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SPACE VEHICLE THERMAL TESTING: PRINCIPLES, PRACTICES, AND EFFECTIVENESS

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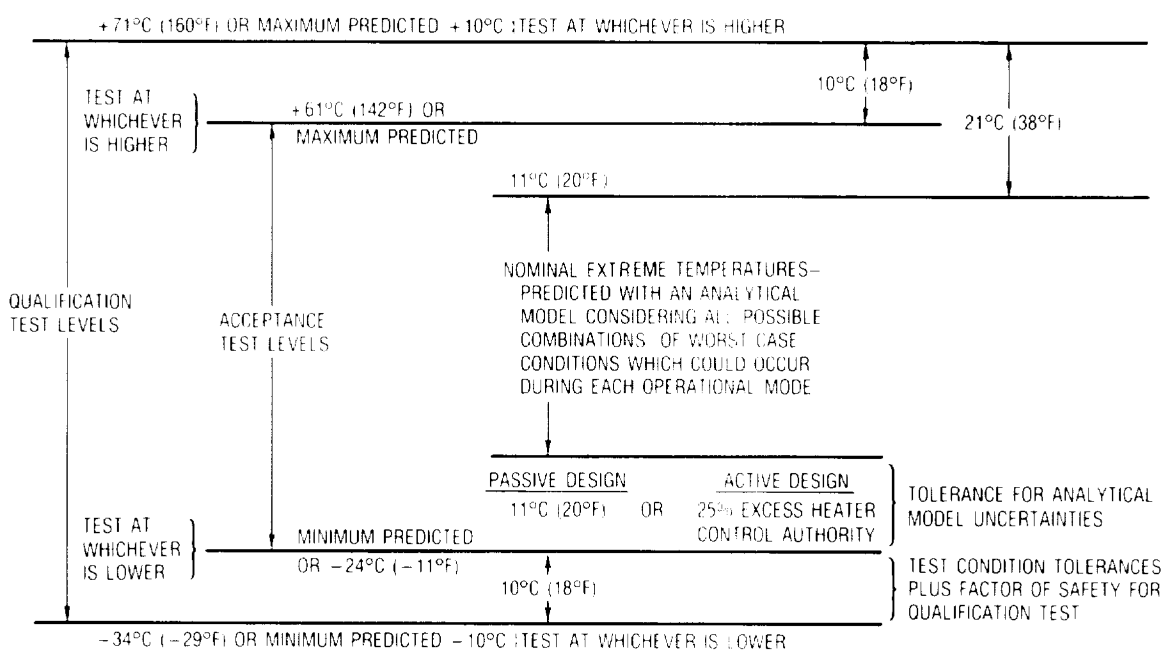
MIL-STD-1540B TEST REQUIREMENTS FOR COMPONENTS

Component qualification and acceptance temperatures are derived from worst case thermal analyses and analytic uncertainty margin subject to certain specified temperature extremes. Nominal extreme temperatures are predicted by applying an analytical model (e.g., SINDA computer program TMM) to each operational mode which considers worst case combinations of equipment operation, space vehicle attitude, solar radiation, eclipse conditions, degradation of thermal surfaces, et cetera. This must be done component by component, as a worst combination of conditions for one component may not prove to be worst for another. To these results an uncertainty margin is added. This margin, which can be quite large at the beginning of a program (e.g., 20 to 40°C), is reduced as the design and analytic process progresses. Following successful correlation of the thermal analysis with thermal balance test data, this uncertainty margin can be reduced to as little as $\pm$11°C. If a component is heater controlled, 25% excess heater control authority is required in lieu of an 11°C temperature margin. These temperatures set component acceptance test levels, subject to the requirement that the mounting plate or case temperature be at least as cold as -24°C and at least as hot as 61°C. These specified extremes are required in order to (a) provide adequate environmental stress screening, (b) demonstrate component survival capability, and (c) assure that temperature-insensitive and high-quality parts and materials are used in component design. Component qualification tests are conducted at temperatures 10°C colder (even if heaters are used for temperature control) and 10°C hotter than the acceptance test temperatures.

For some temperature-sensitive components such as batteries, propellant valves, and inertial reference units, the specified extremes are waived.



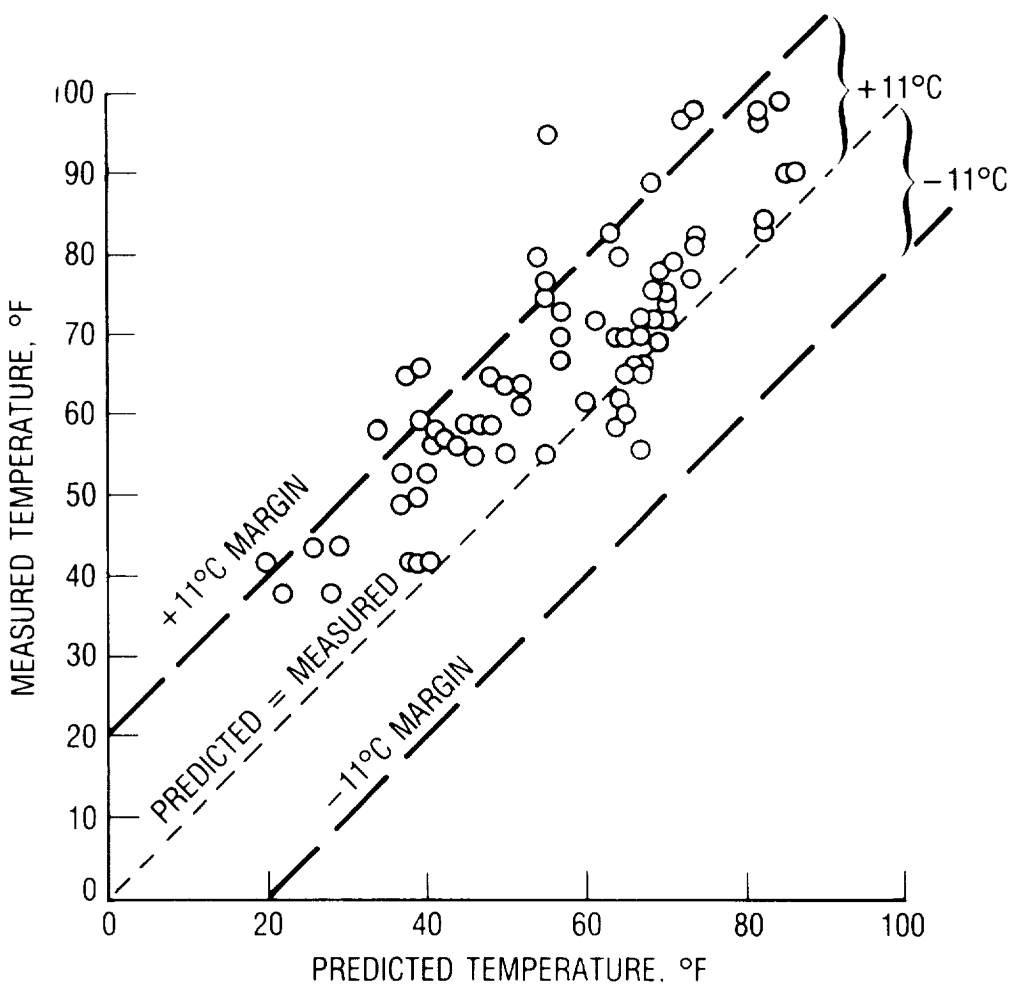
REPRESENTATIVE SPACE VEHICLE THERMAL CONTROL REQUIREMENTS

Temperature requirements are shown for equipment operation within specification and for survival and turn-on (need not operate within specification, but must not experience any degradation when returned to operational range). Temperature excursions for most equipment are seen to be 20 to 50°C above and below room temperature. Components without active electronics which are mounted outboard, such as solar arrays and antennas, are usually designed to withstand wider temperature excursions, particularly at the cold end. Batteries are tightly controlled at cold temperatures to increase life. Payload components such as extremely accurate clocks for precise navigation are controlled over a relatively narrow temperature range.

COMPONENT/SUBSYSTEM	OPERATING TEMPERATURE RANGE (°C)	SURVIVAL/TURN-ON TEMPERATURE RANGE (°C)
DATA HANDLING AND TT&C SUBSYSTEMS	-28.9/60	-28.9/60
ELECTRIC POWER AND DISTRIBUTION SUBSYSTEM	-28.9/60	-28.9/60
EPDS REGULATOR	-28.9/60	-28.9/60
STABILIZATION AND CONTROL COMPONENTS	-28.9/60	-28.9/60
COMPUTER	-28.9/43.3	-28.9/60
DIPOLE RING ARRAY ANTENNA	-150/100	-150/100
CONE ANTENNA	-150/110	-150/110
BICONE ANTENNA	-150/110	-150/110
SOLAR ARRAY	-141/61	-141/61
SOLAR ARRAY DAMPERS	-45.5/55.5	TBD/55.5
PAYLOAD ELECTRONICS	-12.2/43.3	-28/60
PAYLOAD ELECTRONICS	-6.7/43.3	-28.9/48.9 (SURVIVAL) -6.7/48.9 (TURN-ON)
BATTERIES	0 → 5 (TRICKLE CHARGE) 21.1 (DEEP DISCHARGE)	0/30
PROPULSION SUBSYSTEM	-3.9/26.7	TBD/40
THRUSTERS	-3.9/26.7	TBD/40
RUBIDIUM CLOCK	20/45	-19/45
CESIUM CLOCK	20/45	-19/45

FLTSATCOM-F1 PREDICTED TEMPERATURES VERSUS MEASURED TEMPERATURES, EQUINOX DIURNAL EXTREMES

The Aerospace Corporation's Thermal Control Department personnel, B. J. Smith and A. L. Bavetta, compared thermal balance test correlated model predictions with on-orbit measurements for the space vehicle FLTSATCOM-F1. Equinox data showed that measured temperatures were skewed towards being higher than predicted. Of 74 temperature measurements, 65 were within ±11°C of prediction, with a maximum deviation of 22°C. While the skewing was not necessarily experienced on other space vehicles, the pattern and spread were typical.



STP P78-1 SATELLITE (NO THERMAL BALANCE TEST) COMPARISON OF ON-ORBIT TEMPERATURE MEASUREMENT WITH CONTRACTOR ANALYTIC PREDICTIONS

Air Force Space Test Program Satellite P78-1 was launched without a thermal balance test. A comparison has been made of 12th day on-orbit measurements with contractor predictions. The temperature of 10 of 17 components within the wheel (rotating portion of the space vehicle) and 5 of 8 components within the sail (sun-fixed portion of the space vehicle) were within 11°C of the predicted values. The temperature of seven wheel components and three sail components exceeded prediction by more than 11°C, with the largest deviation being 24°C. Agreement between prediction and measurement was substantially poorer than for a typical satellite which had received a thermal balance test.

WHEEL EQUIPMENT, °C

	BATTERIES			ELECTRIC CHARGE CNTR.	POWER SHUNT REQ.	19V REG	TAPE RECORDERS			TELEMETRY DISTRIBU-TION UNIT	TRANS-MITTERS		COMMAND & DATA PROCESSOR	SPIN ASSEMBLY	AZIM. DRIVE ASSEMBLY	REFRIGERATOR PAA ELECTRONICS	
	1	2	3				A	B	C		1	2				GAMMA 3	GAMMA 4
ON-ORBIT TEMPERATURE MEASUREMENT 12th DAY AFTER LAUNCH	17	18	17	18	15	18	20	27	15	16	17	25	31	27	18	7.5	10
CONTRACTOR PREDICTION, NORMAL RESOLUTION	11	11	10	13	6	7	15	15	8	8	0	8	7	10	11	-9	-6

SAIL EQUIPMENT, °C

	REMOTE COMMAND AND DATA PROCESSOR		AMPLIFIER		AUXILIARY CONTROL ELECTRONICS	TRUNNION	NUTATION DAMPER	SOLAR ARRAY
	1	2	SERVO	POWER				
ON-ORBIT TEMPERATURE MEASUREMENT 12th DAY AFTER LAUNCH	21	16	28	24	30	24	25	63
CONTRACTOR PREDICTION, NORMAL RESOLUTION	14	21	15	19	7	16	2	55

THE BASIS OF MIL-STD-1540's TEMPERATURE UNCERTAINTY MARGIN

The table is supported by The Aerospace Corporation's data base. An uncertainty margin of 11°C is used in MIL-STD-1540 for analytic predictions correlated to thermal balance test results. Note that the intent of the standard is to provide 95% confidence that acceptance test temperatures will not be exceeded during mission life.

STANDARD DEVIATION	PERCENT OF CONFIDENCE	TEMPERATURE UNCERTAINTY (°C)	
		UNVERIFIED ANALYTICAL PREDICTIONS	PREDICTIONS VERIFIED BY TESTING
1.0	68%	8.3	5.6
1.4	85%	12.2	7.8
2.0	95%	16.7	11.0
3.0	99%	25.0	16.7

MIL-STD-1540 COMPONENT TEST BASELINE

MIL-STD-1540 defines a component as "a functional unit that is viewed as an entity for purposes of analysis, manufacturing, maintenance, or record-keeping. Examples are hydraulic actuators, valves, batteries, electrical harnesses, and individual electronic boxes such as transmitters, receivers, or multiplexers." Components are made up of modules and assemblies which, in turn, are made up of piece parts. Test and screens are conducted at these lower levels of assembly. However, the lowest level of assembly addressed in MIL-STD-1540 is the component level.

These tables are abstracted from tables in this Standard. Thermal vacuum, thermal cycling, and burn-in are component thermal tests and screens. MIL-STD-1540 requires thermal cycling rather than elevated temperature burn-in. Functional tests, while not considered here as thermal tests, are required at temperature extremes during thermal cycling and thermal vacuum tests.

COMPONENT QUALIFICATION TESTS

TEST	REFERENCE PARAGRAPH	SUGGESTED SEQUENCE	ELECTRONIC OR ELECTRICAL EQUIPMENT	ANTENNAS	MOVING MECHANICAL ASSEMBLY	SOLAR PANEL	BATTERIES	VALVES	FLUID OR PROPULSION EQUIPMENT	PRESSURE VESSELS	THRUSTERS	THERMAL EQUIPMENT	OPTICAL EQUIPMENT
FUNCTIONAL	6.4.1	1(1)	R	R	R	R	R	R	R	R	R	R	R
THERMAL VACUUM	6.4.2	9	R	R	R	R	R	R	R	O	R	R	R
THERMAL CYCLING	6.4.3	8	R	O	O	O	O	O	O	—	—	—	—

COMPONENT ACCEPTANCE TESTS

TEST	REFERENCE PARAGRAPH	SUGGESTED SEQUENCE	ELECTRONIC OR ELECTRICAL EQUIPMENT	ANTENNAS	MOVING MECHANICAL ASSEMBLY	SOLAR PANEL	BATTERIES	VALVES	FLUID OR PROPULSION EQUIPMENT	PRESSURE VESSELS	THRUSTERS	THERMAL EQUIPMENT	OPTICAL EQUIPMENT
FUNCTIONAL	7.3.1	1(1)	R	R	R	R	R	R	R	R	R	R	R
THERMAL VACUUM	7.3.2	7	R(2)	O	R	O	R	R	R	O	R	R	R
THERMAL CYCLING	7.3.3	6	R	O	O	O	O	O	O	—	—	—	—
BURN-IN	7.3.9	8	R	—	O	—	—	R	—	—	R	—	—

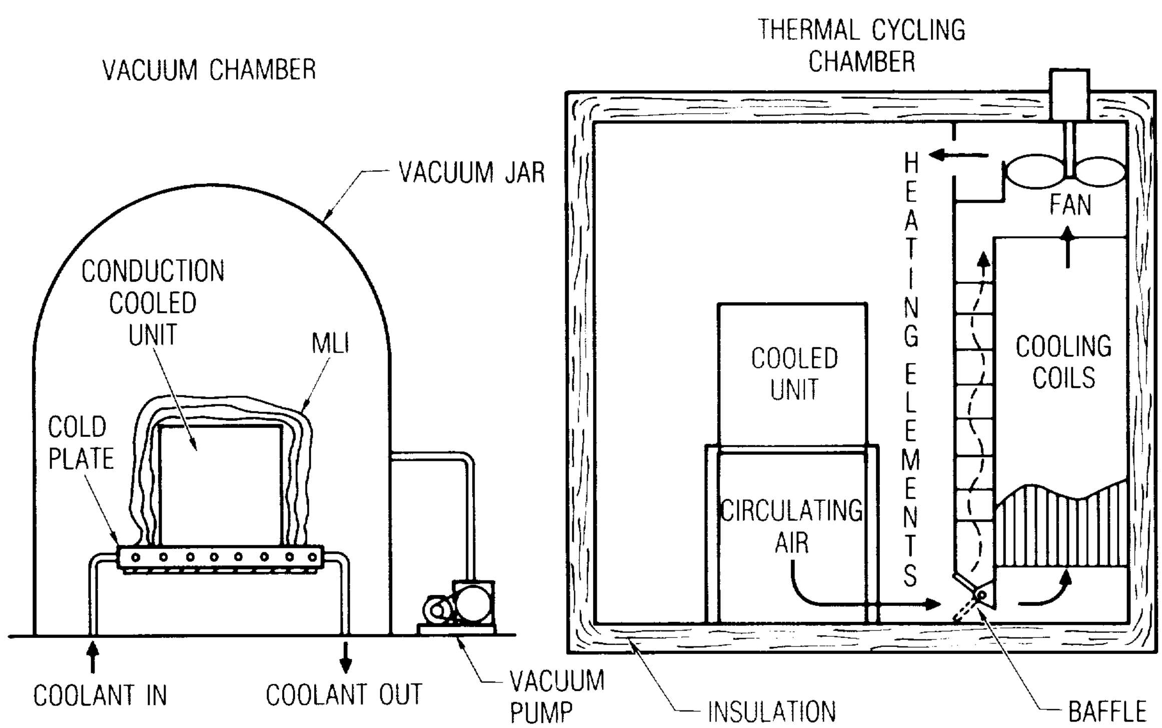
LEGEND R = REQUIRED
O = OPTIONAL TEST
— = NO REQUIREMENT

Notes: (1) Functional tests shall be conducted prior to and following environmental test
(2) Required only on unsealed units and on high power, RF equipment

A wide variety of test chambers are available for thermal cycling tests. Nitrogen or humidity-controlled air is used to prevent water vapor condensation. Heating, cooling, and a rapid air or gas flow are required. A rapid rate of temperature change at the base plate or case of the component of interest is often difficult to achieve. This may be the major technical challenge faced in thermal cycling testing.

Thermal vacuum tests are divided into two categories: (1) those where conduction to a mounting plate is the dominant mode of cooling, and (2) those where radiation to the surroundings dominates or where cooling is by both conduction and radiation. The former has proved to be the more likely occurrence. Conduction cooling is usually accomplished by torquing the component down onto a monolithic, thermally-controlled plate. This is not truly representative of actual component installation, which may, for example, have delron inserts in an aluminum honeycomb with face sheets. However, this is usually acceptable for component testing and buy-off, provided the differences between test mounting and flight mounting are accounted for by analysis and verified by testing at the subsystem or the system level.

Many components are cooled primarily by radiation or by both conduction and radiation. Such components include control moment gyroscopes, horizon sensors, and inertial reference units. Here, control of the heat loss paths should be such that radiation and conduction occur in the same proportion as calculated for the flight environment. This is necessary so that module and piece part temperatures and component temperature gradients duplicate those which occur in actual usage. This can be achieved, for example, by the use of heated baffles and shields and the control of mounting plate temperatures.

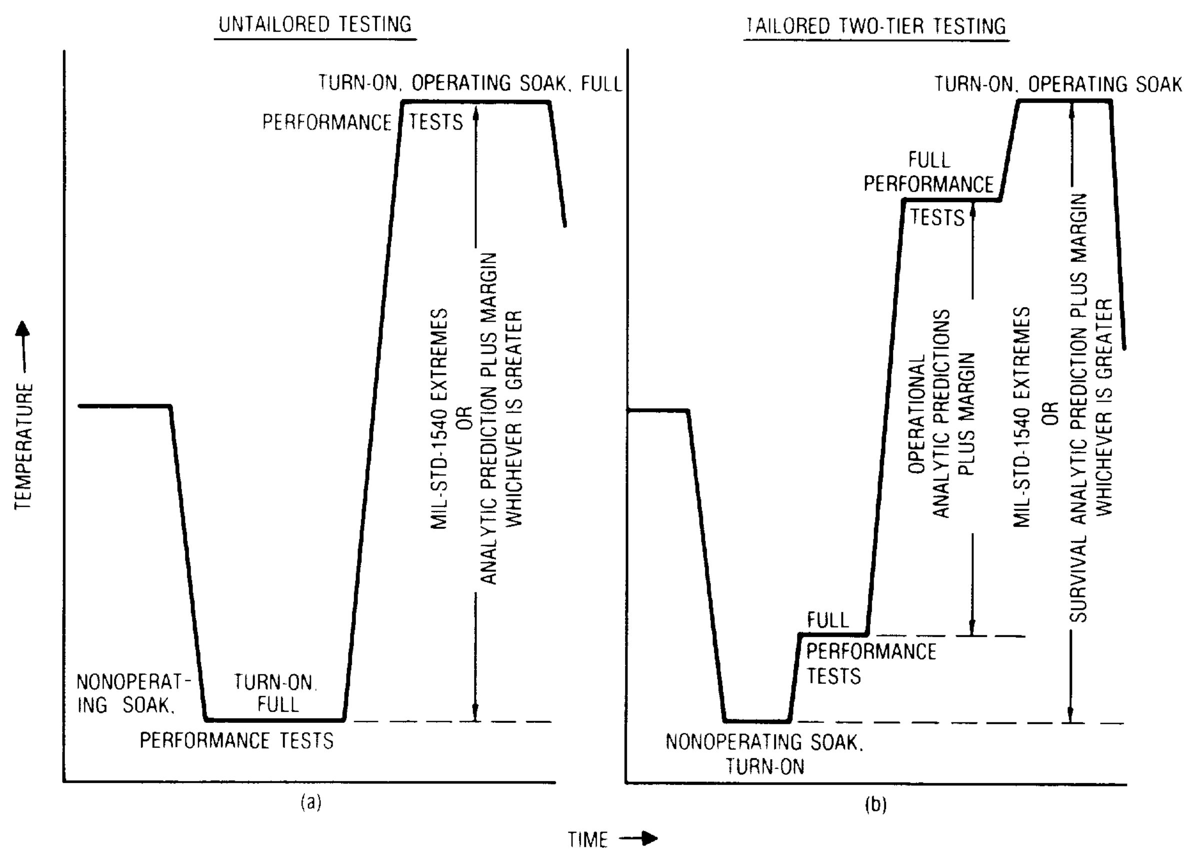


OBJECTIVES OF COMPONENTS THERMAL CYCLING, THERMAL VACUUM, AND BURN-IN TESTS

The specified tests (thermal cycling, thermal vacuum, and burn-in) can be construed as having three functions: environmental stress screening (ESS), demonstration of survival and turn-on capability, and performance verification. ESS, by subjecting hardware to physical stresses, forces flaws which are not ordinarily apparent into observable failures. Flaws are latent defects in design, workmanship, parts, processes, or materials which could cause premature component failure. The defective elements are repaired or removed prior to usage. The intent of the survival and turn-on function is to demonstrate that the equipment can be soaked, started, and operated at cold and hot survival temperature limits without experiencing permanent damage or performance degradation when returned to the operational temperature range. Survival/ turn-on temperature limits derive from ascent, safemode and threat mission phases, and factory and launch site checkout. Finally, the tests verify that the component electronic and mechanical performance is within specification.

- ENVIRONMENT STRESS SCREENING
  - FINDS FAULTS IN COMPONENT DESIGN. WORKMANSHIP, PARTS, MATERIALS, AND PROCESSES
- VERIFICATION OF SURVIVAL AND TURN-ON CAPABILITY
  - DEMONSTRATION THAT COMPONENT CAN BE TURNED ON AND OPERATED OVER SURVIVAL TEMPERATURES WITHOUT EXPERIENCING PERMANENT DAMAGE OR PERFORMANCE DEGRADATION WHEN RETURNED TO OPERATIONAL TEMPERATURE RANGE
- VERIFICATION THAT COMPONENT PERFORMANCE IS WITHIN SPECIFICATION OVER ITS OPERATIONAL TEMPERATURE RANGE

Test temperature limits are the same for performance, screening, and survival/turn-on, if MIL-STD-1540 is applied without tailoring. In this case, component thermal tests are conducted at cold and hot limits determined from analytic predictions plus analytic uncertainty margin or at specified extremes whichever is greater. Some suppliers have requested a waiver for units originally built and qualified before the Standard was issued and for a limited number of new units with special temperature sensitivity; they have proposed, in lieu of the Standard, that tailored two-tier testing be conducted as in Figure b. For such testing, performance is verified over the narrower regime of operational analytic predictions plus margin, while screening is accomplished and survival/turn-on are demonstrated over the wider range of MIL-STD-1540 specified extremes or survival temperature analytic prediction. Unfortunately, this waiver request has propagated, so that it is now being requested for many units regardless of heritage, temperature sensitivity, and the like. Additionally, the outer tier tests and temperature levels have been weakened.



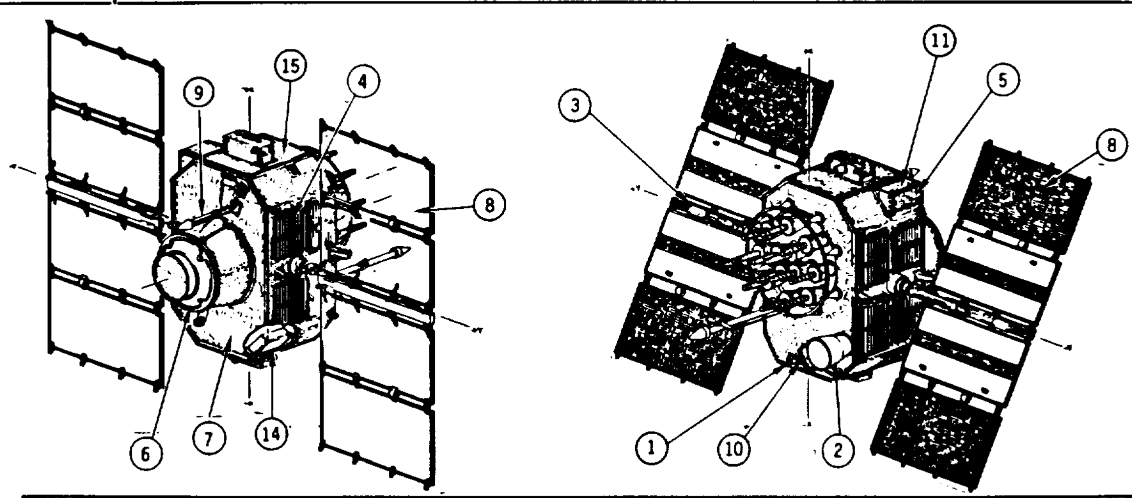
COMPARISON OF MIL-STD-1540 ACCEPTANCE TEST REQUIREMENTS
WITH RECOMMENDATIONS

Key Parameter	Recommendations		MIL-STD-1540B Requirements	Conclusions
	IES Guidelines (Ref. 1)	MMC Study (Ref. 2)		
No. of thermal cycles	12 cycles	Part Count / Recommended No. of Cycles: 100 — 1 500 — 3 2000 — 6 4000 — 10	Thermal cycling -8 cycles Thermal vacuum -1 cycle For TC and TC conduct full functional test at high and low temperature extreme, first and last cycles Burn-in -18 cycles (includes thermal cycling and thermal vacuum)	MIL-STD requirements consistent with industry practice No. of cycles not excessive, may be insufficient
Temp. extremes and range	-40 to +70°C	-54 to +55°C	-24 to +61°C	MIL-STD requirements within design and performance capability and within experience base of suppliers Makes sense for space vehicles because of unattended long-life requirement
Temp. transition rate of change	5°C/minute of surrounding media	-	At least 1°C/minute measured at baseplate of unit	MIL-STD requirements more work is needed on subject Rate of change probably too low; should be at least as great as maximum predicted rate
Operation/non-operation profile	Power ON	-	Power ON during transition Cycles through operational modes Monitor perceptive parameters Cold start/hot start	MIL-STD-1540 requirements are sound and well founded
Dwell	-	-	One hour minimum dwell at high and low temp. extreme, long enough to obtain internal temp. equilibrium	MIL-STD-1540 requirements seem reasonable

Surfaces and finishes are the most basic thermal control hardware. Some are illustrated for a typical space vehicle. Solar absorptance, α, tends to increase with mission life because of contamination and attack by ultraviolet radiation and charged particles. The composite Kapton-H/aluminum film is widely used as the external surface of structure and multilayered insulation because it has good handling and bonding characteristics and experiences relatively minor mechanical damage due to the natural environment. Teflon/silver film has lower values of α/ε than the Kapton film, but it is seeing less use as a flexible second surface mirrors because of mechanical degradation in the natural environment. This satellite did not use the more durable fused silica/silver rigid second surface mirrors commonly called OSRs. White paint such as S13S/LO, composed of zinc oxide pigment and RTV-602 (organic) binder, degrade more rapidly than the newer YB-71 white paint, which is composed of zinc orthotitanate pigment and PS7 potassium silicate (inorganic) binder. The YB-71 paint, sometimes called "ZOT," also appears to have good survival characteristics in some threat environments.

High emissivity white and black paints are widely used for interior surfaces. Polished aluminum, with its low emissivity, is usually employed in applications where there is no direct solar incidence and where low thermal coupling to space and to spacecraft surfaces is desired.

TCS COATINGS

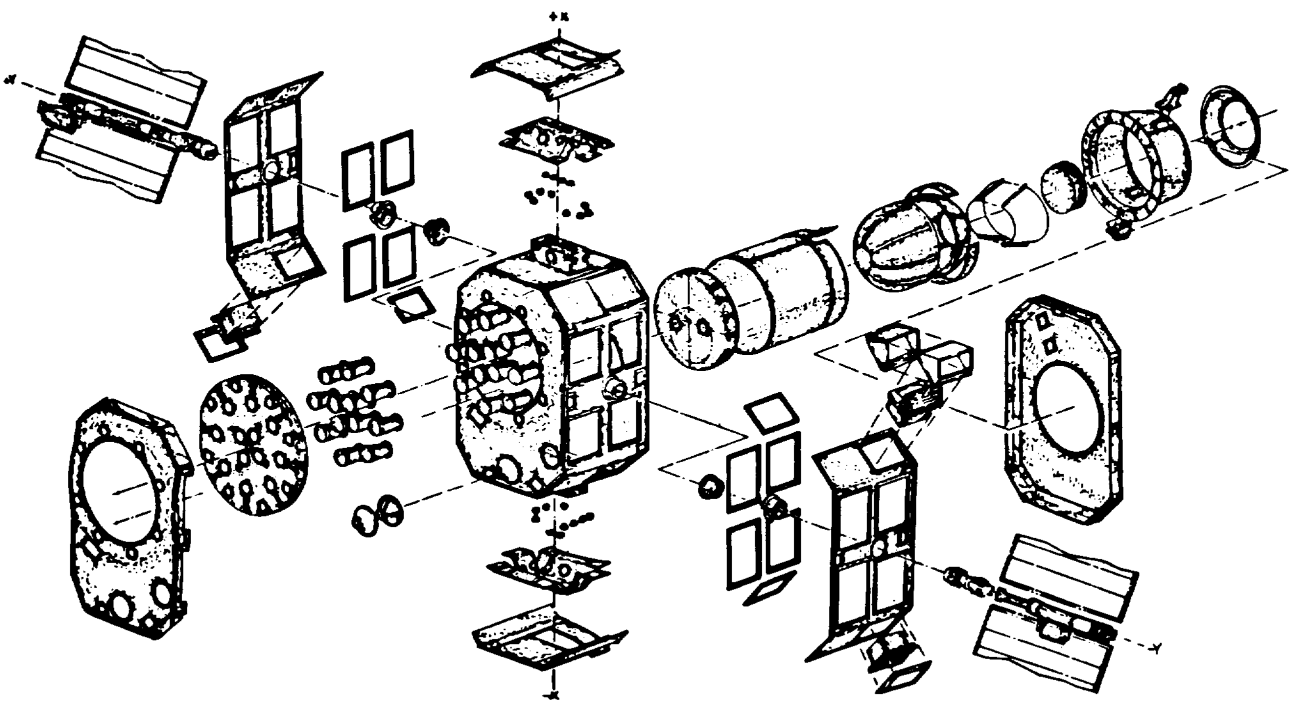


LOCATION – FINISH	OPTICAL PROPERTIES α BOL	α EOL	ε	LOCATION – FINISH	OPTICAL PROPERTIES α BOL	α EOL	ε
1. X SENSOR – MLI-2 MIL AL KAPTON	0.44	0.65	0.72	8. SOLAR ARRAY – ACTIVE SIDE – CELLS	0.67	0.67	0.81
2. Y SENSOR – WHITE (S13G/LO)	0.23	0.70	0.85	SUBSTRATE – GRAPHITE EPOXY	0.93	0.93	0.85
3. S/A BOOM & SHUNTS 5-MIL SILVER TEFLON	0.09	0.54	0.76	9. TTC ANTENNAS – 1-MIL AL KAPTON	0.13	0.13	0.05
BLACK (CHEM GLAZE)	0.96	0.96	0.84	10. CES SHROUD – 2-MIL SILVER TEFLON	0.09	0.54	0.66
4. SHEAR PANEL RADIATORS – WHITE (YB-71)	0.20	0.70	0.91	11. BATTERY 3 SHROUD 2-MIL SILVER TEFLON	0.09	0.54	0.66
5. RADIATORS – BATTERIES – 5-MIL SILVER TEFLON	0.09	0.32	0.76	12. LOUVER/MLI CLOSEOUT – POLISHED ALUM	0.15	0.15	0.05
6. HEAT SHIELD – FIBERGLASS POLYIMIDE	0.75	0.80	0.89	13. SPACECRAFT INTERIOR – BLACK (CHEM GLAZE)	0.96	0.96	0.84
7. EXTERNAL INSULATION – MLI – 2-MIL AL KAPTON	0.44	0.65	0.72	14. ELECTRONIC BOX EXTERIORS (BLACK)	–	–	0.80
				15. ACCESS PANELS – 5-MIL AL KAPTON	0.49	0.70	0.83

INSULATION BLANKETS

The exploded insulation illustration shows the wide variety of multilayered insulation blankets used on space vehicles. These blankets use a layered approach to reduce conduction and radiation heat transfer to low values. Typically, alternate layers of aluminized Mylar or Kapton and a highly porous spacer material control radiation and conduction, respectively.

Blanket construction and installation can cause performance degradation. Heat shorts can be introduced by blanket compression over curved surfaces (especially those with compound curvature or small radii of curvature); penetration of support posts; blanket electrical grounding, venting and outgassing provisions; and stitching, pinning, and binding. Such problems are usually more severe with smaller blankets and those with cutouts, where the ratio of edge length to surface is large. A well-instrumented, properly controlled thermal balance test, using a qualification space vehicle or subsystem which is a true facsimile of the flight article, is necessary to determine blanket effective emissivity.

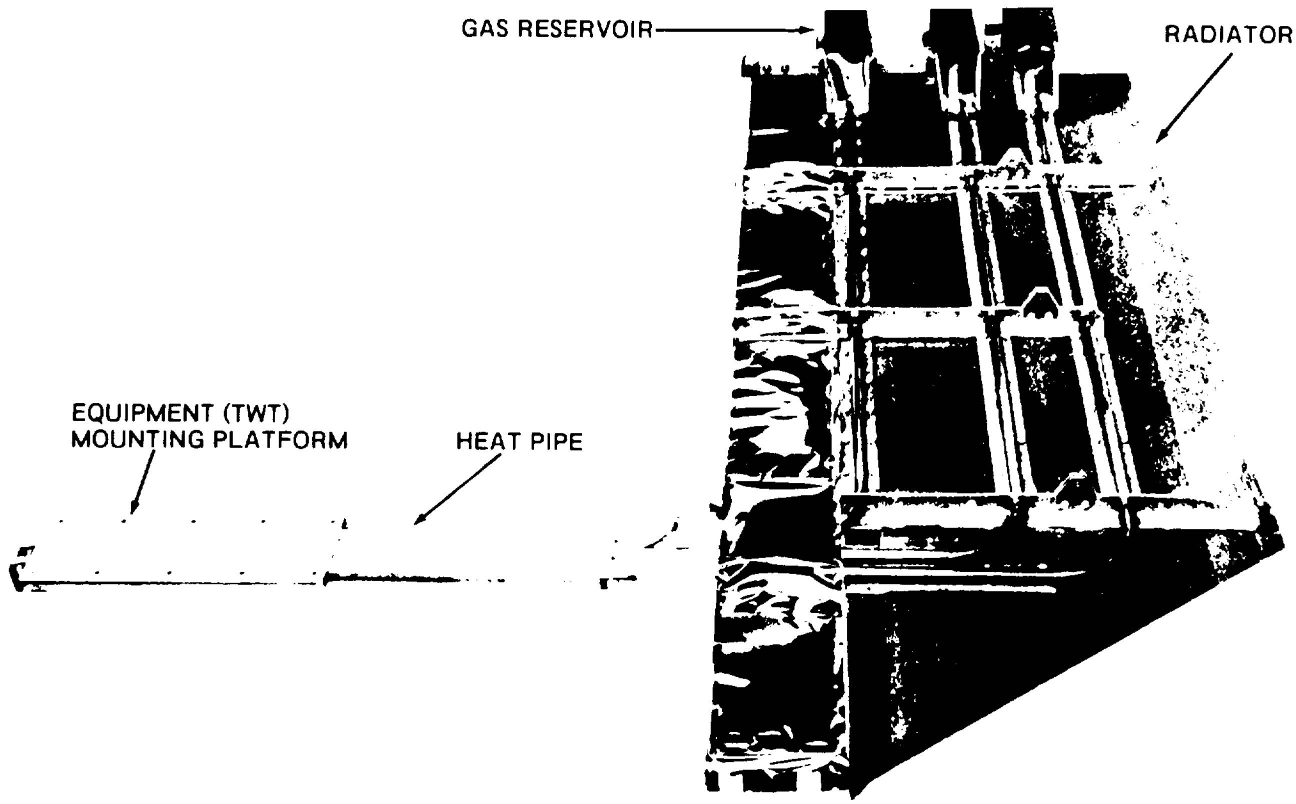


HEAT PIPE ASSEMBLIES

Heat pipes (tubes containing internal wicks and liquid and vapor phase working fluid) are coming into extensive use on space vehicles. Newer vehicles may use more than 100 heat pipes of five to 15 different configurations and types. Evaporation in the region of equipment heat dissipation causes menisci contraction to small radii of curvature. The evaporated vapor condenses in the cold radiator portion of the heat pipe. The differential pressure caused by evaporator menisci pumps the condensed liquid within the wicking grooves to the evaporator end of the heat pipe. A countercurrent convection loop is thereby set up in the pipe which transfers heat at substantially higher rates than a solid aluminum tube of the same diameter. More complex designs offering greater control precision and reduced cold case heater power usage are possible (e.g., the variable conductance heat pipe assembly illustrated here). It employs inert gas within gas reservoirs to block portions of the condenser during mission phases with reduced equipment heat dissipation or environmental loading. For higher heat load applications, capillary pumped loops are receiving consideration. Operation and control of such loops entails yet a higher level of complexity.

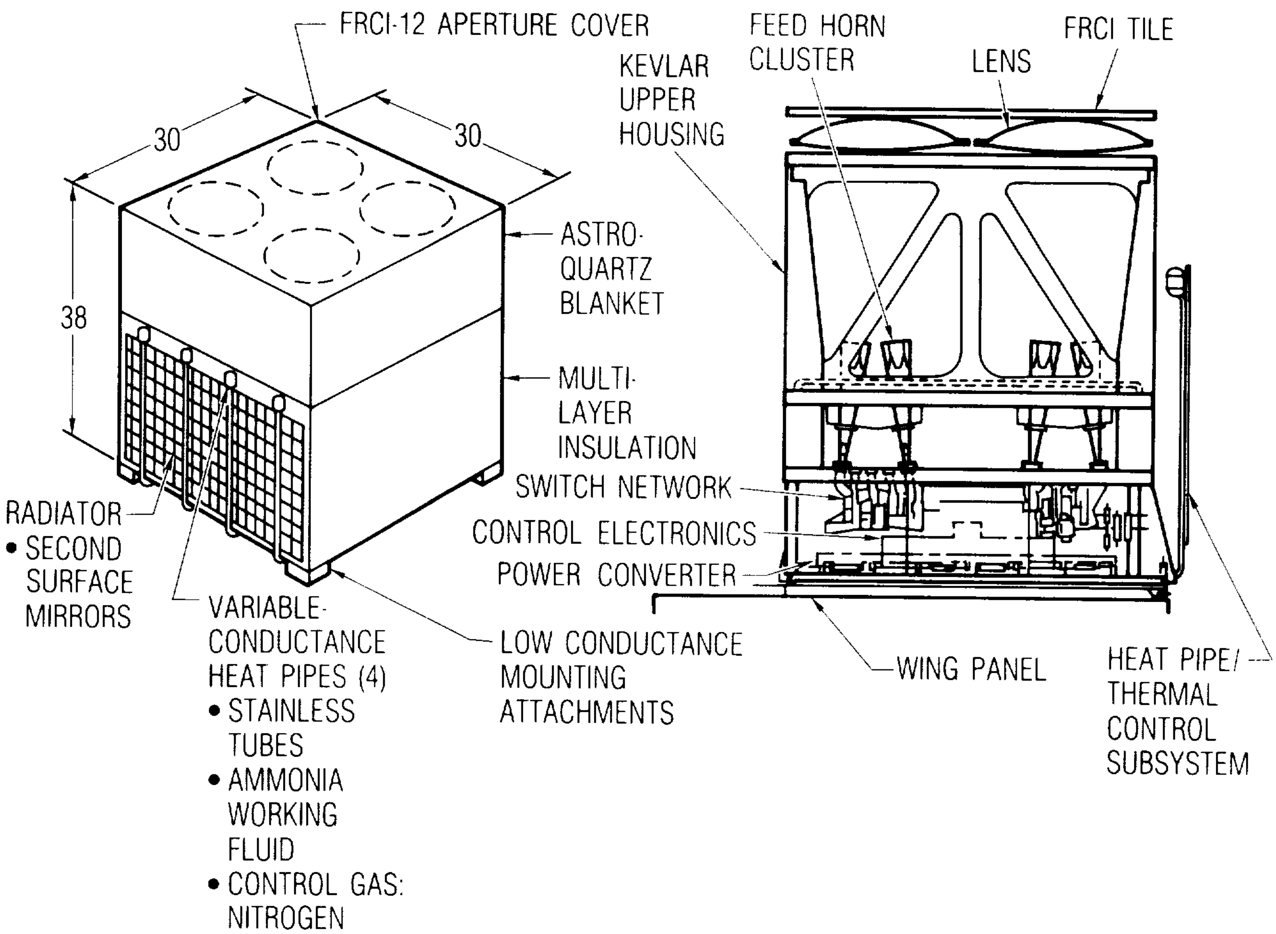
Heat pipe performance, as it depends on relatively small capillary forces, is sensitive to body (gravitational) forces. Consequently, a heat pipe which will work excellently in the near zero gravity space environment, could be rendered inoperative by evaporator height exceeding condenser height by a fraction of an inch during ground tests. The effect on vehicle design and ground testing is profound. Precise tolerance control of the design and the test set up may be required to assure that a heat pipe meets leveling requirements. Because of design requirements and allowable test configurations, some heat pipes cannot possibly be tested in the horizontal configuration during space vehicle tests. The thermal performance of such heat pipe assemblies must be verified at the subsystem level; here, it is often possible to rotate the assembly so that the heat pipes of interest are horizontal. A space test may prove to be the ideal way to verify the performance of new capillary pumped loop designs.

SATELLITE VARIABLE CONDUCTANCE
HEAT PIPE ASSEMBLY



As spacecraft size and complexity has grown, and buildup time has lengthened, the need has developed for intermediate tests between component and space vehicle testing. Such tests may be conducted on all or part of a subsystem. For example, the thermal design of the depicted antenna assembly is sufficiently complex to warrant an assembly level thermal vacuum test. Design features include multilayered insulation, a second surface mirror radiator, conduction coupling to active electronics, variable conductance heat pipes, and heaters and controllers. The test will verify the ability of the thermal design to hold components within allowable temperatures under specified hot and cold conditions.

Subsystem and assembly tests allow use of smaller test facilities than required for the space vehicle tests, and make it easier to tailor the thermal environment to the specific requirements of the components under test. Usually, configuration and leveling requirements can be more readily met in a subsystem, rather than in a space vehicle test. Results are obtained in a more timely manner, facilitating necessary remedial action.



SPACE VEHICLE THERMAL TESTS

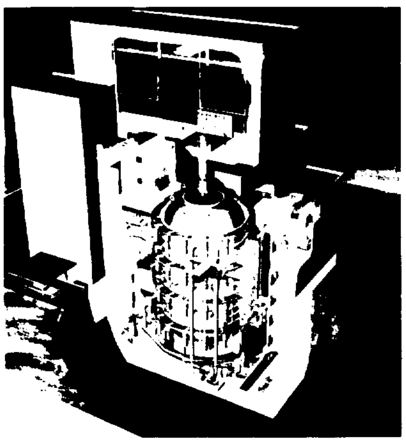
Space Vehicle (SV) qualification thermal tests are more demanding than the SV acceptance tests in that there is a wider temperature range, more thermal cycles, and the inclusion of a thermal balance test. The qualification tests are formal contractual demonstrations that the design, manufacturing, and assembly of hardware have resulted in conformation to specified requirements. The acceptance tests are required formal tests which demonstrate the acceptability of an item for delivery. They are intended to demonstrate performance to specified requirements and to act as environmental screens to detect deficiencies of workmanship, material, and quality. Acceptance test temperature levels should encompass all specified flight environments.

The thermal vacuum test consists primarily of system level functional performance tests (e.g., payload performance, electrical, mechanical, and thermal) between and at temperature extremes. Emphasis is on component and subsystem interaction and interfaces; integrity of mounting, cabling, and connectors; and on end-to-end system performance. An optional thermal cycling test functions as a high level environmental screen. The thermal balance test, conducted as part of the thermal vacuum test for the qualification vehicle, is a dedicated thermal test to correlate the thermal analytic models and demonstrate the design and functional capability of thermal control hardware.

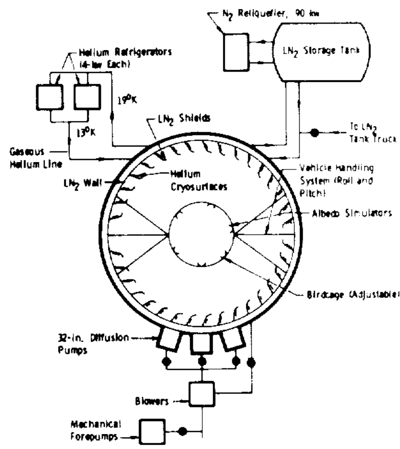
A variety of components, often tested to different temperature extremes during component qualification and acceptance, must be accommodated during SV thermal vacuum testing. The approach taken is to drive as many components as possible (but at least one component per vehicle equipment zone) to their qualification or acceptance temperature extremes, with the constraint that no component should exceed its component level test temperature extremes. This requires pretest analysis, use of test equipment and instrumentation, and local heating or cooling within the chamber. Safeguards are necessary to avoid damage during handling and testing.

ARNOLD ENGINEERING DEVELOPMENT CENTER
AEROSPACE ENVIRONMENTAL CHAMBER (MARK I)

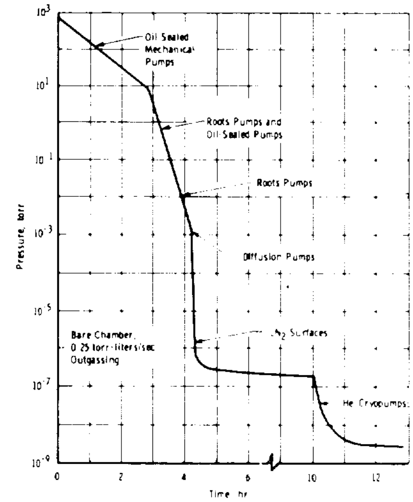
The AEDC Mark I Chamber, in Manchester, Tennessee, is described in order to illustrate a large thermal vacuum facility. The 42-ft diameter, 82-ft high chamber is housed in a 10-story building. It features a 20-ft diameter top hatch for vehicle entry and an 8-ft bottom hatch for personnel access. The cool-down and pump-down systems are shown in the schematic. They feature an 8 kW gaseous helium refrigeration system and a 90 kW nitrogen reliquification system. Diffusion pump capability is 2×10^5 ℓ/sec at 10^{-7} torr and cryopump nitrogen capability is 15×10^6 ℓ/sec.



Mark I Facility Arrangement



Mark I Schematic



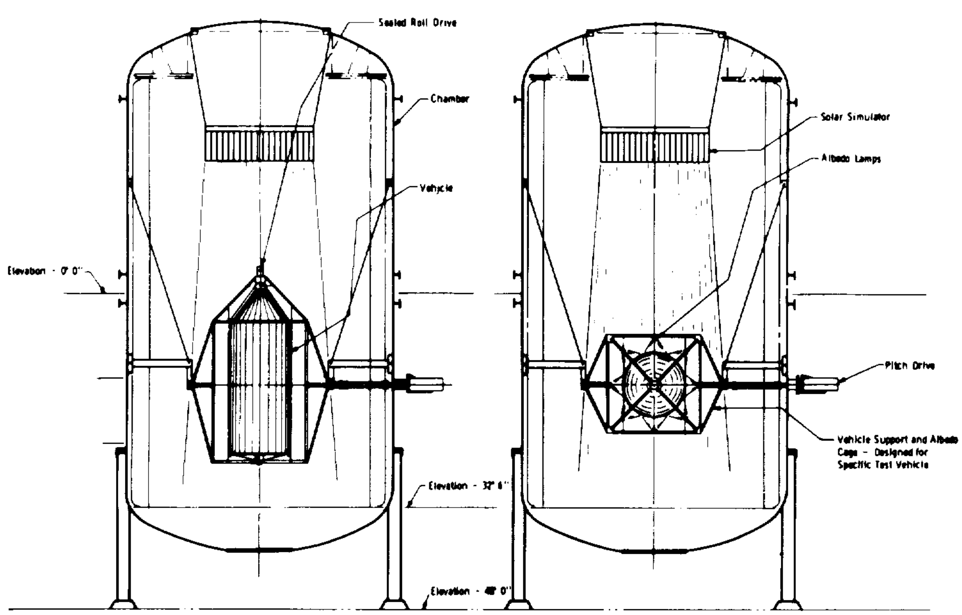
Mark I Pumpdown Curve

ORIGINAL PAGE
BLACK AND WHITE PHOTOGRAPH

The vehicle handling system accommodates moderate size, symmetric shape test articles to 50,000 lb. A pitch drive and slip-ring assembly is used for power transfer. The handling system is capable of simultaneous real time motion about two axes. However, wire bundles and test instrumentation leads may make this impractical. The Block II GPS-NAVSTAR, recently tested in this chamber, utilized motion about one axis to simulate the time-varying solar vector for the beta-equals-zero orbit.

Solar simulation is accomplished using an array of modules, each containing a 1-kW quartz-iodine lamp and a water-cooled collimator tube. As the created spectrum approximates a 3000°K blackbody, with the sun more nearly like a 5800°K blackbody, augmenting xenon short-arc lamps can be used to improve spectral matching. The Mark I system is capable of continuously variable radiation for 0 to 110% of the solar constant with $\pm$ 3% uniformity. Solar simulation is the preferred method of spacecraft heating, as this technique allows the natural blockage and cavity effects to occur, while imposing direct and reflected solar-like radiant heating. This method also creates infrared sources, which can approximate actual self heating by virtue of reradiation of absorbed solar energy. Because of cost and complexity, spacecraft heating is often done by methods that do not simulate the spectral content and directionality of the sun, but do attempt to impose the proper intensity and distribution of heating.

The cold environment of space is well-simulated by a liquid nitrogen-cooled high emissivity internal wall. Because of the fourth power dependence of radiant energy interchange, a wall at 77°K constitutes only a minor radiant energy source for a room temperature spacecraft.



Vehicle in Vertical Position   Vehicle in Horizontal Position

SPACE VEHICLE (SV) THERMAL BALANCE TEST

This test formally qualifies the Thermal Control Subsystem (TCS). It is used to correlate the analytic thermal models; to verify the design and performance of TCS hardware such as insulation blankets, louvers, heat pipes, and heaters/thermostats; and to demonstrate that the TCS maintains all payloads and equipment within allowable temperature limits for all mission phases under worst case environments. This test should be conducted for one-of-a-kind spacecraft; the lead vehicle of a series of spacecraft; and a block change in a series of vehicles, upper stages, and sortie pallets designed to fly with the Shuttle.

The thermal balance test is conducted in a cryogenically cooled thermal vacuum chamber. The tests should simulate worst case combinations of equipment usage (primary and redundant), bus voltage, and solar angles and intensities. During these largely steady state tests all important internal heat flow paths and external radiative surfaces should be exercised. Some tests typically involve simulation of non-operational or transient mission phases: transfer orbit cooldown, eclipse, safemode entry or exit. Large appendages such as solar arrays, booms, and antennas are sometimes not part of the tested configuration. Both stowed and deployed vehicle configurations may be tested, requiring vacuum break. Environmental heating is usually simulated by infrared lamps, heated (radiating) plates, and/or test heaters affixed to external surfaces. Solar simulation is less frequently used.

The contractor should compare pretest temperature predictions with corresponding test data. The Aerospace Corporation has proposed, as a guideline, that those differences that fall outside a $\pm$ 3°C band require either a good explanation or a model adjustment, depending on the size of the deviation. In practice, deviations as large as $\pm$ 6°C are often accepted, with narrower limits for temperature-sensitive or mission-critical components.

A variety of test-related factors contribute to a fairly large residual analytic uncertainty after completion of the thermal balance test. These include imperfect spectral matching, inadvertent test heat losses, end-of-life properties not simulated, test set radiation blockage, and measurement and calibration error.

Model correlation to test data may not be effective if an incorrect heat transfer mechanism is employed. Some design changes that were made because of thermal balance test results are not test verified until the acceptance test of the first flight vehicle and, sometimes, unfortunately there is no test validation.

Overall, the thermal balance test has proved successful in correcting major thermal modeling errors, in reducing the standard deviation between prediction and flight measurements, and in providing physical insight into heat transfer mechanisms.

The thermal balance test and portions of the thermal vacuum test serve to verify the design and performance of thermal control hardware. Primary and redundant heaters and thermostats are exercised and the circuitry is proven, location and response time is verified, and 25% excess heater control authority is demonstrated for the cold case. Radiator surface emissive power and insulation blanket effective emissivity are verified. Performance of louvers and heat pipes (if horizontal) is characterized. The ability of the TCS to maintain SV components within their specified temperature extremes under worst hot and cold case conditions is demonstrated.

THERMAL CONTROL REQUIREMENTS TO SUPPORT FACTORY AND LAUNCH SITE CHECKOUT AND FUNCTIONAL TESTING

Checkout and functional tests are required at various stages during the buildup of a space vehicle. Such tests often are not part of the formal developmental, qualification, and acceptance process. For example, these tests: (1) allow checkout at intermediate stages during the buildup process, (2) can verify that a subsystem has not been damaged or degraded during shipment, and (3) allow continuity, checkout, and limited functional tests during and after assembly at the launch site. Thermal control (i.e., gas or liquid cooling) often is required to ensure that components do not overheat during these tests. Compounding the difficulty of this requirement is the fact that the subsystem or space vehicle configuration and surrounding environment can encumber the cooling process. The cold radiation sink for which the space vehicle is designed is lacking during these tests, and natural convection cooling is not very efficient. Moreover, the subsystem or space vehicle may be oriented so that heat pipes are inoperative and may be enveloped with contamination covers, shrouds or the like, so that there is limited accessibility to fluid cooling.

It is important to identify, early in a program, factory and launch site cooling requirements for checkout and functional tests. This is especially important for sensitive components such as batteries. Space vehicle design accommodations and auxiliary ground equipment which may be required to allow adequate cooling should be specified. This may include ducting and fans, piping and pumps, and leveling hardware and instrumentation.

UNIFIED FAILURE THEORY - DEMISE OF THE BATHTUB CURVE

Bezat and Montague (Ref. 3) have used laboratory and field failure data for the Honeywell Digital Air Data Computer to develop the failure rate curve below. The data base encompassed 6.5 years of revenue service and 11 x 10^9 part hours. The authors point out that the decreasing failure rate with time is consistent with their experience with semi-conductor devices. Herbert and Myron Hecht (Ref. 4) report a similar trend for spacecraft. Their data base was obtained from over 300 satellites, comprising 96 programs, launched between the early 1960s through January 1984. Primary data sources were The Aerospace Corporation's Orbital Data Analysis Program (ODAP) and the On-Orbit Spacecraft Reliability (OOSR) data compiled by the Planning Research Corporation for NASA. This and other data were the basis for Wong's paper, "Unified Field (Failure) Theory - Demise of the Bathtub Curve" (Ref. 5). Wong points out that the same failure pattern is seen in the laboratory, manufacturing screening, in the field, and that failure rate for electronic equipment trends downward (although the path may have some bumps) for all times of practical interest.

The implications for spacecraft testing and reliability, as we see it, are as follows:

1. No amount of testing will prevent infant mortality failures.

2. Testing can reduce the initial failure rate of this downward trending curve.

3. Provided that failures are detected and repaired, electronic equipment cannot be worn out by testing.

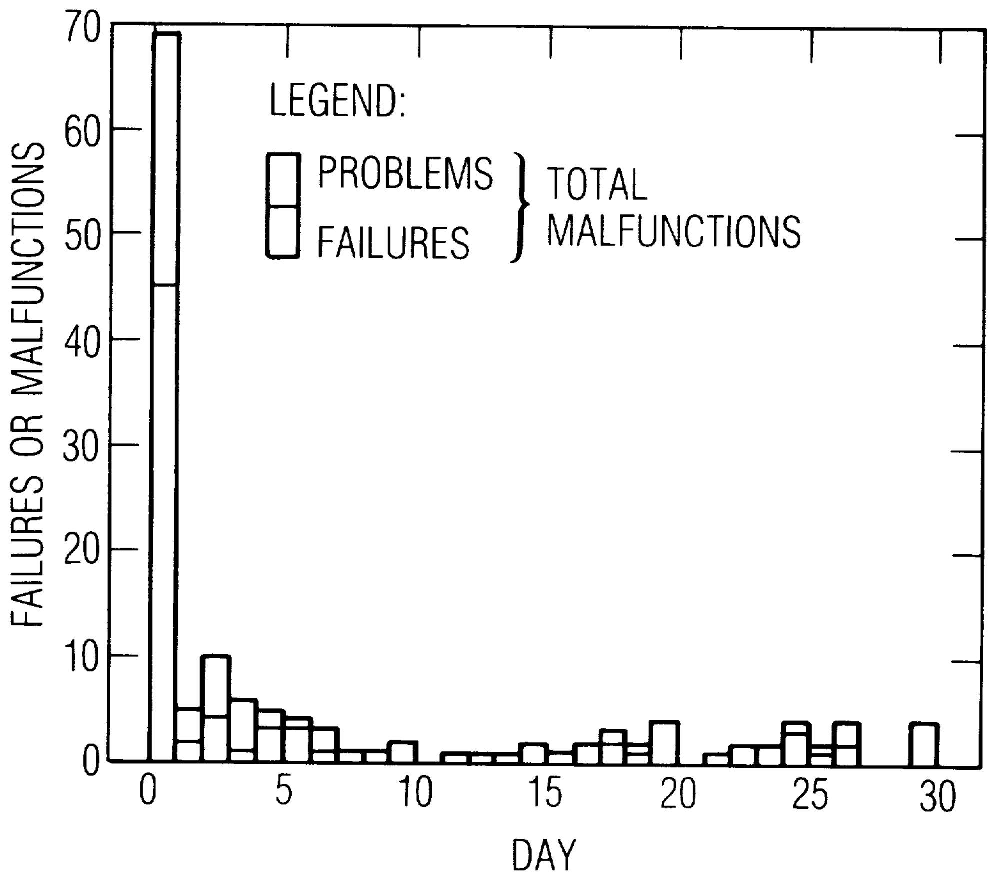
4a. Accelerated testing at high stress levels (even beyond flight levels) may be very beneficial for long term reliability.

4b. Ambient temperature burn-in with little monitoring is ineffective in screening defective equipment.

5. Quality standards and testing requirements fall off very slowly as mission duration decreases.

NASA/GODDARD EARLY ON-ORBIT FAILURE DATA

The work of Timmins (Ref. 7) on NASA/Goddard programs shows that early failures are dominated by first day failures. No corresponding day-by-day failure data has been assembled by The Aerospace Corporation. However, a cursory review by Tosney shows a similar trend, with first day of usage failures quite high.



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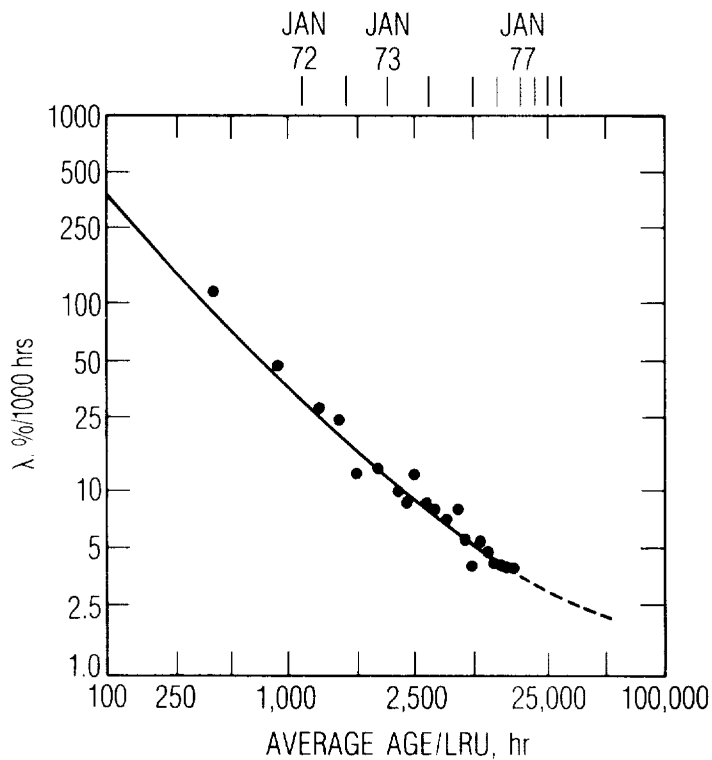
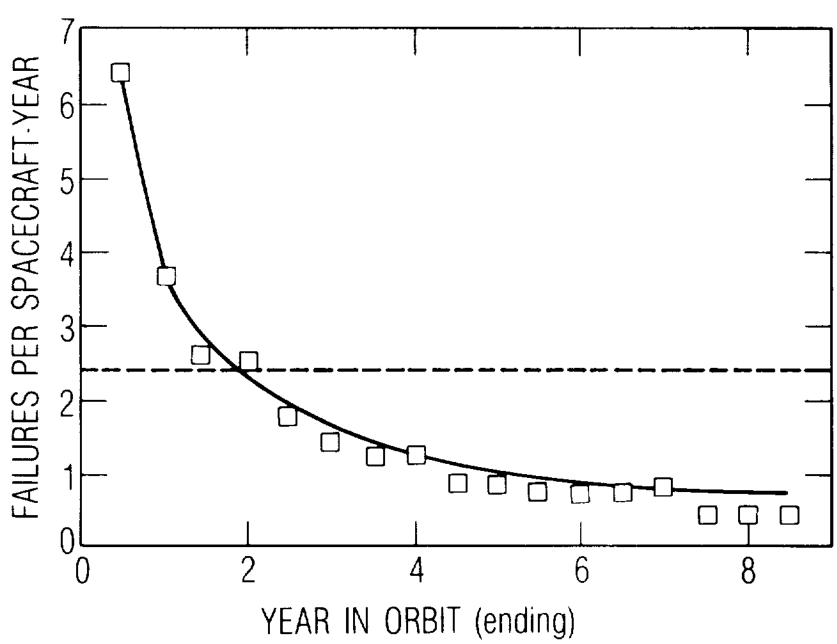
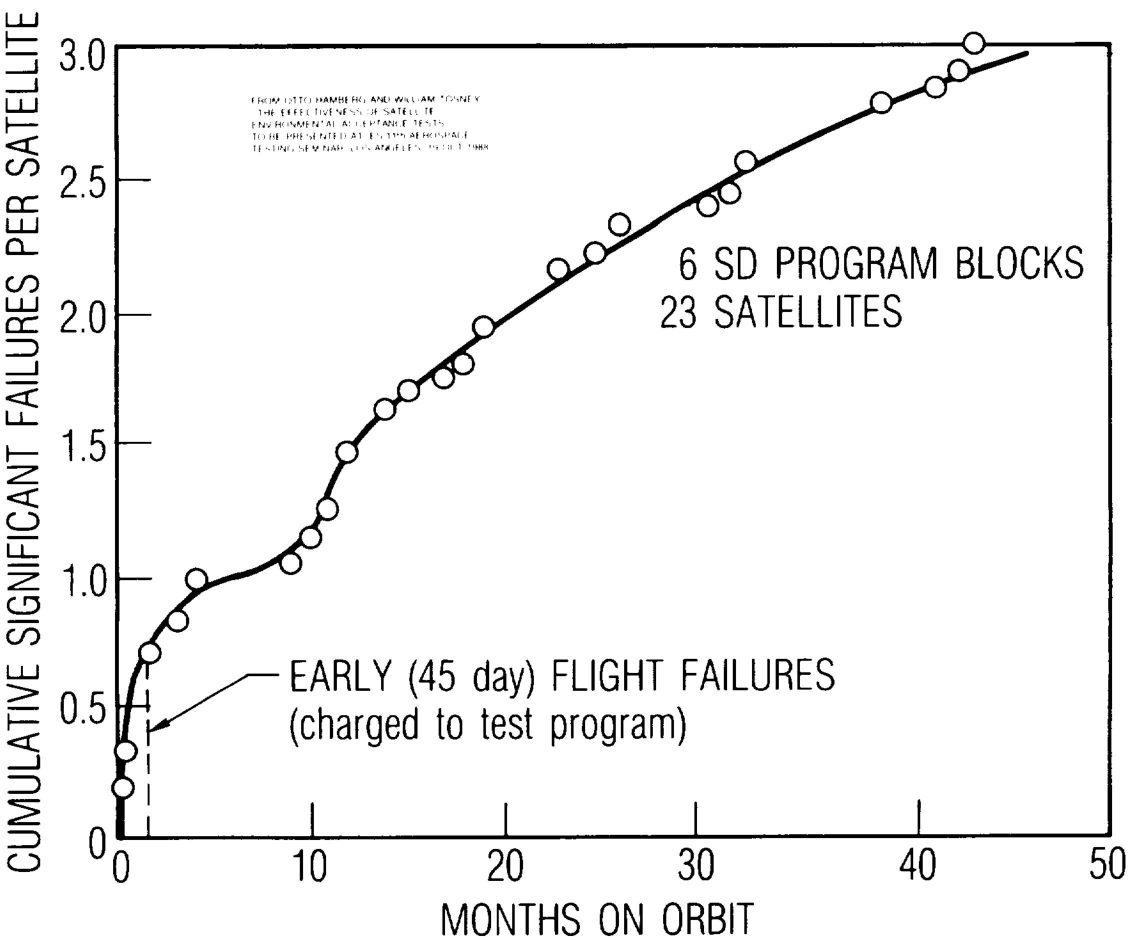
Honeywell Digital Air Computer Failure Rate Curve



Hecht Study-Satellite Failure History



The flight failure history of six Air Force program blocks and 23 satellites is shown (Ref. 6). Only the initial four satellites from each program block were included to minimize the effect of program maturity, and only mission degrading (changes satellite reliability) failures are included. The data were obtained from The Aerospace Corporation's ODAP. It can be noted that the initial high failure rate has moderated somewhat by 45 days. This timeframe coincides with satellite launch, ascent, and the in-orbit operational performance tests. This high failure rate period is considered to be related to the imperfection of the ground test program. The infant mortality period appears to extend out to approximately 12 months of operational flight time. The failure rate after 12 months shows a slowly decreasing rate which is in agreement with the work of the Hechts (Ref. 4).



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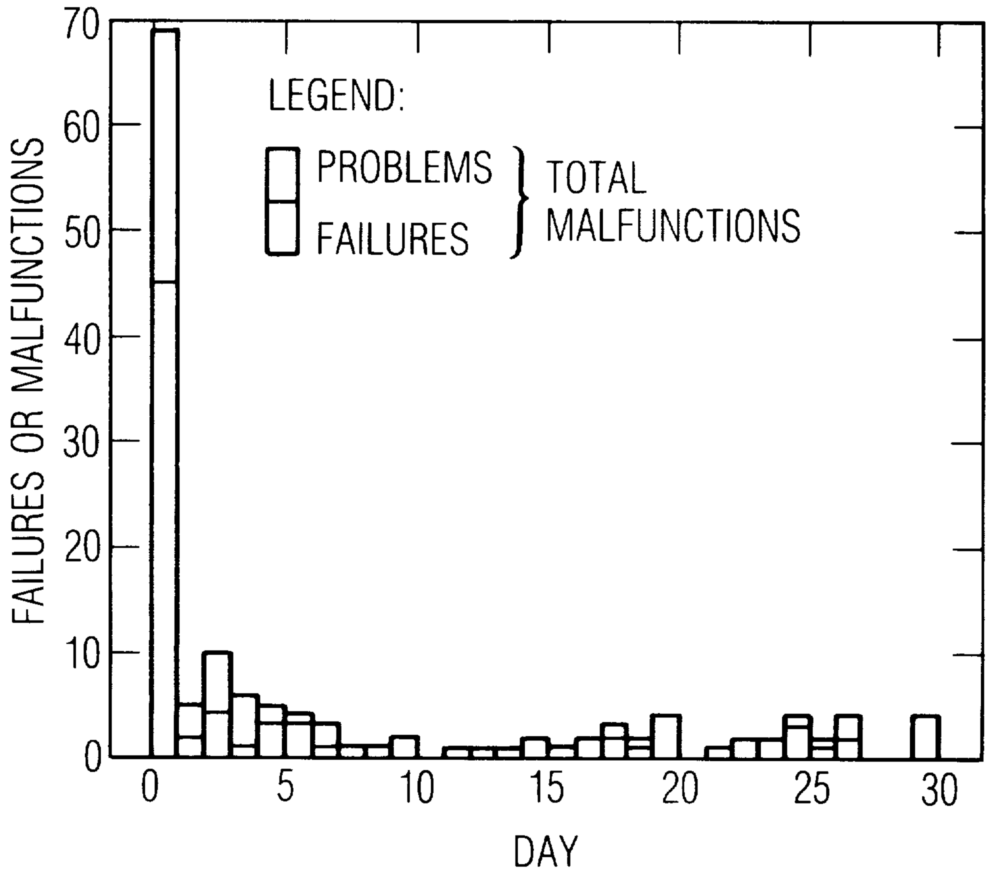


DEFINITION OF TEST EFFECTIVENESS

The premise underlying the definition of test effectiveness (Ref. 8) is that failures found in environmental tests would have occurred early in flight (first 45 days); these early failures are charged to the test program. Aerospace's ODAP data base was used with only significant test and early flight failures considered. Such failures potentially reduce mission life. Generic failures were counted only once and induced failures not counted. This first order method attempts to account for test sequence as illustrated below.

- QUANTITATIVE MEASURE TO EVALUATE/COMPARE TESTS

$$\frac{\text{TEST FAILURES}}{\text{TEST PLUS FLIGHT FAILURES}}$$

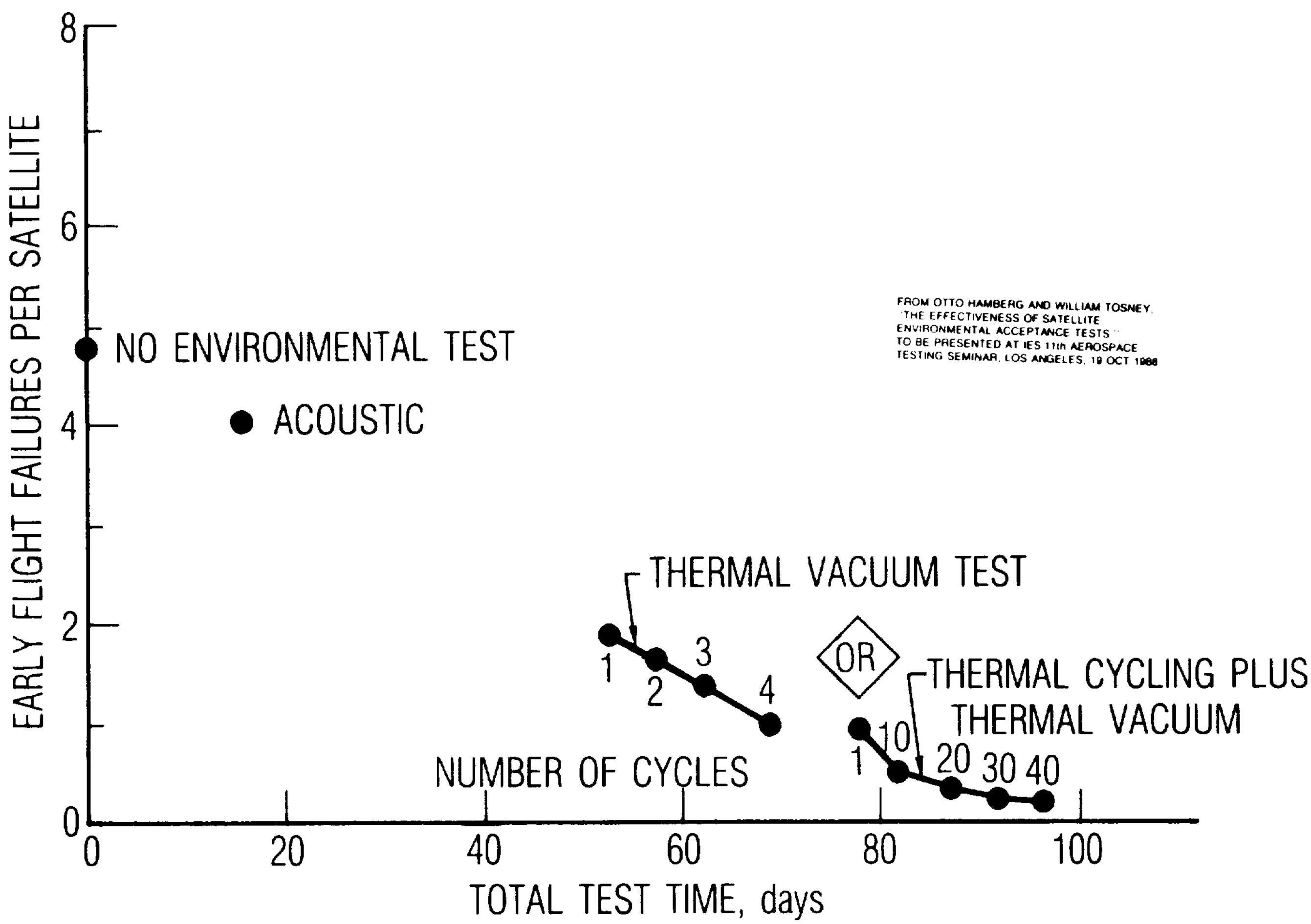
- EXAMPLE PROGRAM A
  FAILURES PER SATELLITE (average of satellite group)

TESTS			FLIGHT
ACOUSTIC	THERMAL CYCLING	THERMAL VACUUM	45 day
0.9	1.4	1.6	0.6

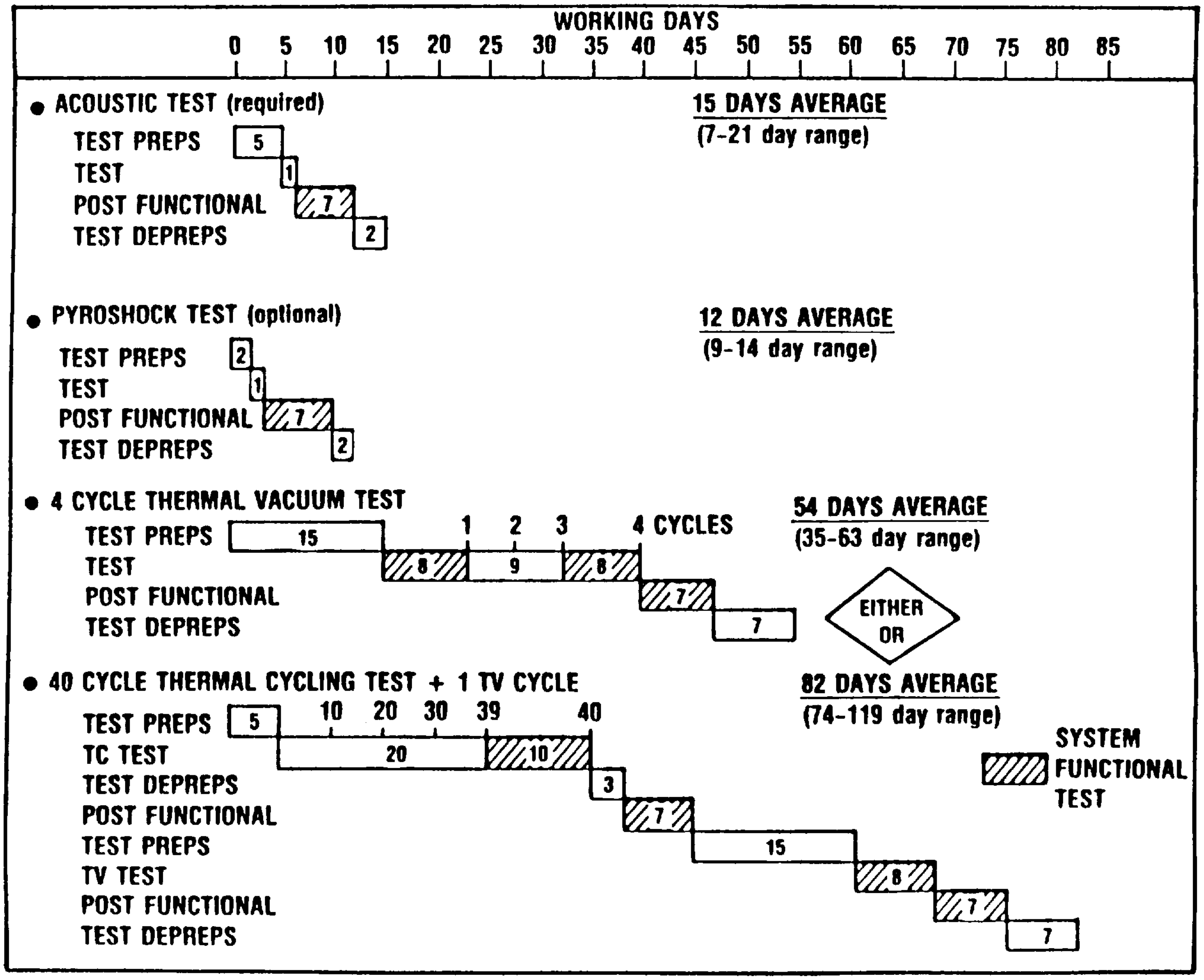
- TEST EFFECTIVENESS

		PERCENT
– ACOUSTIC	= (0.9)(100)/(0.9 + 1.4 + 1.6 + 0.6)	= 20
– THERMAL CYCLING	= (1.4)(100)/(1.4 + 1.6 + 0.6)	= 39
– THERMAL VACUUM	= (1.6)(100)/(1.4 + 0.6)	= 73
– COMBINED	= (0.9 + 1.4 + 1.6)(100)/(0.9 + 1.4 + 1.6 + 0.6)	= 87

The data bank (Ref. 8) developed by Laube has been used by Hamberg and Tosney (Ref. 6) to generalize about the effectiveness of space vehicle environmental acceptance tests in eliminating first-45-day mission degrading failures. On the average, in the absence of any environmental tests, 4.5 early failures per satellite are anticipated. The acoustic test while only moderately successful at eliminating early failures (0.63 per satellite) is a relatively short test, 15 days. The four cycle thermal vacuum test or the optional 40 cycle thermal cycling test plus one cycle thermal vacuum test, while markedly more successful at eliminating early failures, are time consuming. As a rule of thumb environmental testing avoids about 0.05 early flight failures per day of test.



Generalized Failure Data


WORKING DAYS
0 5 10 15 20 25 30 35 40 45 50 55 60 65 70 75 80 85
ACOUSTIC TEST (required)
TEST PREPS
TEST
POST FUNCTIONAL
TEST DEPREPS
5
1
7
2
15 DAYS AVERAGE
(7-21 day range)
PYROSHOCK TEST (optional)
TEST PREPS
TEST
POST FUNCTIONAL
TEST DEPREPS
2
1
7
2
12 DAYS AVERAGE
(9-14 day range)
4 CYCLE THERMAL VACUUM TEST
TEST PREPS
TEST
POST FUNCTIONAL
TEST DEPREPS
15
1 2 3
4 CYCLES
8
9
8
7
7
54 DAYS AVERAGE
(35-63 day range)
EITHER OR
40 CYCLE THERMAL CYCLING TEST + 1 TV CYCLE
TEST PREPS
TC TEST
TEST DEPREPS
POST FUNCTIONAL
TEST PREPS
TV TEST
POST FUNCTIONAL
TEST DEPREPS
5
10 20 30 39 40
20
10
3
7
15
8
7
7
82 DAYS AVERAGE
(74-119 day range)
SYSTEM FUNCTIONAL TEST

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