RWS

~~CONFIDENTIAL~~

Westinghouse Astronuclear Laboratory

SNPO-C

WANL-TME-2745
February 1971

MASTER

EVALUATION OF NSS DESIGN-MISSION INTERACTIONS (U)

~~CONFIDENTIAL~~
~~RESTRICTED DATA~~
~~Atomic Energy Act - 1954~~

DISCLAIMER

[redacted]

(31)

SUBCONTRACT N

WANL-TME-2745

February 1971

Westinghouse Astronuclear Laboratory

CODE IDENT NO. 14683

EVALUATION OF NSS DESIGN-MISSION INTERACTIONS (U)

Classification cancelled (~~or changed to~~)
by authority of H. A. C. TIC, date SEP 11 1973
by

APPROVED BY:

F.D. Retallick 2-26-71

F. D. Retallick, Manager   Date
NERVA Engineering

~~GROUP 1~~
~~EXCLUDED FROM AUTOMATIC DOWNGRADING AND DECLASSIFICATION~~

SPECIAL REREVIEW FINAL DETERMINATION
Class: U
Reviewer: KAW Class. U Date 4-30-82

INFORMATION CATEGORY
~~C-RD~~
R. G. Jones 3-9-71
AUTHORIZED CLASSIFIER   DATE

DISTRIBUTION OF THIS DOCUMENT UNLIMITED

[redacted]

PAGE BLANK

ABSTRACT

This report summarizes the work accomplished under Project 716, Mission Study Reactor Support, during CY 1970. Several analyses were made to evaluate the interaction of NSS design and mission requirements. These studies were related to the effect of cooldown temperature and throttling on propellant use, the endurance and comparative cost of operation of the three primary fuel element candidates, and the reactivity loss due to fuel burnup and fission product poisoning.

PAGE BLANK

TABLE OF CONTENTS

LIST OF TABLES

LIST OF FIGURES

1.0 INTRODUCTION

The studies performed under Project 716, Mission Study Reactor Support, during the period October 1, 1969 to September 30, 1970, were concerned with four primary areas of interest. The first two related to the interaction on propellant utilization of the cool-down temperature and throttling of the NSS. These analyses determined the effect on payload for several typical missions when cooldown temperature was varied and for throttled operation at different power levels and throttling times. The third study investigated the endurance and comparative costs of operation of the three primary fuel element candidates, graphite, composite and hybrid by determining the impact on cargo weight and cost per pound of cargo weight for Mission A-L-U. The final study was concerned with estimating the reactivity loss due to fuel burnup and fission product poisoning to aid in determining control drum span requirements. The calculation was made for Mission A-L-U.

The body of this report summarizes the results and conclusions of each of the four studies. The list of references, given at the end of the report, contain the calculational details of the various studies and the technical backup of the assumptions used in the analyses.

2.0 MISSION STUDY REACTOR SUPPORT

2.1 EFFECT OF COOLDOWN TEMPERATURE ON AVERAGE SPECIFIC IMPULSE AND PAYLOAD[1]

The purpose of this study was to provide data on the interaction of cooldown temperature with average specific impulse and payload. Two representative burns were considered: a low earth orbit to escape trajectory, and a 730-day Jupiter transfer trajectory. For both trajectories, it was assumed that the initial altitude was 608, 952 feet, the inertial velocity was 25, 573 ft/sec and the flight path angle with the local horizon was 0°. The initial vehicle weights were 263, 388 pounds for the earth escape trajectory and 332, 508 pounds for the Jupiter transfer trajectory. It was further assumed that thrust was in the direction of motion, the aftercooling pulses were compressed into a single sustained pulse, and that the stage weight was expressed by:

$$W_{stg} = 33,926 + 0.064\ W_p.$$

The data were obtained by using chamber pressure and temperature profiles during chilldown and bootstrap from Reference (2). The final startup portion was taken as middle of the map. The cooldown and aftercooling pulse temperature and pressure profiles were obtained from the PRECUR computer program[3]. The trajectory equations were solved by using the SG-GEM trajectory program [4](see Appendix 1).

CONCLUSIONS:

Increasing the cooldown temperature results in a payload gain and an increase in average specific impulse for all the cases considered. The payload gain is due to the fact that more decay heat is removed per pound of cooldown propellant with higher cooldown temperatures and, for a given full power burn time, less aftercooling propellant is required. This is partially offset by the fact that total impulse during cooldown decreases with higher chamber temperature. To make up for this lost impulse so that the final energy of the vehicle is the same, the full power burn time must increase which increases propellant consumption. In all the cases considered, the decrease in aftercoolant required was greater than the increase in propellant used during the full power burn.

The results of the calculations are shown in Tables 1 and 2. It can be seen that the greatest gains in payload and specific impulse with an increase in cooldown temperature occur when the cooldown temperature is relatively low (1000 to 1500°R). The gains become proportionately smaller as the cooldown temperature increases because the maximum payload gain cannot exceed the aftercooling propellant required.

The general trend of the results is similar for the throttling and no throttling cases. The differences in these cases are relatively small and tend to become even smaller as the cooldown temperature is increased.

2.2 EFFECT OF THROTTLING ON PAYLOAD (5)

The purpose of this study was to provide data on the interaction of throttling, for the hot bleed cycle, with payload for Missions A and B(6). Mission A is a round trip mission from a 100 nm circular parking orbit to a synchronous equatorial orbit. Mission B is a one-way trip from suborbit conditions to a synchronous equatorial orbit with terminal cooldown to disposal orbit.

In Mission A, the payload consists only of a manned reentry vehicle and chemical-rocket retro-propulsion sufficient to return that vehicle safely to the surface of the earth from a circular rendezvous orbit of 261.9 nm altitude. The total weight of the reentry vehicle, docking structure, adapter and chemical propulsion unit is 22,970 pounds.

In Mission B, a payload of about 101,000 lb. is delivered to a synchronous equatorial orbit.

All calculations were done as explained in Reference (7) except that the thrust and velocity vectors were not colinear for burns 2 and 3 of Mission A and burns 1 and 3 of Mission B. Except where noted, the assumptions and methods are the same as in the previous study.

TABLE 1

SUMMARY OF RESULTS FOR THE EFFECT OF COOLDOWN TEMPERATURE ON PAYLOAD AND AVERAGE SPECIFIC IMPULSE - EARTH ESCAPE

P_T	T_T	T_C	T_{FP}	W_{BO}	W_{AC}	W_{STG}	W_{PL}	ΔW_{PL}	$(I_{sp})_{AV}$
1.0	0	1000	930.0	169480.3	6682.7	39936.1	129544.2	-1270.7	782.6
1.0	0	1200	932.0	170674.6	5700.5	39859.7	130814.9	Base Case	791.4
1.0	0	1500	934.5	171796.1	4675.3	39787.9	132008.2	1193.3	800.0
1.0	0	2000	939.0	172920.9	3604.9	39715.9	133205.0	2390.1	809.2
1.0	0	2500	942.0	173611.1	2927.3	39671.7	133939.4	3124.5	814.7
1.0	0	3000	945.0	173996.8	2443.0	39647.0	134349.8	3534.9	818.2
0.445	120	1000	1000.3	170012.5	6652.2	39902.0	130110.5	-704.4	786.8
0.445	120	1200	1002.0	171031.5	5603.1	39836.0	131194.7	379.8	795.0
0.445	120	1500	1006.5	172002.2	4596.7	39774.7	132227.5	1412.6	802.9
0.445	120	2000	1008.5	173020.3	3585.2	39709.5	133310.8	2495.9	811.0
0.445	120	2500	1011.8	173660.4	2877.4	39668.6	133991.8	3176.9	815.9
0.445	120	3000	1013.8	174059.9	2378.3	39643.0	134416.9	3602.0	819.1

P_T = Throttling Power Fraction

T_T = Throttling Time (sec)

T_C = Cooldown Temperature (°R)

$(I_{sp})_{AV}$ = Average Specific Impulse (# F - sec/ # M)

T_{FP} = Time at Full Temperature (Sec)

W_{BO} = Burnout Weight (lb)

W_{AC} = Aftercooling Propellant (lb)

W_{STG} = Stage Weight (lb)

W_{PL} = Payload (lb)

ΔW_{PL} = Payload gain relative to Base Case (lb)

TABLE 1 (Con't)

P_T	T_T	$\left(\frac{\partial W_{PL}}{\partial I_{sp}}\right)_{AV}$	$\left(\frac{\partial W_{PL}}{\partial T_c}\right)_{AV}$	$\left(\frac{\partial I_{sp}}{\partial T_c}\right)_{AV}$
1.0	0	135	2.40	1.78×10^{-2}
0.445	120	133	2.15	1.61×10^{-2}

TABLE 2

SUMMARY OF RESULTS FOR THE EFFECT OF COOLDOWN TEMPERATURE ON PAYLOAD AND AVERAGE SPECIFIC IMPULSE 730-DAY JUPITER TRANSFER TRAJECTORY

P_T	T_T	T_C	T_{FP}	W_{BO}	W_{AC}	W_{STG}	W_{PL}	ΔW_{PL}	$(I_{sp})_{AV}$
1.0	0	1000	2100.0	125418.2	12364.6	47179.8	78238.4	-1933.7	793.2
1.0	0	1200	2105.5	127235.5	10506.3	47063.4	80172.1	Base Case	800.1
1.0	0	1500	2112.0	129184.3	8649.8	46938.7	82245.6	2073.5	807.5
1.0	0	2000	2116.5	131238.5	6772.1	46807.3	84431.2	4259.1	814.5
1.0	0	2500	2127.5	132124.4	5399.8	46750.6	85373.8	5201.7	819.0
1.0	0	3000	2135.5	133009.4	3934.3	46693.9	86315.5	6143.4	822.1
0.445	120	1000	2168.8	126038.7	12351.0	47140.0	78898.7	-1273.4	794.9
0.445	120	1200	2175.0	127814.2	10492.8	47026.0	80787.8	615.7	801.9
0.445	120	1500	2182.0	129670.1	8623.7	46907.6	82762.5	2590.4	808.9
0.445	120	2000	2190.8	131411.8	6652.9	46796.2	84615.6	4443.5	815.7
0.445	120	2500	2195.2	132451.8	5383.7	46729.6	85722.2	5550.1	819.6
0.445	120	3000	2203.5	133223.6	3893.3	46680.2	86543.4	6371.3	822.5

P_T = Throttling Power Fraction

T_T = Throttling Time (sec)

T_C = Cooldown Temperature (°R)

T_{FP} = Time at Full Temperature (sec)

W_{BO} = Burnout Weight (lb)

W_{AC} = Aftercooling Propellant (lb)

W_{STG} = Stage Weight (lb)

W_{PL} = Payload (lb)

ΔW_{PL} = Payload gain relative to Base Case (lb)

$(I_{sp})_{AV}$ = Average Specific Impulse $(\#_F - sec/\#_M)$

TABLE 2 (Con't)

P_T	T_T	$\left(\frac{\partial W_{PL}}{\partial I_{sp}}\right)_{AV}$	$\left(\frac{\partial W_{PL}}{\partial T_c}\right)_{AV}$	$\left(\frac{\partial I_{sp}}{\partial T_c}\right)_{AV}$
1.0	0	279	4.04	1.45×10^{-2}
0.445	120	277	3.82	1.38×10^{-2}

TABLE 3

PAYLOAD GAINS DUE TO THROTTLING - MISSION A

P_T	T_T	Burn No. 1	Burn No. 2	Burn No. 3	Burn No. 4	Total
270	100	530	705	315	550	2100
270	300	395	730	375	515	2015
270	500	240	615	330	470*	1655
200	100	570	850	345	630	2395
200	300	645	1120	450	720	2935
200	500	335	925	555	655	2470

P_T = Chamber Pressure during Throttling (psia)

T_T = Throttling Time (seconds)

Payload Gain in Pounds

* No full power burn, 468.5 seconds at throttled condition

TABLE 4

PAYLOAD GAINS DUE TO THROTTLING - MISSION B

P_T	T_T	Burn No. 1	Burn No. 2	Burn No. 3	Total
270	300	410	342	568	1320
270	600	388	261	620	1269
200	300	548	671	1068	2287
200	600	489	361	910	1760

P_T = Chamber pressure during throttling (psia)

T_T = Throttling time (seconds)

Payload gain in pounds

CONCLUSIONS:

(U) For all burns of both missions, throttling the engine resulted in a payload gain. Throttling to lower power levels resulted in greater payload gains for all burns of both missions. In most cases, the payload gains exhibited a maximum as throttling time was increased. The results of the study are shown in Tables 3 and 4.

2.3 FUEL COST EFFECTIVENESS STUDY[8]

(U) This study was performed in response to Item 26f of Change Order 70-8 which requested endurance and comparative costs of operation of the three primary fuel element candidates, graphite, composite, and hybrid.

(U) Mission A-L-U was chosen as the reference mission to apply the cost analysis. The Reusable Nuclear Shuttle (RNS) begins its trip from a circular earth orbit of 262 nautical miles (nm), 55° inclination, drops its cargo in a circular 60 nm lunar polar orbit, and returns to its original earth orbit. Stay time and ΔV's are dependent on lunar phasing. The entire mission is simulated by four burns with average ideal impulsive ΔV's. It is assumed that the velocity increments are 10,975 ft/sec for leaving earth, 2935 ft/sec for arriving at the moon, 4555 ft/sec for leaving lunar orbit and 10235 ft/sec for arriving at earth. There are no plane changes.

(U) All missions are assumed to start with a fully loaded propellant tank of 300,000 lbs. A 3.81% contingency or reserve is required so that the engine burns exactly 288,560 lbs. on each round trip at a thrust of 75,000 lbs. and a specific impulse of 822 seconds.

(U) The dry nuclear stage, including instrument unit and a 4,000 lb. disc shield, weighs 45,225 lbs. without the engine. The engine weight with a graphite core is 23,500 lbs. The expendables such as the reaction control system (RCS) and power supply weigh 8,200 lbs. Wet stage return weights is 80,165 lbs. for the graphite core engine, not including any return payload.

(C-RD) Fuel element weights for the three candidate cores are as follows:[9]

Graphite	3,354	lbs.
Composite	5,360	lbs.
Hybrid	4,010	lbs.

(U) The calculation for cargo weight is iterative since the cooldown propellant depends upon burn time and burn time depends upon cargo weight. The initial weight in earth orbit is assumed, the full power burn time of each burn is iterated until the sum of the impulse from full power and cooldown achieves the desired ΔV, and cargo weight is adjusted until exactly 288,560 pounds of propellant used. Shutdown / cooldown has a specific impulse of 475 seconds.

(U) The cost data were obtained from various sources and probably represent a "best guess" rather than hard numbers. These are divided into three main categories:development, hardware, and orbital costs. The non-recurring development costs ranged from \$400,000,000[10] to \$1,900,000,000[11] for the stage. Similar numbers are estimated for the engine. A consensus average for both the engine and the stage is \$900,000,000. Fuel element development and testing was estimated at \$80,000,000[9]. The above three development costs are amortized over the appropriate number of engines, cores, and stages which can complete at least 500 missions[12] in the 1980-1995 time period.

(U) The stage manufacturing and assembly (M and A) charges were \$26,500,000[13] Reference 12 quotes \$19,800,000. Engine M and A costs ranged from \$10,000,000 to \$20,000,000. The price of \$13,000,000 was used by the three potential stage contractors[14, 15, 16] and is used herein. Fuel element costs were taken from Reference 17. Since 11 of 25 months of the schedule are developmental, labor costs were reduced by this ratio. The net dollars for a core of elements are:

Graphite	\$6,889,395
Composite	\$6,354,570
Hybrid	\$7,503,080

It was assumed that the stage and capsule would be orbited together by a Saturn 5 type vehicle with launch costs of \$1,000/lb[18]. Furthermore, the engine, cargo, and propellant could be placed in orbit by the Earth-to-Orbit Shuttle (EOS) for \$100/lb.[18].

Purchase price on propellant is \$0.25/lb[12], on capsule is \$280/lb[11], and on cargo is \$25/lb.

Finally, \$20,000,000 per mission was alotted for total ground activity[19] (range time, tracking, etc.).

Total costs were calculated on the basis of one complete stage lifetime. The number of engines replaced during that lifetime is a specified parameter called L. Values of 2 and 20 engines per stage life were selected for illustration.

The maximum number of missions that each engine is able to fly is dependent on fuel endurance. The expected characteristics of the qualification fuel of 1976 were used herein; namely, 4-hour graphite fuel, and 10-hour composite or hybrid fuel[9]. The total mission burn time is known from the weight calculations and thus the maximum number of missions per engine is known. From this, the required number of engines and stages for a 500 mission program is calculated and development costs are linearly amortized. The unit cargo cost for M missions is defined as,

$$\text{Cost per lb. cargo} = \text{Total cost}/M \times \text{cargo weight} \times (1 + \alpha)$$

where α is an independent parameter for the ratio of cargo weight returned to earth from lunar orbit to cargo weight delivered to lunar orbit. This gives an average cost over the whole length of the program. (Cargo and payload are used synonymously).

The same approach can be taken in a stepwise fashion, calculating and averaging incremental costs for each mission. This results in a sawtooth curve with the jumps in unit cargo cost occurring at points when a new stage or engine is added. As the number of missions approaches 500, the unit cargo cost becomes a continuous curve because the economy of using the RNS for more missions overrides the incremental increase in cost incurred by adding one engine or stage to the system.

CONCLUSIONS:

A number of general conclusions which can be derived from this study are given below:

The lighter weight graphite core can deliver up to 5,700 lbs. more cargo than composite and up to 1,900 lbs. more than hybrid.

Cargo delivery costs are only slightly dependent on the ratio of engine life to stage life, approximately a 10% reduction for a factor of 10 increase.

It is important to have full utilization of the core life. Unit delivery costs decrease nearly a factor of two for small stage lifetimes and a factor of up to 1.2 for longer stage lifetimes, when comparing an engine used only once with an engine used to its full lifetime.

Unit cargo delivery costs to lunar orbit with no return cargo range from \$501/lb. to \$571/lb. with hybrid being the most cost effective.

Fuel costs range from 1-10% of the total program costs depending on their use.

It is 1.4 times as expensive to return cargo from lunar orbit (zero outbound payload) as it is to deliver it to lunar orbit with zero earth return payload.

The ground support costs represent a substantial portion (~ 25%) of the total mission costs.

This study should be continued to evaluate the cost effectiveness of other chamber temperatures and fuel endurance capability.

The delivery costs and fuel costs given in this report should be used in conjunction with failure rates of the individual fuels to produce an overall cost effectiveness.

Figures 1 through 6 summarize the results of the study.

2.4 FUEL BURNUP AND FISSION PRODUCT POISONING [20]

The purpose of this study was to estimate the reactivity loss due to fuel burnup and fission product poisoning to aid in determining control drum span requirements.

The Xenon-135 reactivity loss was calculated at 5 cents based on the maximum Xenon buildup encountered during Mission A-L-M as shown in Table 5. The natural radioactive decay of Xenon-135 precludes any residual effect from buildup previous missions.

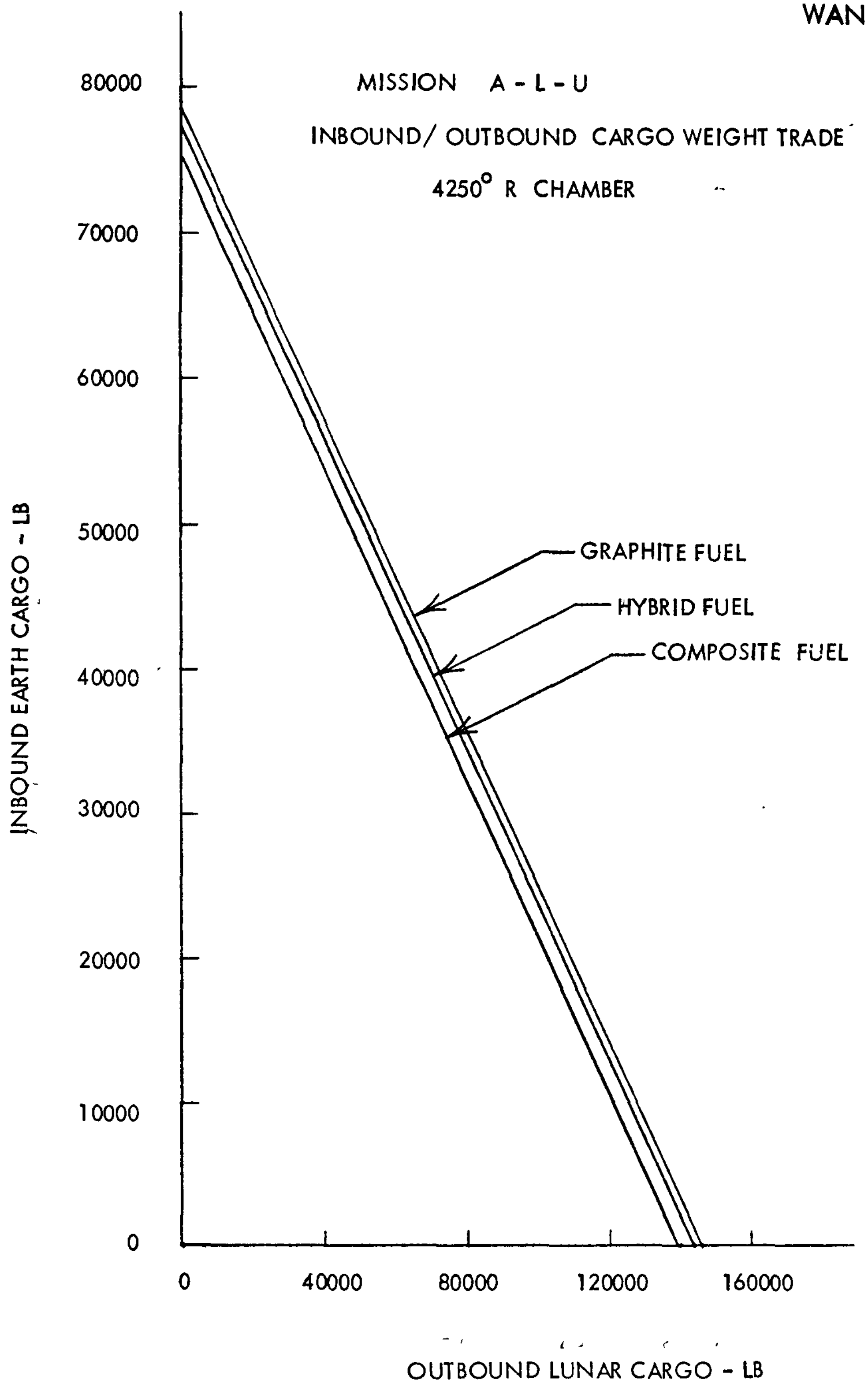

Figure 1. Inbound/Outbound Cargo Weight Trade At Design Rated Conditions for Graphite, Composite and Hybrid Cores

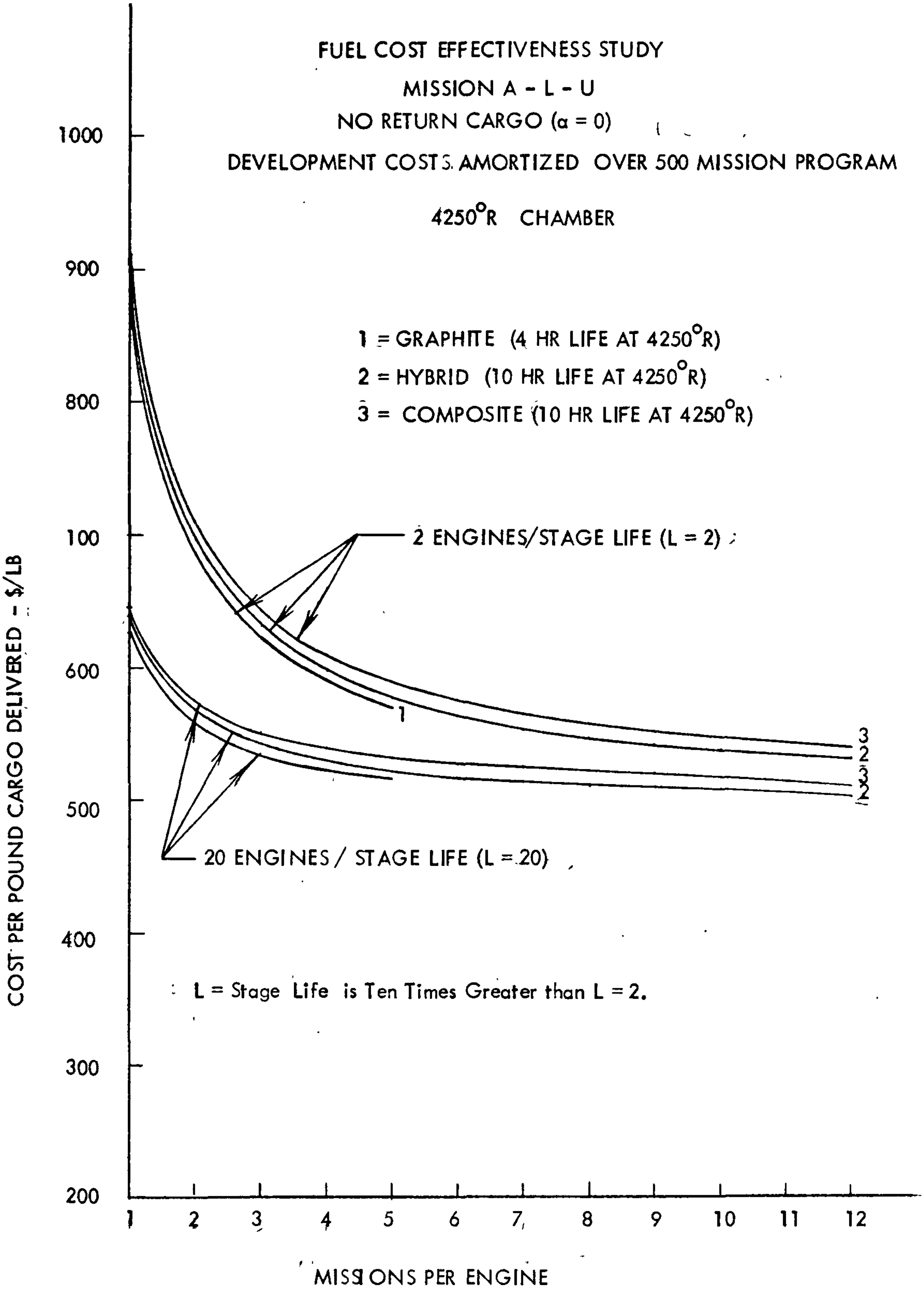

Figure 2. Unit Cargo Delivery Cost vs. Mission per Engine for No Earth Return Cargo

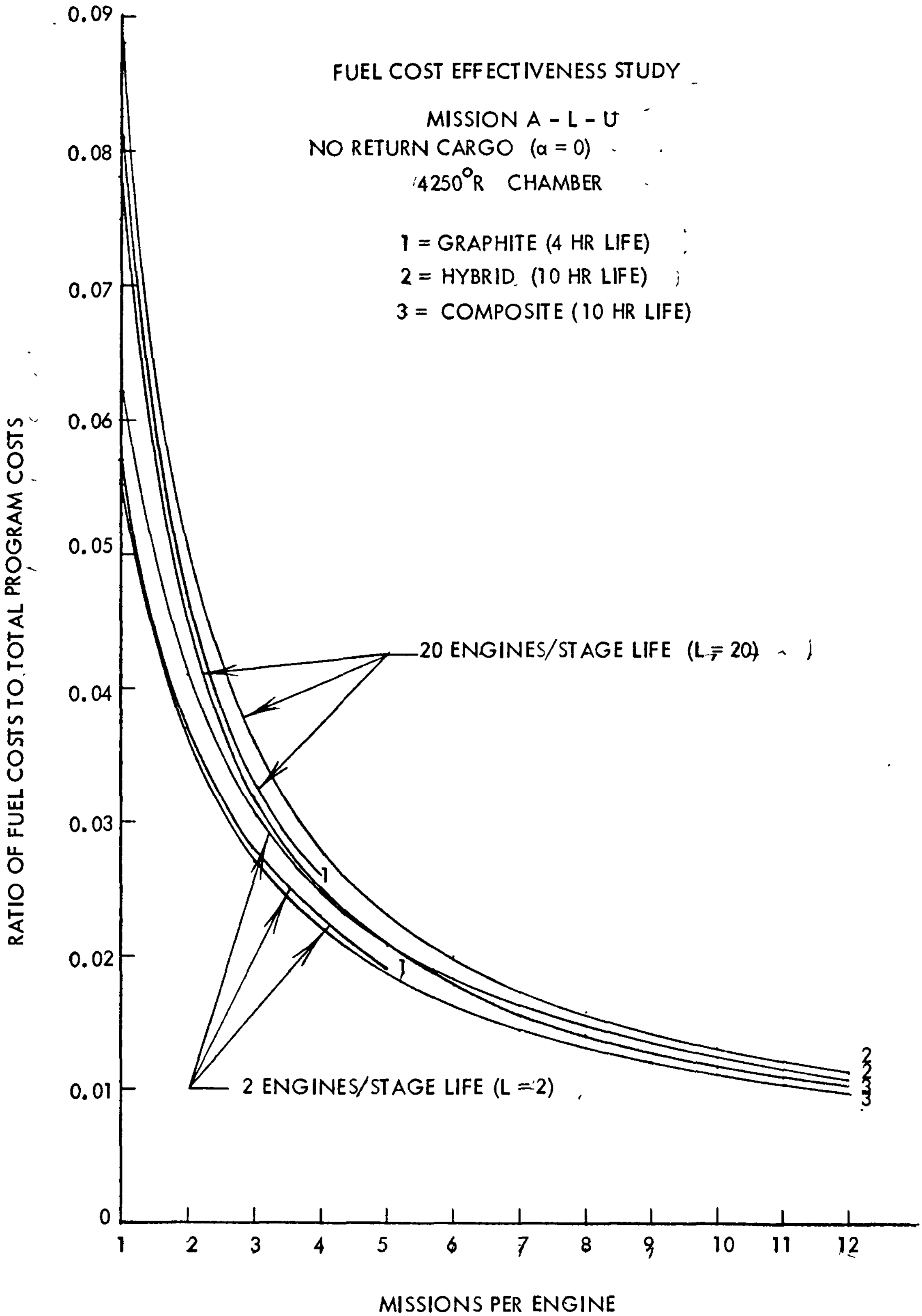

Figure 3. Ratio of Fuel Cost to Total Program Cost vs. Missions per Engine for No Earth Return Cargo

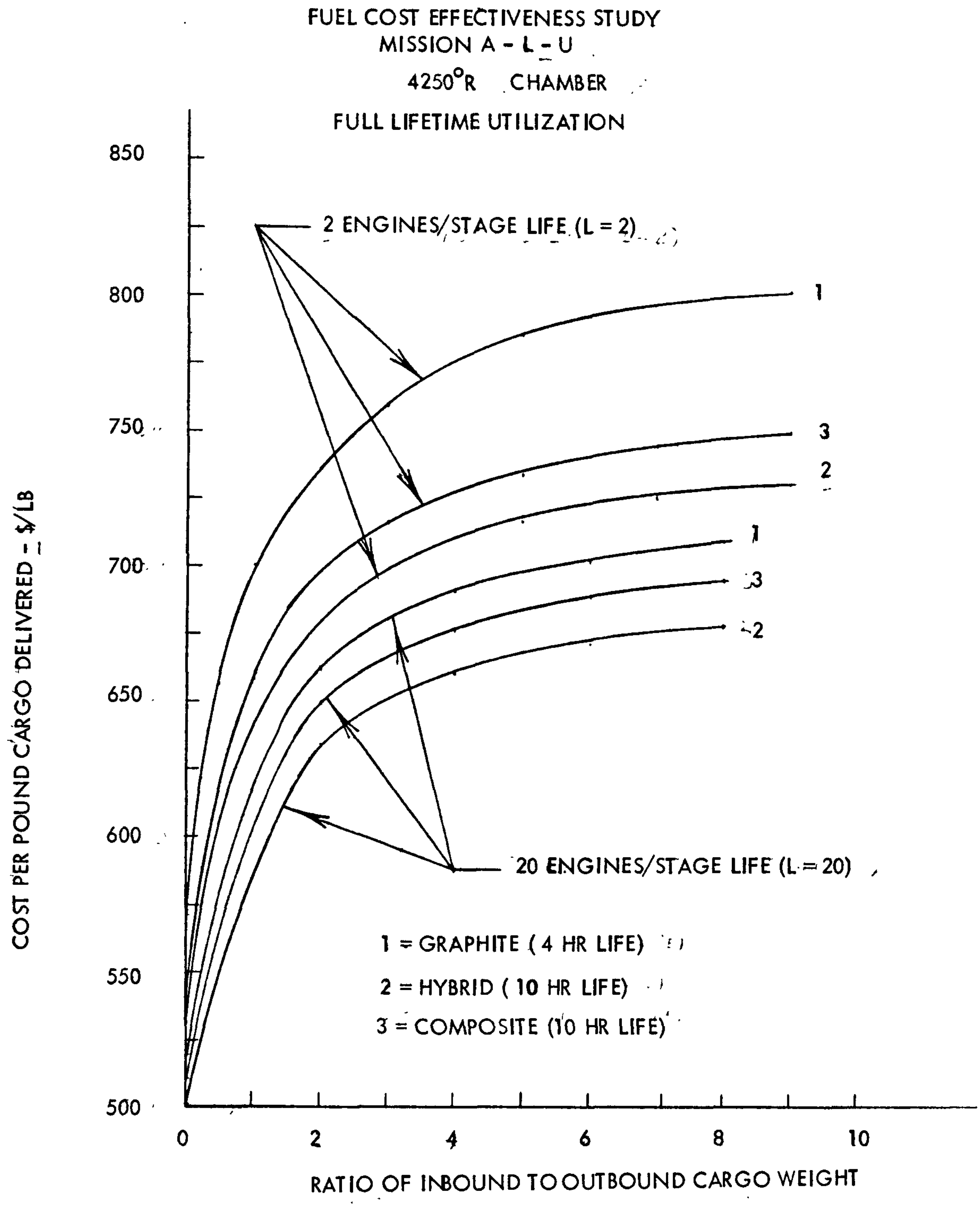

Figure 4. Unit Cargo Delivery Cost vs. Ratio of Inbound to Outbound Cargo Weight

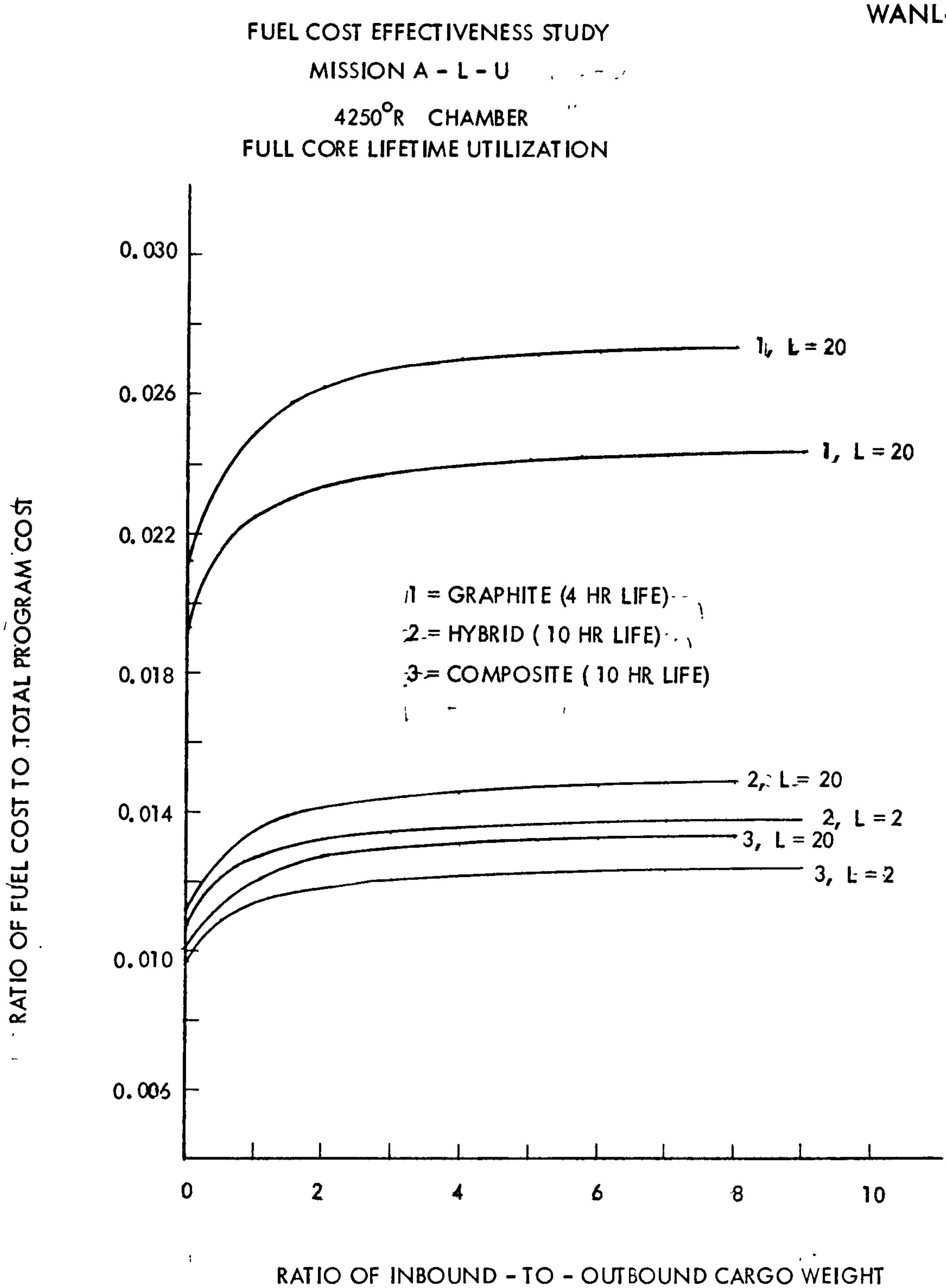

Figure 5. Ratio of Fuel Cost to Total Program Cost vs. Ratio of Inbound to Outbound Cargo Weight

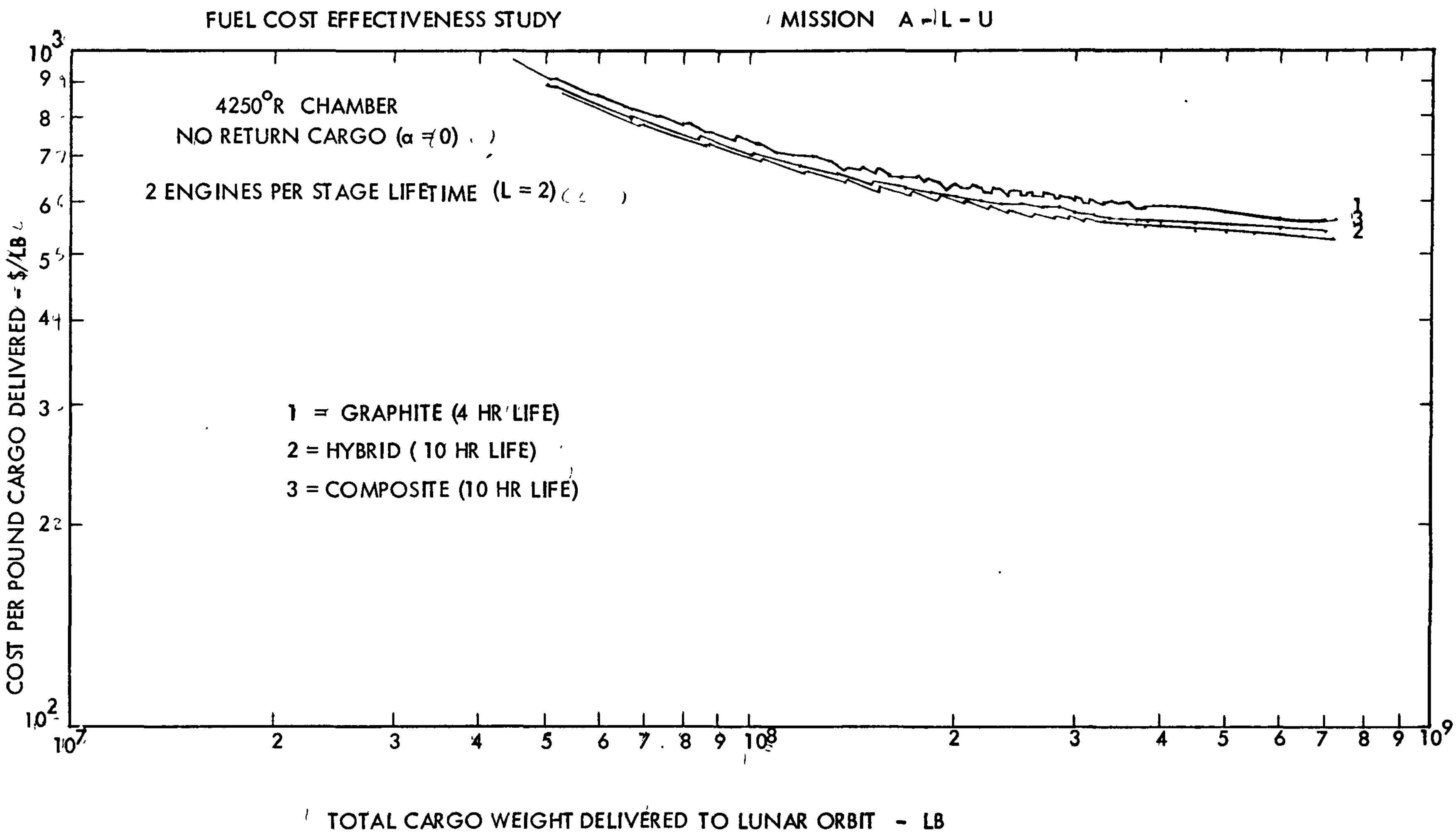

Figure 6. Average Unit Cargo Delivery Cost vs. Total Cargo Weight Delivered to Lunar Orbit

Unlike Xenon-135, the reactivity effects of fuel burnup and stable fission product buildup (e.g., Sm-149) are cumulative throughout reactor operation. The reactivity loss from these effects has been estimated at 20 cents for a 10-hour full rated power operation apportioned as follows:

1. A 13-cent loss due to fuel burnup of 1.05 kg
2. A 6-cent loss due to Samarium-149 buildup
3. A 1-cent loss due to the combined effect of all other stable fission products combined

CONCLUSION:

If Mission A-L-M is the final mission of a 10-hour reactor operating history, a total reactivity loss of 25 cents would be expected.

Table 5

REACTIVITY LOSS DUE TO XENON-135

MISSION A - LUNAR - MANNED

Burn	Steady State Min	Throttling Min	Shutdown Min	(Coast) Cooldown Thrust Null Hours	Xenon Poisoning* At End of Burn	Xenon Poisoning* At End of Coast
1	24.900	0.556	3.210	71.5	3.4	0.9
2	1.980	0.556	1.830	7.7	1.2	3.3
3	1.230	0.556	1.730	7.7	3.4	5.0
4	2.220	0.556	1.850	30.5	5.1	2.8
5	1.930	0.556	1.820	7.7	2.9	4.4
6	0.818	0.556	1.670	7.7	4.5	5.2
7	1.170	0.556	1.725	71.7	5.2	0.1
8	10.000	0.556	2.540	40.3	1.3	3.3

* Xenon poisoning given in cents of reactivity

3.0 APPENDIX 1

THE SG-GEM PROGRAM[4] [21]

The SG-GEM Program is a generalized trajectory computational system that consists of four simulations or packages. Each of the packages in turn contains one or more trajectory programs. The user specifies, by means of input data, which trajectory program within the chosen package is to be executed. The four packages available are:

1. Two-Dimensional Space Simulation
   a. Three Degrees of Freedom
   b. Quasi-Three Degrees of Freedom
   c. Two-Dimensional Commanded Attitude
   d. Two Degrees of Freedom (Particle)
2. Three-Dimensional Space Simulation
   a. Six Degrees of Freedom
   b. Quasi-Six Degrees of Freedom
   c. Three-Dimensional Commanded Attitude
   d. Three Degrees of Freedom (Particle)
3. Multi-Vehicle Simulation

   Three-Dimensional Commanded Attitude
   (Up to Four Vehicles)
4. Interplanetary Simulation

   Three Degrees of Freedom (Includes perturbations from planets, sun, and moon)

The SG-GEM Program permits the running of simulations with any of the above subprograms in any of three available modes viz. normal mode, parameter mode, and boundary value mode. In addition, various output formats are optional in all modes.

The normal mode is by far the one most frequently used. A program operating in the normal mode reads data from the input device, integrates the equations of motion to a specified end condition, and returns to read more data. This sequence of reading data and integrating to termination is called a phase. A case consists of one or more phases, and a run consists of one or more cases.

The parameter mode allows the program to return to an arbitrary point in the normal trajectory, adjust any single input parameter to one or more preset values, and repeat the remaining phases in a case for each value. An option is available to allow linear interpolation and/or extrapolation on an input parameter to reach a given end condition. Two choices are available for the interpolation; either the program will run a series of cases with the input parameter set to given values until the end condition is bracketed, or one step-off value of the input parameter may be run with interpolation and/or extrapolation starting immediately.

The boundary value mode allows the program to return to an arbitrary point in the normal trajectory and adjust up to six initial boundary parameters in order to satisfy the required end conditons for a given trajectory.

Most trajectory simulations involve the numerical integration of functions which are, at best, piece-wise continuous, the discontinuities resulting from such things as abrupt mass changes due to the release of spent motor cases, rocket motor ignition or shut-off, etc. Unless such discontinuities are handled properly, large errors may be introduced if they occur within a single integration step. By subdividing the computation into a number of phases such that no discontinuity occurs within a phase, this problem is eliminated.

At the beginning of each run or phase within a run, it is necessary to specify the initial conditions as well as the condition which determines when that phase ends. Termination of a phase may be accomplished when any quantity computed by the program reaches a prespecified value.

The basic input scheme to start a run, in addition to the mode, simulation and initialization control cards, involves specifying the (1) initial velocity, altitude and flight path angle, (2) position in space, (3) body geometry, (4) body orientation and angular orientation rates, (5) reference body weights and aerodynamic areas and lengths, (6) option control to select the types of aerodynamic calculations, atmosphere models, etc., (7) output quantities and output format, (8) frequency of output, (9) error control parameters, and (10) various control cards to handle the tabular input data.

The evaluation of aerodynamic coefficients, rocket parameters, and any physical parameter such as hypersonic shock wave characteristics is performed by numerical interpolation of tabular data.

Several options are available for specifying position in space, velocity, and body orientation. Output is completely arbitrary in that any quantity computed by the subprogram can be printed.

Due to the flexibility of the program, the input data scheme is inherently somewhat complex. WANL has incorporated a new subprogram, AUDIT, into the SG-GEM program which reads all tabular and non-tabular input data, checks it and denotes the occurrence of any errors, and as part of the output lists all the input data together with appropriate error messages. If input data errors are detected, the AUDIT program will pass over that particular problem and go to the next problem.

4.0 REFERENCES

1. H. J. Stumpf, "Effect of Cooldown Temperature on Average Specific Impulse and Payload for an Earth Escape Trajectory and a Jupiter Transfer Trajectory," DRM-51427, November 18, 1969

2. ANSC Report S054-CP-090194, "Estimated Startup Schedule for 75K NERVA Engine."

3. B. L. Pierce, "User Instructions for PRECUR Program," WANL Systems Analysis Memo SA:5336, February 25, 1970

4. General Electric Report 61 SD 170, "Missile and Satellite Systems Program for the IBM 7090."

5. H. J. Stumpf, "Effect of Engine Throttling on Payload for the Hot Bleed Cycle - Mission A and Mission B," DRM-51434, November 17, 1969

6. ANSC Report RN-S-0506, "NERVA Mission Data Book," February 1969

7. H. J. Stumpf, "Status of Throttling Point and Time Trade Study," DRM-51404, April 10, 1969

8. D. L. Black, "Fuel Cost Effectiveness for Mission A-L-U for Graphite, Hybrid, and Composite Fuel," DRM-52769, March 2, 1970

9. D. C. Thompson, private communication

10. Paper by Sigmund Gronich attached to: R. E. Thompson, "AIF Space Committee Meeting Report," TO:RET:105, 1 July 1970

11. IBM, "Astrionic System Optimization and Modular Astrionics for NASA Mission After 1974," IBM No. 70-238-0001, 14 January 1970

12. "ANSC Maintainability T.S. 1003," February 24-27, 1970 AGC/WANL System Trade Study Meeting Agenda and Vu Graphs, DRM-52357, 11 March 1970

13. Lockheed Missiles & Space Co., "Nuclear Flight Definition Study Second Interim Briefing," LMSC-A965254, 29 January 1970

14. McDonnell-Douglas Corp., "Phase II Final Briefing, Reusable Nuclear Shuttle Concepts," MDC G0594, May 1970

REFERENCES (Continued)

15. NAR, "Space Division, Nuclear Flight System Definition Study, Final Phase II Review," PDS70-223, 19 May 1970

16. Lockheed Missiles & Space Co., "Nuclear Flight Systems Definition Study, Final Briefing," LMSC-A968323, 19 May 1970

17. G. R. Kilp, "Trade Study Cost Estimates," WNCO-(FE)-0741, 19 May 1970

18. ANSC Throttling Trade Study 026, "February 24-27, 1970 AGC/WANL System Trade Study Meeting Agenda and Vu Graphs," DRM-52357, 11 March 1970

19. David S. Gabriel, "The Nuclear Rocket and the Space Future Atomic Industrial Forum," 2 December 1969

20. D. W. Call, "Fuel Burnup and Fission Product Poisoning," WANL Nuclear and Radiation Design Memo NRD-70-93, March 17, 1970

21. H. G. Hargrove, "A Progress Report on the Development of Reentry Body Analysis Techniques," WANL-TMI-1921," July 29, 1968

Astronuclear
Laboratory

TITLE EVALUATION OF NSS DESIGN -
MISSION INTERATIONS (U)

DOCUMENT NO.
TME 2745

DATE
March 1, 1971

ABSTRACT This report summarizes the work accomplished under Project 716, Mission Study Reactor Support, during CY 1970. Several analyses were made to evaluate the interaction of NSS design and mission requirements. These studies were related to the effect of cooldown temperature and throttling on propellant use, the endurance and comparative cost of operation of the three primary fuel element candidates, and the reactivity loss due to fuel burnup and fission product poisoning. (U)

Fold
or Cut

Fold
or Cut

DATA ITEM NO

SECURITY CLASSIFICATION
OF DOCUMENT [redacted]

CDRL LINE NO

TOTAL NO OF PAGES
31

KEY WORDS Mission, Cooldown, Fuel Element Cost Effectiveness, Throttling, Fuel Burnup and Fission Product Poisoning, Specific Impulse and Payload. (U)

AUTHOR/EDITOR
H. J. Stumpf/ D. L. Black

PREPARING DEPT
Systems Analysis

W Form 59943 DOCUMENT INFORMATION SUMMARY

(THIS PAGE IS UNCLASSIFIED)