NASA Project Apollo Working Paper No. 1041

PROJECT APOLLO

THE APOLLO AUXILIARY POWER SUPPLY SYSTEM

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION

MANNED SPACECRAFT CENTER

Langley Air Force Base, Va.

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Prepared by: Robert N. Parker

Robert N. Parker
AST, Flight Systems

for Alfred B. Eickmeier (secy)

Preston T. Maxwell
AST, Flight Systems

Authorized for Distribution:

for Paul E Purser

Robert R. Gilruth, Director

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THE APOLLO AUXILIARY POWER SUPPLY SYSTEM

SUMMARY

This paper documents the basic evaluations and considerations which favor the selection of a full-time fuel-cell system for the Apollo power supply. A comparative analysis is made of a full-time fuel-cell system with two additional systems consisting of combinations of fuel cells and solar cells. The analysis considers concepts such as redundancy, crew participation, mission and environmental restraints, vehicle integration, and available trade-offs. Representative electrical-power profiles are presented, including peak-power levels and emergency-power levels. The Apollo power supply is examined for the earth-orbital, circumlunar, lunar-orbiting, and lunar-landing mission. The selection of a full-time fuel-cell system may or may not offer comparative weight advantages, depending upon the final trade-offs utilized. However, the weight penalties involved are of small consequence when compared to the inherent advantages of utilizing a full-time fuel-cell system for all phases of the Apollo mission.

INTRODUCTION

Earlier assessments (ref. 1) of energy conversion technologies led to a general conclusion that the following systems merited the strongest considerations for the Apollo auxiliary power system:

1. Hydrogen-oxygen fuel cell

2. Hydrogen-oxygen internal combustion engine

3. Silicon solar cell

The above study focused primarily on state of the art and weight comparisons of auxiliary power systems per se; and avoided final considerations attendant to actual spacecraft applications. This restricted approach resulted in an early identification of promising conversion technologies and their related development requirements. The accelerated emphasis on the Apollo program and the establishment of lunar-landing goals alter these initial concepts and precipitate the need for an immediate commitment to an auxiliary power supply system for the program.

Thus, final considerations of redundancy, mission restraints, trade-offs and spacecraft integration must be introduced and their relative merit must be established before a valid selection of a power supply may be made. The selection must also be sensitive to the power profile for the mission, where heat demands are compatible with overload allowances, and where emergency power demands are within capabilities of reserve of redundant components. Examination must be made of all contemplated phases of the Apollo mission for identification of the controlling parameters.

Within this framework, the selection of an auxiliary power supply must be made, a selection which cannot be committed to singular considerations.

DISCUSSION

Electrical Power Profile

Nominal mission.- Estimated electrical power requirements for the nominal, 14-day, lunar-landing mission are tabulated in table I and are depicted graphically in figures 1, 2, and 3. From this information, certain conclusions have been drawn, which serve to establish the basic power-generating requirements for the Apollo electrical power system. These conclusions are summarized as follows:

(a) The total electrical energy requirement for the nominal, 14-day, lunar-landing mission will be approximately 500,000 watt-hours.

(b) The power system must be capable of supplying an average load of approximately 1,500 watts continuously throughout the mission, with intermittent operation at loads up to 2,400 watts for relatively short periods (2 hours or less).

(c) The peak power requirement will be approximately 3,500 watts, occurring during lunar-landing operations for a total period of approximately 150 seconds.

(d) Electrical power requirement during the reentry landing and recovery phases of the mission will be approximately 8,700 watt-hours.

Emergency power.- Emergency electrical power requirements, i.e., the minimum electrical power level which would permit safe return of the spacecraft and occupants to earth from any point in the lunar-landing mission, are tabulated in table II, and depicted graphically in figure 4. The most severe circumstance, that of experiencing a major power system failure while on the lunar surface, was selected for the emergency power study. Under these conditions, it is assumed, of course, that the crew will curtail power use to the fullest possible extent and closely manage the utilization of electrical equipment to minimize peak power loads.

On this basis, it is concluded that the electrical power system must have these emergency capabilities:

(a) Supply a total of 48,000 watt-hours during the emergency trans-earth flight, of which approximately 1,100 watt-hours would be required during the reentry/recovery phase.

(b) Operate at a continuous load of approximately 600 watts throughout the flight. Occasional limited-duration peak loads above 600 watts could readily be absorbed by the reentry/recovery batteries, without necessitating recharging the batteries prior to reentry by utilizing their reserve.

Mission Restraints

Mission compatibility.- Having established an understanding for the electrical power profiles involved, the continued analysis must be directed toward tangible power systems. Before these comparative systems are defined, a brief discussion of specific mission restraints should be established. Such discussion will have direct bearing on the synthesis of composite power systems.

One major criterion for any auxiliary power system considered for Apollo should be the compatibility of that system with all phases of the Apollo program. The case of the lunar-landing mission injects stringent requirements upon the auxiliary power system from both kinetic and thermodynamic areas.

Kinetic restraints.- Considering kinetic problems, a retrograde rocket descent to the lunar surface would impose certain deceleration and impact loads to the vehicle; while the high velocity impingement of the rocket blast on the lunar surface would produce sandblasting and dusting conditions. The alternative of using a retractable solar array would seriously reduce system reliability. In addition, design conflicts would be introduced, particularly where large area solar arrays would compete with thermal-control radiators and communications antennas for the limited external surface area of the space vehicle.

Thermal environment restraints.- The temperature-time variation of the lunar surface will influence the selection of a power system. The logical landing time, assuming the landing point is on the lunar equatorial plane, would be at lunar daybreak. This would compromise the requirement of light for visual observation with the rate of increase of the lunar surface temperature as related to the stay capability of the vehicle. A solar panel, in this situation, would require a differential tracking system relative to the vehicle. This greatly increases the complexity of the deployment mechanism. With the subsolar temperature approaching 250° F within 7 days after landing, reflective insulation must be applied to the back side of the solar-array panels in order to limit the increase in solar-cell temperatures. Since the efficiency of silicon solar cells decreases as a linear function of temperature, an oversize array would be required to compensate for the resultant loss of solar-cell output associated with the lunar temperature environment and may be larger than required for orbit cycling.

Thermodynamic restraints.- Examination must now be made of the auxiliary power systems contemplated and their heat rejection capability on the lunar surface. The hydrogen-oxygen fuel cell and the hydrogen-oxygen internal-combustion engine were previously identified. To this list, a more recent proposal of a thermally-integrated hydrogen expansion engine could be added. As established in reference 2, the thermal control system related to the man-environment interface, will demand specialized treatment for the low temperature heat rejection systems. Every reasonable effort should, therefore, be made to prevent the energy conversion losses of the auxiliary power system from appearing as waste heat in the low temperature heat rejection system. Of the three internal power systems introduced, an order of decreasing heat-rejection capability would exist as follows: A.thermally-integrated cryogenic-hydrogen expansion system, an open-cycle hydrogen-oxygen internal-combustion engine, an intermediate temperature hydrogen-oxygen fuel cell, and a low temperature hydrogen-oxygen fuel cell. However, further considerations are required before relative merits may be discerned for this factor. These considerations are discussed for the above systems as follows:

(a) Thermally-integrated hydrogen system.- While the all-hydrogen system promises adequate thermal capacity, it should be noted that its requirement for return heat loads is mandatory, rather than optimal. A question is, therefore, raised concerning system flexibility. The bulk storage of an all-hydrogen system would be difficult to accommodate in the Apollo vehicle, with 160 cubic feet of propellant storage required as contrasted to 26 cubic feet of propellant storage associated with an equivalent fuel-cell system. In order to realize an acceptable propellant-consumption rate, the hydrogen cycle must be initiated from a high pressure level. This in turn denotes the inability of the system to accept low-pressure boiloffs from other hydrogen storage systems, if existent. Water production would be available only in minute quantities, as associated with a catalytic burner incorporated in the heat regeneration loop.

(b) Hydrogen-oxygen internal combustion engine.- Heat rejection in the open-cycle hydrogen-oxygen internal combustion engine is accomplished by direct expansion of exhaust products into the space environment. An open-cycle heat engine is, therefore, sensitive to discharge pressures, rather than to radiative heat loads. However, where condensation of exhaust water is sought, a condensing radiator must be added. Generally, the optimum oxygen-hydrogen weight ratios for such engines occur at approximately 2 to 1 for dual stage concepts. The high excess of hydrogen gas in the exhaust stream, coupled with low total discharge pressures, results in a low partial pressure and subsequent low saturation temperatures for the exhaust water vapor. Hence, condensation of the water vapor would begin near 70° F and approach 32° F for major extraction of exhaust water. Efforts have been expended to adopt these engines to low-pressure hydrogen and oxygen boiloffs, conceivably available from propulsion systems. However, the advantages of using boiloffs are quickly dispelled by an unacceptable compromise of propellant consumption rates and system complexity. The propellant consumption rate is identified as the controlling factor, since there is no guarantee that propulsion system hydrogen-oxygen boiloffs will be available at the present time. Generally, for the oxygen-hydrogen internal combustion engines to approach the low propellant rates demonstrated for fuel cells, a component complexity is introduced which involves concepts such as small-displacement high-pressure cryogenic pumps, low-temperature hydrogen-gas compressors or other difficult concepts.

(c) Hydrogen-oxygen fuel cells.- At least six major industrial concerns are developing fuel cells which have been identified with possible applications toward the Apollo power supply. It will not be attempted in this paper to discern the technical points of emphasis of these cells, but will remain in a broad thermodynamic concept weighing heat rejection capabilities of the low-pressure hydrogen-oxygen fuel cell. All of these cells are receptive to low-pressure hydrogen and oxygen boiloffs, if available.

Concerning first a "Bacon-type" fuel cell operating near 500° F and 15 psia, heat rejection occurs in a wet hydrogen radiator-condenser loop. During operation at rated power, both hydrogen and excess water vapor are continually recirculated through the fuel cell and radiator-condenser loop. The ratio of these constituents is determined by the fact that the hydrogen will be saturated with water vapor at exit conditions of the condenser radiator. On one side of this balance, the mixture of hydrogen gas and residual water vapor enters the hydrogen compartment of the fuel cell at 100° F. The mixture is subsequently heated to 500° F as it passes over the hydrogen electrode. At the same time, the product water vapor from the over-all cell reaction is formed at the interface of the hydrogen and electrolyte within the porous electrode. The vapor so formed diffuses within the circulating mixture. Since the total pressure of the mixture entering and leaving the fuel cell (and also the condenser) remains unchanged, the relative increase in water vapor content of the mixture leaving the fuel cell and entering the condenser results in an increase in the partial pressure of the water vapor.

Two distinct zones of heat rejection are, therefore, implied: one in which heat is rejected as sensible heat of the total mixture, and one in which latent heat of condensation is predominant. By proper selection of the mixture ratios, over one-half of the total heat load can be rejected as sensible heat from 300° F to the dew point at approximately 135° F, with the remaining heat rejection occurring in a condenser section operated from 135° F to 100° F. The lower cutoff point is matched to the desired partial pressure of the return mixture to the fuel cell commensurate with a balanced water removal rate.

The significant capability noted for the intermediate temperature fuel-cell system is that the heat rejection loop approaches a self-sufficient condition when matched to the lunar thermal environment. Preliminary studies (ref. 2) indicate that the net radiative rejection of a vertical radiator panel, unshielded and operating on the lunar surface near the subsolar point, would rapidly decrease as the panel temperature diminished to 140° F. Below this temperature, either radiator shielding or evaporative cooling would be ultimately required. Since these lower temperatures are associated with the condensing section of the fuel-cell radiator operated at 15 psia, it is probable that an integration will be required of the condensing section with the lower temperature thermal control loop of the environmental control system. However, most of the load will be required for the latent heat in this temperature range. Consequently, the water so condensed may in turn be utilized in open-cycle expansion cooling. The net heat load imparted to the thermal control system, therefore, consists of sensible heat loads of the hydrogen, water vapor, and water mixture from 135° F to 100° F. This sensible heat range is not an excessive burden in that it represents approximately 5 percent of the total heat rejection load of the fuel cell.

Alternate methods of heat rejection are also possible for fuel-cell systems. These would include conductive heat loads to heat sinks, or direct venting of products leaving the cell. The latter method would be at the expense of prohibitive reactant consumption rates, and would preclude collection of product water. The possibility of increasing the total pressure of the cell also exists.

Operational temperatures of the low-temperature fuel cells are generally restricted by material and other limitations to temperatures from 100° F to 180° F. Considerable heat burdens would, therefore, be imposed during lunar stay on the low-temperature thermal control loop of the environmental control system. The effect of this burden will greatly increase the weight and complexity of the low-temperature thermal control system.

Prelaunch and launch restraints.- Discussions thus far have analyzed the restraints of the lunar-landing mission. Common to all missions are the considerations to be given to prelaunch and launch phases. Of prime importance in manned space flight is the capability of placing a power system in complete operation prior to launch. This enhances the crew availability during launch and early mission periods for more critical data assimilations. It also favors a reduction in abort situations attributed to power supply failures. A clear advantage exists for the chemical systems in this area.

In the specific applications of solar panels utilized for any mission involving earth escape, deployment of the panel would not be attempted until final acceleration periods are completed. Hence, failure of a solar panel to deploy would constitute an emergency return condition, which could be faced with minimum return times from approximately 5 hours to many days, depending on the spacecraft abort capability. An excessive reserve power capability would, therefore, be required in the supplementary systems.

Vehicle orientation.- Examination should be made of the demands a power system will place on vehicle orientation. A flat solar-cell panel would require ±10° solar alinement accuracy, whereas no demands are imposed by chemical systems. Applications of fuel cells would not impose any attitude-control propellant requirements and would offer orientation preferences to observation, communications, and navigation functions.

Selection of Auxiliary Power Systems for Further Study

Fuel cells.- In the foregoing discussion, power systems were discussed in relation to mission restraints. Of the various systems reviewed, the fuel-cell system emerges as a system endowed with considerable promise. This confidence is further warranted by examination of the state of the art of the intermediate, low-pressure hydrogen-oxygen Bacon-type fuel cell. In general, this particular fuel cell is considered to be in a more advanced state of development compared not only to other fuel cells, but also compared to the chemical dynamic systems discussed in this paper. The remaining development areas recognized for this fuel cell are associated more with system technology than with the energy-conversion method.

Solar cells.- Silicon solar cells should not be completely dismissed from further considerations. Although these solar cells rated unfavorably in certain mission restraints, many of these objections are overcome where a stored solar array is held in reserve as an emergency transearth power source backing up a fuel-cell system. Solar cells are also adaptive to the orbiting space laboratory and would enhance the longevity of these laboratories without resupply. The strongest advantage of solar cells is their own proven reliability in unmanned space probes.

Synthesis of power systems.- From the logic thus far established, two systems are conceived which should be carefully analyzed. System No. 1 consists of a full-time fuel-cell system with redundant components and tankage. System No. 2 consists of a full-time fuel-cell system backed up by an emergency solar-array panel. For general weight comparisons, a third system, System No. 3 is introduced consisting of solar-cell arrays for primary power, and fuel cells for supplemental power. Each system will be discussed separately, including primary power, reentry, post-landing and emergency power, and supplemental power (if required). Redundant components and trade-offs will be identified for each system. Specific features of placement, safety, system uniformity, and simplicity will be noted for each system. It is not the intent of this paper to establish a reliability figure for the systems studied. However, the concepts of redundancy introduced into the systems are wrought with the full awareness that the Apollo mission will require a high level of reliability for the systems.

Analysis of Auxiliary Power Systems

Basis for analysis.- The three systems to be compared are tabulated in tables III, IV, and V. An analysis is made of each system for four variations of the 14-day Apollo mission, consisting of earth-orbital, circumlunar, lunar-orbiting and lunar-landing. The format used allows differentiation of the system components into fixed or constant units, variable units, and redundant units. The sum of these entries yields a composite system weight without trade-offs. Weights of the electrical power distribution and control systems are not included.

Major trade-offs considered are the propellant boiloffs available from the hydrogen-oxygen lunar-landing engine (if utilized), and the production of water by the fuel cells. Water production is committed only to life systems requirements at 7 pounds per man-day. However, certain power systems will offer water surpluses which could be utilized for evaporative cooling. In such cases, the total water available is noted, but trade-offs are not extended beyond the 7 pounds per day. The trade-offs are reported separately as differential weights that may be discounted from the former totals. Both singular effects and multiple effects of these trade-offs are reported. Where applicable, earth-launch weights and lunar-launch weights are noted.

Although the power profiles indicate that an average power near 1.5 kilowatts should suffice for the lunar-landing mission, a continuous power level of 2 kilowatts is conservatively chosen for the system analysis. Emergency power, reentry power, and postlanding power for all systems are supplied by groups of silver-zinc batteries located in the command module. Such batteries are used basically as primary cells, but advantage is taken of their limited recharge capabilities.

The basis for analysis unique to the individual system is discussed for each of the three systems as follows:

System No. 1 - System No. 1 (table III) consists of a full-time fuel-cell system. The nominal 2 kilowatts of power is supplied by two regular fuel-cell units, plus one redundant fuel-cell unit, each rated at 1 kilowatt. The redundancy of the fuel-cell system external to the cells is increased to 100 percent. Tankage is considered initially without reductions from trade-offs. The 14-day propellant supply is designated as the mission tankage and is generally contained in dual tankage for both hydrogen and oxygen reactants. For earth-orbiting conditions, the loss of one tank would still allow ample time for earth return on the remaining tankage. For the lunar missions, an additional $3\frac{1}{2}$-days of propellant supply in auxiliary tankage is deemed necessary for redundant coverage during earth return. The entire fuel-cell system, including fuel-cell units, controls, radiators, and tankage is located in the service module. This enhances the safety aspect of the crew compartment (command module) and affords a reduced probability of micrometeor damage due to internal packaging. No crew maintenance other than remote switching operations, are anticipated for the fuel-cell system. The entire fuel-cell system is jettisoned as an integral part of the service module during earth reentry.

System No. 2 - System No. 2 (table IV) is almost identical with System No. 1. The basic difference is that System No. 2 utilizes a stored emergency solar-cell array for all lunar missions in lieu of the $3\frac{1}{2}$-day auxiliary fuel-cell propellant tankage backup. This solar panel is rated at the minimum power level of 600 watts and would not be deployed in a normal mission. The net effect of this exchange is to supplant redundant tankage with a redundant emergency-power source for the $3\frac{1}{2}$-day transearth return.

System No. 3 - System No. 3 (table V) is included primarily as a minimum weight reference. In the synthesis of this power system, all mission restraints were discarded and maximum utilization of solar energy was sought. For primary power, four silicon solar-cell panels are utilized, each panel measuring 7 feet by 7 feet. For supplemental power during shadow periods or random positions, the hydrogen-oxygen fuel cell (nonregenerative) was selected in lieu of secondary batteries. This affords a weight advantage, particularly for the earth-orbiting mission, in that the total weight of the fuel-cell system and tankage is less than the composite weight of the additional solar-array size required for battery charging, and the secondary batteries themselves. The solar array is, therefore, sized for an instantaneous output of 2 kilowatts. The same rating is applied to the fuel-cell system. No redundancy is included for the solar-cell array, other than nominal deterioration allowances. The fuel-cell conversion unit and accessories other than tankage remain identical to the applications of Systems No. 1 and No. 2. The reactants for the fuel cells are contained in mission tankage where reasonable quantities are involved. The $3\frac{1}{2}$-day redundant auxiliary tankage is included for all lunar missions.

Results of analysis.- Figure 5 summarizes in graphic form the weight results of this analysis. The systems considered are arranged in vertical columns, and the mission phases are arranged in horizontal columns. The summary also differentiates between the various trade-offs utilized, as established in tables III, IV, and V.

The graphic summary (fig. 5) clearly indicates that the full-time fuel-cell system (System No. 1) is dependent upon the application of trade-offs, before comparative or advantageous weights are realized. Where such a system is backed up with redundant fuel-cell units, accessories, and tankage, continuous water production can be relied upon and utilized in a conservative trade-off. Without this trade-off, the all-fuel-cell system will approach a composite weight of 1,450 pounds. With this trade-off, a weight reduction of 300 pounds can be made. The resulting 1,150-pound weight is approximately 200 pounds heavier than the minimum reference weight of the solar-cell fuel-cell system (System No. 3). These figures are for the lunar-landing mission. This 200-pound approach to the minimum reference weight also applies for the circumlunar and lunar-orbiting mission and reduces to 100 pounds for the earth-orbiting mission.

Concerning propellant boiloffs from other systems, a trade-off is recognized as being possible from the hydrogen-oxygen lunar-landing propulsion engine. If such a boiloff were available, then this trade-off could discount the full-time fuel-cell system by 500 pounds for maximum periods of lunar stay. Since this is an optimistic condition and due to the nebulous definition of the lunar-landing engine, this trade-off is indicated but is not a critical advantage required for justifying the selection of a full-time fuel-cell system.

It is also interesting to note that the full-time fuel-cell system, regardless of trade-offs applied, will represent a more favorable lunar take-off weight, due to reactant consumption prior to this event.

Analysis of the emergency power requirements indicates that two alternative emergency power sources offer promise, i.e., use of a reserve solar array, capable of deployment in an emergency, or redundant reactant tankage, sufficient to provide emergency power for the moon to earth return. No significant weight advantage is attached to either method and present information does not permit a firm choice. It is felt that final selection of an emergency power source should remain contingent on the spacecraft configuration, which will determine the feasibility of the reserve solar array.

Further studies will be required on cryogenic reactant supplies and on integrated radiator concepts. For the most part, weight values assigned to these areas represent an averaging of parameters. However, the revised weights resulting from such studies should not grossly affect the over-all system weights.

The auxiliary power system for the earth-orbiting space laboratory is not included in this analysis.

CONCLUSIONS AND RECOMMENDATIONS

A full-time fuel-cell system is concluded to be a reasonable choice for the auxiliary power supply for all phases of the Apollo mission. This conclusion is based on the compatibility of the system with various mission restraints, and the capability of the system to approach a favorable weight through conservative trade-off.

REFERENCES

1. Parker, Robert N: Some Considerations Toward the Selection of an Auxiliary Power Supply for Apollo. NASA Project Apollo Working Paper No. 1017, May 12, 1961.

2. Guy, Walter W.: Consideration Toward Design of a Space Radiator for Apollo. NASA Project Apollo Working Paper No. 1036, Dec. 29, 1961.

TABLE I.- ELECTRICAL POWER REQUIREMENTS

NOMINAL 14-DAY MISSION

Equipment utilizing electrical power	Power (watts)	Power (watt-hr)	Duty cycle or conditions of use
Communications and Instrumentations			
Nearfield Comm. System	3	30	10 hours (while on moon)
Intercom. System	12	4,032	100 percent
Telemetry System	80	1,600	During major maneuvers, 5 minutes each hour during transit. Total - 20 hours
Television	50	800	Intermittent - principally in vicinity of moon. Total - 16 hours
D. S. Comm. System			
Transponder	17	3,740	Continuous except near earth (8 hrs) and behind the moon. Total - 220 hours
Power amplifier	50	6,400	Turned on when 50 hours away from earth. Total use - 128 hours
Ranging unit	10	2,200	Continuous except near earth (8 hrs) and behind the moon. Total - 220 hours
VHF transmitter	10	200	20 hours near earth
VHF receiver	1	50	50 hours near earth
C-Band transponder	38	190	5 hours near earth
Minitrack beacon	1	20	20 hours near earth
Radar - altimeter	400	2,000	Earth and lunar landing. Total - 5 hours
Tape recorders (2)	60	1,200	Major maneuvers, reentry, behind the moon; intermittent during transit. Total - 2 hours
HF/VHF recovery beacon	12	864	Postlanding
Binary clock (2)	100	33,600	100 percent
General purpose camera	50	150	Major maneuvers, reentry, intermittent during midcourse transit. Total - 3 hours
Telescope-camera	150	450	Intermittent use, principally in vicinity of the moon. Total - 3 hours
Displays	200	67,200	100 percent
Fluorescent lighting	100	25,200	75 percent
Radiation detection equip.	50	16,800	100 percent
Scientific equipment	100	3,000	Vicinity of moon. Total - 30 hours
Propulsion System			
Reaction control	80 to 240	17	2-second pulse each 15 seconds during thrusting maneuvers. Infrequent 2-second pulse during transit
Vernier engines	80	21	Continuous during lunar launch. 10 20-second pulses during transit
Lunar landing engines	1,000	40	1 pulse of 120 seconds; 1 pulse of 30 seconds
Environmental Control System			
Catalytic burner	60	5,040	25 percent (20 min on, 60 min off)
Glycol pump	40	13,440	100 percent
Blower	125	42,000	100 percent
Fan	100	33,600	100 percent
Navigation and Guidance			
	350	117,600	100 percent
	250	10,000	20 periods of 2 hours
Miscellaneous			
Food preparation	200	4,000	Total - 20 hours
Electrical system losses	300	100,800	100 percent - principally comprised of losses in power conversion/inversion
Total	——	496,284	

TABLE II.- EMERGENCY ELECTRICAL POWER REQUIREMENTS

(MOON TO EARTH RETURN)

Equipment utilizing electrical power	Power (watts)	Power (watt-hr)	Duty cycle or conditions of use
Communications and Instrumentations			
Intercommunications System	3	115	40 percent
S-Band, tracking and voice	20	168	10 percent of time until within 8,000 miles of earth
VHF voice	10	5	8,000 miles to reentry
HF/VHF recovery beacon	12	864	Postlanding
Minitrack	3	2	8,000 miles to reentry
Displays	200	4,200	25 percent
Lighting	100	840	10 percent.
Propulsion System			
Reaction control	80 to 240	.17	2-second pulse each 15 seconds during lunar launch. Infrequent 2-second pulse during transit
Vernier engines	80	5	Continuous during lunar launch 10 20-second pulses during transit
Environmental Control System			
Catalytic burner	60	1,260	25 percent
Glycol pump	40	3,300	100 percent
Blower	125	10,500	100 percent
Fan	100	8,400	100 percent
Navigation and Guidance			
	50	4,200	100 percent
	250	1,050	5 percent 10 25-minute periods
	110	185	2 percent 10 10-minute periods
Miscellaneous and system losses	150	12,600	100 percent
Total		47,711	

TABLE III. - 14-DAY APOLLO MISSION

System No. 1

Item	Function	Earth orbital					Circumlunar (dual)					Lunar orbiting (8 days)					Lunar landing				
		Module	No. units	Capability kw	Capability days	Weight lb	Module	No. units	Capability kw	Capability days	Weight lb	Module	No. units	Capability kw	Capability days	Weight lb	Module	No units	Capability kw	Capability days	Weight lb
Constants																					
Fuel Cells	Primary Power	Service	2	2		170	Service	2	2		170	Service	2	2		170	Service	2	2		170
Radiator		Service	1			18	Service	1			18	Service	1			18	Service	1			18
Sys. Piping		Service	1			30	Service	1			30	Service	1			30	Service	1			30
Silver-Zinc Batteries	Reentry and Emergency Pwr.	Command	2	8 (kw-hr)		160	Command	2	8 (kw-hr)		160	Command	2	8 (kw-hr)		160	Command	2	8 (kw-hr)		160
				Subtotal		378			Subtotal		378			Subtotal		378			Subtotal		378
Variables																					
O_2 Reactant		Service		2	14	560	Service		2	14	560	Service		2	14	560	Service		2	14	560
O_2 Tankage		Service	2			56	Service	2			56	Service	2			56	Service	2			56
H_2 Reactant		Service		2	14	70	Service		2	14	70	Service		2	14	70	Service		2	14	70
H_2 Tankage		Service	2			35	Service	2			35	Service	2			35	Service	2			35
				Subtotal		721			Subtotal		721			Subtotal		721			Subtotal		721
Redundancy																					
Fuel Cell	50% Backup	Service	1	1		106	Service	1	1		106	Service	1	1		106	Service	1	1		106
Radiator	100% Backup	Service	1			18	Service	1			18	Service	1			18	Service	1			18
Sys. Piping	100% Backup	Service	1			30	Service	1			30	Service	1			30	Service	1			30
O_2 Reactant	3-1/2 day Back-up						Service	1	2	$3\frac{1}{2}$	140	Service	1	2	$3\frac{1}{2}$	140	Service	1	2	$3\frac{1}{2}$	140
O_2 Tankage							Service	1			14	Service	1			14	Service	1			14
H_2 Reactant	3-1/2 day Back-up						Service	1	2	$3\frac{1}{2}$	18	Service	1	2	$3\frac{1}{2}$	18	Service	1	2	$3\frac{1}{2}$	18
H_2 Tankage							Service	1			9	Service	1			9	Service	1			9
				Subtotal		154			Subtotal		335			Subtotal		335			Subtotal		335
Trade-off No. 1																					
Water Prod. (630 lb avail)	7 lb/man day Life Systems				14	-294				14	-294				14	-294				14	-294
				Subtotal		-294			Subtotal		-294			Subtotal		-294			Subtotal		-294
Trade-off No 2																					
H_2 Boiloff						Not avail.					Not avail.					Not avail.				$10\frac{1}{2}$	-51
O_2 Boiloff																				$10\frac{1}{2}$	-407
ΔH_2 Tank																					-25
ΔO_2 Tank																					-40
Weight summary																					
No trade-off																					
Earth Launch						1,253					1,434					1,434					1,434
Lunar Launch						-					-					-	$\frac{3\ 1/2}{10\ 1/2}$ Days depletion				1,281
																					974
With trade-off no. 1																					
Earth Launch						959					1,140					1,140					1,140
Lunar Launch						-					-					-					680
With trade-off no. 2																					
Earth Launch						-					-					-					911
Lunar Launch						-					-					-					911
With both trade-offs																					
Earth Launch						959					1,140					1,140					617
Lunar Launch						-					-					-					617

TABLE IV.- 14-DAY APOLLO MISSION

System No. 2

Item	Function	Earth orbital					Circumlunar (dual)					Lunar orbiting (8 days)					Lunar landing				
		Module	No. units	Capability kw	Capability days	Weight lb	Module	No. units	Capability kw	Capability days	Weight lb	Module	No. units	Capability kw	Capability days	Weight lb	Module	No. units	Capability kw	Capability days	Weight lb
Constants																					
Fuel Cells	Primary Power	Service	2	2		170	Service	2	2		170	Service	2	2		170	Service	2	2		170
Radiator		Service	1			18	Service	1			18	Service	1			18	Service	1			18
Sys. Piping		Service	1			30	Service	1			30	Service	1			30	Service	1			30
Silver-Zinc Batteries	Reentry and Emergency Pwr.	Command	2	8 (kw-hr)		160	Command	2	8 (kw-hr)		160	Command	2	8 (kw-hr)		160	Command	2	8 (kw-hr)		160
				Subtotal		378			Subtotal		378			Subtotal		378			Subtotal		378
Variables																					
O_2 Reactant		Service		2	14	560	Service		2	14	560	Service		2	14	560	Service		2	14	560
O_2 Tankage		Service	1			56	Service	1			56	Service	1			56	Service	1			56
H_2 Reactant		Service		2	14	70	Service		2	14	70	Service		2	14	70	Service		2	14	70
H_2 Tankage		Service	1			35	Service	1			35	Service	1			35	Service	1			35
				Subtotal		721			Subtotal		721			Subtotal		721			Subtotal		721
Redundancy																					
Fuel Cell	50% Redundancy	Service	1	1		106	Service	1	1		106	Service	1	1		106	Service	1	1		106
Radiator	100% Redundancy	Service	1			18	Service	1			18	Service	1			18	Service	1			18
Sys. Piping	100% Redundancy	Service	1			30	Service	1			30	Service	1			30	Service	1			30
Solar Cell Array	Emerg. Backup	Service	0			0	Service	1 or 2	.6		70	Service	1 or 2	.6		70	Service	1 or 2	.6		70
				Subtotal		134			Subtotal		224			Subtotal		224			Subtotal		224
Trade-off No. 1																					
Water Prod. (Avail)	7 lb/man day Life Systems				14	-294				$10\frac{1}{2}$	-220				$10\frac{1}{2}$	-220				$10\frac{1}{2}$	-220
				Δ Subtotal		-294			Δ Subtotal		-220			Δ Subtotal		-220			Δ Subtotal		-220
Trade-off No. 2																					
H_2 Boiloff						Not avail.					Not avail.					Not avail.				$10\frac{1}{2}$	-51
O_2 Boiloff																				$10\frac{1}{2}$	-407
Δ H_2 Tank																					-25
Δ O_2 Tank																					-40
																			Δ Subtotal		-523
Weight summary																					
No trade-off																					
Earth Launch						1,253					1,323					1,323	$\frac{3\ 1/2}{10\ 1/2}$ Days depletion				1,323
Lunar Launch						—					—					—					$\frac{1,170}{867}$
With trade-off no. 1																					
Earth Launch						959					1,103					1,103					1,103
Lunar Launch						—					—					—					647
With trade-off no. 2																					
Earth Launch						—					—					—					800
Lunar Launch						—					—					—					800
With both trade-offs																					
Earth Launch						959					1,103					1,103					580
Lunar Launch						—					—					—					580

TABLE V.- 14-DAY APOLLO MISSION

System No 3

Item	Function	Earth orbital					Circumlunar					Lunar orbiting (8 days)					Lunar landing				
		Module	No. units	Capability kw	Capability days	Weight lb	Module	No. units	Capability kw	Capability days	Weight lb	Module	No units	Capability kw	Capability days	Weight lb	Module	No. units	Capability kw	Capability days	Weight lb
Constants																					
Solar Cell array (Retr.)	Primary Power	Service	4	2		240	Service	4	2		240	Service	4	2		240	Service	4	2		240
Fuel Cell	Supp Power	Service	2	2		170	Service	2	2		170	Service	2	2		170	Service	2	2		170
Radiator		Service	1			18	Service	1			18	Service	1			18	Service	1			18
Sys. Piping		Service	1			30	Service	1			30	Service	1			30	Service	1			30
Silver-Zinc Batteries	Reentry Emerg., Postlanding	Command	2	8 (kw-hr)		160	Command	2	8 (kw-hr)		160	Command	2	8 (kw-hr)		160	Command	2	8 (kw-hr)		160
				Subtotal		618			Subtotal		618			Subtotal		618			Subtotal		618
Variables																					
O_2 Reactant		Service	1	2	134 hr	215	Service	Combined in redund. tanks	2	1 hr	2	Service	1	2	58 hr	96	Service	Combined in redund. tanks		1	40
O_2 Tankage			1			22							1			10				1	4
H_2 Reactant		Service	1	2	134 hr	37	Service		2	1 hr		Service	1	2	58 hr	12	Service			1	5
H_2 Tankage			1			19							1			6				1	3
				Subtotal		293			Subtotal		2			Subtotal		124			Subtotal		52
Redundancy																					
Fuel Cell	50% Backup	Service	1	1		106	Service	1	1		106	Service	1			106	Service	1			106
Radiator	100% Backup	Service	1			18	Service	1			18	Service	1			18	Service	1			18
Sys. Piping	100% Backup	Service	1			30	Service	1			30	Service	1			30	Service	1			30
O_2 Reactant	3 1/2 day Back-up	Service	0				Service	1	2	$3\frac{1}{2}$	140	Service	1	2	$3\frac{1}{2}$	140	Service	1	2	$3\frac{1}{2}$	140
O_2 Tankage			0					1			14		1			14		1			14
H_2 Reactant	3 1/2 day Back-up	Service	0				Service	1	2	$3\frac{1}{2}$	18	Service	1	2	$3\frac{1}{2}$	18	Service	1	2	$3\frac{1}{2}$	18
H_2 Tankage			0					1			9		1			9		1			9
				Subtotal		154			Subtotal		335			Subtotal		335			Subtotal		335
Trade-off No 1																					
Water Prod.	To Life Systems				134 hr	-224	-			1 hr					58 hr	-106				1	-45
				Δ Subtotal		-224			Δ Subtotal		-			Δ Subtotal		-106			Δ Subtotal		-45
Trade-off No. 2																					
H_2 Boiloff O_2 Boiloff ΔH_2 Tankage ΔO_2 Tankage						Not avail.					Not avail.					Not avail.					Offers No Adv.
Weight summary																					
No trade-off																					
Earth Launch						1,065					955					1,077					1,005
Lunar Launch						-					-					-					960
With trade-off no. 1																					
Earth Launch						821					955					971					960
Lunar Launch						-					-					-					915
With trade-off no 2																					
Earth Launch						1,065					955					1,077					1,005
Lunar Launch						-					-					-					960
With both trade-offs																					
Earth Launch						821					955					971					960
Lunar Launch						-					-					-					915

(1) 2 sec pulse each 15 sec during thrusting maneuvers; infrequent 2 sec pulse during transit. 80 to 240 watts.

(2) 10 - 20 sec pulses during transit. Continuous during lunar launch. 80 watts.

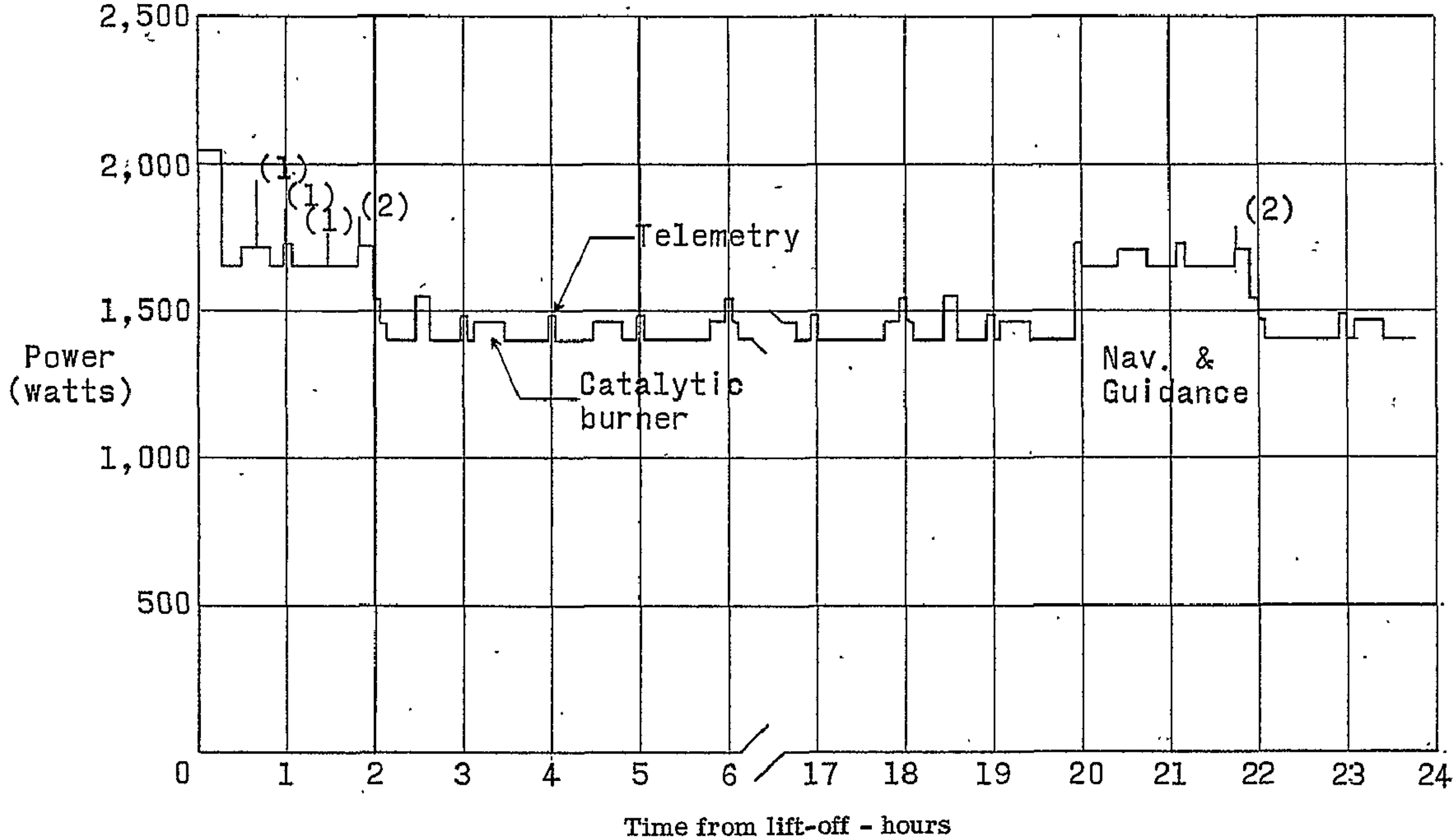

Figure 1. - Nominal 14-day mission - lift-off and midcourse.

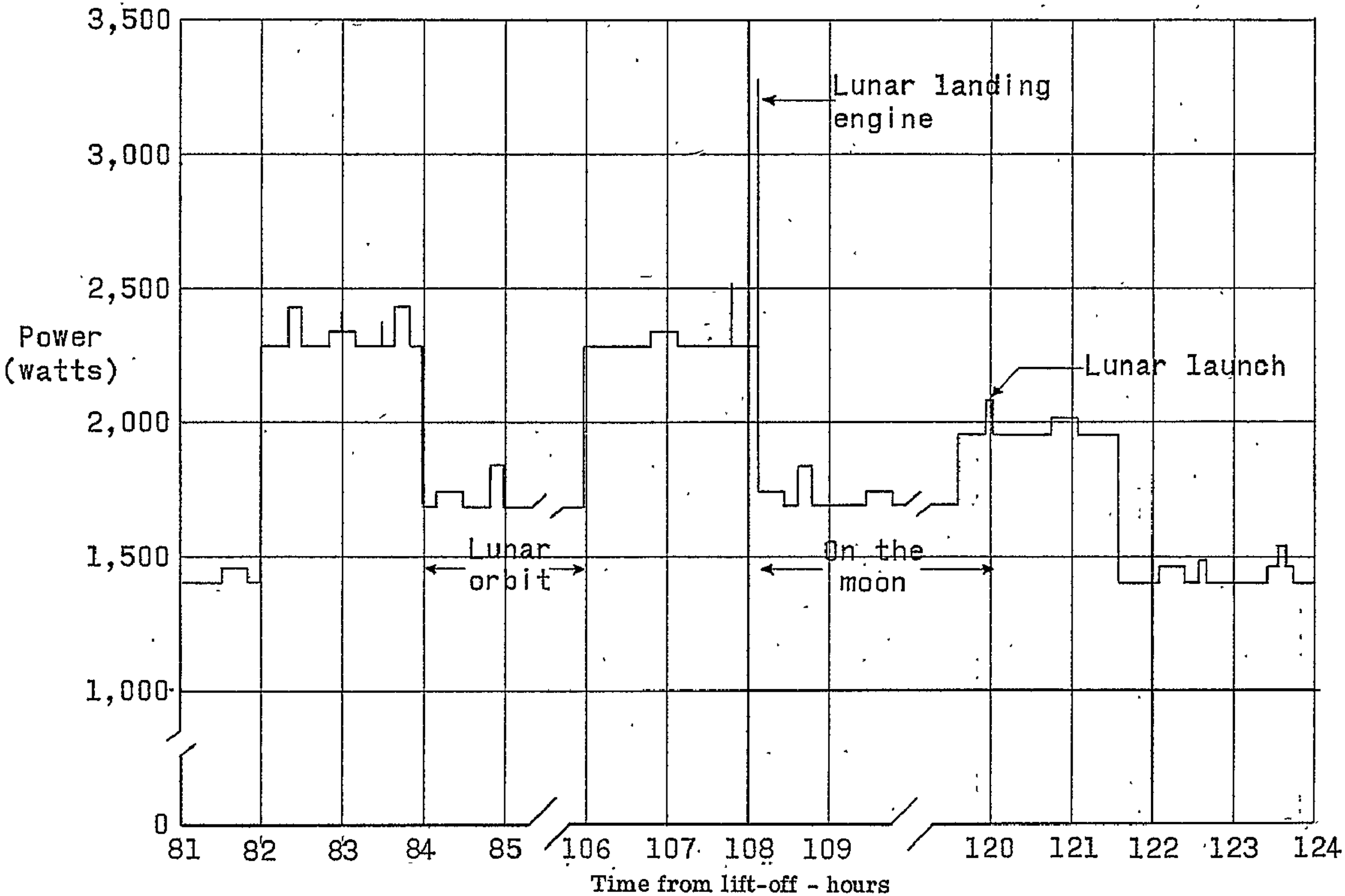

Figure 2. - Nominal 14-day mission - vicinity of the moon.

(1) 2 sec pulse. 80 to 240 watts. Each 15 sec during thrusting maneuvers. Infrequent during transit.

(2) 10 - 20 sec pulses during transit. 80 watts.

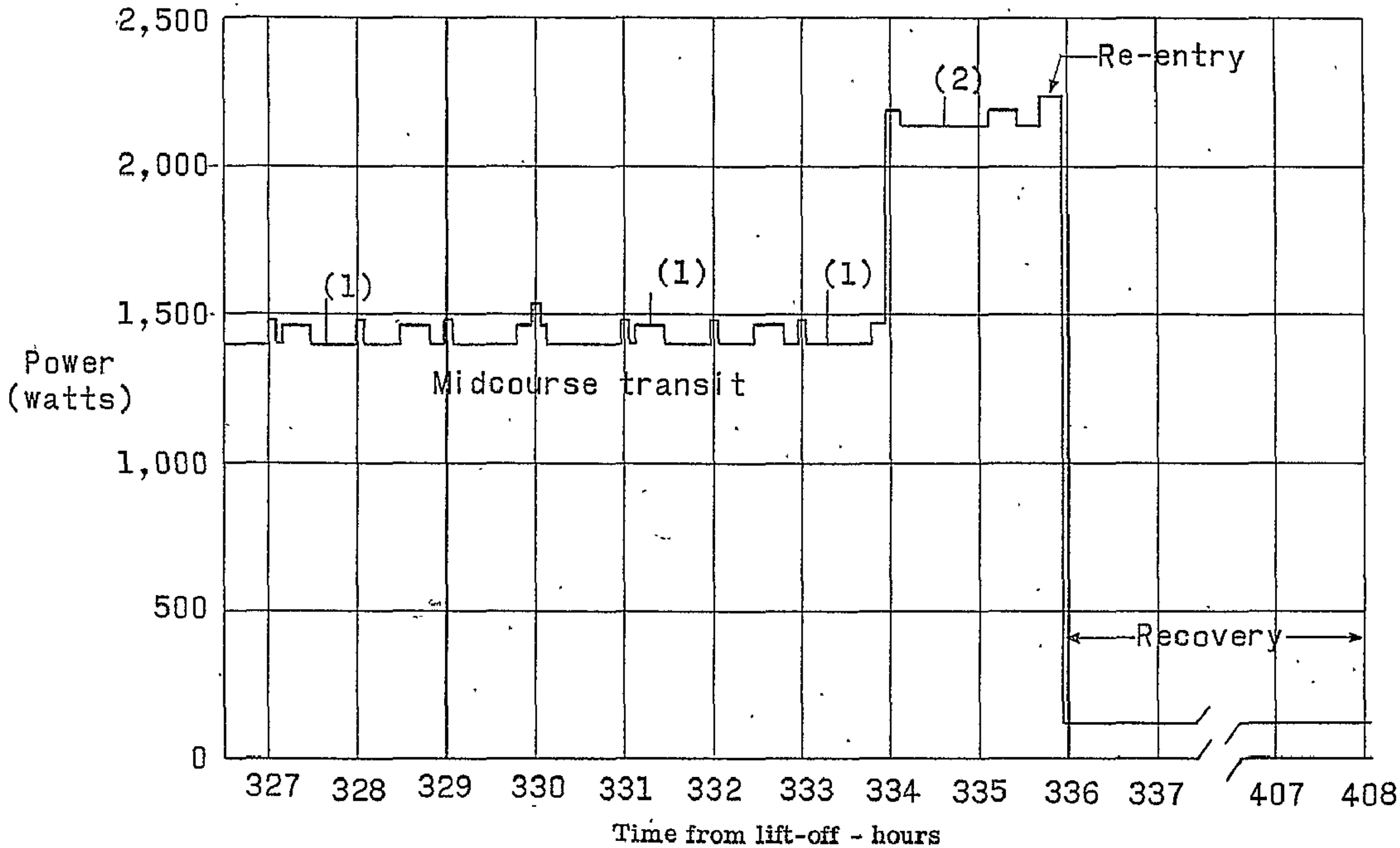

Figure 3. - Nominal 14-day mission - earth approach, reentry, recovery.

(1) 2 sec pulse each 15 sec during lunar launch (80 to 240 watts) intermittent during midcourse transit.

(2) 10 pulses during midcourse transit, 80 watts, 20 sec duration.

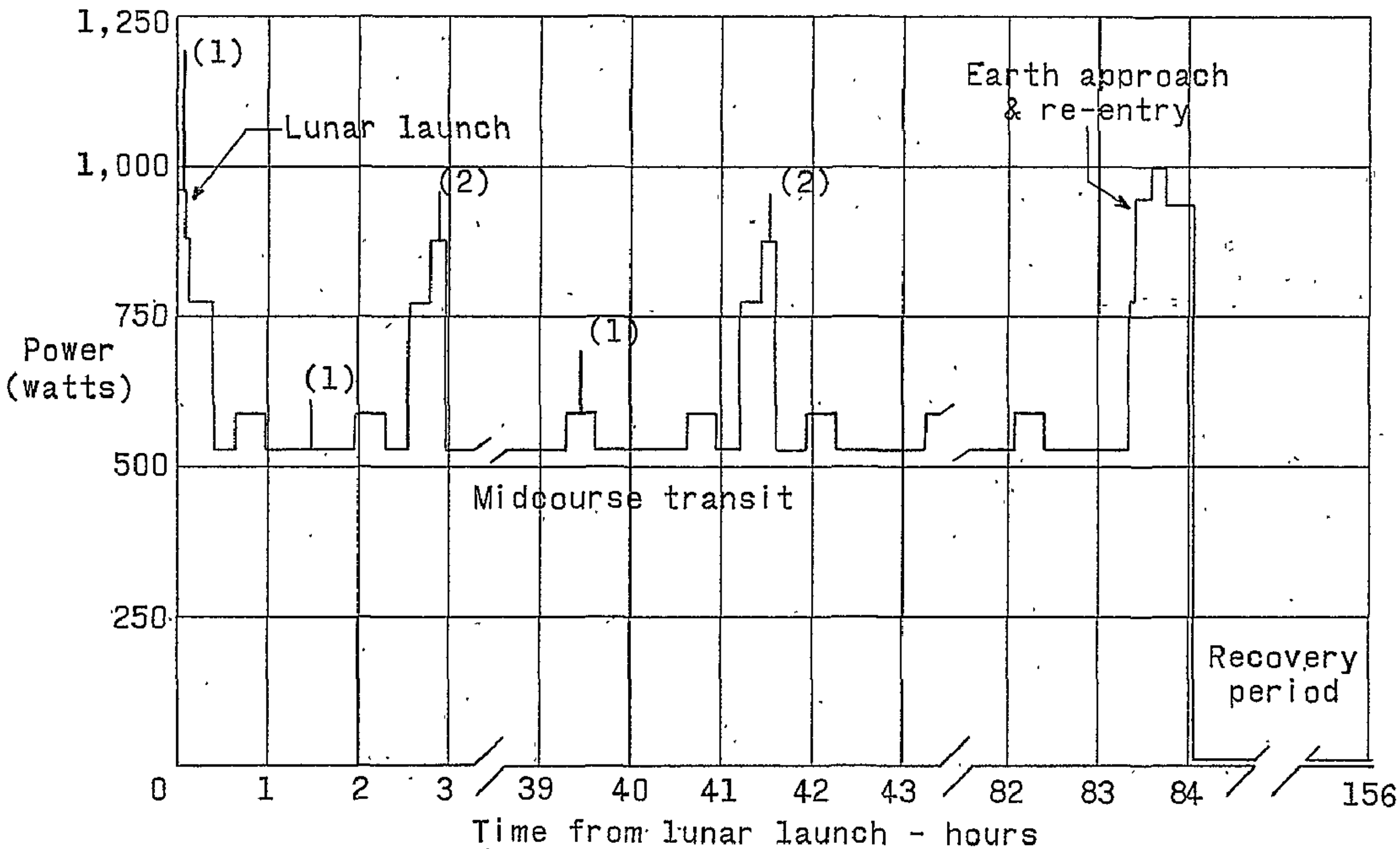

Figure 4. - Emergency power requirements - moon to earth.

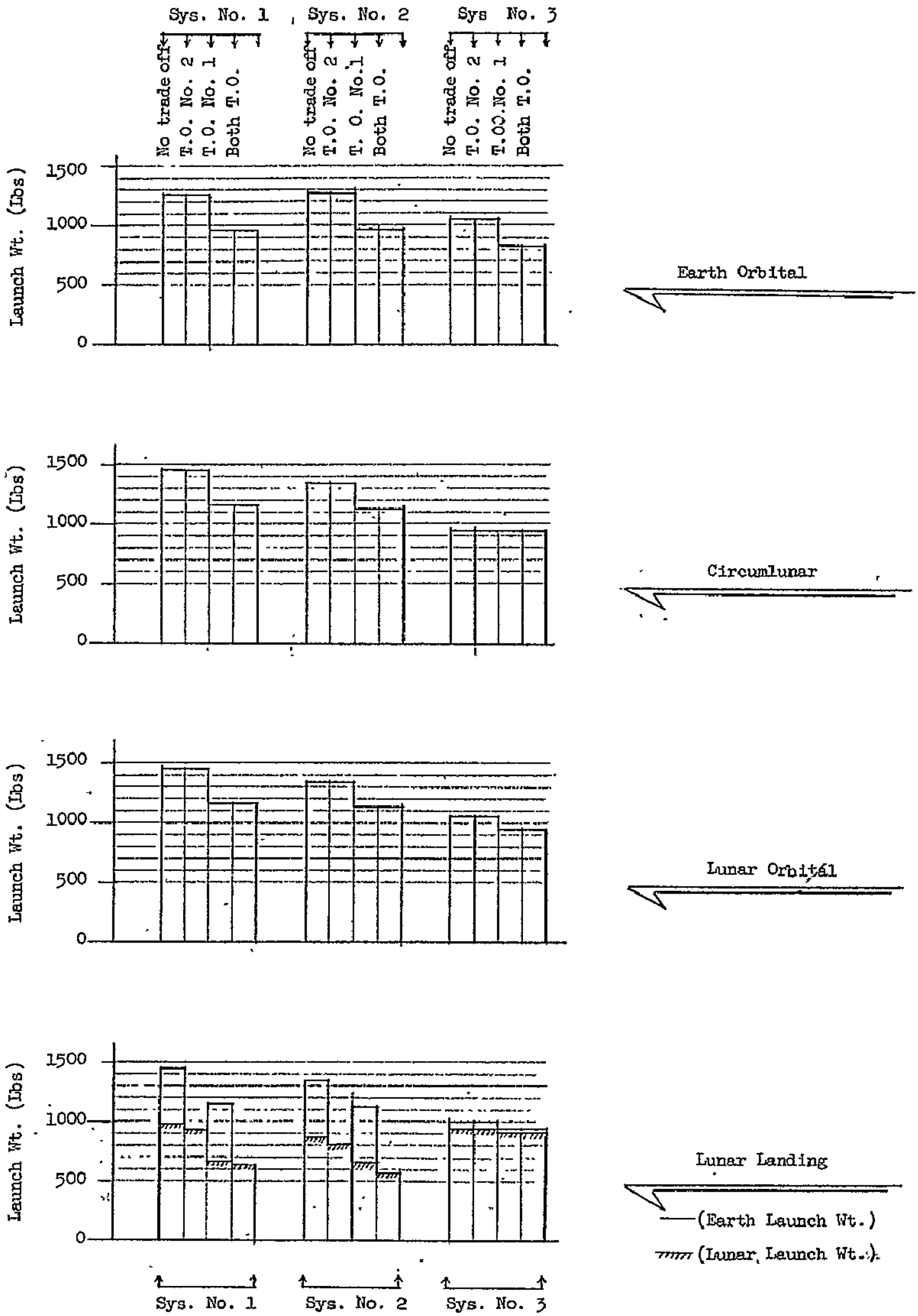

Figure 5. - Weight summary, 2 kilowatt auxiliary power supply.