{\frac{1}{3}} \left[f_c \, w_S \frac{200\,(h + \frac{d}{2})}{d\,h} + 650 \right] \text{ (lb)} \end{aligned} \tag{5-16}$$

where

d = diameter of (cylindrical) radiation shelter

h = height or length of (cylindrical) radiation shelter

T = mission time

n = number of crew members

w_S = area specific weight of shielding material (CH_2 + Boron) = 26.6 lb/ft^2

For the numerical computation, a height of h_o = 5.1 ft or multiples thereof is selected, to assure that all stories of a multiple-story shelter are of the same

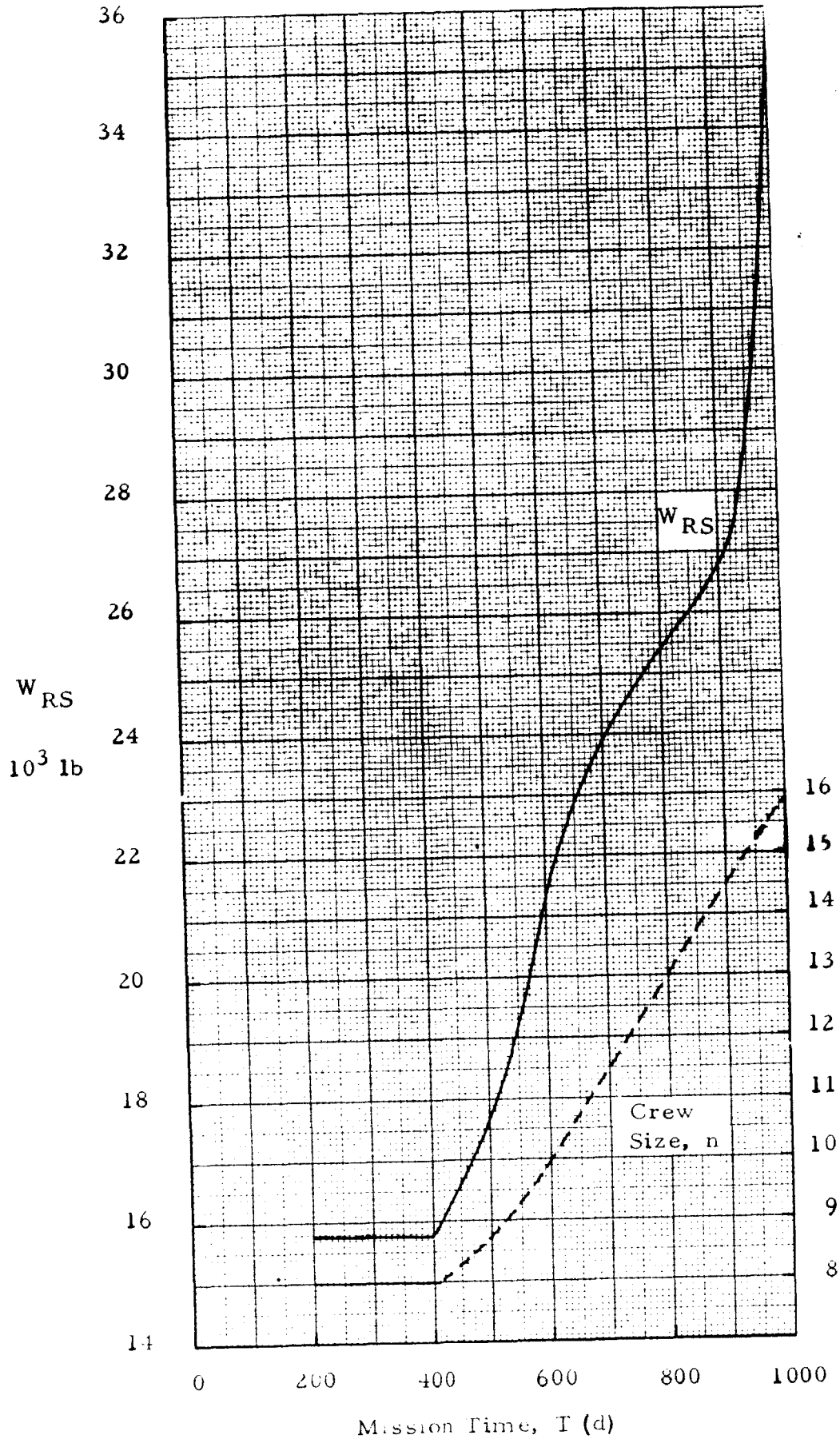

Fig. [illegible] Variation of Radiation Shelter Weight as Function of Mission Time

height. This makes the diameter

$$d = \sqrt{\frac{\pi}{4}\,\frac{V}{h}} = 1.127\sqrt{\frac{V}{h}} = 0.5\sqrt{\frac{V}{h/h_o}} \qquad (5\text{-}17)$$

e.g. for a 2-story shelter, $d = 0.5\sqrt{V/2}$, etc. For the previously tabulated example, then the following results are obtained,

T(d)	200	400	600	800	1000
n	8	8	10	13	16
f_c	1.0	1.0	1.15	1.25	1.3
$f_c\, w_S$ (lb/ft^2)	26.6	26.6	30.6	33.2	34.5
V/n = 50 A (ft^3/p)	50	50	57.5	62.5	65
V (ft^3)	400	400	575	813	1040
h/h_o	1.0	1.0	1.0	2.0	2.0
d (ft)	10	10	12	10.1	11.4
h (ft)	5.1	5.1	5.1	10.2	10.2
Wt. per person (lb/p)	1970	1970	2100	1635	1630
W_{RS} (lb)	15,750	15,750	24,000	26,800	35,400

5.3 Ecological System and Emergency Power Supply

The ecological system is a function of crew size n for $n \leq 8$ and mission period T, approximately as follows:

$$W_{Ecol} = 6500 - \left[27 + 6\,(n-2)\right] T \qquad (5\text{-}18)$$

(where T is measured in days), for the following conditions:

(a) Volume of system: 2600 ft^3

(b) 5.5 psia atmosphere

(b-1) O_2-N_2

(b-2) 2 recompressions

(b-3) 5 lb/d O_2-loss by leakage and use of airlocks

(b-4) 3 lb/d N_2-loss " " " " " "

(b-5) Emergency O_2 = 10% of O_2-loss

(c) Food = 3.3 lb/p/d (80% dehydrated)

(d) CO_2- reduction, O_2-regeneration by H_2-reduction and electrolysis

(e) Water weight = 1200 lb

(f) Emergency power supply : 5 ekw (turbo-generator)

(g) Contingencies: 10%

For $n > 8$ the following equation is tentatively applied(Fig. 16):

$$W_{Ecol} = 6500 + 800\,(n-8)^e + \left[\,27 + 6\,(n-2)\,\right] T \qquad (5\text{-}10)$$

$$e = 0.7$$

5.4 Abort System

The classical abort system consists of an Earth Entry Module (EEM) and associated propulsion system, for rapid separation from the failing vehicle and for entering a suitable orbit if re-entry cannot be effected. This concept obviously cannot be retained as the crew or passenger number keeps increasing. Sections of the vehicle must be considered for abort. This is presently under investigation.

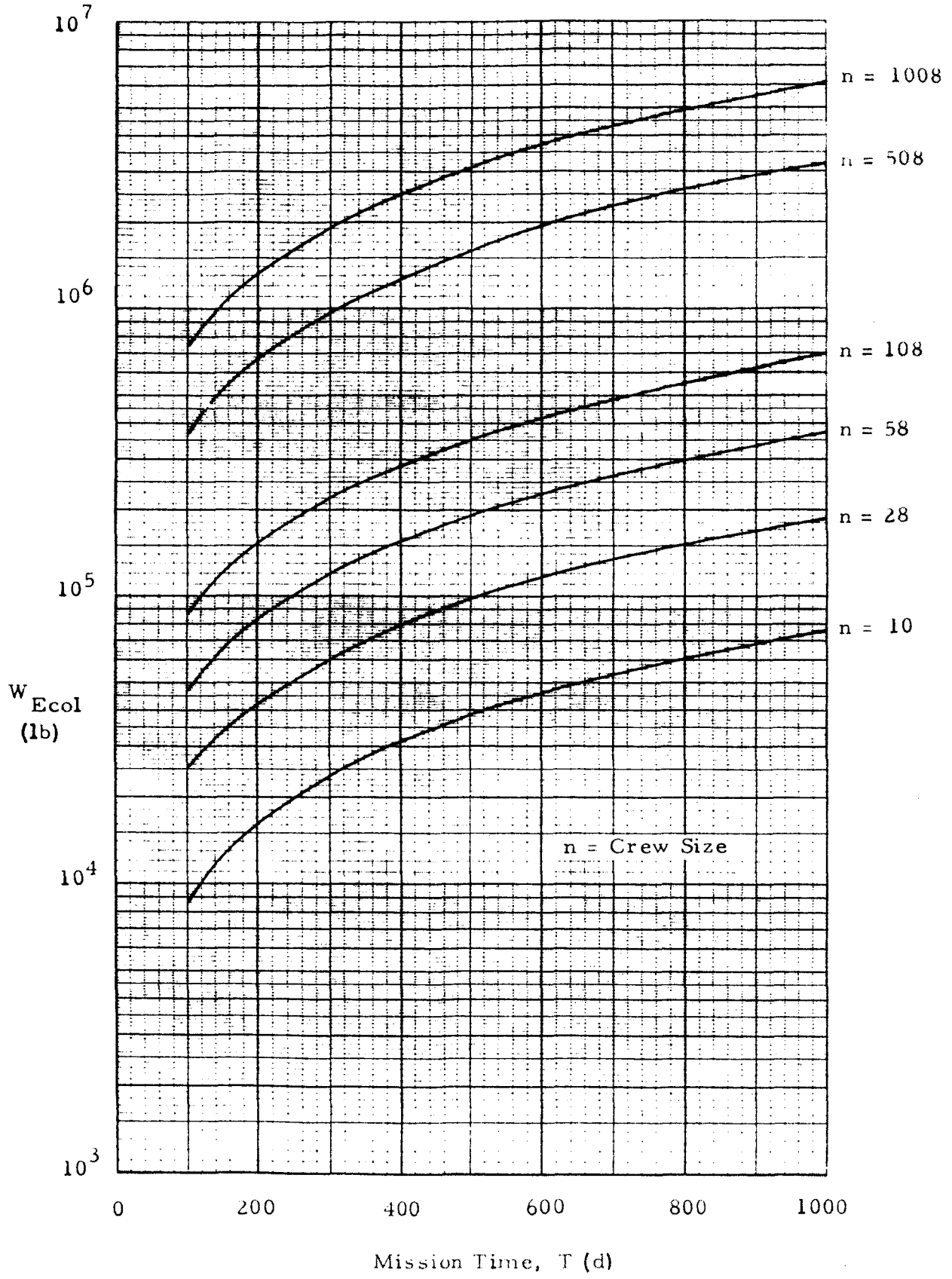

Fig. Weight of Ecological System as Function of Mission Time for Various Crew Sizes

6. SPARE PARTS FOR THE PLANETARY MISSIONS

The length of the missions makes it necessary to carry spare parts along, thereby making use of human skill and judiciousness to keep the vehicles operational, rather than providing added redundancy at considerable weight penalty, as it must be done with an instrumented system.

The kind and quantity of spare parts depends on the vehicle type and length of mission. The weight of the spare parts cannot be determined without a reliability analysis of the vehicle system and its major subsystems and components. Pending this analysis, one can approach the matter parametrically, by allocating a certain weight percentage of the subsystem (e.g. LSS, electric system, communication system, data processing system, etc.) as spare part weight. The spare part weight is a maximum at Earth departure. Before and after each major maneuver, a certain unused portion of this weight can be eliminated. In the first approximation a percentage of each module (M-1 through M-4 plus LSS) can be allocated as spare part weight. Let this percentage be s. This percentage will increase fairly rapidly with the mission time. The reason for this is not only that a piece of equipment of given lifetime to (repairable) failure needs twice as many spare parts if it has to serve twice as long; moreover, as the mission time, say, doubles, the mean-life-to-failure (repairable) of other components is exhausted, whence spare parts or spare modules must be added which (for a given mission probability) did not have to be taken along on the shorter trip.

The vehicle system can be thought of as consisting of N principal subsystems (or modules) of reliability R_{SS}. The product of all subsystem reliabilities is the probability of mission success of the (non-redundant) vehicle. Each subsystem has Z components; each component consists of z parts. The addition of

spare parts

spare components

repair tools and equipment ("workshop")

and the provision of

alternate modes of subsystem or component operation

all raise the probability of mission success of the individual vehicle. The first three cost weight, the fourth approach may cost little or no weight but requires more ingenuity and awareness throughout the design and development period, of the literally vital importance of reliability. Since the fourth approach depends very much on the vehicle type and design details, it does not lend itself to a parametric approach. The reliability R_{SS} of the individual subsystem which contains Z components any one of which could fail, is strongly dependent upon the mission time T (or that number of hours during which the subsystem is used, if it is to be jettisoned prior to mission completion) and upon the average component failure rate F,

$$R_{SS} = e^{-Z\,FT} \tag{6-1}$$

and

$$F = \sum zf/Z \tag{6-2}$$

where z is the number of parts in each component used and f is the component failure rate. Eq. (6-1) clearly shows that, no matter how small F, if T is only long enough, R_{ss} will fall below an acceptable level, requiring spare parts or spare components which are not needed if T is short. Because of the large number of subsystems and the rapidly snowballing number of components and parts in the interplanetary vehicle, the possible seriousness of more or less unlimited growth of T on the spare part weights required to stay above a given mission success probability threshold must be recognized and taken into consideration with longer mission duration.

Since the vehicle is manned, the possibility of failure detection and maintenance must be taken into account. If R_V is the effective vehicle reliability (mission success probability), R_o is the reliability of the non-detectable portion of failures in the vehicle, arising from checkout equipment failure and limitations in locating and distributing checkout equipment, then,

$$R_V = R_o \left[\sum_{i=o}^{N-j} \binom{N}{i} (R_{SS})^{N-i} (1 - R_{ss})^i \right]^m \tag{6-3}$$

N = one plus the number of spares available for each subsystem

j = number of subsystems which must be working for successful vehicle operation. j = 1 for the non-redundant vehicle.

R_{ss} = reliability of the detectable portion of subsystem failures

m = number of replaceable modules or subsystems into which the vehicle is packaged.

$$R_o = e^{-(1-D)FT} \tag{6-4}$$

D = fraction of failures which are detectable and can be isolated on the subsystem or module level

F = failure rate of the module or subsystem

T = mission time (hours)

The value of R_{ss} is

$$R_{ss} = e^{-DFT/m} \tag{6-5}$$

For the case of either a replacement or an alternate mode (until failure has been repaired) for each subsystem or module, the effective vehicle probability of mission success becomes

$$R_{V,1} = R_o \left[\sum_{i=o}^{2-1} \binom{2}{i} (R_{SS})^{2-i} (1 - R_{ss})^i \right]^m$$

$$= R_o \left[2 R_{ss} - R_{SS}^2 \right]^m \tag{6-6}$$

As an initial presentation of trends, Eq. (6-5) is used. This relation shows that, in order to maintain a given subsystem reliability R_{ss} at increasing T, m must increase,

$$\frac{R_{SS,2}}{R_{SS,1}} = \frac{e^{m\frac{DFT_1}{m_1}}}{e^{\frac{DFT_2}{m_2}}} = \exp\left(\frac{DFT_1}{m_1} - \frac{DFT_2}{m_2}\right) \tag{6-7}$$

$$\frac{DFT_1}{m_1} - \frac{DFT_2}{m_2} = \ln\left(\frac{R_{SS,2}}{R_{SS,1}}\right) \tag{6-8}$$

If the subsystem reliability is to remain constant, the right hand side of the above equation becomes zero and

$$\frac{m_2}{m_1} = \frac{T_2}{T_1} \tag{6-9}$$

showing that, for constant reliability the number of replaceable modules or subsystems is directly proportional to the mission time for all those subsystems or modules of equal or similar failure rate. Added to this must be those without spare parts whose reliability according to Eq. (6-1) drops below an acceptable level. However, taking only those increases defined by Eq. (6-9) shows that, if a 400-day (10,000-hr) mission requires an initial spare part weight of, say, 10 percent of the dry weight (about 1 percent of the gross weight), an 800-day (20,000-hr) mission would require at least 20% of the dry weight. Thus, if the weight W_{m1} of the m_1 replaceable modules is

$$W_{m1} = s\, W_{A1} \tag{6-10}$$

(e.g. s = 0.01), then

$$W_{m2} = s\, W_{A1} \frac{T_2}{T_1} \tag{6-11}$$