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SPACE ENVIRONMENTAL EFFECTS ON MATERIALS

By R. J. Schwinghamer
Materials and Processes Laboratory

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16. ABSTRACT

The Space Shuttle will provide a low cost delivery system for Earth orbital payloads by amortizing launch costs through system reusability. This development paves the way for large platforms and structures in space. But successful design of long life platforms and structures for space use requires due consideration of space environmental effects on the materials used. Large space system materials, especially those in geosynchronous Earth orbit (GEO), will be subjected to vacuum, ultraviolet radiation and charged particle radiation which will influence the performance and functional lifetime of the systems. This report describes research oriented toward the acquisition of long term environmental effects data needed to support the design and development of large low Earth orbit (LEO) and GEO space platforms and systems for the next decade.

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INTRODUCTION

The Space Shuttle will provide a low cost delivery system for Earth orbital payloads by amortizing launch costs through system reusability. This development paves the way for large platforms and structures in space. But successful design of long life platforms and structures for space use requires due consideration of space environmental effects on the materials used. Large space system materials, especially those in geosynchronous Earth orbit (GEO), will be subjected to vacuum, ultraviolet radiation and charged particle radiation which will influence the performance and functional lifetime of the systems. This report describes research oriented toward the acquisition of long term environmental effects data needed to support the design and development of large low Earth orbit (LEO) and GEO space platforms and systems for the next decade.

Addressed are the four major considerations of the space environment for LEO and GEO missions: space vacuum, electromagnetic radiation, particulate radiation and space debris. The significance of each environmental element is then explored in turn.

As a part of the vacuum environment parameter, both ascent and orbital vacuum considerations are discussed, as are orbital vacuum evaporation and sublimation of materials, adsorbed gas evolution effects, and multipacting of both electrical insulating and conducting materials.

Relevant electromagnetic radiation effects on typical materials are reviewed. Electromagnetic irradiation of passive thermal control coatings, solar illumination effects on the conductivity of thin film insulating materials and UV irradiation effects on glass materials characteristics are discussed.

The space particulate radiation environment is stipulated, delineating both flux and energy level for: radiation of galactic cosmic, solar cosmic, neutron, solar wind, plasma sheet electron and proton, and geomagnetically trapped, origin. Some generic effects of electron and proton irradiation of materials are given.

Space debris is considered in the context of GEO missions, and probable meteoroid penetrations versus stay time and structure size is given.

Specific LEO and GEO environmental parameters are then postulated, as a prelude to the section dealing with space environmental effects design data requirements.

The data requirements section includes a review of the crucial considerations for space environmental effects experimental testing, and moves rapidly into specific discussions of space environmental effects on solar cells, on composite materials, flexible thin film materials, and materials susceptible to electron radiation induced electrical discharge. The section concludes with a summary of specific component materials radiation environmental effects data and known test conditions, and concludes that additional data from long duration exposure in the combined environment are mandatory.

THE SPACE ENVIRONMENT

Any rational consideration of large space platforms or large space structures must inevitably begin with due consideration of the space environment. For purposes of this discussion, the space environment of primary concern shall be that associated with orbital operations in low Earth orbit (LEO) and in geosynchronous Earth orbit (GEO). The perceived nature of the space environment in these orbits is changing continuously as more information becomes available. Even with currently available data, there still exists some uncertainty about the environmental conditions, especially for long term operation in GEO. This is due to the difficulty associated with obtaining the data, the fact that not enough space environmental effects on materials data are being acquired, and because of the changing nature of some of the parameters.

The space environmental factors affecting the behavior of materials which are of most concern are:

1) Vacuum

2) Electromagnetic Radiation

3) Particulate Radiation

4) Space Debris.

Each of these space environment factors will be discussed.

EFFECT OF SPACE VACUUM ON MATERIALS

Vacuum Considerations Ascent

An immediate physical penalty for travel into space comes into play at liftoff of the launch vehicle. A gross reduction in atmospheric pressure

manifests itself exponentially as the vehicle rises. This gross reduction in ambient pressure is rarely underestimated, or unaccounted for by current aerospace designers, although in the past there have been gears in rocket engines which seized due to evaporation of lubricating oil, and other rapid decompression type mishaps. Another tangible detrimental effect of this atmospheric decompression is the propensity for some types of insulating nonmetallic materials to violently decompress, giving rise to the peculiar result known as "pop-corning" or local area exploding of the materials. Closed cell materials obviously are more prone to this kind of "explosive" degradation. In testing of thermal protection materials for launch vehicles at the Marshall Space Flight Center (MSFC), there is always an initial screening of candidate materials consisting of stimulating liftoff and early flight thermal/vacuum environments. Figure 1 shows the MSFC facility in the Materials and Processes Laboratory during early screening testing of the low cost MSA-1 thermal protection material, which is an intermediate weight, low cost epoxy terminated polyurethane spray-on ablator material used on the Space Shuttle Solid Rocket Boosters. When a TPS material does not survive this simulation of the ascent thermal/vacuum conditions, there is no subsequent aeroshear or other special aerodynamic effects testing done. Perhaps less obvious and more subtle in many respects, are the effects of the high vacuum of the orbital environment on materials.

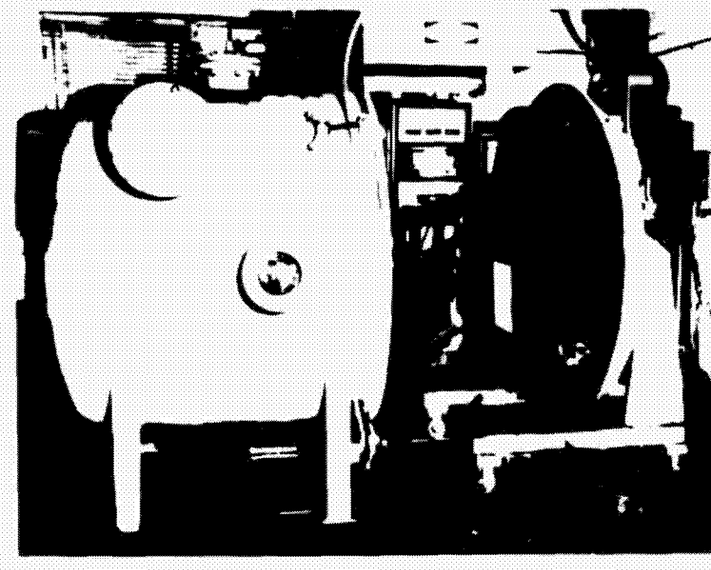


Figure 1. MSFC thermal/vacuum simulator during screening testing of low cost MSA-1 intermediate weight ablator material for use on Space Shuttle Solid Rocket Boosters.

Vacuum Considerations - Orbital

Future space platforms and space stations will be subjected to a high vacuum environment for several years, rather than days or months. The orbital vacuum environment is characterized by extremely low density and a correspondingly low pressure (high vacuum). The space vacuum varies with altitude and with the relative level of sunspot activity, and there is also some diurnal effect. But the actual physical pressure manifestations associated with sunspot and diurnal changes are actually extremely small. For example, at 500 km altitude, the pressure varies due to sunspot activity from 8.25×10^{-10} Torr at sunspot minimum, to 9.0×10^{-9} Torr at sunspot maximum [1]. While this is indeed almost an order of magnitude change, in terms of physical pressure it is quite small. Molecularly speaking though, and in terms of mean free path considerations, very small pressure changes may have dramatic effects.

In the high vacuum orbital environment there are several effects on materials which bear consideration:

1) Evaporation or sublimation of materials

2) Removal of adsorbed gas from the surface of materials

3) Multipacting of materials.

Evaporation or Sublimation of Materials

Evaporation and sublimation in the high vacuum of an orbital environment are crucial not so much from a structural-weakening-of-heavy-structures point of view, but more so from the contamination or "interference" aspect. To better appreciate this, a brief review of high vacuum concepts is helpful.

The mean free path (MFP) concept is central to an understanding of molecular behavior under orbital vacuum conditions. The MFP of a gas is the average distance which a gas particle moves between collisions, and is aptly described by the following equation:

$$MFP = \frac{KT}{Pd_m^2}$$

where T is temperature, d_m is the molecular diameter, P is the pressure, and K is a constant. The mean free path of a nitrogen molecule at constant temperature increases from about 6.7×10^{-6} cm at 760 mm Hg

pressure, to 500 cm at 10^{-5} Torr, a typical hard vacuum in many vacuum systems. A typical MFP of a N_2 molecule at 0°C at a LEO altitude of 500 km however would be about 6 million meters. However, early in the orbital mission, in the immediate vicinity of the space platform, pressures would be higher due to outgassing and hence mean free paths would be proportionately shorter. With long MFP's the propagation of evaporated or sublimed molecules can be considered to be line-of-sight, and shadow shielding techniques are feasible to protect critical optical surfaces or other sensitive materials from having material deposited on them. An estimate of the rate of evaporation of a pure compound material being outgassed from a source aboard a space platform can be made using Langmuir's equation:

$$G = a\, P\mu \sqrt{\frac{M}{T}}\ K\ \text{gm cm}^{-2}\ \text{sec}^{-1}$$

where

a = condensation coefficient

$P\mu$ = vapor pressure

M = molecular weight of substance being evaporated

K = constant

T = temperature.

Clearly, the vapor pressure and operating temperature of materials for use in orbital structures do influence the level of contamination to be expected. For simple evaporation, even when uncomplicated by any sputtering influence, the way in which material contaminant films deposit is a very complex situation, and depends on the substrate temperature, geometry to some extent, and on the energy, mobility, and nucleation relationships between film atoms and substrate [2]. It is frequently possible to operate certain sensitive spacecraft materials at somewhat elevated temperatures, thereby preventing this condensation of contaminating films. Nonmetallic materials in general are the more prolific outgassers, but cadmium, for example, is one of the most potentially contaminating of the metals because it has a relatively high vapor pressure. Calculation of the loss of this metal using the Langmuir relationship above, indicates that at 120°C, cadmium could lose a maximum of 0.040 in. per year, in orbit. Magnesium at 240°C could lose a similar amount. The use of zinc in space platform design, because of its high vapor pressure at 10^{-5} Torr at 211°C, is also to be discouraged. Fortunately, the other metals and alloys of interest for space platforms and structures require very high

temperatures to lose any appreciable weight – so high in fact that for all practical purposes, weight loss is not likely to occur. Sublimation of refractory metals is also essentially nil at typical orbital temperatures. Most ceramics and refractory compounds have very low vapor pressures under orbital temperature conditions, and most of these tend to decompose or sublime rather than vaporize.

Polymers are another matter, and they lose weight by outgassing of both volatile materials and entrapped gasses. The volatiles consist of low molecular weight polymer fractions of monomers, impurities and additives, and except for the very few structural metals noted above, are much more prone to produce contamination problems in space, than are structural metals.

The Langmuir relationship noted previously is less utilitarian for polymer outgassing rate determination simply because the vapor pressure and the molecular weight of the gas species evolved must be known, but unfortunately for most polymeric compositions these parameters are not usually known, and must be determined experimentally. In general, the long-chain, high molecular weight polymers of a high degree of purity are less prolific outgassers. But the general propensity of the organic materials to outgas can be a very serious problem in many respects. The "bad actor" metallic materials can cause a variety of electrical problems such as corona, shorting and arcing, if they vapor deposit. While the organic materials are much less likely to produce electrical problems than vapor deposited metallic films in space vacuum, they do have the potential to seriously impair the performance of optical systems, to interfere with certain measuring systems on a particulate basis, and to degrade thermal control coatings. In terms of general susceptibility, the optical systems have been the most sensitive to these vacuum deposition degrading effects. On the U.S. Skylab experiments, some of the optics were so sensitive to contamination that the materials in these systems were checked for contamination potential as slight as the 300 Å level. A special specification was devised to insure that the Apollo Telescope Mount (ATM) would indeed "fly clean" in Earth orbit [3], and it did so very well. This specification essentially required materials used in the ATM design to have a rate of weight loss during temperature cycling from 25° to 100°C not to exceed 0.2 percent per cm^2/hr, when heated at a rate of 2°C per min at 10^{-6} Torr. Additionally, a residual gas analysis required that no outgassed particles be present which were greater than atomic mass unit 44, nor greater than two orders of magnitude below the base peak in the system. This specification has since been supplemented by SPR-R-0022A [4]. This latter specification currently applies to the NASA Space Shuttle Project, and differs from its former sister-spec only in that it requires materials for space environment use to have a total mass loss no greater than 1 percent of the original mass, and a maximum volatile condensible material content of only 0.1 percent of the original mass of the material. In selecting materials for the Viking Spacecraft design, for example, vacuum contamination criteria were applied. Martin Marietta

Corporation (MMC) developed a test technique which essentially required the materials to meet both isothermal weight loss in N_2 and volatile condensible outgassing requirements [5]. If the isothermal weight loss was greater than 1 percent, or if the condensible outgassing rate was greater than 1×10^{-4} percent per day, the material was rejected. Figure 2 gives the typical results of testing of a variety of polymers for thermal/vacuum compatibility at MSFC. It should be noted that these results can be profoundly affected by seemingly "insignificant" changes in the processing of the polymers, and to some extent by aging, curing or preconditioning at elevated temperature, and most certainly by preconditioning through vacuum baking to outgas volatiles. The fact that the processing of polymers is frequently changed "insignificantly" by the supplier without apprising the user, assuredly gives cause to use outgassing data primarily as a guide, unless the given outgassing test is on the precise material to be physically used in the space hardware. It naturally follows then that sure-fire analytical prediction of outgassing characteristics is impossible.

ILLUSTRATIVE VACUUM OUTGASSING DATA

GENERIC CLASSIFICATION	SPECIFIC MATERIAL	*WEIGHT LOSS @ 100°C & 10^{-6} TORR (%-cm^2-sec)	**TWL/VCM (%)
EPOXY	ECCOBOND 56C/CAT.9	0	0.38/0.03
	EPON828/CAT. D	0.08	0.97/0.02
	SCOTCHCAST 281	0.57 (HIGH N2 CONTENT)	0.36/0.05
	HT 424	0.14	0.18/0.09
URETHANE	SOLITHANE 113	0.17 (HIGH N2 CONTENT)	0.31/0.01
	PR 1538	0.20 (HIGH N2 CONTENT)	0.46/0.02
SILICONE	RTV 560	0.296	1.03/0.68
	RTV 566	0.052 (% TWL)	0.13/0.02
	RTV 8111/CAT. 8881	0.67	1.02/0.28

*MSFC SOMO2442

**JSC 08962, REV. μ

(PER SP-R-0022A TEST CRITERIA)

Figure 2. Weight loss of specific polymers under thermal vacuum conditions.

In the context of outgassing potential, lubricants for orbital applications are deserving of special attention because of the criticality of operating mechanisms such as control moment gyroscopes, shutters, doors, racks and pinions, gears and other devices needed to sustain an orbital platform. Lubricants are needed in orbital operations for a variety of purposes besides minimizing frictional drag. They are needed for reducing wear, sealing against contamination, and carrying away unwanted heat. A lubricant's volatility is a function of its vapor pressure, and vapor pressure is related inversely to molecular weight. In general, lower weight fractions of a lubricant will be lost through volatilization before

the heavier constituents. It does not necessarily follow however that oils with the very lowest vapor pressures and evaporation rates will give the longest life in vacuum service. The only conclusive means of determining vacuum service lifetime is by long term testing. A series of tests is well underway at MSFC which involves 33 grease type lubricants in R-4 size bearings, in five different environments, for periods of 1 year. From these screening data, four lubricants have already been selected for 5 year vacuum tests in selected environments, with 5 repetitions of each test planned to insure statistical validity [6]. Figure 3 shows a summary of the 1 year tests. The superiority of the perfluoroalkylpolyether (PEPE) polymer greases is readily apparent. Figure 4 is a rather simplistic guide to selection of lubricants for use in the vacuum environment. It may be useful in a general sense, for busy non-tribological expert individuals who need some generalized information quickly.

MANUFACTURER DESIGNATION	GENERAL CHEMICAL CLASS OF BASE OIL	VACUUM TESTS AT 38°C			VACUUM TESTS AT 93.3°C			VACUUM TESTS START-STOP			OXIDIZING TESTS			LOW TEMP START TESTS °C	
		①	②	③	①	②	③	①	②	③	①	②	③	①	②
KG-69	MINERAL	11	7605	24	8	8780	35 5	10	8200	17	4	8760	10 6	10	-4 00
SRG-200	MINERAL	10	8760	51	8	8760	40	15	7832	20				21	+0 40
AEROSHELL 5	MINERAL	5	8760	10	7	8760	31 5	6	8760	8 3				17	-15.2
ROYCO 20R	MINERAL	21	1080	77										4	-02 1
ROYCO 40	MINERAL	3	8760	0.8	5	8760	17 4	8	8760	22 8				14	-22 7
AEROSHELL 14	MINERAL	25	320	78.5										5	-59 1
AEROSHELL 18	MINERAL	15	6854	47 5											
APIEZON L	MINERAL	10	4013	4 0											
UNITEMP 600	MINERAL	19	3497	34											
MOBIL GREASE 28	MINERAL	8	8760	22 1				12	7824	30 5					
CONOCO HD #2	MINERAL							8	8760	20 8				15	-21 9
SUPERMIL 00762	DIESTER	14	8200	29 5				14	7200	33 5	3	8760	12	7	51 0
AEROSHELL 17	DIESTER	22	894	63 5											
AEROSHELL 7	DIESTER				17	74	97 1	20	3328	78 5				8	-48 3
L-118	DIESTER	23	834	36 5										8	65 6
EXXON 6102	SYNTHETIC ESTER							5	8760	7 1				11	-44 5
BEACON 326	SYNTHETIC ESTER													12	-41 4
DC NO 33	SILICONE	12	8007	31							7	8760	46 4		
S-361	SILICONE	2	8760	0 5	11	7200	33 5	16	6060	6 5	1	8760	4 0	16	-16 4
SUPERMIL 31002	SILICONE	20	2473	30 0	13	1761	43 5	11	7822	17 5				1	-71 0
S-3300	SILICON				12	8132	27 0								
S-341L	SILICONE							18	5832	21 4				8	-49 2
3427-2	SILICONE	17	5362	36	16	340	16 5				5	8760	40		
FS-1291	FLUORO SILICONE	24	440	15							8	6671	3 5		
FS-1298	FLUORO SILICONE	7	8760	14	10	8273	40 5	17	6072	20 5					
KEL-F NO 90	FLUORO-CARBON				15	569	90 5								
863	PFPEA	8	8760	10	4	8760	17	2	8760	5 8	2	8760	7 8	10	-46 5
3L 30RP	PFPEA	1	8760	8 5	1	8760	14 5	4	8760	7	8	7760	19 2	2	83 4
3L 30RP*	PFPEA							3	8760	9 2				4	-66 8
831A	PFPEA	6	8760	10 5	8	8760	19 5	7	8760	10				20	0 14
240AZ	PFPEA	13	8741	15	14	1587	60	10	5667	74 3	8	8760	47 7	13	35 3
240AB	PFPEA	18	8783	20 3	3	8760	16	13	7270	8					
240AC	PFPEA	4	8760	8 8	2	8760	15 5	1	8760	5 4				18	3 82

* VACUUM BAKED AT 180°C FOR 20 HRS
A PERFLUOROALKYLPOLYETHER
① RATING IN TESTS
② AVERAGE HOURS TO FAILURE OR END OF TEST OF FOUR MOTORS (1 YR = 8760 HRS)
③ PERCENT OF WEIGHT LOSS OF TOTAL WEIGHT OF GREASE ADDED TO TWO BEARINGS OF EACH MOTOR
④ AVERAGE TEMPERATURE OF EIGHT MOTORS (4 MOTORS EACH DAY) TWO DAYS DURATION

Figure 3. Lubricant rating chart according to MSFC 1 year vacuum tests.

Evaporation and sublimation effects on other polymers typically used in orbital satellite design are also jointly under study by MSFC and MMC. Over 40 materials comprising 263 specimens, each having been exposed to thermal vacuum conditions for periods of from 5 years to 8 years, will soon be tested in vacuo. Of these 263 specimens, 63 will be

GENERAL GUIDE FOR VACUUM LUBRICANTS

SOLID FILMS

CLASS	MATERIAL	MFG'S. NAME	COMMENTS	VACUUM RATING
UNBONDED	MoS_2	MOLYKOTE	THREAD LUBE: LOW WEAR LIFE	EXCELLENT
	MoS_2	MIDWEST MIL-1	SPUTTERED: LOW WEAR LIFE	EXCELLENT
	W S_2	DICRONITE	THREAD LUBE: LOW WEAR LIFE	EXCELLENT
BONDED	MoS_2, Sb_2O_3	MIDWEST MLR-2	LOW SPEED, HIGH LOAD, LONG LIFE	EXCELLENT
	MoS_2, GRAPHITE	MIDWEST MLR-4	O_2 COMPATIBLE, LOW WEAR LIFE	EXCELLENT
	MoS_2, GRAPHITE, Sb	NPI 1220, VITRO-LUBE	O_2 COMPATIBLE, LONG WEAR LIFE	EXCELLENT
	MoS_2, GRAPHITE	LUBECO 606	MOD SPEED, LOAD, & WEAR LIFE	EXCELLENT
	MoS_2, GRAPHITE	EVERLUBE 620	MOD SPEED, LOAD, GOOD WEAR LIFE	EXCELLENT
LUBRICATING PLASTICS	FLUORO-CARBONS	TEFLON	LOW LOAD AND SPEED, GOOD WEAR LIFE	EXCELLENT
	POLYIMIDE	VESPEL SP-21	MOD LOAD AND SPEED, GOOD WEAR LIFE	EXCELLENT
	COMPOSITE	DELRIN	LOW LOAD AND SPEED, GOOD WEAR LIFE	EXCELLENT
	COMPOSITE	ARLON	LOW LOAD AND SPEED, MOD WEAR LIFE	EXCELLENT

FLUID & GREASES

MINERAL OILS	GREASE	AEROSHELL 14 OR 15	POOR VACUUM STABILITY, AND LIMITED TEMP RANGE	42% TO 70%
	GREASE	ROYCO 24R		74.6% TO 82%
STRAIGHT CHAIN HYDROCARBON	GREASE	APIEZON L	VACUUM GREASE, POOR LUBRICANT, LIMITED TEMP RANGE	1% TO 11%
HIGHLY REFINED MINERAL OIL	OIL & GREASE	KENDALL NO 60	GOOD VACUUM LUBRICANT, BUT LIMITED TEMP RANGE	21.6% TO 27.6%
	OIL & GREASE	KENDALL GRG 200		30% TO 60%
SILICONES	GREASE	DOW CORNING DC 33	FAIR VACUUM STABILITY, WIDE TEMP RANGE, BUT POOR CORROSION PROTECTION AND	22.6% TO 41%
	GREASE	GE G-351		3.5% TO 12%
	GREASE	SUPERMIL 31062	POOR BALL BEARING LUBRICANT	24.6% TO 62.6%
FLUOROSILICONE	GREASE	DOW CORNING FS1291	FAIR VACUUM STABILITY, WIDE TEMP RANGE, BUT POOR BALL BEARING LUBRICANT	7.5% TO 22.5%
	GREASE	DOW CORNING FS1292		7% TO 21%
DIESTER	GREASE	SUPERMIL 06752	FAIR TO POOR VACUUM STABILITY,	10.6% TO 30.6%
	GREASE	AEROSHELL 17	GOOD TEMP RANGE, BUT POOR	50% TO 72.5%
	GREASE	EXXON 8182	BALL BEARING LUBRICANT	DATA NOT COMPLETE
PERFLUORO-ALKYLPOLYETHER	GREASE	BRAY 803	GOOD VACUUM STABILITY,	10.6% TO 33%
	GREASE	BRAY 24,30AP	WIDE TEMP RANGE, GOOD BEARING	5% TO 8.5%
	GREASE	DUPONT KRYTOX 240AB	LUBRICANT, BUT POOR CORROSION RESISTANCE	3.5% TO 40.5%

* PERCENT LUBRICANT LOSS IN 1 YEAR BALL BEARING TESTS IN VACUUM @ 30°C

Figure 4. Simplistic guide for vacuum lubricants.

also irradiated with charged particles in addition to the thermal vacuum exposure. In vacuo mechanical, electrical, thermal and optical properties measurements will be accomplished. Through the farsightedness of the MMC, and certain MSFC personnel, specimens of interest with this considerable antiquity are available for these tests. Figure 5 shows the polymers, number of specimens, age, temperature condition and test method to be employed on the 8 year old specimens. We confidently expect these data to be valuable to designers in the selecting of materials for LEO and GEO long term orbital missions.

A final objective of these tests will be to determine the validity of proper degradation prediction models. The thermal/vacuum weight loss relationship has been found by MMC to follow the relationship:

$$\frac{dx}{dt} = \frac{K_t}{(Ao-X)}$$

MATERIAL	GENERIC TYPE	NO OF SPEC	TYPE SPECIMEN	MONTHS EXPOSURE TIME (31/88)	COND TEMP	TEST METHOD
F411 LACING CORD	POLYTETRAFLUORO	3	ANDT STRENGTH	100	AMB	CCC T 191
F411 LACING CORD	ETHYLENE/GLASS TAPE	3	TENSILE STRENGTH	100	AMB	CCC T 191
KAPTON H	POLYIMIDE	5	TEAR (MACHINED DIRECTION)	101	150°F	D 1938
ADLOCK 851	PHENOLIC GLASS PREPREG	5	TENSILE ST	101	120°F	D 638
ADLOCK 851	PHENOLIC GLASS PREPREG	5	FLEXURAL ST	101	120°F	D 790
ADLOCK 851	PHENOLIC GLASS PREPREG	5	COMPRESSIVE ST	101	120°F	D-695
ADLOCK 851	PHENOLIC GLASS PREPREG	5	BEARING ST	101	120°F	FTMS 406
ADLOCK 851	PHENOLIC GLASS PREPREG	3	INTERLAMINAR SHEAR	99	120°F	MIL85
EA 934	EPOXY ADHESIVE	5	TENSILE SHEAR	99	180°F	D-1002
EA 934	EPOXY ADHESIVE	5	180° PEEL	99	180°F	D-903
EC 2216		5	180° PEEL	99	150°F	D-903
SCOTCHWELD 2216		5	TENSILE SHEAR	99	150°F	D-1002
BIOSHIELD	POLYIMIDE FILM - GLASS CLOTH LAMINATE	5	TENSION, MEMBRANE	105	120°F	D-882
BIOSHIELD	POLYIMIDE FILM GLASS CLOTH LAMINATE	5	TENSION, SPLICE	106	120°F	D-882
FM 96U	ADHESIVE, FILM	10	LAP SHEAR	97	150°F	ASTM D-1002
HT424	ADHESIVE, FILM	10	LAP SHEAR	97	180°F	ASTM D-1002
HT426	ADHESIVE FILM	10	LAP SHEAR	97	180°F	ASTM D-1002
VESPEL SP 1	POLYIMIDE SHEET	5	TENSILE STRENGTH & ELONGATION	100	150°F	ASTM D-638
VESPEL SP-1	POLYIMIDE SHEET	5	FLEXURAL STRENGTH	100	150°F	D-790
VESPEL SP 1	POLYIMIDE SHEET	5	HARDNESS	100	150°F	D-785
VESPEL SP 1	POLYIMIDE SHEET	5	DIELECTRIC STRENGTH	100	150°F	D-149
VESPEL SP-1	POLYIMIDE SHEET	3	SURFACE & VOLUME RESISTIVITY	100	150°F	D-257
VESPEL SP-1	POLYIMIDE SHEET	3	DIELECTRIC CONSTANT & LOSS TANGENT	100	150°F	D-150
SOLITHANE 113 WITH CAT C113-300	POLYURETHANE CON FORMAL COATING	5	ADHESION TO SUB STRATE	101	AMB	FTMS 6304 1
SOLITHANE 113 WITH CAT C113 300	POLYURETHANE CON-FORMAL COATING	5	FLEXIBILITY	101	AMB	FTMS 141 6221
SOLITHANE 113 WITH CAT C113-300	POLYURETHANE CON-FORMAL COATING	5	BOND STRENGTH	101	AMB	ASTM D-1002
KAPTON H	POLYIMIDE FILM	5	TENSILE (TRANS-VERSE)	101	150°F	ASTM D-882
KAPTON H	POLYIMIDE FILM	5	TENSILE (MACHINE DIRECTION)	101	150°F	ASTM D-882
KAPTON H	POLYIMIDE FILM	3	DIMENSIONAL	101	150°F	ASTM D-1204
KAPTON H	POLYIMIDE FILM	10	TEAR (TRANSVERSE DIRECTION)	101	150°F	ASTM D-1938
ECCO FOAM FPH	POLYURETHANE FOAM ENCAPSULANT	3	COMPRESSIVE STR	98	150°F	D-1621
ECCO FOAM FPH	POLYURETHANE FOAM ENCAPSULANT	3	DIELECTRIC CONST	98	150°F	D-1673
ECCO FOAM FPH	POLYURETHANE FOAM ENCAPSULANT	3	DIELECTRIC CONST STRENGTH	98	150°F	D 149
STYCAST 2850 FT WITH CAT 11	EPOXY ENCAPSULANT	3	DIELECTRIC CONST	98	150°F	D-150
GLOSS GREY SILICONE COATING	SILICONE THERMAL CONTROL COATING	5	EMITTANCE	98	150°F	LION EMIS
FLAT GREY SILICONE COATING	SILICONE THERMAL CONTROL COATING	5	EMITTANCE	98	150°F	LION EMIS
HYSOL C7 4240	EPOXY IMPREGNANT	4	COMPRESSIVE STR	98	150°F	D 695
HYSOL C7 4240	EPOXY IMPREGNANT	3	DIELECTRIC STR	98	150°F	D 149
HYSOL C7 4240	EPOXY IMPREGNANT	3	DIEL CONST	98	150°F	D 150
CHOSEAL 1226	RFI GASKET	3	RESISTIVITY	97	150°F	V71 1200
CHOSEAL 1226	RFI GASKET	3	COMPR SET	97	150°F	ASTM D 395
CHOSEAL 1236	RFI GASKET	3	RESISTIVITY	97	150°F	V71 1200
CHOSEAL 1236	RFI GASKET	3	COMPR SET	97	150°F	ASTM D 395
DC 93 500	SILICONE ENCAPSULANT	3	DIEL CONST	97	150°F	ASTM D 150 60
DC 93 500	SILICONE ENCAPSULANT	4	DIEL STRENGTH	97	150°F	D 149
SOLITHANE 113	POLYURETHANE CON FORMAL COATING	5	INSULATION RESISTANCE	100	AMB	D 257
DC 92 007		1	ABSORPTANCE	99	AMB	
RTV 511		1	ABSORPTANCE	99	AMB	
KAPTON F 910	POLYIMIDE FILM	5	TENSILE	99	150°F	D 882
KAPTON F 910	POLYIMIDE FILM	5	ABRASION	99	150°F	D 1175
KAPTON F 910	POLYIMIDE FILM	5	SURF & VOL RESISTIVITY	99	150°F	D 149
KAPTON F 911	POLYIMIDE FILM	5	TENSILE	99	150°F	D 882
KAPTON F 911	POLYIMIDE FILM	5	ABRASION	99	150°F	D 1175
KAPTON F 911	POLYIMIDE FILM	5	DIELECTRIC STR	99	150°F	D 149
KAPTON F 911	POLYIMIDE FILM	5	SURF & VOL RESISTIVITY	99	150°F	D 257
DC 6 1106	SILICONE SEALANT	5	180° PEEL	100	120°F	D 903
DC 6 1106	SILICONE SEALANT	3	VOL RESISTIVITY	100	120°F	
DC 92 007	SILICONE THERMAL CONTROL COATING	5	EMITTANCE	99	AMB	LION EMIS

Figure 5. long term thermal vacuum test of polymers.

where dx/dt is the rate of weight loss, X is total weight loss at temperature T, Ao is the initial weight, and K_t is a constant at temperature T. The intention is to use this equation, and thermo gravimetric analysis (TGA) and residual gas analysis (RGS) data for a given material, to determine the rate-constant K_t at any temperature. Knowing K_t, the weight loss at any temperature over any time should be calculable for a specific material. This would provide one a means of ultimately predicting long term isothermal weight loss by using test data which can be readily acquired in the laboratory in a matter of a few hours. Again a precautionary note must be added: if the material conditioning history or pretreatment is altered, so might the isothermal weight loss.

We now turn our attention to some space environment effects on materials believed to be associated with adsorbed surface gases.

Vacuum Removal of Adsorbed Surface Gases from Materials

Vacuum or thermal/vacuum effects associated with the removal of adsorbed surface gases from materials are more subtle, and less well documented in the relevant literature. Clearly there are frictional implications in the loss of adsorbed gases from material sliding surfaces, and coefficients of friction can change appreciably. Optical properties of thermal control materials can change because of adsorbed surface gas removal. With regard to metals, the picture becomes less clear. Unfortunately for the designer, much of the early work on the effect of vacuum on the properties of metals was done with academically interesting pure metals. Siren et al. [7] found fatigue in vacuum to have a strain rate dependency on commercially pure polycrystalline aluminum and made an attempt to correlate the phenomenon with changes in the dislocation concentration in the surface debris layer. He concluded that fatigue life is enhanced when the rate of dislocation accumulation in the debris layer is reduced. Sumison [8], in his investigation of vacuum effects on pure magnesium HM 22A and LA 141A at pressures down to 10^{-8} Torr, concluded that the fatigue properties on these materials were substantially better in vacuum than in air. He further postulated that there is strong evidence that the improved fatigue properties of metals tested in vacuum relates to the nonoccurrence of a reaction between the reactive gases normally present in the atmosphere, and the new metal surface created by a fatigue crack.

Tests done at a pressure level of 10^{-7} Torr on 1100 aluminum by Hordon [9] showed up to an order of magnitude increase in fatigue life of the 1100 aluminum below a pressure of about 10^{-2} Torr. He also postulated a surface related phenomenon as the dominant influence in fatigue life improvement, contending that it was related to retardation of the crack propagation phase of the fatigue process. He showed that the fatigue resistance was related to the kinetics of residual O_2 and H_2O gas

adsorption at crack surfaces. Hudson [10] of NASA, in considering relevant engineering alloys (2014, 7075, 2024 aluminum and 316 stainless), noted that most materials indicate an improvement in fatigue life, but cited the case of 316 stainless in Ni-Cr alloys where fatigue life was better in air than vacuum at low strain levels, while the converse was true at high strain levels. He postulated that at low strain levels in air, fatigue cracks fill with oxides which become tensile-load bearing. In vacuum, at high strain levels, there is less O_2 available to react with atoms at the crack tip, hence there is less reduction in the work necessary to break the interatomic bonds.

Some general observations can be made concerning the performance of metals in an orbital vacuum environment; i.e., it can be concluded that vacuum:

1) Has a beneficial effect on the fatigue life of alloys susceptible to stress corrosion (removal of adsorbed molecules hence elimination of the electrolyte),

2) May have a deleterious effect on creep and stress rupture properties of alloys which are subject to preferential evaporative loss of material,

3) Has a beneficial effect in improving resistance to creep and stress rupture of alloys which are prone to reaction with high temperature gases,

4) Has no effect on room temperature ductility.

The final effect of space vacuum to be considered here and which must be considered in materials selection and location for orbiting platforms and stations has to do with the effect known as multipactor breakdown.

Multipacting of Materials

It has long been known that as atmospheric pressure is reduced on materials which have a dc electrical potential across them, there will be increased electron flow between the materials. At some voltage, dependent on the work function of the material, the kind of residual gas molecules and their MFP, and the geometry of the materials, an electrical discharge will finally pass between the materials. As the pressure is reduced even further, the statistical probability of molecular collisions (and hence ionization) decreases, in spite of the fact that the kinetic energy of the remaining molecules is becoming ever higher due to increasing MFP. There comes finally a pressure level (vacuum) where an arc can no longer be sustained. Grossly speaking, this is how dc electrical breakdown of materials usually occurs in the absence of any mitigating circumstances in the launch and flight to orbit condition.

The situation becomes much more complex when the gaps and spacings between materials in the orbital vacuum environment are subjected to the powerful RF fields associated with various communications transmitters used to maintain contact with the ground. When RF voltages are applied to electrical materials, breakdown voltages are applied to electrical materials, breakdown voltages lower than the dc equivalents will occur because of electron resonances and cumulative ionization effects.

Multipactor breakdown occurs when secondary electrons produced at one material surface (electrode) are resonantly accelerated so that they reach the other surface (electrode) in a half-period of the applied RF field. (It should also be noted that the MFP of an electron in a gas, however tenuous the gas may be, is a factor of $4\sqrt{2}$ greater than the MFP of the gas molecules.) If then, these secondary electrons in striking the other electrode surface create a new larger cloud of secondary electrons in time for these to be accelerated back across the gap during the next half-period of field oscillation, there is in effect a resonant build-up or avalanching of electron population in the gap between the material surfaces. This can occur any time the coefficient of secondary emission is >1.0. Multipacting can occur even between insulating materials as well as between metallic electrical conductors because electrical insulators, as a general rule, have even higher secondary emission coefficients than do metals.

The results of multipacting can be quite serious, causing localized heating thereby creating damage such as melting soldered connections and/or causing enhanced outgassing of materials involved, which in turn can cause damaging vacuum redeposition of materials and even full fledged electrical breakdown between the materials. The existence of multipacting does not however necessarily imply a catastrophic failure; there can be a power loss, deterioration of components and associated materials, and electrical noise and nonlinear effects co-existing for a time without any uncontrolled or rapidly avalanching catastrophic multipacting occurring. Satellites have already failed because of the multipacting phenomenon [11], and there has been enough bad experience to warrant careful examination of orbital platform and satellite designs to preclude such an event.

By examining the relationship describing the multipacting effect, one can postulate means of materials selection and placement to eliminate multipacting. Researchers have done this at MSFC [12]. The equation for multipacting breakdown voltage V is

$$V = \frac{(2\pi)^2 (fd)^2}{\frac{(e)}{(m)}} \left[\frac{1}{\frac{(k-1)}{(k+1)} (\pi \cos \theta + 2 \sin \theta)} \right]$$

where f is frequency, d is gap width, (e)/(m) is electron charge to mass ratio, θ is the phase angle of secondary electron with respect to the RF

field, and K is the ratio of electron arrival velocity to its initial emission velocity in the direction of the electric field.

Since the permissible values of these parameters have a minimum and maximum for a given frequency and spacing, there is a discrete range of voltages that permits multipacting. From a materials perspective, some possible solutions are:

1) Use of potting materials or foaming in materials to fill the gap with high dielectric strength material. This works because the electron MFP in dielectric materials is orders of magnitude smaller than that of typical "inter-electrode" gap spacing in Earth orbit.

2) Pressurization of the "inter-electrode" gap space – that is, keep the MFP of the gas too small to allow RF resonance in the gap. Not much can be accomplished by contemplating materials selections based on the electron work function since work function does not vary appreciably over the materials gamut. There are electrical tricks such as choice of operating voltage out of critical range, dc bias and other schemes, but these are beyond the scope of a materials treatise and are left to the electrical designers to solve.

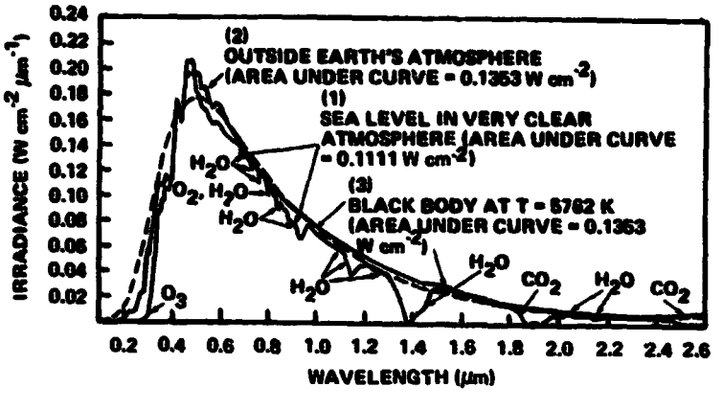
Having considered some of the primary considerations associated with space vacuum effects on materials, we now turn our attention to the second of the ... influences on orbital platforms – electromagnetic radiation.

ELECTROMAGNETIC RADIATION

In terms of electromagnetic radiation significant for LEO and GEO orbital missions, only that electromagnetic radiation emitted by the Sun will be considered in this paper, although there are other sources of electromagnetic radiation.

Solar radiation covers the wavelength region from about 0.01 μ to greater than 10^4 μ. This region includes low energy gamma ray, x ray, UV, visible, infrared, and microwave photons, and represents an air mass zero energy density of 1353 W/m^3 (1 solar constant). And although about 99 percent of the solar spectrum energy lies between 0.3 and 4 μ wavelength, the UV wavelength region from 0.1 to 0.4 μ, which contains only about 9 percent of the total energy, actually has the most effect on materials (Fig. 6).

X, gamma, and beta rays interact almost exclusively with orbital electrons in the structure of solids. The result of knocking electrons out of orbit is ionization, or charged ions. In metals, ionization causes almost no damage because the electronic imbalance is quickly neutralized



(1) NORMALLY INCIDENT SOLAR RADIATION AT SEA LEVEL ON VERY CLEAR DAYS

(2) SOLAR SPECTRAL IRRADIANCE OUTSIDE THE EARTH'S ATMOSPHERE AT 1 AU

(3) BLACK BODY SPECTRAL IRRADIANCE CURVE AT T = 5762 K (NORMALIZED TO 1 AU)

Figure 6. Solar irradiance versus wavelength.

by the large number of free electrons of high mobility in the lattice. In molecular, ionic, and covalent compounds such as ceramics and organic polymeric materials, ionization can have serious and permanent consequences. Ionization of these kinds of materials may disrupt chemical bonds of the structure and thereby render the compounds chemically active. The result could be the formation of new types of bonds either within the material or by interaction with the environment. Partial or total destruction of the original material can take place. Effects of solar UV radiation are especially potent because the UV photons are adsorbed within 1 μ of the surface by the ionization interaction. Thus, the UV photons have the ability to break chemical bonds and affect material properties to that depth. UV radiation can also produce significant degradation of optical and thermal control materials whose performance is primarily dependent on surface properties such as reflectance and transmittance. For very thin polymer films, even the bulk mechanical and electrical properties can be affected. UV irradiation is known also to enhance the vacuum outgassing of some organics and to polymerize recondensed sublimates. Critical platform or station materials applications which are potentially susceptible to UV radiation are thermal control coatings, solar cell glass cover slips, adhesives, and electrical insulation.

Solar Irradiation of Passive Thermal Control Coatings

Passive thermal control coatings are of perpetual interest to space designers because they provide for the space platform or the space station

a measure of temperature control without the direct expenditure of energy. It is perhaps, in a gross sense, the space denizen's equivalent of terrestrial solar heating and cooling systems. Because of their energy saving potential, passive thermal control coatings must be capable of long time operation without appreciable degradation in performance. The importance of thermal control coatings can be perceived from the following: An orbiting space platform typically receives energy from the Sun. It is also irradiated by the Earth's infrared radiation and by albedo, or Earth reflected solar energy. The platform itself contributes internal energy, and the radiated energy is therefore actually the sum of all the previously mentioned sources. Of these energy sources, the incident radiation can be strongly influenced by passive control. The temperature of the platform surfaces involved to a first approximation will vary as the fourth root of the ratio of the solar absorptivity to the infrared emittance of those surfaces:

$$T \propto \sqrt[4]{\frac{\alpha s}{\varepsilon}} .$$

Some typical examples of $\alpha s/\varepsilon$ ratios of materials familiar to all of us are polished aluminum foil with an $\alpha s/\varepsilon$ of about 5.0 which would theoretically stabilize in space at about 150°C, a painted black body has an $\alpha s/\varepsilon$ of about 1.0 and would stabilize at 25°C, while a painted white body has an $\alpha s/\varepsilon$ of about 0.2 and would develop a space thermal equilibrium of about -50°C. Many other factors, practical and otherwise, must be considered in the selection of thermal control coatings besides UV coating stability in space. Some of these other factors are geometry, color, surface finish, bakeout requirements, prelaunch environmental durability and cleanability, ascent environment, possible synergistic coating effects with substrate, coating repair characteristics, and the response integrity to thermal and vacuum environment. On balance, the sine qua non in the passive thermal control discipline is still UV coating stability. Without that, the other factors are meaningless.

There are enormous amounts of information and data available on the effect of UV radiation on thermal control materials, but generally the most important factors are the effects of intensity, integrated dose, and spectral distribution of incident energy on the material. As noted in the Space Materials Handbook [13], some have postulated an exposure time intensity reciprocity law for certain organic materials, but many others hold that simple reciprocity laws are inapplicable. One frequently ignored factor which casts doubt on much of the body of existing data is the bleaching effect factor. The UV reflectance of dielectric pigmented thermal control materials is seriously degraded after relatively short exposure to UV/vacuum, and does not recover substantially after the pigment is subsequently exposed to air. However, when untreated semiconductor pigmented samples are removed from the vacuum environment for the $\alpha s/$ measurements, there is a beneficial bleaching or a reoxidation effect which causes at least a partial recovery of the infrared emittance

characteristic and hence the $\alpha s/\epsilon$ ratio. Therefore the semiconductor pigmented sample measurements must be made in situ unless the material has been specially formulated to provide protection from photodesorption of oxygen and the subsequent degradation of the infrared reflectance.

The UV stability of several classes of materials is shown in Figure 7. Here we see the change in absorptance with exposure time. The realistic inflight conditions however can vary widely due to contaminating effects from outgassing materials, scheduled dumps of volatiles, and operation of various types of propulsion and attitude control systems. Quartz or teflon second surface mirror materials are very stable in UV environments, as are the zinc oxide/alkali silicate paints such as Z-93 and S-13GLO. The S-13GLO material (LO refers to low outgassing improvement) shows somewhat less stability than the second surface mirror materials, but it is a highly useful and much used thermal control material because of its easy processing and better cleaning and durability characteristics. It is of interest to note that besides being used extensively on spacecraft, the S-13G predecessor of S-13GLO was the material selected for the MSFC thermal shield called SAIL, which was deployed over the Skylab to salvage it for further use. This material is reactively

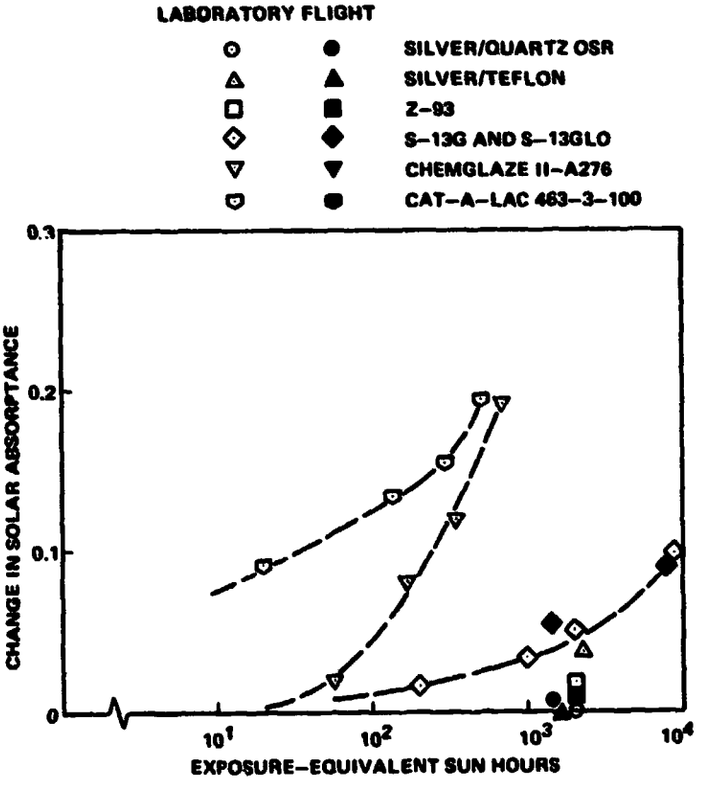


Figure 7. Stability of thermal control coatings in UV/vacuum exposure.

encapsulated ZnO in molecularly distilled RTV-602, a poly dimethyl siloxane. A final constituent, tetramethylguanidine and mixed amines, about 0.2 percent by weight, rounds out the formulation. The intrinsic propensity of semiconductor materials (ZnO) to lose their ability to reflect infrared radiation after exposure to UV in vacuum has been circumvented by stabilizing the ZnO through reactive treatment of the pigment's particle surfaces with potassium silicate. This treatment is believed to prevent photodesorption of oxygen, a process that is known to degrade the infrared reflectance of ZnO in the orbital environment. The ZnO semiconductor has band gaps of about 3.2 electron volts and it absorbs radiation less than 3800 Å wavelength, hence it absorbs nearly 10 percent of the total extraterrestrial radiation, including the UV. In spite of this inherent deficiency, and on balance, the S-13G and GLO formulations have proven to be superior thermal control materials for other reasons.

The great flexibility and advantages attendant to the use of this material are evident in a report on performance of thermal shields for Skylab [14]. The S-13GLO material currently differs from the material used for Skylab only in that vacuum stripping in a molecular still is carried out to strip various fractions of low molecular weight material from the RTV-602 vehicle.

The performance of the S-13G material as the thermal control material on the thermal shield for Skylab (called SAIL) was excellent. The material is inherently flexible and permitted the SAIL to be folded into a small package for subsequent deployment in orbit. There was more degradation in performance during the Skylab mission than anticipated, but through analysis of recovered specimens we concluded that this degradation was incurred primarily because of contamination. Early in the Skylab mission there was a loss of thermal coolant loop material called Coolanol 15 (a silicate ester) which deposited over large areas of Skylab, including the SAIL. Analysis of returned witness specimens and actual photographs confirmed that there had indeed been such contamination. The lesson to be learned here is that any prolific sources of orbital outgassing must be considered as potential thermal control coating degraders, whether their origin is from poor materials selection, system dumps, or malfunctions. When compared with the specially developed good-to-excellent stability in vacuum/UV environment materials, many off-the-shelf commercial paint products are rapidly degraded. In general, the commercial products do not have specially tailored pigments or vehicles. Referring again to Figure 7, it is evident that the epoxy based Cat-A-Lac white paint 463-3-100 (a dielectric material) suffers an immediate increase in solar absorptance. But even so, there is some usefulness associated with these commercial coatings where moderate stability for only a few hundred hours is a requirement. Figure 8 shows a general material characteristic table comparing two important classes of thermal control materials, and summarizes some of the more important information previously noted.

	SEMI – CONDUCTOR MATERIALS	DIELECTRIC MATERIALS
	(ZnO, TiO_2)	(LITHIUM ALUMINUM SILICATE, ZrO_2, Al_2O_3)
BAND GAP	3.2 ev	6.3ev
RADIATION ABSORBED	<3800 Å	<2500 Å
% RADIATION ABSORBED	10%	$<10\%$
FUNCTION PRIMARILY DEGRADED	ϵ	α_s
FUNCTION RECOVERY (IN AIR)	YES	NO
INCIDENT UV SCATTER & BINDER DAMAGE PRONE	NO	YES

Figure 8. Thermal control material characteristics.

The actual performance of materials in space was a test objective of the OSO-II satellite [15]. Optical stability of coatings exposed to 4 years of space environment was determined. Solar absorptance and infrared emittance were measured for a variety of materials. OSO-II was in a near circular orbit at an altitude of approximately 550 km (LEO) at an inclination of about 33° relative to the equator. The experiment specimens were "barbecued" (rotated at 37 rpm). The salient conclusions of this 4-year orbital test were:

1) Five coatings were stable over 7500 ESH (equivalent sun hours) – Z-93, OSR, barrier layer anodic, 2000 Å Al and V-groove.

2) The near-Earth micrometeoroid environment does not seriously affect thermal control of optical surfaces.

The Z-93 is a ZnO semiconductor material, OSR is fused silica with Ag/Inconel second surface, and the V-groove was a Mg alloy with concentric V-groove machined into it.

While the optical stability of materials in orbit is crucial, there are a few other materials changes in the electromagnetic radiation sensitivity band which must be considered.

Solar Illumination Effects on Conductivity of Materials

The electrical conductivity properties of some thin film insulating materials are significantly altered by exposure to illumination. Recent work at Stanford Research Institute [16] for NASA investigated the dark and illuminated electrical conductivities of Kapton V and polyvinylidene fluoride (PVF_2) films, and the changes in insulating properties produced

in Kapton H, Kapton V, PVF_2 and FEP Teflon films as a result of prolonged exposure to solar illumination.

Kapton, a thermoplastic aromatic polymer and widely used spacecraft material, shows a bulk electrical conductivity increase by four orders of magnitude during brief periods of illumination by a xenon lamp with a power density of 100 MW/cm^2 at the sample surface (99.8 percent of the lamp output power at the 200 to 2100 nm wavelength range).

In tests in which a 5-kV voltage was applied to Kapton samples during illumination, the resulting high bulk currents produced voltage breakdown and severe physical damage to the samples before the end of the 24-hour test period. The investigators also found that the dark bulk conductivity of Kapton increases by up to two orders of magnitude when the sample temperature is increased from 22° to 100°C. By contrast, FEP Teflon, a fluoroplastic material, retained its electrical insulating properties without significant change after 24 hr of illumination and also at increased temperatures. Interestingly, the dark bulk conductivity of polyvinylidene fluoride (PVF_2) was about two orders of magnitude higher than that of Kapton or FEP Teflon prior to illumination, but it did not increase at all (decreased slightly) with illumination and did not increase significantly with temperature.

Based primarily on leakage -- current considerations, FEP Teflon apparently is the best choice for use as a high-voltage solar array substrate. In this report, the use of FEP Teflon is advocated as the best choice among the materials tested for use as a solar-cell cover in a high-voltage array due to its higher leakage resistance and dielectric strength under illumination, and because of its relatively high optical transmission properties throughout the visible portion of the spectrum. Clearly then, the electrical space systems designer must keep in mind that design for long term orbital platforms must consider the degradation of bulk conductivities associated with materials exposed to elevated operating temperatures and/or solar illumination.

UV Irradiated Glass Characteristics

A final effect worthy of mention is the effect of UV radiation on glasses. Large space platforms, or space stations will undoubtedly have large windows and other optical apparatus in the habitable area and long term optical apparatus exposed to space, so some basic information about glass in UV radiation fields seems appropriate.

The transmission of glasses in the UV region of the spectrum is largely determined by the content of Fe_2O_3 [17] although cerium oxide also has a pronounced absorbing effect. Vitreous silica transmits well below 0.2 μ.

When sufficiently pure, silica and silicate glasses are well suited for UV transmission in the short-wave regions. Ordinary window glass is typically of the soda-lime-silica composition, and in ordinary single strength thickness (0.087 to 0.100 in.) it is opaque to UV radiation below about 0.310 μ, and hence absorbs the components of solar radiation in the 0.28 to 0.32 μ portion of the UV spectrum, which typically is the region beneficial to biological processes (suntan effect). The loss in UV transmission, or absorption of the UV, is generally attributed to the oxidation of ferrous iron to ferritic iron, and, as previously noted, this is the determining factor for UV transmission. Glasses are also affected adversely by particulate irradiation. Absorption of irradiation causes a decrease in transmission by the formation of color centers. In general, the glasses rank order grossly, as follows, in decreasing order of resistance to degradation:

1) Crystalline quartz, sapphire

2) Fused synthetic, s[illegible]a quartz

3) Ordinary glass.

A plethora of information is available on electromagnetic radiation effects on glasses, and for LEO application it is highly probable that enough hard data exist to do a long term LEO design considering the glass materials needed. However for the synergistic effects of electromagnetic and particulate irradiation in GEO environment there is much less confidence. The particulate irradiation is the key variable here, and that is the subject of the next section.

PARTICULATE RADIATION

The particulate radiations in space include charged and neutral energetic particles. The sources and prevailing types of particles are as follows.

Galactic Cosmic Radiation

This radiation originates outside the solar system and consists of a continuous, essentially isotropic flux of protons and comparatively fewer heavier nuclei. Protons of 10^{18} eV energy have been identified from this source. Protons (hydrogen nuclei) constitute about 85 percent of this radiation, alpha particles (helium nuclei) about 13 percent, and heavier nuclei the remaining few percent. However, it should be noted that the proton flux is relatively low, and is only about 2 to 5 protons cm^{-2} sec^{-1}.

Solar Cosmic Radiation

Solar activity occurs in approximately 11-year cycles and is characterized by solar flares, which develop rapidly with intense activity, but last only about 30 to 50 min. The solar particles continue to arrive near Earth for a few hours to several days after visible activity has ceased. Solar cosmic rays consist chiefly of protons, some electrons and alpha particles and a few heavier nuclei of atomic number Z up to 9 or 10. The fluxes are low, on the order of 10^5 protons cm^{-2} sec^{-1} $ster^{-1}$, but the proton energies can range from 1 to greater than 100 MeV.

Neutrons

Neutrons are produced from proton interactions with the atomic nuclei of spacecraft materials, and are significant as a secondary or induced radiation that is a function of the high energy proton fluence. They are not persistent, having a 12-min half-life after which they decay into protons. The total flux is hence negligible when compared with other particle types.

Solar Wind Particles

The Sun continuously ejects plasma called the solar wind. This plasma consists of low energy protons, electrons, and alpha particles (helium nuclei) which stream continuously outward from the Sun. The particle flux intensity is relatively high, being on the order of 10^8 to 10^9 particles cm^{-2} sec^{-1}, but the energies are only on the order of a few keV.

Plasma Sheet Electrons and Protons

These particles are located in the "magneto-tail" (Fig. 9). Kinetic energies of these particles range from less than 20 keV for electrons to about 30 keV for protons. Fluxes are on the order of 10^8 particles cm^{-2} sec^{-1}.

Geomagnetically Trapped Electrons and Protons

There are two belts of geomagnetically trapped radiation around the Earth (the Van Allen belts). The Earth's magnetic field constitutes a trap for high energy electrons and protons – the geomagnetic field causes such an interaction. The directionality of the electrons and protons is related to the orientation of the Earth's magnetic field. Because the orientation of a spacecraft or a space station varies with

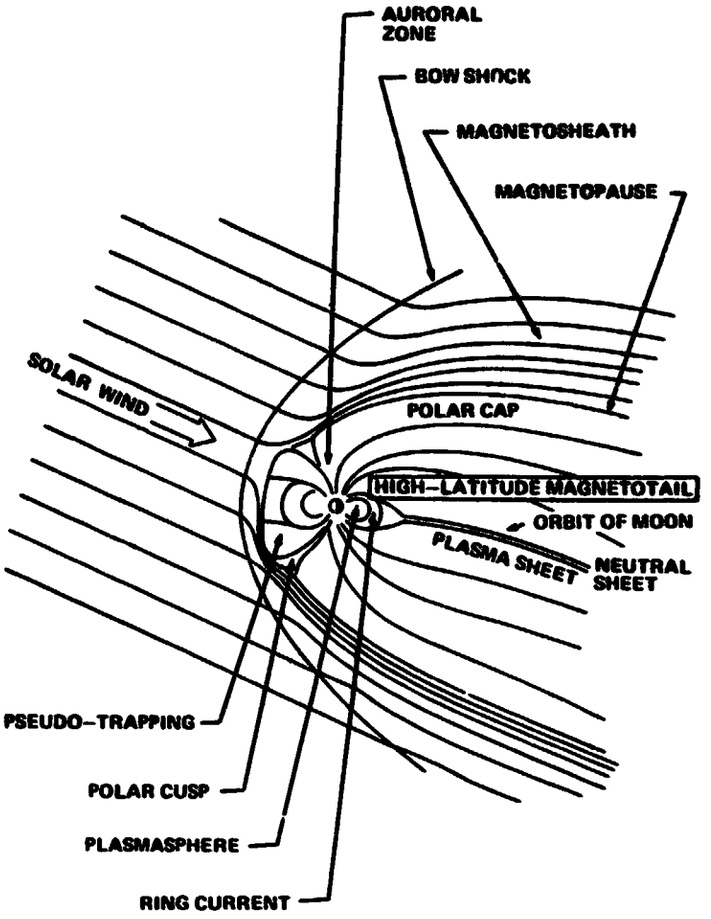


Figure 9. Regions of the magnetosphere shown in the noon-midnight meridian plane.

respect to the Earth's magnetic field during the course of a mission, particle fluxes are usually considered to be isotropic. Because of different trapped particle flux and energy characteristics there are, as previously noted, two radiation belts: an inner belt at 1.2 to 3.2 Earth radii, and an outer belt at 3 to 7 Earth radii, as shown in Figure 10. High energy electrons and protons are contained in the inner belt, whereas high energy electrons and lower energy protons are found in the outer belt. The relatively high fluxes and energies associated with these trapped particles make them the primary source of radiation damage for spacecraft or platforms operating in orbits all the way from 200 km t geosynchronous orbit, the orbits with which this paper is concerned. The Van Allen radiation belts are not entirely symmetrical, however. In the South Atlantic Anomaly, extending from 0 to 60° west longitude and 20 to 50° south latitude, the trapped proton intensity for energies more than 30 MeV is the equivalent at a height of 100 to 200 miles altitude, to that at 800 miles altitude elsewhere. This is due to a

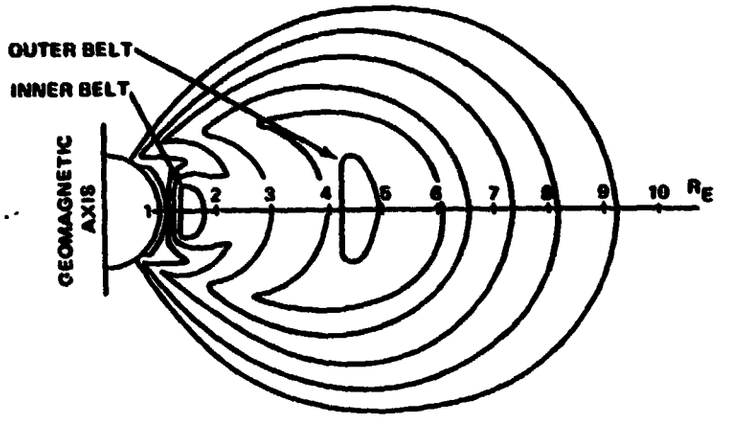


Figure 10. Van Allen trapped radiation belts.

perturbation of the Earth's geomagnetic field. For flight paths of LEO space platforms of 30° inclination from the equator or greater, there will be approximately five traverses through this anomalous area each day. Experience with orbital missions shows that a major portion of the accumulative radiation has been attributable to passage through this geomagnetic anomaly.

Electron Irradiation Effects

Electrons interact with matter and lose energy primarily through ionization of the atoms or molecules in the absorbing material. Another mechanism of energy loss that is significant for high energy electrons is the generation of bremsstrahlung radiation (x-rays resulting from the deceleration of the electrons) which is more penetrating than the original electrons, and is an additional source of radiation change. The amount of bremsstrahlung radiation generated is directly proportional to the atomic number (Z) of the absorbing material and the electron energy. However, for electrons in the energy range found at GEO which are being absorbed by low Z materials, the energy loss due to this process is small (< 10 percent). Thus, the predominant energy loss mechanism for GEO plasma sheet and radiation belt electrons is ionization. The range penetration of electrons is dependent on their energy. For the plasma sheet electrons with energies of a few keV, the penetration range is less than 200 μ. Thus, these electrons can affect the surface properties of absorbing materials, and if the materials are very thin then the bulk properties (e.g. mechanical) can also be affected. Since the radiation belt electrons at GEO span a broad energy range (from <100 keV to 6 MeV), both surface and bulk property changes can be induced in materials with thicknesses up to approximately 1 cm. Hence, these electrons could, for example, have a deleterious effect on the mechanical properties and the dimensional stability of the coefficient of thermal expansion of critical composite structural materials and on the power output of solar cells.

Proton Irradiation Effects

Protons deposit energy in materials principally through ionization and atomic displacement effects. However, for LEO and GEO environments, only the South Atlantic Anomaly protons have sufficient energy to produce a measurable number of atomic displacements. The plasma sheet protons in the 30 keV energy level have a penetration range less than plasma sheet electrons; therefore, ionization produced will affect surface properties primarily. Even the high energy radiation belt protons have limited penetration range and their effects are restricted mainly to surfaces unless the material is extremely thin.

SPACE DEBRIS

The risk of collision with space debris is an operating environmental consideration that assumes greater significance as orbiting structures (spacecraft, platforms, stations) volume and stay time increase. The definition of space debris herein has been expanded to include not only that debris of extraterrestrial origin (meteoroids) but also orbital man-made debris. As the Space Shuttle becomes operational, satellite emplacement and other structure insertions into orbit will become commonplace, thereby greatly compounding the difficulty of orbital payload discrimination and subsequent debris tracking. Extraterrestrial debris of interest concerns meteoroids. Meteoroids are generally conceded to have a mass density of approximately 0.5 gm cm^{-3}, and velocities of 11 km sec^{-1} to approximately 72 km sec^{-1}. Their size may range from less than 1 μ (micrometeroid size) to more than 300 cm (great fireball). Fortunately, as the size increases, the velocity and flux decrease. Thus the probability of impact by a large low velocity meteoroid is negligible compared with the probability of an impact with a high velocity micrometeoroid. For very large structures with long stay times in orbit, it is probable that there will be a large number of impacts with micrometeoroids. Whether or not these penetrate the structure is a function of the structural material, its properties and thickness, and the velocity and mass of the impacting projectile. The resulting damage is dependent on the number, size, and location of the penetrations. Figure 11 shows a statistical assessment of the number of penetrations that may be expected for structures of various sizes as a function of time in GEO.

We now turn our attention to the space environmental conditions to be expected for both the LEO and GEO missions.

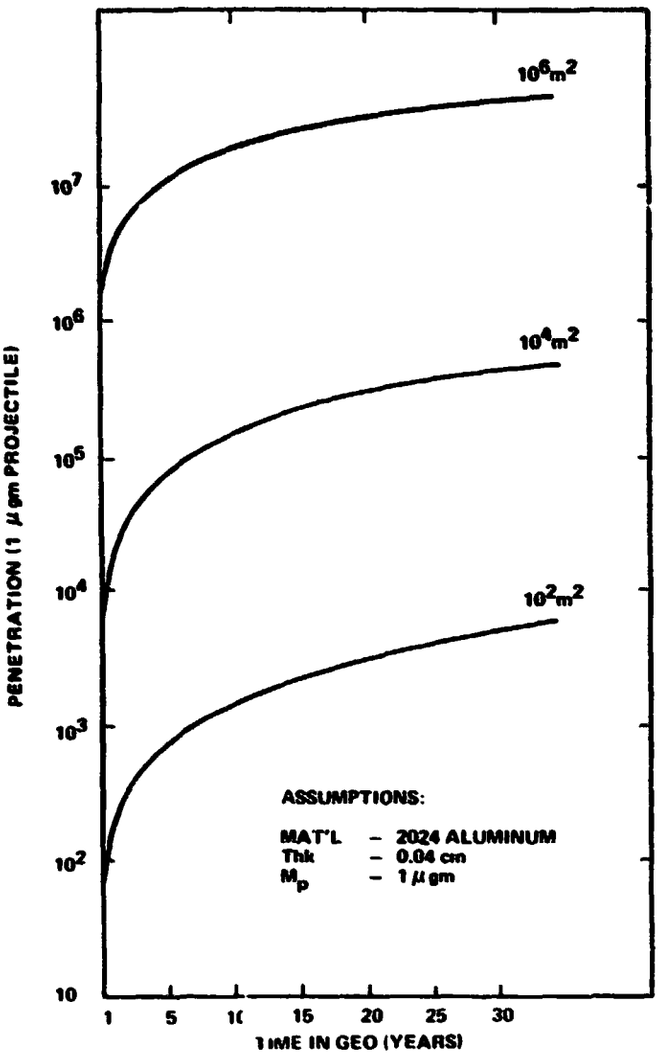


Figure 11. Micrometeoroid penetration as a function of spacecraft area and time in GEO.

THE LOW EARTH ORBIT (LEO) ENVIRONMENT

Based on the previously noted considerations, one may infer those environmental parameters which constitute the operating environment of primary concern for space platforms or a spacecraft in LEO:

1) High vacuum - approximately 10^{-8} Torr.

2) Solar electromagnetic radiation - UV through near UV at one sun (solar constant) intensity for air mass zero.

3) South Atlantic Anomaly (SAA) trapped particles.

4) Space Debris – probability of impact with micrometeoroids of microgram mass range, dependent upon spacecraft volume and stay time in orbit.

In the SAA Region, the magnetic flux lines reach a low point at about 30°S latitude, which manifests itself as a dip in the inner radiation belt. Figure 12 shows the altitude variation of field strength (flux density) in the SAA. And the energy spectrum for the electrons and protons trapped in this region are shown in Figures 13 and 14.

The situation becomes somewhat more complicated when one considers GEO.

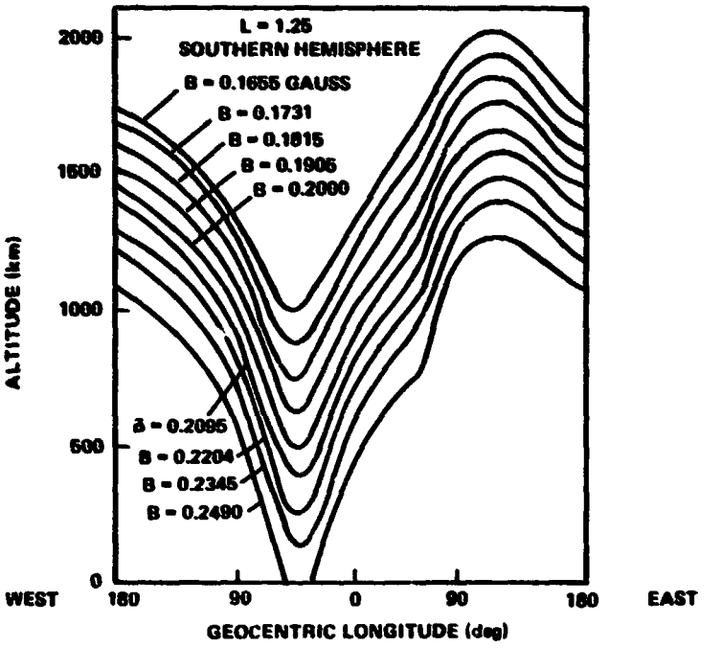


Figure 12. Altitude variation of field strength in the South Atlantic Anomaly (SAA).

THE GEOSYNCHRONOUS EARTH ORBIT (GEO) ENVIRONMENT

The operating environment for a space platform in GEO (35,900 km altitude) is considerably more hostile than that encountered in LEO due to the higher radiation fluxes in GEO. The trapped electrons and protons are expected to be the prime contributors to space platform

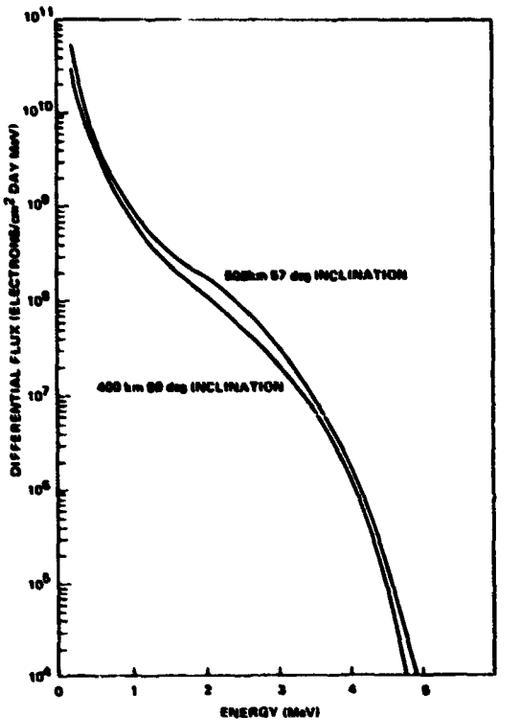


Figure 13. LEO electron differential energy spectra.

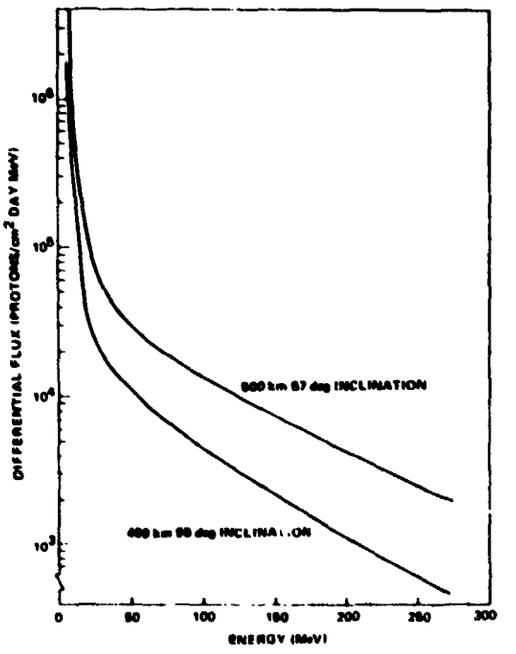


Figure 14. LEO proton differential energy spectra.

materials degradation. Other environmental parameters such as cosmic rays, gamma rays, X-rays, neutrons and higher Z particles are considered to have negligible probability for creating measurable degradation effects on materials due to their significantly lower fluxes and/or energy deposition rates. Thus, the critical environmental material degradation considerations from a space platform or a spacecraft performance and lifetime perspective can be characterized by seven parameters:

1) High vacuum – $<10^{-14}$ Torr.

2) Solar electromagnetic radiation – For UV through near UV, at one solar constant intensity for air mass zero.

3) Radiation belt electrons – Electron fluence in e cm^{-2} yr^{-1} > E, as seen in Figure 15.

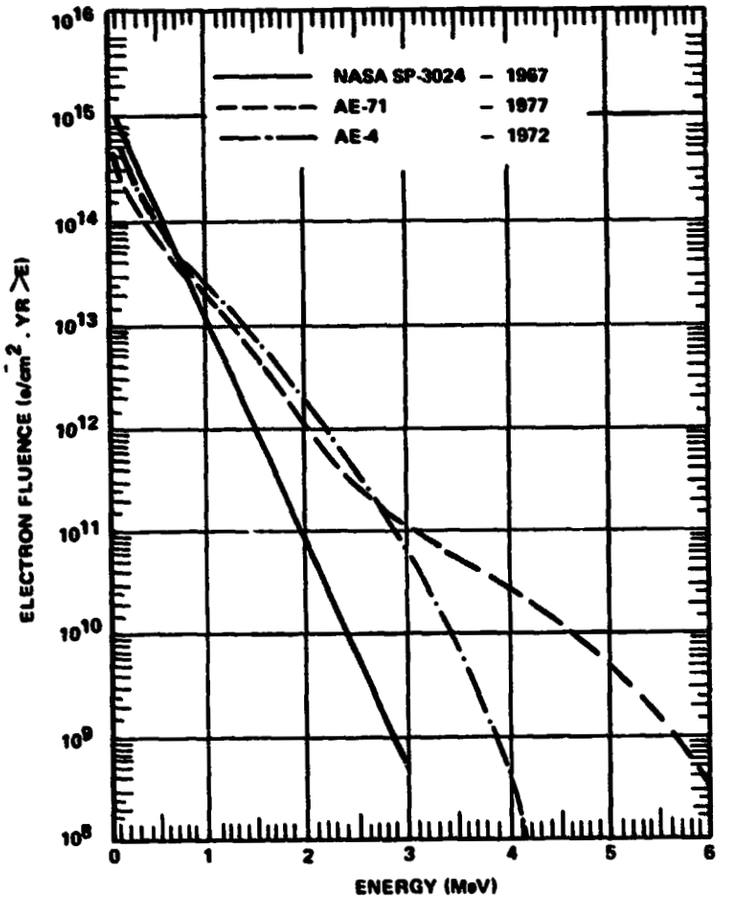


Figure 15. Radiation belt electron environment model.

4) Radiation belt protons – Proton energy distribution as shown in Figure 16, in p cm^{-2} sec^{-1}.

ENERGY RANGE (MeV)		INTEGRAL ($E > E_1$)	DIFFERENTIAL ($E_1 < E < E_2$)
E_1	E_2		
0.10	0.25	1.2×10^{7}	1.0×10^{7}
0.25	0.50	1.7×10^{6}	1.5×10^{6}
0.50	0.75	1.6×10^{5}	1.5×10^{5}
0.75	1.00	1.7×10^{4}	1.4×10^{4}
1.00	1.25	1.7×10^{3}	1.5×10^{3}
1.25	1.50	1.8×10^{2}	1.5×10^{2}
1.50	1.75	1.8×10^{1}	1.6×10^{1}
1.75	2.00	1.9×10^{0}	1.7×10^{0}
2.00	2.25	1.9×10^{-1}	1.7×10^{-1}
2.25	2.50	2.0×10^{-2}	1.8×10^{-2}
2.50	2.75	2.2×10^{-3}	2.0×10^{-3}
2.75	3.00	2.2×10^{-4}	2.0×10^{-4}

Figure 16. Radiation belt omnidirectional proton flux at synchronous altitude (p/cm^2-sec).

5) Plasma sheet electrons – Energies from 6 to 13 keV, and fluxes on the order of 10^8 to 10^9 e cm^{-2} sec^{-1}.

6) Plasma sheet protons – Energies near 30 keV and flux levels approximately 10^8 p cm^{-2} sec^{-1}.

7) Space Debris (primarily micrometeoroid) – There is reasonable probability of impact with many high velocity low mass micrometeoroids for a large space platform.

From the foregoing, it is apparent that spacecraft operating for long periods of time (years) in LEO and GEO will be subjected to environments that can potentially degrade the functional properties of critical materials and components, thereby limiting the service life of the spacecraft. It is necessary, therefore, that the design of these vehicles be based on materials environmental performance data derived from simulation or flight testing, from analytical predictions based on materials environmental performance data derived from simulation or flight testing, from analytical predictions based on validated performance degradation models, or suitable combinations of both. Only by these means can materials selections be made which will provide a high probability that the mission requirements will be met.

SPACE ENVIRONMENTAL EFFECTS DESIGN DATA AND REQUIREMENTS

To meet the requirements of ongoing and future projects which require long term space environment exposure, environmental effects design data for materials and components are required to support the design and materials selection efforts in all project phases.

Review of the flight systems required for these projects discloses that there are certain materials whose long term functional integrity is critical to achieving the economic operating life of the hardware. Some of these materials which have been identified as critical to the success of each project, their most critical radiation environments, and the most likely environmental effects are listed in Figure 17.

MATERIALS	MOST DAMAGING ENVIRONMENTAL PARAMETER	ACCELERATED RATE SINGLE PARAMETER EXISTING DATA	ENVIRONMENTAL EFFECT
SOLAR CELLS	LOW ENERGY e LOW ENERGY p UV	e p UV	REDUCED POWER OUTPUT
COVER GLASSES	LOW ENERGY e LOW ENERGY p	e p	REDUCED POWER OUTPUT
FLEXIBLE SUBSTRATES	HIGH ENERGY p UV	e p UV	EMBRITTLEMENT AND BREAKAGE
STRUCTURAL COMPOSITES (e.g. G/E)	LOW & HIGH ENERGY p & e UV (THIN MATERIALS)	VERY LIMITED FOR THIN MATERIALS	MICROCRACKING DIMENSIONAL INSTABILITY STRUCTURAL FAILURE
MIRROR COATINGS	LOW ENERGY p & e UV	VERY LIMITED	LOSS OF REFLECTANCE
REFLECTORS (METALLIZED FILM)	LOW ENERGY p & e UV	p e UV	EMBRITTLEMENT FRACTURE AND TEARING, OPTICAL DEGRADATION
THERMAL CONTROL COATINGS	LOW ENERGY p & e UV	p e UV	OVERHEATING
ADHESIVES (CELLS, MIRRORS, STRUCTURES)	HIGH ENERGY e HIGH ENERGY p	e p	DEBONDING DELAMINATION
INSULATIONS (THERMAL, ELECTRICAL)	LOW ENERGY p LOW ENERGY e UV	LIMITED	ELECTRICAL FAILURE
LUBRICANTS	HIGH ENERGY p HIGH ENERGY e	LIMITED	GALLING SEIZING
SEALS	HIGH ENERGY e HIGH ENERGY p	e p	LEAKAGE
PROPELLANTS	HIGH ENERGY e VERY HIGH ENERGY p	VERY LIMITED	BECOME SENSITIZED BECOME INERT

* LONG DURATION EXPOSURE IN THE COMBINED ENVIRONMENT IS NEEDED FOR THE ABOVE LISTED COMPONENTS.

Figure 17. Component radiation environmental effects summary.

Experimental evaluation of the long term environmental effects on materials is required for the following reasons:

1) Current knowledge of basic radiation damage mechanisms generally limits theoretical evaluations to microscopic changes in simplistic materials produced by a single well-defined environmental parameter.

2) Engineering materials are not simplistic but are complex compositions whose specific chemistry and processing are sometimes proprietary and subject to change by the vendor.

3) Hardware design is based on functional properties and not on atomic or molecular changes which cannot be directly correlated quantitatively to macroscopic effects.

4) Spacecraft materials are exposed to several environmental stresses simultaneously. These stresses which, when considered singularly, may evoke a particular time dependent response, but when combined, may produce a response that cannot be predicted based on the summation of the singular responses. The latter result is termed a synergistic effect. Because of these unknown synergistic effects, it generally is not possible to theoretically predict the total response of a material (even a simplistic one) to a combined environment exposure.

5) Nonlinear effects cannot be theoretically predicted. These may be caused by (a) unknown and varying levels of impurities/contaminants, (b) irradiation rate, (c) degree of cure or (d) in the case of a composite, layup configuration.

6) Time-dependent effects typically are theoretically intractable; that is, long term effects cannot be predicted based on short term behavior.

7) The present understanding of the behavior of engineering materials exposed to complex environmental stresses precludes the quantitative prediction of the time dependent property changes that will be produced by this exposure.

To illustrate the lack of current understanding and predictability of environmental effects on engineering materials, data for some typical types of materials are discussed as follows.

Solar Cells

Solar cells are of utmost importance as sources of electrical energy for spacecraft. Thus, the critical functional property is power output. Figure 18 shows the effect of 1 MeV charged particle irradiation on the power output of two different resistivity silicon cells, 10 ohm-cm and 2 ohm-cm. The decrease in power output evidenced in these data is nonlinear and is a function of (a) the type of cell, (b) the type of radiation (e or p), and (c) the integrated flux. Note that the proton irradiation produced greater degradation than did the electron exposure.

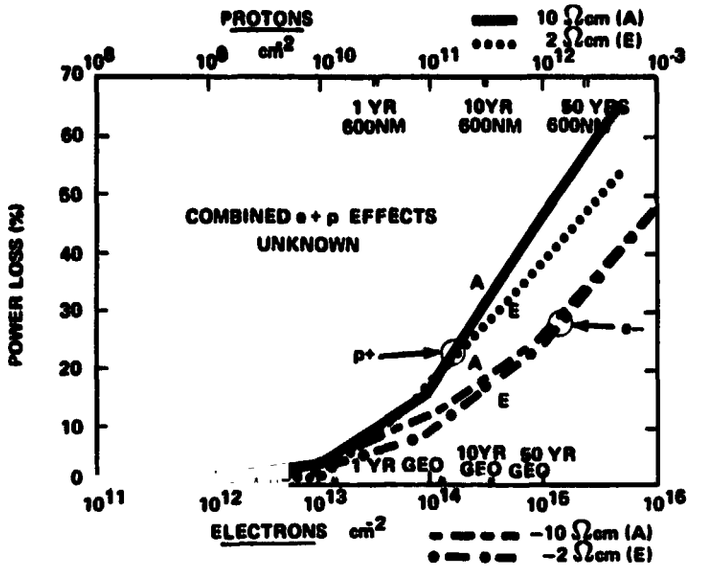


Figure 18. Electron and proton degradation of silicon solar cells.

These data were obtained using a single particle energy and one flux (particles cm^{-2} sec^{-1}). Prior to obtaining these data, these effects could not be quantitatively predicted. However, with these data and data for other energies and fluxes, a prediction model could probably be developed for these specific cells. The problem of synergism can also be illustrated with these data. Assume a 10-year stay time in GEO where both electrons and protons are encountered. The power degradation by protons alone is approximately 33 percent whereas for electrons alone the loss is on the order of 20 percent. If these effects are strictly additive, then the total loss is 53 percent in 10 years. However, the effects probably are interactive for this type of material and the total effect may be greater or less than each individual effect or their sum. Only through a series of parametric combined exposure tests can this question be resolved.

It should be noted, however, that at least one other researcher has reported no synergistic effects detected for simultaneous electron and proton irradiation [18]. The investigator also reports that the synergistic effects under consideration might be rate sensitive, and that his research was carried out using accelerated particulate irradiation rates which might well have influenced the results.

McGlathery [19] has done a detailed analysis of the problems involved regarding degrading space environmental effects on the performance and missions of a Solar Electric Propulsion Stage (SEPS). He has devised an analytical approach, converting the Vette NASA/Goddard Space Flight Center time averaged differential energy spectra for protons

and electrons into a 1 MeV electron equivalent environment as a function of spatial position and thickness of various shielding materials and solar cell cover slides. He also reports a hazardous near-Earth charged particle operating zone of from 1852 to 14000 km which is a "hot" radiation zone especially destructive to solar cell arrays. Other work has been done at MSFC on irradiation of polymeric materials for high performance solar arrays [20]. The investigators reported a need for additional higher level particulate and electromagnetic radiation testing to further define the synergistic relationships.

It should also be noted that there are many variables in solar cell design and manufacture that can influence cell response to particle irradiation, thereby making experimental evaluation even more strongly justified and much more tedious to execute.

Composite Materials

For many years, strength improvements in metal alloy systems have had to come through unconscionable costs in fatigue-life, fracture-toughness and corrosion resistance. The biggest penalties have been in elastic modulus-to-weight ratios. Filamentary composites, however, can be made with specific strength 2 to 5 times that of competitive metal alloys, and with specific moduli from 4 to 8 times as great as the metals. For aerospace use the ideal fibers for composites are those having light atoms with strong chemical bonding forces such as oxides, nitrides, borides, and especially boron and carbon. However, mother nature does not relinquish these remarkable properties without any attendant disadvantage – the fibers are brittle and fracture sensitive and have to be incorporated in a host or matrix material. There is also a high level of anisotropy associated with the fibers, and practical working composite materials have fibers optimally oriented to obtain the result the designer seeks.

Graphite-epoxy (G/E) composites in particular are being widely used where high specific modulus and strength, and attendant dimensional stability are major considerations. High levels of stability are being attained and "error budgets" maintained, for example in the NASA Space Telescope, by tailoring the materials for very low coefficients of thermal expansion (CTE). The NASA satellite ATS-F has already proven the tremendous advantages associated with its G/E truss structure.

G/E structures can absorb as much as 4 percent by weight of moisture which can cause as much as a 3 percent volumetric change – in spite of the fact that G/E composites contain about 65 volume percent of fibers [21]. In practice, 0.2 to 0.3 percent water weight is present in a thermosetting composite immediately after removal from the curing oven, and storage at 50 to 70 percent relative humidity and 70°F can result in a moisture content of 0.7 to 0.9 percent by weight. Hence the designer worries not only about CTE, but also about CME (coefficient of

moisture expansion), because the loss of this moisture in space is accompanied by a corresponding dimensional change. CME is also anisotropic and depends on fiber layup.

In the space environment, organic matrix materials can be subject to a wide range of radiation effects, depending on the molecular structure of the material and the amount of energy absorbed from the radiation sources. The resulting formation of free radicals and ions in the material can lead to crosslinking, chain scission, chain polymerization, shock copolymerization, unsaturation and possibly chain transfer. Recent tests at MSFC have shown evidence of several of these effects. G/E materials with differing fiber orientations were irradiated with 2.6×10^{12} cm^{-2} particulate radiation, and some materials lost as much as 25 percent in tensile strength while in others the modulus increased by 13 percent. Additional verifying data are being taken and testing is continuing, particularly with regard to an apparent fiber orientation dependence, an effect that was not quantitatively predicted. Other investigators have noted changes in mechanical properties of electron irradiated composites as well. Figure 19 shows data on von Bassewitz and also NASA data indicating changes in mechanical properties due to electron radiation.

DOSE RADS (C)	COMPOSITE MATERIAL	MECHANICAL STRENGTH PROPERTY	PERCENT CHANGE IN MECHANICAL PROPERTY	REFERENCE
10^9	GRAPHITE/TETRAGLYCIDYL EPOXY	TENSILE	+6	
10^{10}	GRAPHITE/TETRAGLYCIDYL EPOXY	TENSILE	-18	vonBASSEWITZ,
10^9	GRAPHITE/TETRAGLYCIDYL EPOXY	FLEXURAL	-28	FRANCE
10^9	GLASS/DIGLYCIDYL EPOXY (ALIPHATIC AMINE CURE)	FLEXURAL	-20	
10^9	GLASS/DIGLYCIDYL EPOXY (AROMATIC AMINE CURE)	FLEXURAL	-20 TO -50	NASA-SP-5053
10^9	GLASS/DIGLYCIDYL EPOXY (ALIPHATIC AMINE CURE)	FLEXURAL	-20 TO -50	
10^9	GLASS/DIGLYCIDYL EPOXY (AROMATIC AMINE CURE)	FLEXURAL	-90	

Figure 19. Changes of mechanical properties of fiber/epoxy composites due to electron radiation.

Because thin composites, on the order of 10 to 20 mils thick, are being considered for use on Large Space Systems, concern over both UV and proton radiation is probably warranted. Such radiation may cause significant changes in the bulk properties of these materials. As noted earlier, any radiation induced change in CTE which alters the dimensional stability characteristic, can have a major impact on "error budgets" associated with high accuracy, high precision orbiting structures such as telescopes, antennae, reflectors, etc. Currently there is no theoretical capability for accurately predicting the change in CTE (or CME either, for that matter) as a function of radiation exposure. This critical data must be obtained experimentally.

A final consideration in the use of composite materials in the space environment has to do with their use in proximity to high intensity electromagnetic fields [22]. In nonmetallic composite materials, the matrix is a dielectric and any electrical current flow will take place through its fibers. In composite materials containing graphite fibers, the fibers are thin (7 to 10 μ in diameter) and resistivity is high enough to cause heating, which can result in arcing and structural damage at high current levels in high level electrical environments. Such materials also exhibit substantial variations in intrinsic EM parameters and lack the shielding qualities of conventional metal structures. Additional variations occur when nonmetallic and metallic materials are used in combination, i.e., when multidirectional lay-ups are used in combination with metallic structures. Because complex structures contain multi-layers of fibers having different fiber orientations -- and therefore multiple laminate geometric angles – it is difficult to derive intrinsic EM measurement/ analytic technologies.

To determine how the energy sources encountered in the natural space environment affect the electromagnetic properties of composites, and thus develop the necessary protective measures, it will be necessary to collect evidence of their vulnerabilities to EM hazards and then devise a means of simulating the energy sources and measuring the effects. Such measurements and analyses should simulate the cumulative effects over the operational life of the space structure. This is another example substantiating the need for space environmental effects testing to be sure we do not encounter unanticipated effects which would jeopardize the mission.

Flexible Thin Film Materials

All past and future space systems use, to some extent, flexible thin film materials on surfaces exposed to the space environment. I refer here to film materials such as Teflon, Kapton, Mylar, Tefzel, etc., and second surface thermal control films made by various coatings on these materials. These materials are used as thermal control materials, or as primary portions of inflatable or unfurlable structures. In GEO, these thin films are especially vulnerable to radiation effects. The accumulated radiation surface dosage in GEO is considered to be about 10^9 to 10^{12} rads, and hence quite capable of damaging polymer films which have a damage threshold in the range 10^5 to 10^8 rads. Even for bulk penetration in GEO, the absorbed dosage is between 10^8 to 10^9 rads, with a corresponding penetration depth of 0.1 to 10 mils. Since most of the polymer film thicknesses used are on the order of a few tenths of 10 mils, one can see that a substantial fraction of the material will be affected by the absorbed energy in the GEO environment.

Unfortunately, much of the available literature is filled with contradictions as far as radiation effects on thin films is concerned. Most of

these disparities are associated in one way or another with the inadequacy of testing, and an inability tc simultaneously test for synergistic effects. One recent effort to improve the situation and derive good data is described in a report by TRW for the USAF [23]. Twenty-three different flexible film materials were tested to simulate the effect on optical, thermophysical and tensile properties of a 5-year GEO equatorial environment.

A major portion of this study effort was directed toward the development of a laboratory facility that would provide a suitable combined simulation of the critical space environmental conditions, do simultaneous irradiation of specimens, and allow in situ testing of the specimens.

The results of these tests convincingly demonstrate the initial analytical unpredictability of the synergistic effects of electrons, protons, and UV encountered in the GEO environment. The findings are summarized as follows:

1) Teflon materials (FEP and PFA) undergo substantial changes in optical, thermophysical, and mechanical properties when exposed to the long-term space radiation characteristic of a synchronous orbit. Solar directional absorptance increases by a factor of 3 or 4. The specimens yellow and begin to lose their transparent and specular qualities. They become brittle to the touch and retain no residual flexibility. Therefore, their continued use for thermal control purposes or in applications requiring flexibility after exposure to the environment is not recommended.

2) Kapton material undergoes changes in optical, thermophysical, and mechanical properties due to long-term exposure to synchronous orbit space environment. Its solar directional absorptance doubles and it becomes black. Kapton experiences a 25 to 40 percent reduction in ultimate elongation, a slight reduction in tensile strength, and negligible change in modulus. These changes, however, are not deemed catastrophic enough to eliminate its continued use on space systems.

3) Changes in optical and thermophysical properties for Teflon and Kapton are due primarily to low energy proton radiation and/or its synergistic combination with other environmental components. This conclusion is based on shielding from proton radiation a portion of each long-term exposure optical test specimen.

4) The discoloration of Teflon and Kapton is limited to the upper layer of exposed material. Light scratching of the surface removes the discoloration, revealing clear material underneath. This result supports the contention that protons are the primary cause of the optical degradation [proton penetration depth is less than 0.3 μm (≈0.01 mil)].

5) Catastrophic changes in mechanical properties for Teflon are due primarily to near UV radiation and/or its synergistic combination

with high energy electrons. Specimens exposed to 5-year equivalent electron radiation retain a percentage of their original ultimate elongation, while all specimens subjected to combined near UV and high energy electron radiation become brittle after 17 to 30 months of exposure.

6) Changes in mechanical properties for Kapton are due primarily to near UV radiation and/or its synergistic combination with high energy electrons. Specimens exposed to 5-year equivalent electron radiation show a negligible change in mechanical properties; however, combined near UV and high energy electron radiation results in a measurable, but not catastrophic, reduction in ultimate elongation.

7) Long-term space radiation has a negligible effect on the optical and thermophysical properties of dark mirror materials and carbon-loaded materials. However, the carbon-loaded polyester materials lose much of their flexibility.

8) Based on screening tests consisting of accelerated exposure to charged particles, materials such as Tefzel, Mylar, Aclar, and polysulfone are subject to major changes in optical and thermophysical properties. No tests were conducted to evaluate the radiation effects on their mechanical properties.

9) The annealing effects of short time air exposure on the irradiated optical and thermophysical properties of polymer films are small.

10) There is some evidence to suggest that the return of irradiated specimens to atmospheric conditions prior to testing may affect the measured mechanical properties. The primary effect is an increased rate of degradation if tests are made ex situ.

Aluminized Kapton has been a widely used material for spacecraft passive environmental control and for optical signature control. In LEO where UV is the prime consideration, the material has enjoyed an untarnished reputation [14]. As previously noted and as seen in Figure 20, the time dependent effect on spectral reflectance of a combined UV, electron, and proton environment can be quite severe. The reflectance decreased from 0.28 to 2.5 μ, with the greatest change occurring in the range of 0.5 to 2.0 μ. An important fact is that this response as a function of wavelength was not theoretically predictable, indicating again the necessity for combined UV electron proton exposure to drive out the synergistic effects.

Materials Susceptible to Electron Radiation Induced Electrical Discharge

Many satellites in GEO have experienced, and are continuing to experience, anomalous behavior in their electronic systems at various times during the operational life of the satellite. This is perceived as

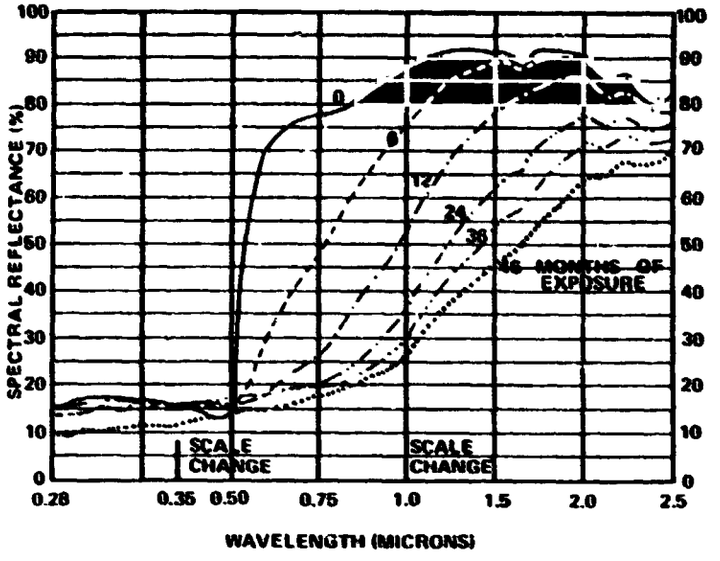


Figure 20. Directional spectral reflectance of 2 mil second surface aluminized kapton.

resulting from environmental charging of the insulator surfaces to high enough potentials to cause electrical breakdown, or discharge. These uncontrolled discharges can be quite harmful by coupling with other circuitry and thereby overstressing electrical components and causing failure, or by creating a steady leakage path through the discharging material itself.

These potentials (voltages) acquired by different areas of a space platform or structure are determined by several parameters: characteristics of the local charged particle population, presence or absence of solar illumination, material properties with respect to secondary emission, backscatter and photo emission, and by the characteristic impedance of the various conducting paths between materials either directly or through the surrounding environment.

Potentials as high as several kilovolts have been noted in sunlight, and as high as 10 keV have been reported in GEO. Appropriate materials selection coupled with judicious and astute electrical discharge circuitry design might possibly be used to provide a controlled discharge, or safety valve function, to preclude uncontrolled and potentially more devastating voltage breakdowns and subsequent damage.

The NASA/Lewis Research Center has tested a number of materials frequently used in spacecraft in a simulated substorm environment [24], i.e., monoenergetic electron energies ranging from 2 to 20 keV at a current density of 1 ma/cm^2, on samples from 300 to 1000 cm^2 in area. They reported that the electrical insulator materials behave as if "they were capacitors," with one surface at ground potential while the surface

facing the electron beam equilibrated at a potential dependent upon secondary emission and backscattering properties, and upon leakage currents. They noted strong edge effects with strong voltage gradients there.

Of considerable interest to the materials specialists is the fact that ZnO pigmented RTV silicone thermal control paint S13G, other conductive paints, quartz cloth and conductive coverslide solar array segments, did not discharge. The Teflon, Kapton, standard solar cells on fiberglass substrate, and solar cells on flexible substrate samples did discharge under the test conditions (Fig. 21). The authors note that some sample scaling up to flight hardware must still be done, but it is clear that electrical discharge effects on materials caused by particulate radiation must be taken into account when selecting materials for long life space structures, especially in the GEO environment.

We now turn our attention to the probable susceptibility of specific space structure components.

MATERIALS TESTED	INDUCED DISCHARGE UNDER GIVEN TEST CONDITIONS
SPACECRAFT PAINTS	
S-13G NON-CONDUCTIVE PAINT	NO
BLACK, WHITE AND YELLOW CONDUCTIVE PAINTS FROM GSFC	NO
SILVERED TEFLON	YES
THERMAL BLANKET MATERIALS	
KAPTON OUTER LAYER	YES
QUARTZ CLOTH OUTER LAYER	NO
SOLAR ARRAY SEGMENTS	
10 MIL 10 OHM-CM SOLAR CELLS WITH 12 MIL FUSED SILICA COVER SLIDE	YES
8 MIL 1 OHM-CM SOLAR CELLS WITH 4 MIL CERIUM DOPED COVER SLIDE	YES
11 MIL 15 TO 45 OHM-CM SOLAR CELLS WITH THIN TRANSPARENT CONDUCTIVE COATING	NO

Figure 21. Spacecraft materials electron radiation induced discharge susceptibility.

Present and Future Projects Environmental Effects Data Requirements

In Figure 22 we see an attempt to survey some present and future representative projects which do require long term space environmental effects data to support conceptual design, trade studies and later phase

Project Status	Mission Life	Tentative Availability	Indigenous Environment*	Critical Materials
Approved				
Space Telescope (OSS)	15 yrs	Nov 83	LEO	Dimensionally Stable Structural and Optical Materials Low Degradation Coatings (Mirror, Thermal Control) Ultrastable Mirror Bonding Adhesives Vacuum Compatible Lubricants
Definition Stage				
Solar Array (OSTS)	5 yrs	1984	LEO/GEO	Radiation Resistant Cover Glasses Long Life Solar Cells Space Stable Cell Adhesives, Substrates, Padding Extended Life Energy Storage Materials (Batteries) High Reliability Deployment Mechanism Materials Vacuum Compatible Lubricants
25 kW Power Module (OSTS)	5 yrs	1984	LEO/GEO	Radiation Resistant Cover Glasses Long Life Solar Cells Space Stable Cell Adhesives, Substrates Padding Extended Life Energy Storage Materials (Batteries) High Reliability Deployment Mechanism Materials Vacuum Compatible Lubricants
Solar Electric Propulsion System (OSTS)	5 yrs	1985	LEO/GEO/ IP	Radiation Resistant Cover Glasses Long Life Solar Cells Space Stable Cell Adhesives, Substrates, Padding Extended Life Energy Storage Materials (Batteries) High Reliability Deployment Mechanism Materials Low Degradation Radiator Coatings Long Life Thruster Materials Vacuum Compatible Lubricants
Advanced X Ray Astrophysics Facility (OSS)	15 yrs	1987	LEO	Ultrastable Structural and Optical Materials Low Degradation Coatings (Mirror Thermal Control) Ultrastable Mirror Bonding Adhesives Long Life Solar Power System Vacuum Compatible Lubricants
Possible Future Projects				
Large Space Structures (OSTS)	30 yrs	1984	GEO	
Deployable Antenna	15 yrs	1984		Dimensionally Stable Reflector Structural Materials Space Stable Insulations and Coatings Radiation Resistant Surface Sensor Materials Vacuum Compatible Lubricants

Figure 22. Present and future projects requiring space environmental effects data.

Project Status	Mission Life	Tentative Availability	Indigenous Environment*	Critical Materials
Large Power Module		TBD		Dimensionally Stable Structural Materials Ultra Low Degradation Solar Cell/ Array Materials Radiation Resistant Electrical Insulation Vacuum Compatible Lubricants
Science and Applications Platform		1986		Dimensionally Stable Structural Materials Radiation Resistant Thermal Control Coatings Space Stable Adhesives Vacuum Compatible Lubricants
Geostationary Platform (OSTS)	30 yrs	1987	GEO	Dimensionally Stable Structural Materials Radiation Resistant Thermal Control Coatings Space Stable Adhesives Vacuum Compatible Lubricants
Orbital Transfer Vehicle (OSTS)	10 yrs (multi mission)	1988	LEO/GEO/ LGT	Radiation Resistant Thermal Insulation Environmentally Stable Propellants Zero Leakage Seals Space Stable Lubricants Vacuum Compatible Lubricants
Science and Applications Platforms (OSTS)	10 yrs	1984	LEO	Dimensionally Stable Structural M Materials Radiation Resistant Thermal Control Coatings Space Stable Adhesives Vacuum Compatible Lubricants
Satellite Power System (OAST)	30 yrs	TBD	GEO	Dimensionally Stable Structural Materials Radiation Resistant Thermal Control Coatings Space Stable Adhesives Vacuum Compatible Lubricants
Nuclear Waste Management (OAST)	30 yrs	1987	IP (Internal Nuclear Radiation)	High Temperature Corrosion and Radiation Resistant Structural Materials Vacuum Compatible Lubricants

MISSION	THERMAL VACUUM	ELECTROMAGNETIC RADIATION	ENERGETIC PARTICLE RADIATION	DEBRIS
LEO	X	UV	South Atlantic Anomaly p	X
LGT LEO GEO Transition	X	UV	Radiation Belt, e p	X
GEO	X	UV	Radiation Belt, e Plasma Sheet e p Solar Wind e p	X
IP (Interplanetary)	X	UV	Solar Wind e p Cosmic Ray Particles	X

Figure 22. (Concluded)

design and materials selection. The economic life of the flight hardware for these projects ranges from 5 to 30 years, which is much longer than most of the payloads previously flown. All of the spacecraft listed in Figure 22 will be subjected to a combined UV/thermal/vacuum/particulate radiation environment for which the specific values of each parameter (flux, energy, particle type, etc.) are determined by the mission flight profile.

A review of the flight systems required for each project discloses that there are certain materials whose long term functional integrity is critical to achieving the economic operating life of the hardware. Some of these materials which have been identified as critical to the success of each project are listed in the table. It is apparent from this listing that there is considerable materials commonality among the projects. Thus, parametric effects data for these common materials would satisfy the needs for several projects. To identify the time frame in which materials data are needed it is typically necessary to know when the so-called phase activity is to be started, since specific property data are then required for design. If the data are to be obtained experimentally, then testing must be initiated in advance of Phase C/D to ensure the timely availability of the required design data base. To obtain long term environmental effects data, testing should begin at least 2 years prior to Phase C/D start. For most of the projects listed in Figure 22 this testing should already be underway.

SUMMARY

Experimental evaluation of the long term environmental effects on materials and components is required for several reasons. Current knowledge of basic radiation damage mechanisms generally limits theoretical evaluations to microscopic changes in simplistic materials produced by a single well-defined environmental parameter. Engineering materials are not simplistic but are complex compositions whose specific chemistry and processing are sometimes proprietary and subject to change by the vendor. In the real world, hardware design is based on functional properties and not on atomic or molecular changes which cannot be directly correlated quantitatively to macroscopic effects. Spacecraft materials are exposed to several environmental stresses simultaneously. These stresses, when considered singularly, may evoke a particular time dependent response that cannot be predicted by a simplistic summation of the singular responses. This result is a typical synergistic effect. Because of these unknown synergistic effects, it generally is not possible to theoretically predict the total response of a material (even a simplistic one) to a combined environment exposure. Further, nonlinear effects cannot be theoretically predicted. These synergisms may be caused by (1) unknown and varying levels of impurities/contaminants, (2) irradiation rate, (3) degree of cure, or (4) in the case of a composite, layup configuration. Time-dependent effects typically are theoretically intractable; that is,

long term effects cannot be predicted based on short term behavior. The present understanding of the behavior of engineering materials exposed to complex environmental stresses largely precludes the quantitative prediction of the time dependent property changes that will be produced by this exposure. Hence there is still a great need for simultaneous environment exposure testing.

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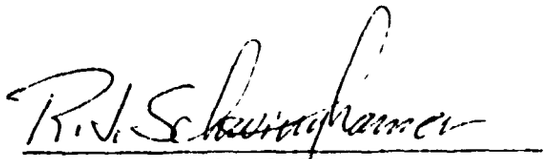
APPROVAL

SPACE ENVIRONMENTAL EFFECTS OF MATERIALS

The information in this report has been reviewed for technical content. Review of any information concerning Department of Defense or nuclear energy activities or programs has been made by the MSFC Security Classification Officer. This report, in its entirety, has been determined to be unclassified.



R. J. SCHWINGHAMER
Director, Materials and Processes Laboratory

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