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SPACE MATERIALS HANDBOOK

Supplement 1 to the Second Edition
Space Materials Experience

Published under Sponsorship of
NATIONAL AERONAUTICS AND SPACE ADMINISTRATION
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FOREWORD

This report was prepared by the Lockheed Palo Alto Research Laboratory, Lockheed Missiles & Space Company, a Group Division of Lockheed Aircraft Corporation, Sunnyvale, Calif., under Contract AF 33(615)-2060. It was initiated under Project 7381, "Materials Application," Task No. 738103, "Data Collection and Correlation." The work was administered jointly under the direction of the AF Materials Laboratory, Research and Technology Division, Mr. E. L. Horne, project engineer, and under NASA Office of Advanced Research Technology, Materials Research Branch, Mr. George Deutsch, Chief.

This report covers work conducted from October 1964 to September 1965. This manuscript was released by authors September 1965 for publication as a RTD Technical Report.

Since this report constitutes a Supplement to the Space Materials Handbook, 2nd edition, some of its elements apply to both the Supplement and to the Space Materials Handbook, 2nd edition. Specifically the appended Glossary and Table of Conversion Factors contain terms and factors drawn both from this Supplement and from the Space Materials Handbook, 2nd edition. The Errata refer only to the second edition. Two indexes are included; the first indexes only the Supplement while the second is a combined index for both this Supplement and the Space Materials Handbook, 2nd edition.

The authors wish to acknowledge the assistance of those colleagues and co-workers at Lockheed who helped in the progress of this work. Our thanks are due to Dr. M. A. Steinberg and Dr. E. C. Burke of the Materials Sciences Laboratory under whose direction this project was carried out. Mr. William G. Jurevic of the Materials Sciences Laboratory was most helpful in the collection and interpretation of data for the materials tables. Our sincere appreciation is due personnel of the LMSC Technical Information Center, namely to Mrs. H. M. Abbott for searching the published literature and to Mr. C. G. Gros for preparing the index. Finally, we are indebted to Mr. J. Todd of the Research Publications Staff for his invaluable assistance in the production of the report.

This technical report has been reviewed and is approved.

D. A. Shinn
Chief, Materials Information Branch
Materials Applications Division
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ABSTRACT

This report consists of two parts. In the first, materials used in successfully orbited spacecraft are reviewed and tabulated. These materials are discussed, insofar as information is available, as to their contribution to spacecraft performance. Data on the effects of the space environment on materials, as measured in space, are tabulated and discussed; particular attention is given to correlating these data with measurements made with simulated environments. The second part of the report recommends specific materials for use in various functional categories on a spacecraft. These recommendations are based on results from both simulated environmental testing and the performance of materials in space as given in the first part. Reasons for the materials choices are discussed and some of the further limitations on these choices due to environmental factors are pointed out.

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Section 1

SPACECRAFT MATERIALS EXPERIENCE

1.1 INTRODUCTION

This section surveys the materials used in the construction of successfully orbited spacecraft for which information has become available since the completion of the 2nd edition of the Space Materials Handbook. It covers the period between October 1963 and June 1965 during which approximately 184 space vehicles were put into either Earth or solar orbit. The present compilation is intended to supplement Chapter 20 of the Space Materials Handbook.

After increasing at a geometric-progression rate for the first 5 yr of the space era (Ref. 1), launchings actually decreased during 1963, with 1964 running slightly ahead of the peak of 1962. Both U.S. and USSR launches followed this pattern of a build up to 1962 as a peak year although USSR launches showed less pause for 1963 than did the U.S. After launching no more than 6 satellites and probes a year before 1962, the USSR launched 20 in 1962, 17 in 1963, and 30 in 1964. Unfortunately little or no materials data are available on Soviet space vehicles.

Information on the effects of the space environment on materials used in spacecraft must still be obtained mostly by inferences gleaned from the overall systems performance of the space vehicle since materials recovered from space are generally not available except for the short-term exposures obtained in Mercury and Gemini spacecraft and in certain recoverable classified payloads. In a few cases, however, instrumentation has been flown to obtain direct information on environmental effects, such as the solar-cell shielding experiments carried out on certain of the Explorer satellites. Generally, the reported performance of a given spacecraft when related to the materials known to have been used in its construction enables the designer to deduce some useful information about the behavior of the materials used in a given orbital environment. Particular attention is paid in this supplement to relating orbital performance, where available, to the results of space simulation testing in the laboratory.

The information presented in this supplement has been obtained from published technical literature, from Government-sponsored reports, and from direct interviews with responsible scientists and engineers connected with the various spacecraft projects at government agencies, laboratories, and industrial organizations. The volume of information obtained through interviews has made necessary the use of a large number of "personal communication" references. It is felt that the timeliness of including such late information through this device outweighs the disadvantage of these references not being immediately available to the reader.

1.2 RESUMÉ OF SUCCESSFULLY LAUNCHED SPACECRAFT

Spacecraft launched since October 1963 are summarized in the first four tables and include their orbital parameters and some launch data (Refs. 2 and 3). Table 1-1 lists those spacecraft which were launched primarily to gather scientific data, to report on the Earth's weather conditions or to serve as a link in long-range communications systems and for which a fair amount of data have been released. These spacecraft are mostly sponsored by the National Aeronautics and Space Administration (NASA). Spacecraft sponsored by the U.S. Department of Defense through one of the armed services are collected in Table 1-2. This collection includes a number of vehicles identified only by launch vehicle and orbital parameters; no information has been released on their payloads and missions. Also included in this table are other satellites, which, while built under sponsorship of one of the services, have had their payloads reported to greater or lesser extent. Some materials information is available on some of these satellites and is included in this supplement.

The U.S. manned spacecraft and direct precursors are listed in Table 1-3. Table 1-4 shows available information on satellites and space probes launched by the USSR.

The environmental factors affecting a spacecraft designed to operate at a given orbital altitude and inclination are necessarily complex, but, in a general manner, can be correlated with these orbital parameters. As an aid in this correlation, Table 1-5 presents spacecraft arranged according to altitude at apogee and inclination of their orbits from the equator. By use of this table, the attention of the spacecraft designer can thus be focused readily on those satellites which have encountered the same space environment as that for which he is designing a new satellite.

Following the practice established in Ref. 4, the categories into which spacecraft are sorted are as follows:

- Earth orbiting vehicles having apogees < 600 sm
- Earth orbiting vehicles having apogees < 600 sm and inclinations > 70 deg
- Earth orbiting vehicles having apogees between 600 and 10,000 sm
- Earth orbiting vehicles having apogees between 600 and 10,000 sm and inclinations > 70 deg
- Earth orbiting vehicles having apogees between 10,000 and 45,000 sm
- Earth orbiting vehicles having apogees > 45,000 sm
- Lunar probes and interplanetary vehicles

The space environments to be encountered by spacecraft in these various orbits are discussed in Chapters 4 through 8, pp. 19–88 and on pp. 603–604 of Ref. 4.

Table 1-1

U.S. SATELLITES AND SPACECRAFT

Name	Project Direction	Date	Site(a)	Vehicle(b)	Weight (lb)	Period (min)	Perigee (sm)	Apogee (sm)	Inclination From Equator (deg)	Status
Explorer 18	NASA	11/26/63	AMR	Delta	138	96.3 hr 93.4 hr	119 2,402	122,522 119,523	33.3 36.4	In orbit: IMP-1, discovered high-energy radiation region beyond Van Allen belt, transmitting on 136.111 Mc
Explorer 19	NASA	12/19/63	PA	Scout	18	115.9 115.4	366 382	1,487 1,454	78.6 78.6	In orbit: 12-ft balloon identical to Explorer 9, for atmospheric density studies in near-polar orbit
Tiros 8	NASA	12/21/63	AMR	Delta	265	99.3 99.4	430 437	473 467	58.5 58.5	In orbit: first Tiros to carry APT system, transmitting on 136.233 and 136.924 Mc
Relay 2	NASA	1/21/64	ETR	Delta	172	194.7 194.7	1,298 1,296	4,606 4,607	46.0 46.3	In orbit: active-repeater comsat, provided first Japan-France comsat link transmitting on 136.14[illegible] and 136.621 Mc
Echo 2	NASA	1/25/64	WTR	Thor-Agena B	547	108.8 108.5	642 676	816 766	81.5 81.5	In orbit: passive reflector comsat, first cooperative program with USSR, beacon transmitting on 136.021 and 136.170 Mc
Ranger 6	NASA	1/30/64	ETR	Atlas-Agena B	804	Flight time: 65.6 hr				Impacted on moon: closeup photographic experiment failed to provide data
Ariel 2	NASA/UK	3/27/64	WI	Scout	150	101.3 100.6	180 180	843 800	51.6 51.7	In orbit: returning data from British galactic noise, ozone and micrometeoroid experiments, transmitting on 136.577 Mc
Ranger 7	NASA	7/28/64	ETR	Atlas-Agena D	806	Flight time: 68.6 hr				Impacted on moon: closeup photographic experiment returned 4316 high-quality photographs in last 13 min
Syncom 3	NASA	8/19/64	ETR	TAD	86	1436.2 1436.5	22,164 22,239	22,312 22,244	0.1 0.1	In orbit: first truly synchronous comsat, now stationed at 180° W; communication tests successful
Explorer 20	NASA	8/25/64	WTR	Scout	97	103.9 103.9	540 540	634 634	79.9 79.9	In orbit: ionosphere research satellite part of Topside Sounder Program, transmitting on 136.680 and 136.350 Mc
Nimbus 1	NASA	8/28/64	WTR	Thor-Agena B	830	98.3 98.4	263 267	579 580	98.6 98.7	In orbit: second-generation weather satellite, 27,000 photos returned until 8/28/64 when solar paddles locked
OGO 1	NASA	9/4/64	ETR	Atlas-Agena B	1,073	64.0 hr 64.0 hr	175 801	92,827 92,216	31.1 32.3	In orbit; spin-stabilized rather than earth-oriented because of boom erection malfunction, returning data from 20 geophysical experiments on 400 MC

Table 1-1 (cont'd)

Name	Project Direction	Date	Site[a]	Vehicle[b]	Weight (lb)	Period (min)	Perigee (sm)	Apogee (sm)	Inclination From Equator (deg)	Status
Explorer 21	NASA	10/3/64	ETR	Delta	136	35 hr 34.7 hr	122 414	59,253 58,734	33.5 33.8	In orbit: IMP-2, apogee much lower than planned 126,500 sm, transmitting solar wind data on 136.145 Mc
Explorer 22	NASA	10/9/64	WTR	Scout	116	104.7 104.8	549 552	669 671	79.7 79.7	In orbit: ionosphere and geodetic research satellite, being tracked by laser at Wallops, transmitting on 136.171 Mc
Mariner 3	NASA	11/5/64	ETR	Atlas-Agena D	575					In solar orbit: shroud failure prevented planned Mars flyby
Explorer 23	NASA	11/6/64	WI	Scout	295	99.2 99.2	288 289	610 606	51.9 52.0	In orbit: four micrometeorite experiments wrapped around Scout fourth stage, transmitting on 136.080 and 136.857 Mc
Explorer 24	NASA	11/21/64	WTR	Scout	19	116.3 116.3	344 336	1,551 1,544	81.4 81.4	In orbit: 12-ft balloon, to continue Explorers 9 and 19 studies, first NASA dual payload launch, beacon on 136.710 Mc
Explorer 25	NASA	11/21/64	WTR	Scout	90	116.3 116.2	345 329	1,547 1,550	81.4 81.4	In orbit: radiation satellite, results integrated with Explorer 24 air density data, transmitting on 136.292 and 136.860 Mc
Mariner 4	NASA	11/28/64	ETR	Atlas-Agena D	575					In solar orbit: five interplanetary experiments operational; took first close-up pictures of Mars on July 14, 1965, passing within 5600 sm of the planet
San Marco 1	Italy	12/15/64	WI	Scout	254	94.9 94.8	128 126	510 499	37.8 37.8	In orbit: flight test of Italian ionosphere-air density satellite, transmitting on 136.536 and 136.738 Mc
Explorer 26	NASA	12/21/64	WTR	Scout	101	456 458	190 193	16,280 16,326	20.2 20.2	In orbit: repeat of Explorer 15, intended to study natural and artificial radiation, transmitting on 136.275 Mc
Tiros 9	NASA	1/22/65	ETR	Delta	305	119.2 119.2	435 437	1,602 1,606	96.4 96.4	In orbit: first "cartwheel" Tiros, elliptical rather than planned circular orbit
OSO 2	NASA	2/3/65	ETR	Delta	545	96.5 96.5	343 341	393 388	32.9 32.9	In orbit: six of eight experiments returning good solar x-ray, γ-ray, and UV data, transmitting on 136.712 Mc
Pegasus 1	NASA	2/16/65	ETR	Saturn I	23,000	97.0 97.0	308 310	462 453	31.7 31.8	In orbit: meteoroid detection satellite with 2300 ft^2 of sensors, attached to S-IV second stage, transmitting on 136.41 and 136.89 Mc

(a) AMR – Atlantic Missile Range; PA – Point Arguello;
ETR – Eastern Test Range; WTR – Western Test Range; WI – Wallops Island.
(b) TAD – Thrust-Augmented Delta.

Ranger 8	NASA	2/16/65	ETR	Atlas-Agena B	809	Flight time: 64.9 hr				Impacted on moon: returned 7137 photos, landed in Sea of Tranquility at 2.59° N, 24.77° E
Ranger 9	NASA	3/21/65	ETR	Atlas-Agena B	809	Flight time: 64.5 hr				Impacted on moon: returned 5814 photos, landed in crater Alphonsus at 12.91° S, 2.38° W
Early Bird 1	COMSAT Corp.	4/6/65	ETR	TAD	88	1437	21,748	22,732	359.86	In orbit: transmitting on 136.440 and 136.980 Mc
Explorer 27	NASA	4/27/65	WI		132	107.8	595	818	41.16	In orbit: transmitting on 136.739 Mc
Pegasus 2	NASA	5/25/65	ETR	Saturn		97.29	314.1	465.9	31.77	In orbit: transmitting on 136.890 and 136.410 Mc
Explorer 28	NASA	6/7/65		Delta		8558.1	120	164,000	34	In orbit: transmitting on 136.125 Mc

(a) ETR – Eastern Test Range; WI – Wallops Island.
(b) TAD – Thrust-Augmented Delta.

Table 1-2
DOD-SPONSORED SPACECRAFT

Name	Project Direction	Date	Site[a]	Vehicle[b]	Weight (lb)	Period (min)	Perigee (sm)	Apogee (sm)	Inclination From Equator (deg)	Status
None	USAF	8/24/63	VAFB	TAT Agena D		89.5	104	202	75.0	Decayed 9/12/63: classified payload
None	USAF	8/29/63	VAFB	Thor-Agena D		90.8	183	202	81.9	Decayed 11/7/63: classified payload
None	USAF	8/29/63	VAFB	Thor-Agena D		92.0	195	262	81.9	Decayed 9/28/63: classified payload
None	USAF	9/6/63	PA	Atlas-Agena D						Decayed 9/13/63: classified payload
None	USAF	9/23/63	VAFB	TAT Agena D		90.6	100	274	74.9	Decayed 10/12/63: classified payload
None	USAF/USN	9/28/63	VAFB	Thor-Able Star	160	107.4 107.4	676 664	714 710	89.9 89.9	In orbit: classified payload included 27-lb SNAP-9A nuclear power supply
None	USAF/USN	9/28/63	VAFB	Thor-Able Star	120	107.4 107.3	667 664	705 710	89.9 89.9	In orbit: classified payload, transmitting on 136.652 Mc
Vela 1	USAF	10/16/63	AMR	Atlas-Agena D	297	105 hr 108.0 hr	63,441 62,906	70,631 72,279	38.3 38.1	In orbit: nuclear detection satellite, to provide radiation background data
TRS 5	USAF	10/16/63	AMR	Atlas-Agena D	4.5	39 hr 38.7 hr	129 592	64,531 63,610	36.7 35.9	In orbit: pickaback returned radiation data for two weeks
Vela 2	USAF	10/16/63	AMR	Atlas-Agena D	297	108.7 hr 109.9 hr	62,806 64,474	72,974 72,379	38.0 37.0	In orbit: second nuclear detection satellite, initially lagged 140 deg behind Vela 1, continues to transmit
None	USAF	10/25/63	PA	Atlas-Agena D		90.0	89	206	99.1	Decayed 10/29/63: classified payload (Initial orbital data as of 10/26/64)
None	USAF	10/25/63	PA	Atlas-Agena D		88.7	75	181	99.1	Decayed 10/29/63: classified payload
None	USAF	10/29/63	VAFB	TAT-Agena D		90.9	173	218	89.9	Decayed 1/21/64: classified payload
None	USAF	10/29/63	VAFB	TAT-Agena D		93.4 91.5	193 170	349 263	89.9 90.0	In orbit: classified payload
None	USAF	11/27/63	PA	Thor-Agena D		90.1	109	236	70.0	Decayed 12/15/63: classified payload
None	USAF/USN	12/5/63	VAFB	Thor-Able Star		107.2 107.1	665 667	690 693	90.0 90.0	In orbit: classified payload included SNAP-9A nuclear power supply, transmitting on 150 and 400 Mc
None	USAF/USN	12/5/63	VAFB	Thor-Able Star		107.2 107.1	666 667	689 693	90.0 90.0	In orbit: classified payload, transmitting on 54, 163, 324, and 648 Mc
None	USAF	12/18/63	PA	Atlas-Agena D		88.8	76	165	97.9	Decayed 12/20/63: classified payload (initial orbital data as of 12/19/63)
None	USAF	12/21/63	VAFB	Thor-Agena D		89.3	107	189	64.9	Decayed 1/9/64: classified payload

None	USAF	12/21/63	VAFB	Thor-Agena D		91.7	196	245	64.5	Decayed 11/7/64: classified payload
None	USAF/USN	1/11/64	WTR	TAT-Agena D		103.5 103.4	563 567	578 579	69.9 69.9	In orbit: classified payload (initial orbital data as of 1/15/64)
GGSE	USAF/USN	1/11/64	WTR	TAT-Agena D		103.5 103.4	560 567	585 579	70.0 69.9	In orbit: gravity gradient stabilization experiment
EGRS	USAF/USN	1/11/64	WTR	TAT-Agena D		103.5 103.4	563 567	578 579	69.9 69.9	In orbit: payload designed to precisely locate other satellites, transmitting on 136.804 Mc
Greb 5	USAF/USN	1/11/64	WTR	TAT-Agena D	100	103.5 103.5	563 567	578 579	69.9 69.9	In orbit: solar radiation satellite monitoring x-ray and UV emissions, transmitting ing on 136.887 Mc
None	USAF/USN	1/11/64	WTR	TAT-Agena D		103.5 103.5	555 569	591 578	69.9 69.9	In orbit: classified payload
None	USAF	1/19/64	WTR	Thor-Agena D		101.3 101.3	500 501	518 518	99.0 99.1	In orbit: classified payload (initial orbital data as of 1/31/64)
None	USAF	1/19/64	WTR	Thor-Agena D		101.3 101.3	501 501	514 521	99.1 99.1	In orbit: classified payload
None	USAF	2/15/64	WTR	TAT-Agena D		90.9	119	278	75.1	Decayed 3/9/64: classified payload
None	USAF	2/25/64	WTR	Atlas-Agena D		88.2	107	118	95.7	Decayed 3/1/64: classified payload (initial orbital data as of 2/26/64)
None	USAF	2/27/64	WTR	TAT-Agena D		94.6 94.6	302 304	319 318	82.1 82.1	In orbit: classified payload
None	USAF	3/11/64	WTR	Atlas-Agena D		89.8	89	240	95.8	Decayed 3/16/64: classified payload (initial orbital data as of 3/12/64)
None	USAF	4/23/64	WTR	Atlas-Agena D		89.4	93	209	103.6	Decayed 4/28/64: classified payload (initial orbital data as of 4/25/64)
None	USAF	4/27/64	WTR	TAT-Agena D		90.8	109	277	80.0	Decayed 5/26/64: classified payload (initial orbital data as of 4/30/64)
None	USAF	5/19/64	WTR	Atlas-Agena D		101.1	88	236	89.7	Decayed 5/22/64: classified payload (initial orbital data as of 5/20/64)
None	USAF	6/3/64	WTR	Scout		103.1 103.1	531 537	594 588	90.4 90.5	In orbit: classified payload transmitting on 150 and 400 Mc (initial orbital data as of 6/5/64)
None	USAF	6/4/64	WTR	TAT-Agena D		90.2	93	267	80.0	Decayed 6/18/64: classified payload (initial orbital data as of 6/7/64)
None	USAF	6/13/64	WTR	TAT-Agena D		91.7 91.1	219 199	225 211	115.0 115.0	In orbit: payload included Star Flash experiment (initial orbital data as of 6/15/64)
None	USAF	6/17/64	WTR	Thor-Agena D		101.6 101.6	514 513	523 524	99.8 99.8	In orbit: classified payload (initial orbital data as of 6/18/64)
None	USAF	6/17/64	WTR	Thor-Agena D		101.6 101.6	515 513	523 524	99.8 99.8	In orbit: classified payload
None	USAF	6/19/64	WTR	TAT-Agena D		91.0	109	287	85.0	Decayed 8/16/64: classified payload (initial orbital data as of 6/30/64)

(a) VAFB – Vandenberg Air Force Base; PA – Point Arguello; AMR – Atlantic Missile Range; WTR – Western Test Range.
(b) TAT – Thrust-Augmented THor.

Table 1-2 (cont'd)

Name	Project Direction	Date	Site[a]	Vehicle[b]	Weight (lb)	Period (min)	Perigee (sm)	Apogee (sm)	Inclination From Equator (deg)	Status
None	USAF	7/2/64	WTR	TAT-Agena D		94.9 94.9	311 309	329 328	82.1 82.1	In orbit: classified payload (initial orbital data as of 8/3/64)
None	USAF	7/6/64	WTR	Atlas-Agena D		92.9	75	215	89.4	Decayed 7/8/64: classified payload (initial orbital data as of 7/7/64)
None	USAF	7/6/64	WTR	Atlas-Agena D		90.9 89.3	179 146	216	93.0 93.0	In orbit: classified payload (initial orbital data as of 7/15/64)
None	USAF	7/10/64	WTR	TAT-Agena D		91.0	112	286	85.0	Decayed 8/6/64: classified payload
Vela 3	USAF	7/17/64	ETR	Atlas-Agena D	319	100.3 hr 101.5 hr	63,369 65,061	62,024 65,349	39.5 39.1	In orbit: experimental nuclear detection satellite to test satellite-borne sensor system, continues to transmit
Vela 4	USAF	7/17/64	ETR	Atlas-Agena D	319	100.1 hr 101.2 hr	58,766 58,772	69,482 70,292	40.9 40.9	In orbit: icosahedron identical to Vela 3, lags Vela 3 by 140 deg, continues to transmit
TRS 6	USAF	7/17/64	ETR	Atlas-Agena D	4.5	39.2 hr 39.2 hr	120 198	64,886 64,637	36.7 38.3	In orbit: tetrahedral research satellite intended to return radiation data, transmitting on 136.771 Mc
None	USAF	8/5/64	WTR	TAT Agena D		90.7	271	4,076	80.0	Decayed 8/31/64: classified payload
None	USAF	8/14/64	WTR	Atlas-Agena D		89.0	93	191	95.5	Decayed 9/23/64: classified payload (initial orbital data as of 8/15/64)
None	USAF	8/14/64	WTR	Atlas-Agena D		127.4 127.1	163 167	2,332 2,315	95.6 95.7	In orbit: classified payload
None	USAF	8/14/64	WTR	TAT-Agena D		91.6 90.8	217 118	226 199	115.0 115.0	In orbit: classified payload (initial orbital data as of 8/22/64)
None	USAF	9/14/64	WTR	TAT-Agena D		90.8	119	286	85.0	Decayed 10/6/64: classified payload (initial orbital data as of 9/15/64)
None	USAF	9/14/64	WTR	Atlas-Agena D		89.0	90	188	92.9	Decayed 9/28/64: classified payload (initial orbital data as of 9/25/64)
None	USAF	10/5/64	WTR	TAT-Agena D		90.4	109	243	80.0	Decayed 10/26/64: classified payload (initial orbital data as of 10/7/64)
None	USAF/USN	10/6/64	WTR	Thor-Able Star		106.6 106.6	657 656	673 674	89.9 89.9	In orbit: classified payload (initial orbital data as of 10/15/64)
None	USAF/USN	10/6/64	WTR	Thor-Able Star		106.6 106.6	655 657	674 672	89.9 89.9	In orbit: classified payload

None	USAF/USN	10/6/64	WTR	Thor-Able Star		106.6 106.6	657 659	673 672	90.0 89.9	In orbit: classified payload
None	USAF	10/17/64	WTR	TAT-Agena D		90.6	117	258	75.0	Decayed 12/4/64: classified payload (initial orbital data as of 10/19/64)
None	USAF	10/23/64	WTR	Atlas-Agena D		88.6	86	168	95.6	Decayed 10/28/64: classified payload (initial orbital data as of 10/19/64)
None	USAF	10/23/64	WTR	Atlas-Agena D		91.1 90.5	193 178	214 194	95.5 95.5	In orbit: classified payload (initial orbital data as of 10/31/64)
None	USAF	10/23/64	WTR	Atlas-Agena D						Decayed 10/29/64: classified payload
None	USAF	11/2/64	WTR	TAT-Agena D		90.7	112	278	80.0	Decayed 11/28/64: classified payload (initial orbital data as of 11/3/64)
None	USAF	11/3/64	WTR	TAT-Agena D		95.1 95.0	318 316	327 328	82.0 82.0	In orbit: classified payload (initial orbital data as of 11/5/64)
None	USAF	11/18/64	WTR	TAT-Agena D		89.7	112	211	70.0	Decayed 12/6/64: classified payload (initial orbital data as of 11/21/64)
None	USAF	12/4/64	WTR	Atlas-Agena D						Decayed 12/5/64: classified payload
None	USAF/USN	12/12/64	WTR	Thor-Able Star		106.3	639	672	90.0	In orbit: classified spacecraft, transmitting on 136.651 Mc
None	USAF/USN	12/12/64	WTR	Thor-Able Star						In orbit: classified spacecraft
None	USAF	12/19/64	WTR	TAT-Agena D		90.3	114	237	75.0	In orbit: classified spacecraft
None	USAF	12/21/64	WTR	TAT-Agena D		89.4	144	155	70.1	In orbit: classified payload
None	USAF	1/15/65	WTR	TAT-Agena D		90.5	112	261	75.0	Decayed 2/9/65: classified payload (initial orbital data as of 1/21/65)
None	USAF	1/18/65	WTR	Thor-Altair		97.7 97.6	293 285	511 519	98.8 98.8	In orbit: classified payload, first use of Thor-Altair launch vehicle (initial orbital data as of 1/20/65)
None	USAF	1/23/65	WTR	Atlas-Agena D		88.9	91	181	102.5	Decayed 1/28/65: classified payload (initial orbital data as of 1/25/65)
LES 1	USAF	2/11/65	ETR	Titan 3A	69	145.7 145.7	1,726 1,727	1,744 1,744	32.2 32.1	In orbit: Transtage ejected experimental communications satellite after two re-starts (Initial orbital data as of 2/15/65)
None	USAF	2/25/65	WTR	TAT-Agena D		90.1	110	234	75.1	Decayed 3/18/65: classified payload (initial orbital data as of 2/26/65)
Surcal 1	USN/USA	3/9/65	WTR	Thor-Agena D		103.5	564	584	70.1	In orbit: surveillance calibration satellite first 8-payload launch
GGSE 2	USN/USA	3/9/65	WTR	Thor-Agena D		103.5 103.5	562 566	583 583	70.1 70.1	In orbit: gravity gradient stabilization experiment, transmitting on 136.739 Mc orbital data as of 3/15/65)
GGSE 3	USN/USA	3/9/65	WTR	Thor-Agena D		103.5 103.5	562 563	583 585	70.1 70.1	In orbit: gravity gradient stabilization experiment, transmitting on 136.776 Mc

(a) WTR – Western Test Range; ETR – Eastern Test Range.
(b) TAT – Thrust-Augmented Thor.

Table 1-2 (concl'd)

Name	Project Direction	Date	Site[a]	Vehicle[b]	Weight (lb)	Period (min)	Perigee (sm)	Apogee (sm)	Inclination From Equator (deg)	Status
Greb 6	USN/USA	3/9/65	WTR	Thor-Agena D		103.4 103.5	562 564	583 585	70.1 70.1	In orbit: solar radiation satellite, transmitting on 136.801 Mc
Secor 3	USN/USA	3/9/65	WTR	Thor-Agena D	40	103.4 103.5	562 564	583 585	70.1 70.1	In orbit: geodetic satellite, transmitting on 136.84 Mc
Surcal 2	USN/USA	3/9/65	WTR	Thor-Agena D						In orbit: surveillance calibration satellite
Oscar 3	USN/USA	3/9/65	WTR	Thor-Agena D	33					In orbit: amateur radio satellite, transmitted for 16 days
None	USN/USA	3/9/65	WTR	Thor-Agena D						In orbit: classified payload
None	USAF/USA	3/11/65	WTR	Thor-Able Star		97.8 97.9	184 175	634 642	89.9 90.0	In orbit: classified payload (initial orbital data as of 3/15/65)
Secor 2	USAF/USA	3/11/65	WTR	Thor-Able Star	40	97.9	178	642	90	In orbit: geodetic satellite, transmitting on 136.840 Mc
None	USAF	3/12/65	WTR	Atlas-Agena D		88.8	93	178	107.6	Decayed 3/17/65: classified payload (initial orbital data as of 3/15/65)
None	USAF	3/17/65	WTR	Thor-Altair		97.6	329	471	99.0	In orbit: classified payload
None	USAF	3/25/65	WTR	TAT-Agena D		88.9	112	147	96.0	In orbit: classified payload

(a) ETR – Eastern Test Range; WTR – Western Test Range.
(b) TAT – Thrust-Augmented Thor.

Table 1-3

U.S. MANNED ORBITAL SPACECRAFT

Name	Project Direction	Date	Site[a]	Vehicle	Weight (lb)	Period (min)	Perigee (sm)	Apogee (sm)	Inclination From Equator (deg)	Status
Gemini-Titan 1	NASA	4/8/64	ETR	Titan 2	11,400	89.2	100	204	32.6	Decayed 4/12/64: unmanned boilerplate Gemini, plus second stage
Gemini-Titan 3	NASA	3/8/65	ETR	Titan 2	7,000	88.2	100	140	32.5	Re-entered 3/23/65: first manned orbital maneuvers, V. Grissom and J. Young landed after 3 orbits
Gemini-Titan 4	NASA	6/3/65	ETR	Titan 2	7,000		87	158		Re-entered 6/7/65: first US extra-vehicular activity by astronaut in orbit

Table 1-4

USSR SPACECRAFT

Name	Project Direction	Date	Site	Vehicle	Weight (lb)	Period (min)	Perigee (sm)	Apogee (sm)	Inclination From Equator (deg)	Status
COSMOS VEHICLES										
Cosmos 20	USSR	10/18/63	Tyuratum			99	123	186	65.0	Re-entered or decayed 10/30/63: unannounced payload
Cosmos 21	USSR	11/11/63	Tyuratam			88.5	121	142	64.9	Re-entered or decayed 11/14/63: unannounced payload
Cosmos 22	USSR	11/16/63	Tyuratam			90.3	127	245	64.9	Re-entered or decayed 11/22/63: unannounced payload
Cosmos 23	USSR	12/13/63	Kapustin Yar			92.9	149	381	49.0	Decayed 3/27/64: unannounced payload
Cosmos 24	USSR	12/19/63	Tyuratam			90.5	131	254	65.0	Re-entered or decayed 12/28/63: unannounced payload
Cosmos 25	USSR	2/27/64	Kapustin Yar			92.3	169	327	49	Decayed 11/21/64: unannounced payload
Cosmos 26	USSR	3/18/64	Kapustin Yar			91.1	168	250	49	Decayed 9/28/64: unannounced payload
Cosmos 27	USSR	3/27/64	Tyuratam			88.4	119	147	64.8	Re-entered or decayed 3/28/64: unannounced payload
Cosmos 28	USSR	4/4/64	Tyuratam			90.4	130	245	65	Re-entered or decayed 4/12/64: unannounced payload
Cosmos 29	USSR	4/25/64	Tyuratam			89.5	127	192	65.1	Re-entered or decayed 5/2/64: unannounced payload
Cosmos 30	USSR	5/18/64	Tyuratam			90.2	128	238	64.9	Re-entered or decayed 5/26/64: unannounced payload
Cosmos 31	USSR	6/6/64	Kapustin Yar			91.6	142	316	49	Decayed 10/20/64: unannounced payload
Cosmos 32	USSR	6/10/64				89.8	130	207	51.3	Re-entered or decayed 6/18/64: unannounced payload, new inclination for Cosmos program
Cosmos 33	USSR	6/23/64	Tyuratam			89.4	130	182	65	Re-entered or decayed 7/1/64: unannounced payload
Cosmos 34	USSR	7/1/64	Tyuratam			90	127	223	65.0	Re-entered or decayed 7/9/64: unannounced payload
Cosmos 35	USSR	7/15/64				89.2	135	166	51.3	Re-entered or decayed 7/23/64: unannounced payload

Cosmos 36	USSR	7/30/64	Kapustin Yar			91.9 90.7	161 150	313 231	49	In orbit: unannounced payload
Cosmos 37	USSR	8/14/64	Tyuratam			89.5	127	186	65	Re-entered or decayed 8/22/64: unannounced payload
Cosmos 38	USSR	8/19/64				95.2	130	544	56.2	Decayed 11/8/64: unannounced payload, first Soviet triple payload launch
Cosmos 39	USSR	8/19/64				95.2	130	544	56.2	Decayed 11/17/64: unannounced payload
Cosmos 40	USSR	8/19/64				95.2	130	544	56.2	Decayed 11/18/64: unannounced payload
Cosmos 41	USSR	8/22/64	Tyuratam			715 714.7	245 311	24,765 24,672	64 65.5	In orbit: unannounced payload, first Tyuratam-launched Cosmos with extended lifetime
Cosmos 42	USSR	8/22/64	Kapustin Yar			97.8 96.8	144 138	683 610	49 49.0	In orbit: unannounced payload, first double payload launch in Cosmos program
Cosmos 43	USSR	8/22/64	Kapustin Yar			97.8 96.8	144 141	683 606	49 49.0	In orbit: unannounced payload
Cosmos 44	USSR	8/29/64	Tyuratam			99.5 99.5	384 377	534 538	65 65.1	In orbit: unannounced payload
Cosmos 45	USSR	9/13/64	Tyuratam			90	128	203	64.9	Re-entered or decayed 9/18/64: unannounced payload
Cosmos 46	USSR	9/24/64				89.2	134	168	50.3	Re-entered or decayed 10/2/64: unannounced payload
Cosmos 47	USSR	10/6/64	Tyuratam			90.0	110	257	64.8	Re-entered or decayed 10/7/64: unannounced payload, believed to be final unmanned flight test of Voskhod
Cosmos 48	USSR	10/14/64	Tyuratam			89.4	122	177	65	Re-entered or decayed 10/20/64: unannounced payload
Cosmos 49	USSR	10/24/64	Kapustin Yar			91.8 91.6	162 160	304 280	49.0 49.0	In orbit: unannounced payload
Cosmos 50	USSR	10/24/64				88.7	122	150	51.3	Decayed 12/5/64: unannounced payload
Cosmos 51	USSR	12/10/64	Kapustin Yar			92.5 92.5	164 162	344 331	48.8 48.8	In orbit: unannounced payload, 27th Cosmos orbited in 1964
Cosmos 52	USSR	1/11/65	Tyuratam			89.5	127	189	65.0	Re-entered or decayed 1/19/65: unannounced payload
Cosmos 53	USSR	1/30/65	Kapustin Yar			98.7 98.3	141 138	741 700	48.8 48.7	In orbit: unannounced payload
Cosmos 54	USSR	2/21/65				106.2 104.9	174 162	1,153 1,064	56.1 56.0	In orbit: unannounced payload, second Soviet triple satellite launch
Cosmos 55	USSR	2/21/65				106.2 105.1	174 163	1,153 1,078	56.1 56.0	In orbit: unannounced payload
Cosmos 56	USSR	2/21/65				106.2 104.4	174 162	1,153 1,036	56.1 56.0	In orbit: unannounced payload

Table 1-4 (cont'd)

Name	Project Direction	Date	Site	Vehicle	Weight (lb)	Period (min)	Perigee (sm)	Apogee (sm)	Inclination From Equator (deg)	Status
Cosmos 57	USSR	2/22/65	Tyuratam			91.1	109	318	64.8	Re-entered or decayed 2/22/65: unannounced payload, exploded into more than 150 pieces
Cosmos 58	USSR	2/26/65	Tyuratam			96.8 96.8	361 349	409 403	65.0 65.0	In orbit: unannounced payload
Cosmos 59	USSR	3/7/65	Tyuratam			89.8	130	211	65.0	Decayed 3/15/65: unannounced payload
Cosmos 60	USSR	3/12/65	Tyuratam			89.1 89.0	125 121	178 164	64.7 64.7	Re-entered or decayed 3/17/65: unannounced payload
Cosmos 61	USSR	3/14/65				105 105.1	170 163	1,141 1,077	56.0 56.0	In orbit: unannounced payload, third triple satellite launch in Cosmos series
Cosmos 62	USSR	3/14/65				105 104.8	170 161	1,141 1,063	56.0 56.0	In orbit: unannounced payload
Cosmos 63	USSR	3/14/65				105 104.4	170 164	1,141 1,035	56.0 56.0	In orbit: unannounced payload
Cosmos 64	USSR	3/25/65	Tyuratam			89.2 89.1	128 127	168 155	65 65.0	In orbit: unannounced payload
Cosmos 65	USSR	4/17/65								Re-entered 4/25/65
Cosmos 66	USSR	5/7/65								Re-entered 5/15/65
Cosmos 67	USSR	5/25/65				90.1	126	218	51.86	Decayed 6/2/65
Cosmos 68										
Cosmos 69	USSR	6/25/65				89.7	126	200		
Cosmos 70	USSR	7/2/65				98.3	142	717	48.8	
MANNED SPACECRAFT										
Voskhod 1	USSR	10/12/64	Tyuratam		11,070	90.1	100	255	65	Re-entered 10/12/64: first three-man crew – V. Komarov, K. Feoktistov, and B. Yegorov, landed after 16 orbits
Voskhod 2	USSR	3/18/65	Tyuratam			90.9	107	308	65	Re-entered 3/19/65: A. Leonov spent 10 min outside spacecraft, landed with B. Belyayev after 17 orbits

OTHER SATELLITES AND SPACE PROBES										
. Polyot 1	USSR	11/1/63				102.3	211 212	329 868	59.0	In orbit: first Soviet spacecraft reported to have extensive maneuver capability
Elektron 1	USSR	1/30/64				169 169.3	252 251	4,412 4,421	61 60.9	In orbit: scientific satellite to study inner Van Allen radiation belt, first dual Soviet launch
Elektron 2	USSR	1/30/64				1360 1356.3	286 557	42,377 41,958	61 59.2	In orbit: intended to provide data on outer radiation belt, configuration differed from Elektron 1
Zond 1	USSR	4/2/64				Heliocentric Orbit				In solar orbit: only the third announced Soviet probe in long, unsuccessful inter-planetary program
Polyot 2	USSR	4/12/64				92.5 92.0	193 185	311 278	58.1 58.1	In orbit: reported to have completed series of maneuver tests during first day of orbit
Elektron 3	USSR	7/11/64				168.1	251 247	4,365 4,368	61 60.8	In orbit: second Elektron dual launch, research satellite to monitor radiation in inner Van Allen belt
Elektron 4	USSR	7/11/64				1313.8	285 379	41,076 41,078	61 60.1	In orbit: intended to make simultaneous radiation measurements in outer belt and magnetosphere
Zond 2	USSR	11/30/64								In solar orbit: Mars probe, tested plasma engine in flight
Molniya 1	USSR	4/23/65				720.5	308	23,800	65.19	
Lunik 5	USSR	5/9/65			3,500					Moon probe, failed to carry out soft landing, crashed on moon 6/12/65
Luna 6	USSR	6/8/65								Moon probe, missed moon

Table 1-5

APOGEE AND INCLINATION OF VARIOUS SPACECRAFT

Spacecraft	Launch Date	Apogee (sm)
Earth Orbiting Vehicles With Apogees < 600 sm and Inclinations < 70 deg		
Gemini-Titan 3	3/23/65	140
Gemini-Titan 4	6/3/65	158
Gemini-Titan 1	4/8/64	204
OSO 2	2/3/65	393
Pegasus 1	2/16/65	462
Pegasus 2	5/25/65	466
Tiros 8	12/21/63	473
San Marco 1	12/15/64	510
Greb 5	1/11/64	578
EGRS	1/11/64	578
GGSE	1/11/64	585
Earth Orbiting Vehicles With Apogees < 600 sm and Inclinations > 70 deg		
Nimbus 1	8/28/64	579
Surcal 1, 2	3/9/65	585
GGSE 2, 3	3/9/65	585
Greb 6	3/9/65	585
Secor 3	3/9/65	585
Oscar 3	3/9/65	585
Earth Orbiting Vehicles With Apogees Between 600 and 10,000 sm and Inclinations < 70 deg		
Explorer 23	11/6/64	610
Explorer 27	4/29/65	818
Ariel 2	3/27/64	843
Tiros 9	1/23/65	1,602
LES 1	2/11/65	1,744
Relay 2	1/21/64	4,606
LES 2	5/6/65	8,009

Table 1-5 (cont'd)

Spacecraft	Launch Date	Apogee (sm)
Earth Orbiting Vehicles With Apogees Between 600 and 10,000 sm and Inclinations >70 deg		
Explorer 20	8/25/64	634
Secor 2	3/11/65	642
Explorer 22	10/9/64	669
Echo 2	1/25/64	816
Explorer 19	12/19/63	1,487
Explorer 25	11/21/64	1,547
Explorer 24	11/21/64	1,551
Earth Orbiting Vehicles With Apogees Between 10,000 and 45,000 sm		
Explorer 26	12/21/64	16,280
Syncom 3	8/19/64	22,312
Early Bird 1	4/6/65	22,372
Earth Orbiting Vehicles With Apogees >45,000 sm		
Explorer 21	10/3/64	59,253
TRS 5	10/16/63	64,531
TRS 6	7/17/64	64,886
Vela 3	7/17/64	65,624
Vela 4	7/17/64	69,482
Vela 1	10/16/63	70,631
Vela 2	10/16/63	72,974
OGO 1	9/4/64	92,827
Explorer 18	11/26/63	122,522
Explorer 28	6/7/65	164,000
Lunar and Interplanetary Probe Vehicles		
Ranger 6	2/15/64	Lunar Probe
Ranger 7	7/28/64	Lunar Probe
Mariner 3	11/5/64	Mars Probe (failed)
Mariner 4	11/28/64	Mars Probe
Ranger 8	2/17/65	Lunar Probe
Ranger 9	3/21/65	Lunar Probe

1.3 MATERIALS USED IN SUCCESSFULLY ORBITED SPACECRAFT

Materials used in the construction of successfully launched satellites and spacecraft are tabulated in Tables 6 through 14, pp. S-30–S-52. The materials are arranged according to their class of functional application with separate tables covering thermal control materials, optical materials, materials for lubricated systems, adhesives, sealing materials, organic structural materials, materials for antennas, electronic materials, and inorganic structural materials. Within the tables, a further breakdown is made to show materials used in particular satellite and spacecraft vehicles. Because of the nature of the materials tabulation scheme followed in these tables, it has been deemed most practicable to group together all references used in the preparation of Tables 1-6–1-14; these sources are accordingly listed alphabetically by author in Sec. 1.4.

The accompanying paragraphs are grouped according to spacecraft project name and are intended to form, in effect, a discussion of Tables 6 through 14 giving the results obtained from the spacecraft and correlating its performance insofar as possible with the choice of materials made during its design and the behavior of these materials under simulated and actual space environment.

1.3.1 Alouette

This satellite, the first to be built in Canada, was designed to measure electron densities and their temporal and spatial variations in the Earth's ionosphere by means of a variable frequency radio probe which was periodically swept from 1.6 to 11.5 Mc. In terms of lifetime and reliability of performance, it must be counted as a remarkably successful effort. Although it was launched in September 1962 and thus may have been affected by the artificial electron belt set up by the Starfish high-altitude nuclear experiment of July 1962 and the subsequent high-altitude nuclear experiments of the USSR, Alouette performed thousands of measurements of the ionosphere and 2-1/4 yr after launching was reported to have suffered not a single failure in any of its 7000 parts (Ref. 5).

Perhaps the most interesting materials application aboard Alouette was that for the antennas used to receive reflected radio signals from the ionospheric layers. After injection into orbit, the antennas were deployed from a tape of spring steel which had been previously heat-treated, opened flat, and wound on a drum for storage. When this material was pulled off its drum and deployed into space, it formed into a tube with about 180 deg of overlap, forming a structure with considerable bending strength. On Alouette, four of these antennas were extended from the equator of the satellite to form a crossed dipole; the maximum extension was 75 ft. Much longer extensions than this are apparently possible and DeHaviland Aircraft of Canada, patent holder on this device, has designed antennas up to 1000 ft in length (Ref. 6). However, with a tube of even 75 ft in length and 0.85 in. in diameter, as in Alouette, the effect of solar heating of one side of the tube with consequent differential expansion and distortion of antenna alignment may become serious. On the Alouette, the antenna material selected was spring steel, but the current state-of-the-art choice would be beryllium copper alloy with a highly reflective silver plate coating (Ref. 6). Basically for high strength and good repeatability (of deployment), a material of low Young's modulus and highest possible yield strength is required; high conductivity is also required to minimize thermal distortion. The beryllium copper alloys seem to represent a reasonable compromise among these demands.

The power supply of Alouette utilized 6,480 p-on-n type silicon solar cells coupled with sealed nickel-cadmium batteries. Covers of 12-mil thick glass protected the solar cells. Power output from the solar cell array had been degraded by 20% in the first 11 days (Ref. 7), by 33% within 3 months (Ref. 8) and by 37% by the end of 10 months in orbit (Ref. 9). Part of this degradation could result from darkening of glass or adhesive cementing the glass to the silicon module. In the follow-on vehicles to Alouette, power will be obtained from the more radiation-resistant n-on-p type solar cells.

Several items may be mentioned in regard to correlation between simulation testing and performance in orbit. The solar cell degradation rate detailed above followed predictions made by NASA-Goddard scientists on the basis of previous testing and experience with p-on-n cell assemblies. Temperatures as measured on the instrumentation decks of the spacecraft followed closely the predicted values. In a 66% sun orbit the mean temperature was measured as 56° F as opposed to a prediction of 45° F; for the 100% sun orbit the spacecraft ran at 86° F as compared with a prediction of 84° F (Ref. 8).

Initial testing of a structural model of Alouette revealed a severe resonant response to vibration but the installation of eight struts between the equipment shelf and the thrust tube eliminated this problem. In a test of the antenna extension made after thermal-vacuum testing, it was found that prolonged exposure to vacuum had caused a nylon sleeve in the deployment mechanism to shrink. This shrinking combined with possible small clearances had resulted in the binding of the mechanism. The parts were modified to provide greater clearance; thus in this case the application of knowledge regarding space environmental effects on materials to the simulation situation resulted in a design change critical to the ultimate spacecraft mission. Another change made during the environmental testing program was to replace aluminum bushings with stainless steel bushings on certain aluminum shafts in the antenna drive train idler mechanism to prevent binding due to nylon gears running on aluminum shafts.

1.3.2 Ariel

The Ariel satellite program, which so far has seen two vehicles launched, is the first international cooperative space program. The shells of these vehicles were designed and built by NASA –Goddard; the vehicles carried various scientific experiments which were planned and instrumented by British scientists. Ariel 1 measured cosmic rays, solar radiation, and the structure of the ionosphere. Ariel 2 carried instruments to measure ozone concentrations in the upper atmosphere, micrometeoroid impacts, and galactic radio noise.

After launch on 26 April 1962, Ariel 1 performed successfully all its functions with the exception of a Lyman-alpha ultraviolet detector which was evidently damaged during launch. On 12 July 1962 just 3 days after the Starfish high-altitude nuclear tests of 9 July, Ariel began to malfunction. Periods of non-transmission and transmission only of the unmodulated carrier signal alternated with periods of good transmission of data. It was deduced that the undervoltage cutout system was in operation, pointing to a reduced power output from the solar cells (Ref. 10). The correlation between this performance and radiation damage to the solar cells by Starfish fission electrons trapped in the Earth's radiation belts seems clear. There is reasonable agreement between the observed performance of the power supply on Ariel 1 and the predicted degradation of 20% in output based on characteristics of the radiation and of the solar cells as protected by 12-mil glass covers (Ref. 10).

The solar cells of Ariel 1 were of the p-on-n type but Ariel 2 carries n-on-p cells in line with the current trend of switching most satellite power systems to these more radiation resistant elements.

Another early casualty was the tape recorder of Ariel 1 which stopped permanently about 1 August 1962 (Ref. 11). However, this failure may not necessarily be related to a materials problem or to radiation damage since the tape recorder performed so erratically, with many failures such as electrical shorting and loose belts during both prototype and flight acceptance environmental testing. This behavior made it advisable to include a current-sensing overload relay in the tape recorder power circuit (Ref. 12).

Environmental testing results are also correlated with flight performance in the case of the electron density experiment. During thermal-vacuum ground tests, this experiment malfunctioned because of low temperature attained by its mounting boom. To remedy this problem the boom was subsequently coated with evaporated gold and black paint to maintain the temperature of the boom electronics between 11 and 51° C during orbital flight. The electron density experiment did function until March 1963 and data were obtained by direct transmission although $\sim 75\%$ of it was lost due to the failure of the tape recorder low-speed data store. The average satellite temperature ranged from 20 to 50° C depending on percent solar irradiation of the vehicle.

Despite its difficulties with radiation damage, Ariel 1 transmitted some data until as late as November 1964.

The second Ariel satellite, launched 27 March 1964 into a slightly higher orbit than Ariel 1, has successfully transmitted data from all its experiments since launch.

1.3.3 Courier

Representing one of the earliest of the communications satellites, Courier was launched 4 October 1960 and performed real-time communications tests for 17 days. Its beacon transmitter, powered by a solar-cell power supply, functioned until 16 September 1962 with no degradation in receiving signal strength until it stopped completely on that date, apparently because of loss of solar charging capability or failure of the acquisition transmitter (Ref. 13).

The passive temperature control performed well within design limits and, indeed, within a few degrees of the estimated values. Temperatures after 17 days in orbit were reported in Ref. 4.

1.3.4 Early Bird

The Early Bird satellite, although not the first privately developed spacecraft since that designation must go to Telstar, may truly be characterized as the first commercial satellite since it now has available communications channels for rent. Launched on 6 April 1965 by NASA for the Communications Satellite Corp., Early Bird was successfully maneuvered into a synchronous orbit and, as of 27 June 1965, has been approved for commercial operations (Ref. 14). One of its earliest jobs has been the transmission across the Atlantic of Tiros 10 weather maps at significantly improved resolution and speed (Ref. 15).

These accomplishments in themselves indicate a high degree of reliability in the space environment of the materials and designs used in the Early Bird satellite. Unfortunately, little more concrete information is available on these materials. In general, materials selection may be presumed to have followed rather closely that used for the Syncom series of satellites since several components such as the hydrogen peroxide attitude control system and the Jet Propulsion Laboratory solid-propellant apogee motor are the same as for Syncom (Ref. 16). Early Bird carries 6000 n-on-p solar cells to provide a maximum power of 45 W. Since its communications operations will require only about 27 W, the extra margin should contribute to long life for this satellite.

1.3.5 Echo

The performance of Echo 1 has continued to surprise space scientists. Originally designed for a useful life of about 2 weeks, the satellite, a 100-ft diameter aluminized Mylar balloon, is still in orbit after almost 5 yr and is providing long-term data on solar pressure and air drag effects on its orbit as well as on the effects of the space environment on its construction materials.

The long-term orbital behavior of Echo 1 and its rocket casing has been particularly watched because Echo 1 is the first satellite to be appreciably influenced by solar pressure. The rocket casing also in orbit is relatively unaffected and furnishes a "control" orbit. Data compiled for the first 500 days of the life of Echo 1 and reported in Ref. 17 indicate that the shape of the orbit has been changing, with the eccentricity varying between 0 and 0.08. The changes in the parameters which define the orbit appear to be periodic with similar conditions recurring at approximately 310 to 340 days. It also appears that the orientation of the satellite orbit with respect to the sun is a factor in the changes in eccentricity and energy.

In another study of the light reflected from Echo 1 and reported in Ref. 18, some interesting correlations were drawn between exposure of materials to the space environment and the expected degradation of them. This study showed, by means of visual, photographic, and photoelectric observation of the satellite, that its original property of specular reflectivity had been degraded very little by the effects of space environment. In 1963, when the data were taken, the satellite was almost 4 yr in orbit and still showed approximately 96% specular reflection. In addition the study showed that the mean radius of curvature of the sphere had remained near the design value, although locally varying, and that its total reflection coefficient was near the value at the time of launch. These results were interpreted to indicate that the space environment had not appreciably removed or modified the reflectivity of the vapor-deposited aluminum coating of Echo 1 and that therefore the space environment does not degrade aluminized Mylar as rapidly as originally postulated. Also the orbital forces, such as solar and meteoroid pressure, have not appreciably affected the satellite's overall geometry.

The environment, to which Echo is subjected, is near the maximum of dosage from the trapped radiation belts, since its orbit is roughly circular and at an altitude of 900 to 1000 sm. (Initial parameters were perigee – 941 sm, apogee – 1052 sm, and the inclination to the equator was 47.2 deg.)

The Echo 2 satellite differed from Echo 1 in several important respects. Echo 1 was primarily planned as a passive communications satellite and was deployed simply by blowing up a balloon in space with enough gas to hold it in shape for ~ 2 weeks during which measurements would be made. A rather unexpected result was that even after the inflation gas had presumably leaked out, the satellite still held its shape reasonably well. For Echo 2, a sufficient quantity of subliming material was included so that the resultant gas would stress the inflated balloon past its yield point and effect rigidization. Calculations by Fichter and associates (Ref. 19) at Langley indicate that the buckling pressure of the satellite under these conditions would be at least 142 times the solar radiation pressure, which is expected to be the largest deforming factor, and thus the satellite could be expected not to buckle unless its orbit brings it too far into the atmosphere so that dynamic pressure comes into play. The primary purpose of the Echo 2 was to serve as a test of this concept of deployment of rigid inflatable structures in space, although it too could and did serve as a passive communications satellite also.

Echo 2 was launched on 25 January 1964 and successfully placed into an orbit of 816-sm apogee and 642-sm perigee. Some question seems to remain as to whether the sphere was fully inflated. However, good experimental results have been obtained and communications tests with the satellite continue.

1.3.6 Explorer

Satellites carrying the designation of Explorer include a variety of vehicles of diverse shapes and weights and many different missions. In general, the name has been applied to scientific data gathering satellites developed and built by either Goddard Space Flight Center or Langley Research Center (some of the earlier Explorers were built by the Jet Propulsion Laboratory and by Space Technology Laboratories) and which have as their mission the determination of more information about the space environment.

The first successfully orbited U. S. satellite was Explorer 1 on 31 January 1958; the latest of the series, Explorer 28, was orbited 29 May 1965. Most of the Explorers launched are still in orbit. Information on Explorers 1 through 17 is given in Ref. 4. Tables 6 through 14 of this supplement include materials information on some of these vehicles which was not available previously in addition to the coverage of later spacecraft.

Explorers launched within the last 3 yr may be broadly grouped into about six categories:

- Atmospheric Density Explorers or small balloon satellites including Explorers 9, 19, and 24
- Atmospheric Structure Explorers or sealed steel sphere satellites including so far only Explorer 17
- Energetic Particles Explorers including Explorers 12, 14, 15, and 26
- IMP series, closely related to the EPE vehicles and including Explorers 18, 21, and 28
- Ionospheric and geodetic satellites including both the Fixed Frequency Topside Sounder, Explorer 20, and the Beacon Explorers 22 and 27
- Meteoroid detection satellites including Explorers 13, 16, and 23

Several other satellite programs are closely related to Explorer. The Pegasus meteoroid detector satellites are the successors to Explorers 13, 16, and 23 in purpose and technology. The Canadian Alouette satellites and the British Ariel as well as the Italian San Marco are also related to some of the Explorer programs.

Atmospheric Density Explorers. Satellites of this series are inflatable balloon structures of 12-ft diameter constructed of a four layer laminate of two 1/2-mil layers of vapor-deposited aluminum and two 1/2-mil layers of Mylar cemented together with GT-301 adhesive.* Use of Mylar for these satellites is based on its successful use in the Echo satellites as well as its low rate of degradation in the space environment as shown by laboratory testing (Ref. 20).

These satellites are constructed as two hemispheres with a dielectric gap between them. On Explorer 9 this gap (~1 in.) was also of Mylar, but on Explorers 19 and 24 it was changed to H-film because of a discharge phenomena noted in Mylar due to ionizing radiation (Ref. 21).

The thermal control coatings on these satellites furnish another example of design changes made on the basis of environmental experience in materials degradation. Thermal control is effected by means of a system of white dots spotted on the aluminum outer surface. The dots, amounting to ~ 17% of the surface area, were of epoxy paint for Explorer 9, but for Explorers 19 and 24 were changed to the less radiation – sensitive silicone rubber LTV-602.

*Manufactured by G. T. Schjeldahl Company.

Simulated environmental testing of both the epoxy and silicone paints (Ref. 22) had shown that the white titanium dioxide pigmented epoxy paints was not stable for long duration use inasmuch as the solar absorptance increased 70% when exposed to an equivalent 1000 hr of simulated solar UV radiation. On the other hand the silicone paint is reasonably stable with an increase of 6 to 15% of the initial value of solar absorptance when exposed to an equivalent 1000 hr of simulated solar UV radiation.

Explorer 9 was successfully tracked by optical observation to correlate upper atmosphere density with solar activity. It was originally estimated to have a lifetime of ~ 1 yr but decayed after ~ 3 yr (Ref. 23). The other two small balloon satellites, Explorers 19 and 24, are still in orbit and being tracked.

Explorer 9 carried a radio beacon for tracking purposes but this failed within a few hours after successful injection into orbit and before the inflation gas leaked out of the satellite. The most probable cause of the transmitter failure is believed to be excessive voltage applied to it, with consequent destruction of the transistors. This failure has been linked to space environmental effects through the following conjectures (Ref. 24). The temperature of the satellite skin and the radio beacon became very low when the satellite was in the Earth's shadow. Then on entry of the satellite into sunlight, higher than normal voltage from the cold solar cells and higher internal resistance of the cold batteries caused excessive voltage to be applied to the transmitter.

As a result of the Explorer 9 transmitter failure, the beacon transmitter on Explorer 19 was modified by the incorporation of additional current and voltage regulators and by thermal isolation of the transmitter unit and battery-control unit (Ref. 25).

A detailed comparison has been made of ground tests and orbital launch results for the Explorers 9 and 19. The two flight tests have demonstrated that, in general, the in-flight operation of the ejection and inflation system was similar to that experienced during environmental tests and that a lightweight delicate structure can be erected in space after withstanding the severe conditions imposed by an orbiting spin-stabilized launch vehicle (Ref. 25).

Atmospheric Structure Explorers. So far this series includes only Explorer 17 although a second spacecraft is planned for launch within the near future. Since this Explorer was designed to measure the physical parameters and chemistry of the atmosphere prevailing at a few hundred miles altitude, the entire spacecraft was carefully sealed to prevent contamination of the local region by materials evolved from the vehicle. For this reason, not many space environmental effects on materials are expected to be of interest here. The spacecraft was constructed as a sphere of AISI 321 stainless steel sealed with OFHC copper gaskets. The mass spectrometers, vacuum gages, and electrostatic probes of the scientific payload were likewise sealed with copper, gold, or glass-to-metal seals. The power supply for Explorer 17 was a primary battery system inside the sphere; solar cells were not used because of their unknown gas evolution characteristics (Ref. 26).

Data on the structure of the atmosphere were successfully telemetered back with no failures in the spacecraft during the 100 days prior to the loss of reception due to the anticipated exhaustion of the battery (Ref. 26).

Energetic Particles Explorers. This series includes Explorers 12, 14, 15, and 26, all of which carried instrumentation to measure concentrations, energies, and directions of electrons, protons, and ions in the regions of space between a few hundred and several thousand miles above Earth. These spacecraft also measured magnetic fields and carried solar cell damage experiments.

Explorer 12, launched 15 August 1961 carried p-on-n type solar cells, both bare and with various thicknesses covers. During the first 3 days in orbit, the bare cells were degraded 50% in power output; no degradation was observed for cells covered with 3-mil or with 20-mil glass covers (Ref. 27). Explorer 14 also carried a solar cell experiment which after about 8 days in orbit showed 70% degradation of unshielded p-on-n cells and 40% degradation of unshielded n-on-p cells. Cells of both types protected by a 3-mil thick glass cover degraded by 10% (Ref. 28).

Explorer 15 was launched 27 October 1962 to investigate the enhanced radiation situation following the Starfish nuclear explosion. Its orbit of 194 sm perigee and 10,760 sm apogee placed the satellite in the heart of the newly created radiation belt. Radiation damage to a component in the electronic system of the satellite caused failure of telemetry after 95 days in orbit (Ref. 29). This deduction is based on simulated environmental tests of electronic components exposed to radiation during which a number of parts that were used in Explorer 15 failed (Ref. 29).

Explorers 12, 14, and 15 also made use of silicone paints (black, white, and leafing aluminum) formulated by the Naval Research Laboratory (NRL) and manufactured by Andrew Brown Co. Because of problems with paint adhesion on Explorer 15, this practice was changed on Explorer 18 to Cat-a-lac black and to white epoxy undercoat with the NRL silicone white for white surfaces. Due to the indicated high degradation rate of the NRL white paint, a methyl silicone paint with TiO_2 pigment was used on Ariel 2, Explorers 21, 23, and 26. These spacecraft also used Cat-a-lac black epoxy and some NRL leafing aluminum paint. Small areas on all these satellites have been covered with aluminized tape or foil tape.

Interplanetary Monitoring Platforms. The IMP series of spacecraft, including Explorers 18, 21, and 28 are designed to make measurements of cosmic radiation, magnetic fields, and solar wind out to distances of 100,000 to 200,000 sm from Earth. The IMPs are closely related to the EPE spacecraft and share with them the same basic octagonal shape of the main spacecraft body.

The first IMP, Explorer 18, was launched 26 November 1963 and all electronic and mechanical systems in the spacecraft operated in a near perfect condition through the early part of 1964 (Ref. 29) but about June 1964 its signal became too weak to be picked up by NASA tracking stations (Ref. 30). However, beginning 17 Sept 1964, the spacecraft again started transmitting useful data. It is considered that a more favorable sun angle contributing to more power from its solar panels is responsible for this rejuvenation (Ref. 31). Explorer 21, the second in this series, was launched successfully on 3 October 1964 but failed to make the highly eccentric orbit planned for it. This failure is attributed to a lack of sufficient thrust in the last stage of the Delta launch vehicle (Ref. 32). Explorer 28 was successfully orbited 7 June 1965.

Ionospheric and Geodetic Explorers. The Explorer 20 spacecraft is closely related in mission to the Canadian Alouette satellite, since it is designed to probe the ionosphere by radiofrequency pulses reflected back to the satellite. Explorer 20 used a set of fixed frequencies for this purpose whereas Alouette used a continuously variable frequency probe.

The long antennas used to receive the reflected signals from the ionosphere were made of beryllium-copper alloy (typically alloy 25) in Explorer 20 but made use of the same extension principle described in Sec. 1.3.1 for Alouette (Ref. 33).

Explorers 22 and 27, also called Beacon Explorers, are intended to provide data on the ionosphere by transmitting signals through it to a network of ground stations in thirty-two countries. A second purpose of these spacecraft was to conduct laser and Doppler-shift geodetic tracking experiments.

To accomplish this purpose, a passive optical laser reflector was mounted on the forward face of the satellite. This reflector was a mosaic built up of forty corner cube prisms of fused silica. A ground-based laser source tracking the satellite transmitted pulses of light to it which were reflected from the corner cube prisms back to a detector near the source. Laser tracking was first accomplished 11 October 1964 on Explorer 22, two days after its launch on 9 October, although the returned signal was weaker than expected.

Meteoroid Detection Explorers. This series of Explorer spacecraft includes Explorers 13, 16, and 23. The type meteoroid detectors used in this group of satellites has evolved from the resistance measuring card type through one-shot puncture cans to the more versatile capacitor type detectors which have now been applied to large areas on the Pegasus vehicles.

Explorer 13 was launched 25 August 1961 but due to an error caused by the Scout launch vehicle it achieved a very low perigee and thus remained in orbit only for 2.5 days instead of the designed 1 yr. The spacecraft carried six types of meteoroid detectors. The primary sensors were pressurized cells of beryllium-copper (typically alloy 25). The design of these cells limited detection by the loss of pressurization to only one meteoroid hit per cell. A total of 160 of these cells were carried in thicknesses of 1-mil, 1.5-mil, 2-mil, 2.5-mil, and 5-mil. Other sensors included two types of resistance detectors, two types of piezoelectric impact detectors, and cadmium-sulfide detectors for very small particles (Ref. 34). During launch an aluminized Mylar diaphragm in the cadmium-sulfide detectors was ruptured rendering it inoperative; this was presumably due to a poor design for venting the sensitive chamber during ascent. During the flight, no meteoroid hits were recorded by the resistance detectors and the pressurized cell detectors (Refs. 34 and 35). A number of events were recorded by the impact detectors; however, correlating with other experiments and taking into account the low perigee of the spacecraft, the experimenters deduced that the events were probably not meteoroid hits and therefore the experiment was inconclusive (Ref. 34).

Correlation of the temperature data recorded during flight of Explorer 13 with laboratory testing and predicted temperatures shows that surface temperatures, pressurized cell temperatures, and telemeter temperatures were higher than expected. These high flight temperatures, although still within the prescribed limits for operation, have been ascribed to the existence of free-molecular flow heating at the low perigee (Ref. 34). This heating source would not have been present had the Explorer 13 orbit been as expected.

Fused quartz covers 1/16-in. thick were used to protect Explorer 13 solar cells from possible damage by space radiation and micrometeoroids. This material was selected on the bais of simulated testing in which it was found to transmit 93% in the spectral range of the solar cells. Sandblasting the outer surface of the quartz covers to simulate micrometeoroid impingement was found to reduce this figure to only 89%. Proton irradiation tests showed that the quartz should not be appreciably darkened in the expected orbit (Ref. 34). No degradation of the power supply was noted during the short lifetime of the satellite.

Explorer 16 was launched 16 December 1962 carrying the same complement of meteoroid detectors as did Explorer 13 (Ref. 36); it stopped transmitting data 22 July 1963 and is still in orbit. The basis of this failure has not been determined; however, the telemetry data indicate that the power supply system was working properly up to the time of failure and that the failure cannot be attributed to the command receivers, the solar cells, the nickel-cadmium batteries, or the transmitters (Ref. 37).

The effect of the space meteoroid environment on the detector materials has been summarized in a series of progress reports on Explorer 16 (Refs. 37 through 40). Through the date of telemetry failure, punctures were recorded in forty-four of the one hundred 1-mil thick beryllium-copper pressurized cells and in eleven of the forty 2-mil thick cells. Corresponding puncture rates are 0.031 and 0.016 punctures/ft^2-day, respectively. No punctures were measured through the 5-mil cells. These data have been reduced to equivalent puncture of aluminum and compared in Fig. 1-1 (Ref. 37) with theoretical curves derived by several workers and taken from Ref. 41.

The copper-wire resistance type detectors registered one break of a 3-mil wire and one break of a 2-mil wire but no further interpretation of these data has been issued. Preliminary data through 13 January 1963 (Ref. 38) indicated three punctures of the cadmium cell which is designed to measure particles down to 25 μ in diameter. Again no further interpretation has been made.

Results from the impact detectors of Explorer 16 show a wide discrepancy of several orders of magnitude from the results of the pressurized cell experiments (Ref. 37). The calibration of impact detectors suffers from a lack of complete ground simulation correlation with behavior in space, whereas the penetrating experiments give a simple direct indication of materials damage.

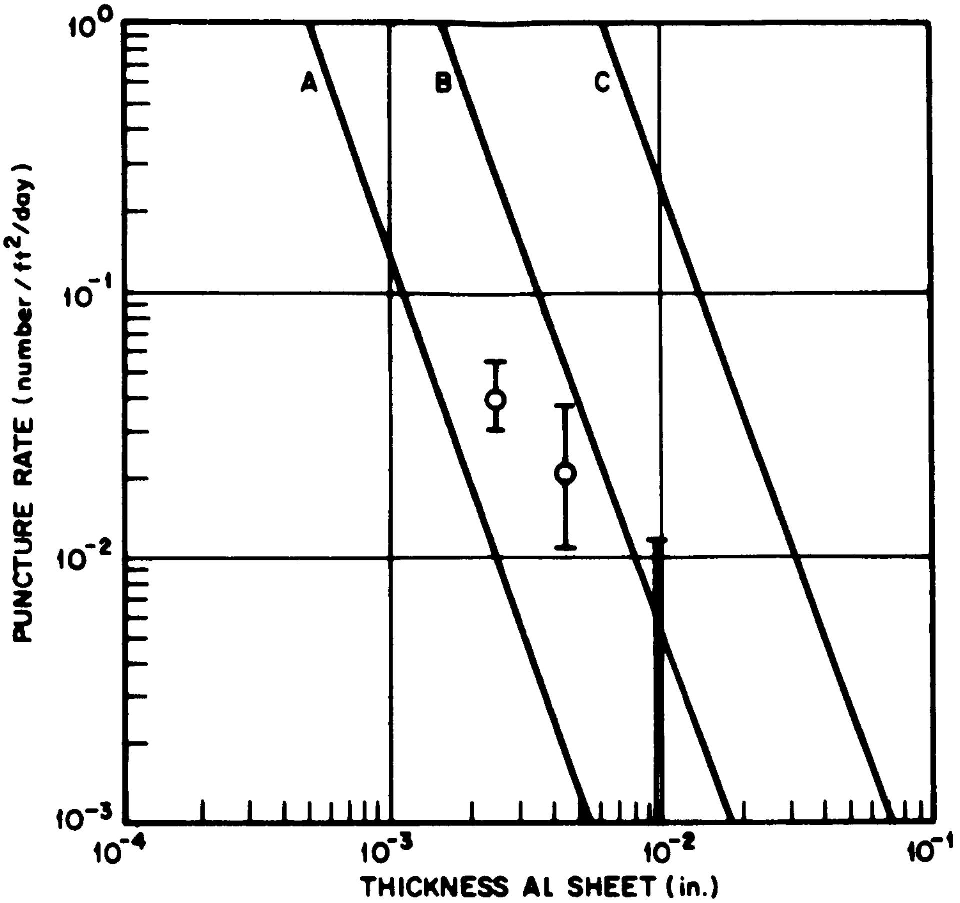

Fig. 1-1 Most probable rate of puncture of aluminum skin as a function of skin thickness showing experimental points from Explorer 16 pressurized cells experiment. Bars on data points represent 95% confidence levels. Curves are derived as follows:
A– Application of Bjork penetration theory to Watson (1956) flux
B– Application of Bjork penetration theory to Whipple (1961) flux
C– Application of Charters and Summers penetration theory to Whipple (1957) flux

The remaining meteoroid detector was that which measured penetrations through stainless steel sheets by means of rupture of a gold foil resistance network lying directly beneath the steel. An area of 1.5 ft^2 of 1-mil thick AISI 304 stainless steel was punctured six times in 221 days of the satellite life while a 2-ft^2 area of 2-mil thick steel experienced one penetration in the same time and 0.25 ft^2 of the same material 6-mil thick was not punctured at all.

A considerable effort was put into developing radiation shielding for the solar cells of Explorer 16 to insure adequate life in the artificial electron belt where it was to operate. Cells were irradiated in the laboratory with 1.2 MeV electrons bare, with 1/16-in. thick quartz covers and with 3/16-in. thick quartz covers. On the basis of correlating the observed laboratory damage with the known intensity of the trapped radiation belts, the 3/16-in. covers were used in the spacecraft. This choice was vindicated by the performance of a group of test cells which were flown on Explorer 16. After 100 days in orbit a group of unprotected solar cells showed a degradation of ~30% of power compared with a predicted value of 40%. Another test group, protected with 3/16-in. thick quartz covers in the same manner as the power cells, showed a degradation of ~20% compared with a predicted value of 23%. The lower degradation than predicted is partly explained by the decay of the electron trapped belt between the time simulation tests were run and the satellite experience (Ref. 42).

A surprising feature noted during the above described simulation experiments was the great difference in the darkening of four samples of quartz windows under irradiation. Each had a different trade name but all were manufactured by fusing crystalline quartz. For a dose of about 2×10^{15} electrons/cm^2 of 1.2 MeV, vapor-deposited quartz showed no loss in transmission while the various fused quartzes were degraded to levels of transmission of from 60 to 95% of their original value.

Explorer 23, the next of this series of spacecraft, was launched successfully on 6 November 1964. It was similar in configuration to the previous members of the series but with some changes in the meteoroid detectors. The pressurized cells were made of stainless steel, a larger number of aluminum impact detectors were added, and the spacecraft also carried two of the capacitor-type detectors. Meteoroid strikes on these capacitor detectors were registered when the particle ionized the material of the dielectric in the capacitor sheet thus allowing it to discharge and form an electrical pulse in the telemetered signal.

Capacitor detectors are not quickly used up as are the pressurized cell type detectors but do have the disadvantage that the capacitors tend to accumulate electrons when exposed to high radiation levels and these electrons can cause capacitor shorts and millisecond current pulses. Ground tests with this type capacitor panel in a simulated radiation environment showed that an accumulation of electrons on the panels caused current pulses very similar to those anticipated from meteoroid strikes (Ref. 43).

No information on the meteoroid punctures experienced by Explorer 23 has been published by NASA.

Table 1-6

THERMAL CONTROL MATERIALS

Alouette. Several surfaces were used to achieve the desired passive thermal control including optically coated glass-covered solar cells, anodized aluminum (α/ϵ = 0.95), black epoxy-based FeO enamel (α/ϵ = 1.0), ultraviolet resistant white polyurethane-based TiO_2 enamel (α/ϵ = 0.2), and polished aluminum (α/ϵ = 2.5).

Ariel 1. Evaporated-gold coating over varnish, lacquer, conducting silver paint, electroplated copper, and lacquer substrates applied in that order. About 25% of the surface area was covered with a combination of silicone paints pigmented with either carbon black or zinc sulfide (white).

Courier 1B. TiO_2-pigmented paint

Echo 2. Outer layer of balloon – Alodine 401-45, an amorphous compound consisting mainly of chromium and aluminum phosphates, of thickness 0.02 mil
Inner layer of balloon – carbon black of thickness 0.02 mil

Explorer 10. Evaporated aluminum and aluminum paint designed to have an α/ϵ of 1.6 and to hold the magnetometer sphere at 40°C in a highly eccentric orbit

Explorers 9, 19, 24. (Atmospheric Density Explorers). Passive thermal control accomplished by means of a system of white dots spotted on the aluminum outer surface of the balloon. The dots made up about 17% of the surface and consisted of an epoxy (epichlorohydrin bisphenol-4) paint pigmented with titanium dioxide for Explorer 9 and a methyl silicone elastomer pigmented with zinc oxide for the Explorers 19 and 24.

Explorer 17 (Atmospheric Structure Explorer). Vapor-deposited aluminum covered with silicon oxide coating

Explorers 12, 14, 15, and 26 – Energetic Particles Explorers (EPE). Coated first with lacquer to get a mirror finish, then finished with vapor-deposited aluminum and silicon oxide coating. Other parts of the spacecraft covered with silicone paints (black, white, and leafing aluminum pigmented) formulated by NRL and manufactured by Andrew Brown Co. Small areas were covered with aluminized tape or foil tape.

Explorers 18, 21, and 28 – Interplanetary Monitoring Platform (IMP). Some parts of Explorers 18 and 21 used the lacquer, deposited aluminum, and silicon oxide finish as described for the EPE vehicles. Because of problems with paint adhesion on Explorer 15, some painted surfaces on Explorer 18 onward were changed to Cat-a-lac black epoxy and a white epoxy undercoat. Due to the indicated high degradation rate of the NRL white paint, a methyl silicone paint with titanium dioxide pigment was used on Explorers 21 and 28 as well as on Explorer 26 of the EPE series. The Cat-a-lac black epoxy and the NRL leafing aluminum paint were also used on Explorers 21 and 28. Small areas were covered with aluminized or foil tape. Active thermal control, heaters in the magnetometer package, was also used.

Gemini. Adapter section – silicate paint pigmented with zirconium dioxide on outside, Lockspray gold over epoxy paint on the inside
Other interior paints – Andrew Brown Co. XA silicone paints
Instrument coolant – Monsanto silicate ester MCS-198

LES (Lincoln Experimental Satellite). Outer panels, where not supporting solar cells, were covered with glass microsheet backed with aluminum. Interior surfaces were painted black.

Mariner 2. Exposed wiring wrapped with Teflon-coated aluminum foil.

Mariner 3 and 4. Surfaces of some electronic boxes in main bus were covered with 12-mil polished aluminum shielding. Five of the eight boxes had aluminum foil louvers over a surface of TiO_2 white paint for active thermal control. Some of the shields contain cutouts which allow heat to be radiated from local "hot spots." Upper and lower surfaces of bus were shielded with 30 layers of crinkled aluminized Mylar. The outer and inner layers of each multilayer were covered with either black Dacron, if the shield is in the sunlight, or aluminized Teflon. Cable harnesses were wrapped with aluminized polyvinylfluoride. Some exposed wiring was wrapped with aluminized Teflon.

Nimbus 1. Two multilayer assemblies insulate the spacecraft control subsystem. One consists of 11 layers of 1/4-mil thick crinkled aluminum foil separated by 10 layers of 4-mil thick matted glass fiber paper. The assembled stack is covered on both sides by an additional layer of glass fiber paper and is enclosed with a 2-mil thick woven glass fiber fabric which is impregnated with Teflon. The second assembly consists of 34 layers of 1/4-mil aluminized Mylar. Teflon-impregnated glass fiber fabric is also used as an external cover for this assembly.
Solar power paddles – front side, solar cells, $\alpha = 0.773$, $\epsilon = 0.942$; back side PV 100 or white epoxy, $\alpha = 0.2$ and $\epsilon = 0.88$

Faying surfaces of components and panels were coated with Eccobond 56C for more efficient conductive heat transfer.

Sensory assembly crossbeam structure on which the camera system components are mounted was painted with absorptive flat black paint to achieve an adiabatic structure.

Vita Par PV 100 silicone was used on the outboard faces of compartments and on the bottom surfaces of batteries.

Aluminum-pigmented silicone alkyd paint (D4D) was used on the quadraloop antenna, separation ring, struts, lower section of command antenna, outer surface of insulation on three compartments and on the insulation blanket sheet.

Amfab (TV 20-60, a Teflon-impregnated glass fabric manufactured by AM&F) was used as the outer coating of the top insulation and on the control system surfaces.

In addition to passive control, Nimbus also incorporated active control in the form of louvers constructed of multilayer aluminized Mylar. They are actuated by a stainless steel AISI 302 bellows and provide control for the vidicon electronics and the attitude control servomechanisms. When open, the louvers uncovered the magnesium alloy container cover which was painted white. The expansion of Freon 114 in the vidicon thermal control mechanism and Freon 11 in the attitude control instrument system thermally actuated the louver mechanisms.

The interior of the structure was black anodized over aluminum or black painted over magnesium with a GE proprietary carbon black pigmented formulation ($\alpha/\epsilon = 0.94$) where high temperature stability was required or with a Lowe Bros. glycerol phthalate black in other less sensitive locations.

The truss structure was wrapped with aluminized Mylar and aluminized Teflon tape, as were the electrical cable harnesses.

OGO. Back surfaces of solar panels – ZrO_2 pigmented potassium silicate inorganic paint

Except for the 2 surfaces covered with louvers and the solar panels, multiple layer aluminized Mylar is used. Five to ten layers of 1/2 mil material is used plus an inner and outer layer of 3 mils each. Most of the insulation is applied by ultrasonic welding.

Heat sink – lithium filled, limits transistor temperatures.

OSO-1. Coating experiment – TiO_2 in epoxy, TiO_2 in silicone, white fused porcelain enamel, aluminum powder in silicone, Al-SiO-Ge films, and Al-SiO-Ge – SiO films were tested.

OSO-2. Paints – a base of phenylmethyl silicone resin with the white version pigmented with TiO_2
Outer rim of wheel – leafing aluminum
Top and bottom of wheel – polished aluminum
Internal parts – black anodized
Heat sink – Coors BeO for power transistors

Ranger 6 through 9. External – completely passive thermal control system used; black Cat-a-lac paint used in all surfaces where sunlight might be reflected to other parts of the spacecraft; polished surfaces used for other surfaces to reflect sunlight away from the spacecraft.
External – aluminized Mylar used for wrapping cable harnesses.
Electronics boxes – white TiO_2-pigmented silicone paints used on exteriors of some boxes; also gold plating used on exteriors and interiors of some boxes.

Syncom 1, 2, and 3. Internal surfaces – black epoxy paint or vapor deposited aluminum
Electronics packages – gold plated
Traveling wave tube – finished white, internal portions encapsulated in filled polyurethane for heat dissipation; silver filled epoxy between electronics packages and bulkheads for better heat conduction.

Pegasus 1 and 2. Back of solar panels – PV-100 white paint
Saturn S-IV stage – S-13 paint
Electronics boxes – PV-100, S-13, and aluminum pigmented silicone (Fuller 172-A-1) paints
Tubing near electronics boxes – wrapped with aluminized Mylar tape
Louvers on bottom of electronics canister – polished aluminum
Other miscellaneous surfaces – Lowe Bros. black paint
Aluminum surfaces of meteoroid detectors – adaptation of chromium aluminum phosphate coating used on the Echo. The change involved addition of phosphoric acid in processing. Fairchild-Hiller designation is MTL-3.
Canister area – superinsulation (aluminized Mylar) except for the bottom which is louvered.

Telstar 1 and 2. Outer skin – about 60% total area covered with a plasma sprayed aluminum oxide coating.

Miscellaneous Satellites

Oscar 3. External surface – non-leafing aluminum paint on Mystic tape

Nav Sat 2. Equipment section insulation – alternate layers of aluminum foil and fiberglass filter paper

TRS (Tetrahedral Research Satellite). External – Cat-a-lac flat black epoxy paint for high-altitude polar orbits. Silver Chromatone paint for low-altitude polar orbits
Internal – solid state commutator mounted on a boron nitride block to maintain thermal stability of commutator.

Table 1-7

OPTICAL MATERIALS

Alouette. Solar cell covers – 12-mil thick glass

Ariel 2. Solar cell covers – 12-mil optical microslides with UV filter on one surface and antireflecting coating peaked at 0.7 μ on the other surface

Explorer 19. Solar cell covers – 1/8-in. thick quartz glass (selected since this satellite was placed in polar orbit and would pass through the radiation belts)

Explorer 12. Solar cell covers – some cells flown bare and others with glass covers of various thicknesses 3 mil, 6 mil, and 20 mil

Explorer 18. Solar cell covers – 12-mil thick glass

Explorer 22. Laser reflectors – 360 1-in. corner cube prisms of fused silica

Explorer 16. Solar cell covers – one group of cells flown bare and others with 6-mil glass covers and with 3/16-in. fused silica covers (see p. S-29 for results)

Explorer 23. Solar cell covers – some cells bare, others covered with 3/16-in. thick quartz

Gemini. Viewing windows – Vycor panes, unsealed; aluminum silicate glass, sealed; MgF_2 coating on window surfaces
Antenna windows – Vycor, fused silica

Mariner 3 and 4. TV camera telescope – an f/8 Cassegrainian system with beryllium primary and secondary mirrors; primary mirror is 1.62 in. in diameter. Canopus tracker assembly – lenses made of highest grades of synthetic quartz and light flint glass. Edges of lenses were polished and painted with Cat-a-lac black paint to cut down stray light. Assembly incorporates a baffle made of black-anodized aluminum coated with 3M Black Velvet paint and fabricated with sharp knife edges (1-mil radii).

Nimbus 1. The horizon scanner of the Nimbus control subsystem contains a filter window designed to limit energy transmission to the wavelength band of 13 to 18 μ. The window consists of a plate of germanium coated on both sides with 45 to 50 alternating layers of vapor-deposited zinc sulfide and germanium. Individual filter coating layer thicknesses range between 50 and 1000 Å. The windows are manufactured by the Optical Coatings Laboratory, Inc.
IR sensor housing window – single germanium crystal
Prism wedge – single crystal germanium, 300 HM-CM
Sun sensor window – Corning fused silica No. 7940
Solar cell covers – 6-mil Corning type 7940 silica with UV filter
Camera lenses – conventional optical glass; protected by cover plates of fused silica

OGO. Solar cell covers – coated 6-mil-thick Corning 0211 glass

OSO-1. Solar sensor lenses – made from borosilicate crown glass
Solar cell covers – 6-mil glass

OSO-2. Solar sensor lenses – cerium glass with coarse sensors further protected by a cerium glass filter.
Solar cell covers – 20-mil Spectrolab quartz cover

Pegasus 1 and 2. Solar cell covers – 60-mil-thick fused silica with polished edges and antireflection and UV blocking coatings
Solar sensor – fused quartz block backed by a special pattern of solar cells (manufactured by Adcole Corp.)
Earth sensor – bismuth-antimony thermopile in nitrogen atmosphere behind germanium lens coated with Optical Coatings Laboratory, Inc. interference filter (proprietary).

Ranger 6 through 9. Sun sensors – cadmium sulphide
Earth sensors – three end-on photomultiplier tubes

Soviet Spacecraft (Vostok 3 and 4). Windows – heat-resistant glass with protective blinds which are activated mechanically or electrically.

Syncom 2. Solar cell covers – 6-mil Corning 0211 microsheet

Syncom 3. Solar cell covers – 12-mil Corning 7940 fused silica quartz

Tiros. Medium resolution scanning radiometer: thermistor bolometer detectors. Various combinations of lenses and filters were used to define the wavelength regions to be observed in each of the five channels. They are as follows:
Channel 1 – germanium lenses coated with ZnS and a multilayer filter incorporating Al_2O_3, germanium, and SiO.
Channel 2 – ZnS-coated germanium lenses, filters of InSb and As_2S_3 glass.
Channel 3 – a barium fluoride lens and sapphire lens, no filters
Channel 4 – not used
Channel 5 – one quartz lens and a sapphire lens; filters made by Infra-Red Industries of three types: Type 259, 011, 513; Type 259, 011, 608; and Type ON-20.

Miscellaneous Satellites

Oscar 3. Solar cell covers – 0.020-in. fused silica glass with blue coating

TRS (Tetrahedral Research Satellite). Solar cell covers – 0.029-in. Corning 7940 quartz with blue filter for satellite power; test solar cells bare and with 20- and 40-mil Corning 7940 covers.

Table 1-8

MATERIALS FOR LUBRICATED SYSTEMS

Alouette. Stainless steel bushings replaced aluminum bushings on certain aluminum shafts in the antenna drive train idler mechanism to prevent binding due to nylon gears running on aluminum shafts.

Explorers 12, 14, 15, 26 – Energetic Particles Explorers (EPE). The solar paddle deployment shaft runs in a Fafnir AN200 KP3 bearing on the first three EPE'S. A Teflon bushing was used in Explorer 26.

Gemini. Cabin area – silicones, fluorocarbons, and baked dry film lubricants
Valves and fittings in propulsion system – fluorocarbons and heat-cured dry film lubricants
Mechanical systems operated in the orbital environment – sodium silicate bonded dry-film lubricant
Antifriction bearings – G-300 silicone grease or CLD 5940 dry-film lubricant

Mariner 3 and 4. Solar panel actuator – silicone oil sealed internally; MoS_2 coating externally
Solar panel hinges – single ball bearing Teflon coated
Latch mechanisms – MoS_2 coatings
Temperature control louvers – Teflon coated bushings
Solar sail deployment pivots – Nylotron bushing
IR, UV, and TV scanner mechanisms – GE-300 grease and F50 oil lubricants

Nimbus 1. Solar cell paddle bearings – large ball bearings lubricated with G-300 grease
Slip ring assembly – brushes, 88% gold and 12% MoS_2; slip ring rhenium plated
Thermal shutters – sintered bronze bushing impregnated with F-50 oil (Shaft material is aluminum.)
Iris drive motor brushes – Stackpole carbon brush 566 (lithium carbonate)
Radiometer – Windsor Lube L245X (diester base oil, MIL-L-6085A) dispensed from sintered nylon reservoirs surrounding rotating shaft.
Gear motor used to drive solar panel mechanism – lubricated with G-300 grease (F-50 silicone oil in lithium soap base)
Separation ring lubricant – DC-55 silicone grease
Other applications – DC-4 silicone grease
Paddle deployment hinge pins – MoS_2
Tape recorder – BarTemp bearings, gold-plated slip rings
Solar panel drive mechanism – beryllium-copper alloy output shaft pinion mating with a Micro Mach 416 nitrided gear lubricated with G-300 grease mounted on an aluminum alloy 2014-T4 tubular shaft which was connected to a backlash clutch mechanism and to the solar panels; bearings were 440C stainless steel balls and races with crown retainers with a 25% pack of G-300 grease.

For the yet-to-be launched Nimbus 2 this has been changed to AISI 4340 alloy steel hardened to 46 to 50 Rockwell for the output shaft pinion. The pinion mates with a hard anodized 2014-T4 aluminum alloy gear on the tubular aluminum shaft. The external ring gears are now nitrided 135 M nitralloy instead of the Micro Mach 416 previously used. The R4 shaft bearings and the R2 motor bearings are 440 C stainless steel races and balls with paper base phenolic retainers. A 25% pack of G-300 grease is in the bearings and the same lubricant is used for the gears.

OGO. Thermal louvers – sleeve bearings made of 80% Ag and 20% MoS_2 with a shaft made of aluminum coated with Everlube 811B (sodium silicate bonded MoS_2).
Solar panel orientation devices – gold-plated balls and races burnished with MoS_2.
Wabble drive – MoS_2 impregnated sintered bronze driver gear and gold-plated stainless steel output gear
Orbital plane experimental package – gold-plated balls and races, MoS_2 and G-300 grease in various applications
Shaft assembly bearing – Synthane L retainer impregnated with 5% by volume Apiezon K.
Elgiloy springs in boom deployment mechanism – Everlube 811 B
Solar array drive and shaft support bearings:
 Balls and races – 440 C CRES, conforming to QQ-S-763, gold plated
 Retainer – 416 CRES, conforming to QQ-S-763, gold plated
 Gold plate – 24-carat gold plating preceded by nickel strike 0.00004-in. thick maximum in conformance with MIL-G-45204, Type 1, Class 1, 0.00008 to 0.00010-in. thick including nickel strike.
 MoS_2 burnishing – application of dry, unbonded MoS_2 films.

OSO-2. Main bearing – proprietary (Ball Bros.) lubricant impregnated into retainer.
Hinge pins – Apiezon L and bonded MoS_2
Slip rings – 90% Ag-10% Cu impregnated with proprietary lubricant.
Brushes – 75% Ag-20% graphite-5% MoS_2 impregnated with proprietary lubricant.

Pegasus 1 and 2. Deployment mechanism – Everlube 811 and Molycote X-15
Thermal control louvers – Fluorosint bearings
Hinge pins on wings – not lubricated but contain Teflon bushings.
Deployment mechanism bearings – 52100 and 440 C
Output shaft bearings – Gold-plated 52100 steel races with Rulon C ball retainers and 440 stainless steel
Gear and pinion shaft radial and thrust bearings – Torrington needle bearing with 440 C needles in 1012 steel cups (electroless nickel plated) and lubricated with burnished MoS_2.

Ranger 6 through 9. (Same as Ranger 1 through 5). Solar panel actuator – silicone oil, sealed internally; MoS_2 coating externally.
Solar-panel hinges – single ball bearing, silicone grease
Gamma-ray extension boom – silicone oil, sealed internally.
Lyman-alpha telescope scanner – low-speed stepping motor, MoS_2; high-speed actuator, F-50 silicone oil, sealed
Latch mechanisms – MoS_2
Omni-antenna hinge – MoS_2; actuator dash pot used silicone oil, sealed.
High-gain antenna drive – sealed actuator gears, MoS_2; shaft bearing, silicone oil; worm gears, Aeroshell 7 grease

Tiros. Camera shutter – aluminum shutter sliding in channel of graphite-impregnated nylon.
Radiometer – Windsor Lube L245X (diester base oil, MIL-L-6085A) dispensed from sintered nylon reservoirs surrounding rotating shaft.

Table 1-9

ADHESIVES

Courier 1B. Bonding solar cells to module skin – Dow-Corning Q-3-0120
Bonding modules to satellite – Dow-Corning Q-3-0120
Bonding solar cells covers – Dow-Corning Q-3-0040

Echo 2. Bonding Mylar to aluminum in skin laminations – GT-301 adhesive (a modified polyester cement) of thickness 0.3 mil

Explorers 9, 19, 24 (Atmospheric Density Explorers). Bonding gores in construction of balloon – Goodyear Pliobond and GT-301 (made by G. T. Schjeldahl Co.)
Bonding aluminum to Mylar for skin lamination – GT-301

Explorer 22. Bonding corner cube silica prisms to aluminum mounting bracket – epoxy cement

Gemini. Bonding layers of reentry heat shield and bonding heat shield to titanium substrate – HT-424

Mariner 3 and 4. 913 Epon used for bonding face sheets to core of solar panels. RTV-60 used to bond dielectric face plate for solar cell insulation. RTV-612 used to bond solar cell covers to cells.

Nimbus 1. Bonding solar cell covers to cells – PD-454
Camera lens cover glass adhesive – RTV-60 silicone with SS 4004 primer
Other adhesives – FM-1000; Epon 815, 828; Epibond 123; Eccobond 57C (Emerson and Cumings); Eastman 910; Dow-Corning A 4000
Video target mount – RTV-731
Leadwire tack bonding – RTV-102
Solar panel honeycomb construction – F 100

OGO. FM 100 for bonding phenolic fiberglass face sheet to phenolic fiberglass honeycomb. EC 1469 for bonding 2024 Al alloy face sheets to 3003 Al alloy honeycomb. These laminates used for instrument and electronic module containers. EPDX 74 used to bond solar cell covers to cells. Versamid 140, Epon 6, and Epon 8 used for miscellaneous bonding.

Pegasus 1 and 2. Laminating Mylar dielectric on capacitive detectors – GTS A-29
Cementing detector assembly to foam substrate – GTS A-49
Electrical connectors on detector panels – Pittsburgh Plate Glass Co. M-690
Diode boards to detector panels – RTV-577
Foam end plugs in structural tubing – RTV-102
Solar cell cover plates – RTV-602
Solar cells to substrate – RTV-40

Ranger 6 through 9. RTV-602 for bonding cover glass to solar cells and bonding solar cells to substrate; Versamid 125 and Epon 828 for bonding electronic components; also used Armstrong A-2/A epoxy adhesive.

Syncom 1 and 2. Solar cells to satellite skin – RTV-731

Syncom 3. Solar cell covers – LTV 602
Solar cells to satellite skin – proprietary Hughes adhesive
Other uses – Borden Epiphen 825A and a polyamide cured alicyclic epoxy

Table 1-10

SEALING MATERIALS

Explorer 17. OFHC copper gaskets are used in the twenty-two instrument ports, and the main shell seal is formed by a 0.05 in. copper gasket.
Mass spectrometer – gold O-rings
Other seals – glass-to-metal with rubber O-rings as backups

Gemini. Hatch seals and window seals – silicone rubber

Nimbus 1. Control pneumatic subsystem – Epon 8 filled epoxy, polyester base Loctite
Cross beam edge member sealant – Stafoam 110
Other locations in spacecraft – Loctite Sealant Grades A and C, Buna-N elastomer, glyptol, RTV silicone rubber, Type IV, mica filled epoxy PRH-103

Tiros. Bondmaster M648 as a fillet and sealant around solar cells and module boards to stop vibration and fill voids.

Table 1-11

STRUCTURAL MATERIALS – ORGANIC

Alouette. Component mounting platform in electronic packages – glass base epoxy laminate, 1/16-in. thick
Encapsulant for electronic units – polyurethane foam of density 10 lb/ft^3

Ariel 1. External skin of the spacecraft – epoxy fiberglass bonded with Epon 828; top dome is 1/16-in. thick, bottom is 1/32-in. thick, and center cylinder is 1/16-in. thick by 23 in. in diameter
Inertia booms – epoxy bonded fiberglass tubing 30-in. long
Encapsulants for electronic subsystems – Eccofoam
Harness wire insulation – Teflon

Courier 1B. Equipment shelves – fiberglass reinforced plastic honeycomb, the reinforcement consisting of a continuing length of tubing molded into the shelf.
Outer shell – two hemispheres fabricated of fiberglass plastic honeycomb between two skins of epoxy-impregnated fiberglass. Outer skin is coated with aluminum to function as a ground plane for the antennas.
Encapsulating compounds – RTV-60 silicone rubber, Eccofoam of density 2 lb/ft^3
Wire insulation – Teflon

Echo 2. Main skin of balloon consists of a laminated sandwich construction with 0.35-mil thick Mylar (polyethylene terephthalate) film between two layers of 0.18-mil thick 1080 aluminum.

Inflation was achieved by means of sublimation of 38 lb of pyrazole crystals packed in wax-sealed plastic bags inside the balloon and arranged to be unsealed by heating and inflating the balloon at a slow uniform rate.

Explorers 9, 19, 24 (Atmospheric Density Explorers). This series of spacecraft were spheres 12 ft in diameter and constructed of a four-layer laminate of two 1/2-mil layers of aluminum and two 1/2-mil layers of Mylar cemented together with GT-301 adhesive. The outside surface was aluminum and the inner Mylar. The balloons were constructed as two hemispheres with a dielectric gap between the halves. On Explorer 9 this gap (~ 1 in.) was of Mylar but on Explorers 19 and 24 it was changed to H-film.

Ten fiberglass stiffeners were equally spaced around the balloon equator and bridging the antenna gap to compensate for the weakness of the dielectric material in buckling.

Ejection bellows – flexible body, Fairprene elastomer; fixed bulkhead, fiberglass; sliding piston, aluminum

Explorers 12, 14, 15, 26 (Energetic Particles Explorers). Equipment shelf – octagon shaped platform made from nylon honeycomb with facings of 181 glass cloth bonded with Epon 828.

Explorers 18, 21, 28 (IMP). Equipment shelf – platform of nylon honeycomb and fiberglass similar to the EPE satellites

Supporting struts and one part of the magnetometer boom were also made of fiberglass laminate.

Encapsulating materials – Eccofoam FP-12-6

Explorer 22. Outer shell – nylon honeycomb and fiberglass

Explorers 13, 16, 23 (Meteoroid Detection Explorers). Mylar strips 5-mil thick were used to insulate the heat-transfer band from the forward shell.

On Explorer 13, the stainless-steel meteoroid sensor assemblies were mounted on silicone rubber. For Explorer 16, the mounting was changed to 1/8-in. thick urethane foam.

Encapsulating materials – silicone rubber used to fill wiring channels

Gemini. Re-entry heat shield – one layer of Refrasil phenolic honeycomb plus another layer of the same honeycomb filled with DC-325 ablative material. The two layers are bonded together with HT-424 and to a titanium back-up plate with the same adhesive.
Heat shield edge ring – silica-cloth phenolic resin composite
Circuit boards – epoxy glass laminate coated with epoxy
Astronaut couches – nylon netting
Wiring insulation inside crew cabin – Teflon
Wiring insulation outside crew cabin – irradiated polyolefin
Encapsulating compound – RTV silicone

Mariner 3 and 4. Antenna feed – epoxy fiberglass tubing
Solar sails – Mylar, aluminized on one side, coated with black dye on the other
Ascent fairing shroud – fiberglass honeycomb construction on Mariner 3 (changed to magnesium on Mariner 4)

Nimbus 1. Control box top cover – fiberglass laminate
Thermal control shutters – 30 layers of 1/4-mil aluminized Mylar
Wiring insulation – silicone rubber-impregnated braided fiberglass (MIL-I-1805A). Some polyvinyl insulation sleeving was used but in more lately constructed equipment, change has been made to fiberglass. Other low-voltage wiring insulation was irradiated polyethylene.
Target lamp encapsulant – PD-454 (Code L), a transparent epoxy
High-voltage wiring – silicone rubber insulated (MIL-W-16878, Type FF)
Electronic modules encapsulants – GE formulations MP 49 and MP 50 low-density filled epoxy resins (filler was silica micro-balloons). Also used for circuits carrying less than 200 V was the transparent polyurethane PR-1530 or the non-transparent silicone resin RTV-60.
Insulation of solar cells from structure – epoxy fiberglass cloth

OGO. Outboard end of short booms – fiberglass reinforced epoxy laminate torque box, rigid support for experiment assembly
Self-locking nuts – pure nylon or nylon with non-outgassing dye
Instrument boom insulation – 181 glass cloth reinforced Epon 828 brackets
Electronic modules; instrument boxes – fiberglass reinforced phenolic honeycomb construction
Thermal insulation blanket – aluminized Mylar Type W; 10 to 12 layers, ultrasonically bonded
Electronic cable lacing – Dacron
Wire insulation – irradiated polyolefin, also some TFE Teflon
Encapsulants – Solithane 113 polyurethane for conformal coating, silicone RTV-11 and RTV-60 on cable clamps, relays, and switches; Scotchcast epoxy and Stycast 1090 on connectors; Epon 826, TETA cure impregnated with Al for thermal conducting applications

OSO-1. Circuit board – epoxy glass laminate materials
Potting applications – Epon 828, RTV-11, RTV-40
Wire insulation – Teflon
Shrinkable tubing – Polyolefin
Circuit board damper – silicone foam
Encapsulant – Rigidlock foam to hold tuned coils of diplexer
Gas bottle (attitude control system) – glass reinforced epoxy

Pegasus 1. Meteoroid detecting surfaces – the capacitor sandwich has a center insulator of Mylar, laminated from 3 layers of 0.15-mil-thick Mylar. For the 1.5-mil-thick aluminum detector panels, the sandwich is attached to a 1-in.-thick core of Delta-T rigid polyurethane foam. For the 8- and 16-mil panels, the sandwich is cemented to a 0.25-in.-thick layer of soft Scott foam which in turn is cemented to a core of 0.5-in.-thick rigid NOPCO G-304 polyurethane foam. On the other side of the rigid foam an identical arrangement completes the double-sided detector.

Pegasus 1 and 2. Potting compound – PR-1538 urethane
Isolation mounts – silicone rubber
Cable harness insulation – Raychem Corp. Novathene on Pegasus 1 and Raychem Corp. Spec 44 wire insulated with irradiated polyolefin and polyvinylidene fluoride on Pegasus 2.
Wire and cable ties – silicone rubber tape and nylon sta-straps
Conformal coatings – X-81 epoxy and PR-1538 urethane
Center section supporting angles – fiberglass

Ranger 6 through 9. Supports for high gain antenna feed are four fiberglass tubular struts. Container for hydrazine fuel in the midcourse rocket motor is a rubber bladder. (Fargo Rubber compound FR-6-60-26)
Attitude control system valve seat – Teflon
Junction connectors – silicone rubber filled in Ranger 7 and above instead of polyurethane
Circuit boards – G11-FR4 epoxy-fiberglass

Potting materials – Stycast 1090/11 epoxy, RCA 688 epoxy, and PR-1527 polyurethane
Lacing cords – nylon for Ranger 6, changed to Dacron for Ranger 7.
Connector conformal coatings – Solithane 113
Shrink fit tubing for cables – Thermofil type CRN
Wire insulation – Rayolin-N and some Teflon
Cable clamp cushion – silicone rubber
Electrical insulation between structure and cells in solar panel – laminar sheet of 4-mil-thick epoxy fiberglass
Fasteners – KEL-F was used to minimize galling and nylon caps were bonded to the back of nuts as a dust cover.
Linings for wire cable clamps – silicone rubber
Encapsulant – for connector pins, Silastic 881; flexible, epoxy compound Bondmaster 688; rigid, Stycast 1095, an epoxy compound filled with glass microspheres. Ranger 7 through 9 and Mariner 2 and 4 organic structural materials were given a vacuum outgassing treatment prior to application in spacecraft.

Relay 1. Dry-friction damper – polished, phenolic fiberglass rod sliding in a stainless steel tube.

Syncom 1, 2, and 3. Spacecraft skin – 3-mil glass fabric with aliphatic amine curing epoxy (2 plies aft end, 3 plies forward and, and 1 ply on the cylindrical surface as a face sheet on the aluminum honeycomb.
Electrical terminal boards – copper-clad epoxy laminate with gold or silver plating
Apogee motor nozzle – Raybestos-Manhattan Refrasil
Potting in electronic boxes – CPR-23 polyurethane foam
Conformal coatings on terminal boards – epoxy polyamide

Syncom 2. Hookup wiring insulation – 6-mil-thick Teflon FEP with a coating of 0.5- to 1-mil modified polyimide resin

Tiros. Low-resolution infrared radiometer – Two highly aluminized truncated Mylar cones mounted on a 3-in. diameter, gold-plated, aluminum plate. At the base of each cone is mounted a thermistor detector fastened to the Mylar base by a grid of fibers of low thermal conductivity.

Miscellaneous Satellites

Oscar 3. Encapsulation material – polyurethane foam

Satellite 1963 22A. Encapsulant – to prevent damage to the damping spring, it was encapsulated in biphenyl. After boom was extended, biphenyl sublimed releasing one coil of the spring at a time.

TRS (Tetrahedral Research Satellite). Solar cell substrate – 0.007-in. copper-clad fiberglass.

Table 1-12

ANTENNAS

Alouette. The sounding antenna system consists of crossed dipoles, one 150 ft tip to tip, the other 75 ft. The antennas are made of thin flat strips of heat-treated spring steel, 0.004-in. thick and 4-in. wide, wound tightly on 3-in. long spools. After the satellite was inserted into orbit, the spools unwound with the flat strips curling as they passed through a set of guides to form tubular antennas.

Telemetry antennas were four whip antennas.

Ariel. Thin wall aluminum tubing whip antennas (four) which unfold to 21-3/4 in. long in orbit.

Explorer 20. Beryllium-copper 2-mil thick forming a tube 1/2-in. in diameter when unfurled.

Explorer 13. 6061-T6 aluminum alloy tubing, spring loaded to erect after jettisoning of fourth-stage heat shield.

Mariner 3 and 4. High-gain antennas – aluminum honeycomb with 4-mil aluminum foil facings and honeycomb core of 1/4-in. cells fabricated from 0.7-mil foil.

Nimbus 1. S-band antenna – a fiberglass cone containing on its surface two logarithmic spirals of copper.
Quadraloop antennas – dielectric was Fluorosint, a sintered Teflon-based material; conductor, a special treatment for magnesium, DOW 23 or MD-678, consisting of a coating of magnesium stannate and tin on the magnesium surface.

Ranger 6 through 9. High-gain antenna – a circular dish-shaped structure constructed of fabricated sheet aluminum-alloy (2024) ribs emanating radially from the center and supported at midpoint and the outer diameter by sheet metal rings (aluminum alloy). The dish surface is covered by a black anodized aluminum alloy (5052) mesh, held in conformation by the radial ribs and the mid and outer rings.

Miscellaneous Satellites

Oscar 3. Measuring tape steel with silver plating

TRS (Tetrahedral Research Satellite). 0.5-in. wide ribbon measuring tape steel stiffened by concave forming and gold plated

Table 1-13

ELECTRONIC MATERIALS

Alouette. Solar cells – 6480 p-on-n type
Batteries – nickel-cadmium, hermetically sealed

Ariel 1. Solar cells – silicon p-on-n type

Ariel 2. Solar cells – 5376 silicon n-on-p type gridded cells 1 cm by 2 cm qualified to withstand 10^{16} electrons/cm^2 of 1 MeV.

Courier 1B. Solar cells – 19,152 Hoffman Type 120 C silicon cells 1 cm by 2 cm

Early Bird. Solar cells – n-on-p type, 45-W array
Batteries – nickel-cadmium

Explorer 17. Batteries – silver-zinc batteries of the modified Yardney HR type

Explorers 12, 14, 15, 26. Solar cells – 6144 p-on-n silicon cells, 12% efficient
Batteries – silver-cadmium type
Windows of Geiger tube detectors – mica, 0.17-mil thick

Explorers 18, 21. Batteries – silver-cadmium
Solar cells – 11,520 p-on-n type cells, 12.5% efficient

Explorer 22. Solar cells – p-on-n type
Batteries – nickel-cadmium

Explorer 16. Solar cells – p-on-n type, 8% efficient

Explorer 23. Solar cells – n-on-p type
Capacitor type detectors for electron accumulation – sandwich of Mylar layer between copper and stainless steel electrodes

Mercury. Voice cable insulation – Teflon

Gemini. Connectors – cadmium plated (but kept out of line of sight with horizon scanners)

Injun 1 and 3. Windows of Geiger tube detectors – mica, 0.17-mil thick

Mariner 3 and 4. Earth sensor detector – S-11 photocathode Dumont photomultiplier tube
Sun sensor detector – cadmium sulfide
Solar cells – 28,244 silicon p-on-n cells
Batteries – silver-zinc with 1200 W-hr rating
Insulation between solar panel structure and solar cells – epoxy fiberglass sheet

Nimbus 1. Solar cells – Approximately 11,000 2 cm by 2 cm silicon cells connected by means of beryllium-copper interconnecting strips
Batteries – hermetically sealed nickel-cadmium type
Radiometer detector – lead selenide operating in 3.4- to 4.2-μ range
Sun sensors – silicon p-on-n cells

OGO. Solar cells – 32,256 gridded p-n silicon, 1.9 cm^2, 10.5% efficiency
Batteries – nickel-cadmium, 12-A-hr capacity, two packs each consisting of 22 cells.
Sun sensors – p-n silicon cells

OSO-1. Solar cells – Hoffman p-on-n silicon cells
Circuit boards – use gold plating.

OSO-2. Solar cells – Heliotek n-on-p solar cells
Circuit boards – use gold plating.

Pegasus 1 and 2. Hookup wiring to detector panels – Methode Plioduct ribbon cable
Solar cells – 1 cm by 2 cm n-on-p silicon cells (Hoffman and Texas Instruments

Pegasus 1. Batteries – nickel cadmium
Earth sensor detector – thermocouples in a thermopile

Ranger 6 through 9. Batteries – two silver-zinc batteries with capacity of 84 A-hr
Solar cells – p-on-n type silicon cells, 4,896 used on Ranger 7.

Relay 2. Solar cells – n-on-p cells

Syncom 1 and 2. Solar cells – p-on-n type, 3,840 in number
Batteries – nickel-cadmium

Syncom 3. Solar cells – n-on-p silicon cells of 10 Ω-cm resistivity
Batteries – nickel cadmium

Tiros 9. Solar cells – n-on-p type instead of p-on-n type used in previous Tiros.

Tiros. Attitude control coil – aluminum wire (250-turn coil) which when energized interacts with Earth's magnetic field to control the attitude of the satellite.

Miscellaneous Satellites

Oscar 3. Solar cells – n-on-p silicon cells

TRS (Tetrahedral Research Satellite). Solar cells – n-on-p cells

Table 1-14

STRUCTURAL MATERIALS – INORGANIC

Alouette. The basic structure consists of seven parts. An aluminum central load-bearing column or thrust tube supports two payload platforms. A spun aluminum half shell is attached to the periphery of each platform. Finally, a thermal radiation shield is attached to the top of each shell.
Electronic packages – aluminum
Battery packages – aluminum castings

Ariel. End flanges for fiberglass midsection of spacecraft – aluminum alloy 6061-T6
Instrument shelf and base assembly – aluminum alloys 7075-T6 and 6061-T6 plates, bars, and billets
Paddle arms – 7075-T6 aluminum alloy channel
Fasteners – gold-plated aluminum machine screws
Battery case – stainless steel

Courier 1B. Basic frame of the satellite – aluminum alloy tubing assembled in the shape of an octahedron.
Central band to which fiberglass hemispheres are attached – magnesium; this band also carries the four v-h-f whip antennas.
Transmitter cases – magnesium, hermetically sealed by welding
Sealed box for recorder-reproducer – aluminum alloy 6061-T6 sheet

Explorer 9. Inflation bottle – AISI 4340 steel normalized after welding
Sliding piston of the ejection bellows – aluminum

Explorer 17. This spacecraft is a completely sealed sphere of 321 stainless steel 25-mil thick and 35-in. diameter.

Explorers 12, 14, 15, 26. All have the same basic configuration of an octagonal box atop a truncated cone. Four solar cell paddles are attached to the base of the cone and a flux-gate magnetometer stands off from the box on a long tube. Running along the symmetry axis of the spacecraft is the main structural element, a tube of ZK60A-T5 magnesium alloy. Forming the top cover of the box is an aluminum core honeycomb sheet with an outer skin of 3-mil and an inner skin of 8-mil aluminum bonded with Bondmaster M-690 adhesive. The magnetometer support is a tube of 6061-T6 aluminum alloy. This tube was 14-mil except in Explorer 15, in which it was made 22-mil thick in order to provide a better conductive path to the magnetometer which was expected to be more shadowed and thus run colder. The lower surface of the spacecraft, covering the conical portion, is 6061-T6 aluminum alloy 20-mil sheet. Other metallic elements include the separation spring seat and the antenna cover support which are ZK60A magnesium, the damping springs on the solar paddles which are Elgiloy (nonmagnetic), and a beryllium-copper actuator for turning on experiments after launch. Helicoil beryllium-copper thread inserts were used in magnesium parts. Stainless steel fasteners were used in the smaller sizes and aluminum for larger ones.

Explorers 18, 21, 28. A magnesium center tube, magnesium and fiberglass struts, and a bottom shelf and cover of aluminum honeycomb constitute the basic structure. In IMP a rubidium vapor magnetometer is carried on a telescoping tube, one element of which is 40-mil thick aluminum and the other is fiberglass laminate.

Explorer 22. Side and end panels – honeycomb sheet metal sandwiches with inserts for local strengthening and fastener support
Solar cell panels – corrugated core laminate assembly

Explorers 13, 16, 23. Forward shell of spacecraft – Type 410 stainless steel, 31-mil thick
Heat-transfer band – aluminum alloy 1100-H14, 0.1-in. thick and 5-in. wide
Telemetry canisters – aluminum alloy 2024-T4, machined from bar stock
Telemetry bases – machined from mild steel and inside surfaces silver plated to prevent rusting
Detector mounting structure – forward and aft rings were machined from AZ31B magnesium alloy; rings were riveted to an aluminum alloy 2024-T3 cylinder of 16-mil thick material
Meteoroid detectors – stainless steel targets of both 3-mil and 6-mil thickness; gold grid detectors bonded to rear of steel by means of an intermediary layer of 1/2-mil thick Mylar

Explorer 16 also exposed stainless steel meteoroid targets of thickness 1-mil in addition to the 3-mil and 6-mil ones. Pressurized cells of beryllium-copper were also included in 1-mil, 2-mil, and 5-mil thicknesses.

Gemini. Substructure of spacecraft – commercially pure unannealed titanium, Ti-8Mn, Ti-5Al-2.5Sn. and Ti-6Al-4V alloys; the pure titanium was used for welded and severely formed parts.
Exterior covering – beryllium and René 41 shingles (beryllium shingles cover the rendezvous and recovery section and the reentry control section)
Crew compartment items, ejection, instrument panel, electrical consoles, and storage compartments – aluminum alloys 2024-T4, 2024-T6, and 7075-T6
Adapter section skin – magnesium alloy HK31A
Adapter section reinforcing stringers – magnesium HM31A
Spacecraft attitude and trajectory control system pressure vessels – Ti-6Al-4V
Spacecraft attitude and trajectory control system piping – 304L stainless steel
Retrorocket case – Ti-6Al-4V forging
Fuel cell tubing – 304 stainless steel
Tubing other than fuel cell and control system – 5052 aluminum alloy
Flare tubing nuts – 7075-T73

Mariner 3 and 4. High-gain antenna – aluminum honeycomb with 4-mil aluminum foil facings and honeycomb core of 1/4-in. cells fabricated from 0.7-mil foil.
Pressure vessels for nitrogen gas for attitude control system – titanium 6Al-4V alloy in the stabilized annealed condition

Solar panels – truss-core construction with 5-mil 3003 H14 aluminum alloy facings and 3.5-mil 5052L aluminum alloy core
Top and bottom octagonal plates and longerons in bus – ZK60-T5 magnesium alloy forgings
Platforms for scientific experiments – 6061-T6 aluminum alloy, fasteners of 6Al-4V titanium and of A286
Electronic module containers – ZK60 magnesium
Mast for supporting instruments and waveguide – 6061-T6 aluminum, 25-mil thick, 6-ft long, 6-in. diameter
Solar sail frame – 6061-T6 aluminum tubing
Solar sail deployment mechanism – Be-Cu springs

LES (Lincoln Experimental Satellite). Frame is a tubular aluminum structure in the shape of a rhombocuboctahedron. Tubing is dip-brazed into cast aluminum corner hubs.

Nimbus 1. Sensory ring – ring is a 57-in. toroid, 40-in. diameter, 13-in. tall, and 8-in. deep, fabricated of magnesium alloy AZ31B-H24. The outboard webs in the sensory ring are also magnesium and the V-shaped separators are castings of magnesium alloy ZH62A.
Infrared sensor housing – ZK60A-T5 magnesium
Coarse sun sensor structure – aluminum alloys 6061-T6 and 5052-0
Command antenna – sheet and tubing of aluminum alloys 5052-0, 6061-T6, and 2024-0
Thermal control shutter – aluminum 2024-T3
Sensory subsystem structure – magnesium AZ31B-H13 – ZK60A-T5 and aluminum 7075-T6 – 2024-T3
Upper torus structure – magnesium AZ31B-H24 – ZH62A-T5 and aluminum 2014-0
Separator casting supporting the solar platform support – magnesium casting ZH62A-T5
Pneumatic subsystem nitrogen gas storage tank – titanium 6Al-4V
Solar array panel – aluminum alloy 3003 honeycomb material with 1-mil core and 3-mil skin
Electronic equipment containers – magnesium ZH62A-T5 castings
Tubing to attitude control jets – aluminum alloy 6061

OGO. A double-walled corrugated all aluminum structure was used for the box. On two sides (experiment door), the box wall consists of three sheets (two panels and core) of 0.016-in. 2014-T6 aluminum alloy. On the remaining sides, there are 0.016-in. outer sheets and cores and 0.020-in. inner sheets.
Experiment platform – 2024 aluminum fittings and sheet metal parts
Hinge joints – standardized Elgiloy latching spring
Fasteners – A286 in some special nuts, others from aluminum or 6Al-4V titanium alloy
Solar panels – Substrate 0.04-in. cross-rolled sheet beryllium under solar cell modules; 2024 aluminum alloy channel reinforcing members vary from 0.020 to 0.032 in.

Louvers for thermal control are made of two thin hydroformed sections of 1145-H19 aluminum alloy that are spot-welded together. Louver operated by bimetallic spring.

Main spar of solar panel – extruded square tube of 2024-T42 aluminum alloy
Booms – 6061-T6 aluminum alloy 0.020-in. wall thickness; four short, single-element booms 1.25-in O.D.; two long, 3-element booms 1.50-in. O.D. The OPEP booms machined in form of I beam from 2024-T351 aluminum alloy.

OSO-1. Panels, cover plates, and other sheet applications – 2024, 5051, 5052, 6061, and 7075 aluminum alloys, mostly in T3 condition
Castings – 356 aluminum alloys
Hinge pins – chrome-plated steel
Fasteners (electronic assemblies) – A286 stainless steel (for lower magnetic permeability)
Solar cell panel substrate – AVCO aluminum honeycomb panel

OSO-2. Same as OSO-1, except 7075 aluminum alloy replaced by 5051 aluminum alloy.
Pneumatic tubing – silver brazed
Gas bottle (attitude control system) – titanium alloy 6Al-4V

Pegasus 1 and 2. Structural frame, wings – tubing of 6061-T6 aluminum alloy connection fittings of 2024-T4 aluminum alloy
Center section – extruded 2024-T3 aluminum alloy
Deployment gear train housing – A356 aluminum casting with zinc chromate coating
Lightly loaded gears – 4340 steel, black oxidized
Drive – 302 stainless steel
Heavily loaded gears – 9312 steel, black oxidized
Meteoroid detecting surfaces – sandwich consisting of an aluminum alloy outer layer in various thicknesses up to 16 mil, a layer of Mylar, and a layer of OFHC copper (~ 25 μin.) on the back of the Mylar:

- 1-1/2-mil sheets are 1100-0 aluminum and cover 80 ft^2.
- 8-mil sheets are 2024-T3 aluminum and cover 160 ft^2.
- 16-mil sheets are 2024-T3 aluminum and cover 1700 ft^2.

Solar panels – 0.75-in. thick aluminum honeycomb sandwich, 41 in. by 63.5 in.

Ranger 6 through 9. Solar panel construction – aluminum alloys; 2024-T6 frame; sandwich, truss-core, corrugated 5052-H38 foil, 3003-H14 face sheet; longitudinal and transverse cross bracing, 6061-T6 angles. Spot and seam welded frame and bracing; truss-core, FM 1004 adhesive bonded.
TV tower shroud – aluminum sheet 2024-T4 braced with aluminum ring support T-sections and rivets.
Main spacecraft bus – (mostly same as Ranger 1 through 5); however, the support legs which were gold-plated magnesium in Ranger 1 through 6 were changed in Ranger 7 through 9 to 6061-T6 aluminum alloy.
Gas tank for attitude control subsystem – titanium 6Al-4V alloy
Plumbing for attitude control subsystem – all welded stainless steel

Ranger 7 through 9. Vidicon camera structure – right cone made from 2024 T4 aluminum alloy sheet 1/32-in. thick with T-sections for rings and longerons.
Brackets for base – machined from 2024-T3 bar stock.
Camera support ring – aluminum alloy 356-T6 casting
Fasteners – A286 steel Longlock screws in Davis nuts
Containers for power supplies and other electronic circuitry – 2024-T4 aluminum alloy sheet and bar stock
Wire cable clamps – stainless steel

Relay 1. Dry-friction damper – polished phenolic fiberglass rod sliding in a stainless steel tube.

Soviet Spacecraft (Sputnik 2). Animal cabin and spherical container – AMT's AM sheet aluminum 2-mm thick (~ 0.080 in.). Surface was polished and subjected to special processing for thermal control purposes.
Sanitary tank – sheet aluminum alloy

Syncom 1, 2, and 3. Basic structure – cylindrical forging of ZK60 magnesium alloy
Bulkheads – 6061-T6 aluminum alloy
Electronics packages – 6061 aluminum alloy with a zincate pretreatment plated with electroless nickel, then copper flashed and silver plated.
Skin of cylindrical surface – 1/4-in. aluminum honeycomb with glass fabric facing
Apogee motor case – 410 stainless steel
Attitude control gas tank – titanium-7Al-4Mo on Syncom 1 and 2, and 1060-H12 aluminum alloy on Syncom 3

Miscellaneous Satellites

Oscar 3. Shell – magnesium-lithium alloy

Nuclear Detection Satellite. Equipment platform – aluminum honeycomb sandwich
Injection engine housing – chem milled magnesium sheet

Satar. End sections – hemispherical domes of drop hammered forged duraluminum
Structure – largely aluminum

Satellite 1963 22A. Attitude stabilization – damping rods of Permalloy (47.5 Ni-52.5 Fe)
Electromagnet core – unannealed electrolytically pure iron
Extensible boom – tape material, beryllium-copper, 2-mil thick, 2-in. wide, when deployed. The tape forms a cylinder 0.45 in. in diameter and 100-ft long
Damping spring – 70 turns (7.6 in. in diameter) of 8-mil beryllium-copper wire (typically Beryllco Alloy 25) with 0.8-mil cadmium plating followed by 0.2-mil coating of gold.

TRS (Tetrahedral Research Satellite). Structure – aluminum 6061-T6 sheet and tubing

1.3.7 Gemini and Mercury

Spacecraft in the current Gemini series and its immediate predecessor, the Mercury program, constitute one class of that rare group of objects recovered from space and available for examination. Thus far, however, little information has been published on results of examination of the materials orbited in these vehicles.

Some items of interest that have been studied are the periscope lenses from some of the Mercury spacecraft. In a microscopic examination of the surfaces of these lenses, an effort has been made to correlate certain surface spots with artificially produced sites on a control lens in order to check the possibility that the Mercury lenses were bombarded with meteoroids while in orbit (Ref. 44). Several suspected sites were examined in detail although the number of supposed meteoroid impact sites cannot as yet be estimated due to the fact that an insufficient area of the lenses has yet been examined under high magnification. The sites reported in this work range from $\sim 5\mu$ to ~ 1 mm in diameter. By impacting high-velocity particles on a control lens with an electrostatic accelerator, artificial craters have been produced whose appearance can be correlated with those from the flight.

Experience gained with space environmental effects on the materials of the Mercury spacecraft has been utilized in selection of materials for Gemini wherever applicable. Much of the metallic structure of Gemini is the same as Mercury, such as the main structure skin of commercially pure titanium and the René 41 and beryllium shingles on the conical and cylindrical afterbody surfaces (Ref. 45). Where new structural materials have been introduced, the change has not been due to space environmental effects.

Electrical wiring insulation on the Mercury vehicle was irradiated polyolefin, while the Gemini was wired with Teflon insulated wire. This change was the result of tests in 5-psia O_2 atmosphere in which polyolefin burned vigorously when ignited while Teflon would burn but was self-extinguishing (Ref. 46). The Gemini continues to use polyolefin in areas outside the manned cabin.

The heat shield of the Gemini spacecraft has been tested in the actual space environment and found to perform well. This shield is a composite system consisting of a layer of Refrasil phenolic honeycomb plus another layer of the same honeycomb filled with Dow-Corning DC-325 ablative material. The two layers are bonded with HT-424 adhesive made by Ruberoid Co. and bonded with the same adhesive to a titanium back-up plate. An edge ring around the shield is made of silica-cloth phenolic resin. The successful trial in space of this shield was on the GT-2 suborbital unmanned launch of 19 January 1965; no cracks were formed in the material and an even regression of the char line into the shield was noted. Charred material as formed was held in place by the honeycomb (Ref. 47).

Tubing for the fuel and oxidizer system on the Mercury was aluminum with flared seals. Due to the difficulty of getting leakproof seals, this was replaced with AISI 304L stainless steel tubing with all brazed joints for the Gemini. All other tubing remains 5052 aluminum alloy as it was on the Mercury spacecraft except that in future Geminis carrying fuel cells, AISI 304L stainless will be utilized for the cell solutions.

One of the areas in which materials environmental testing has been extensively carried out for the Mercury and Gemini programs has been that of screening for toxicity of materials to be used in the manned cabin. Although only indirectly related to the space environment, this aspect of materials selection is obviously of great importance to the success of the spacecraft mission. Certain materials of obvious toxicity are easily spotted and eliminated. Many others have been tested by a simple odor test after the material was subjected to simulated environmental conditions of the space capsule. With this test, 125 non-metallic materials were examined for the Mercury cabin by subjecting them to 5 psia, 100% O_2 atmosphere and 150 to 300° F. Of this number, thirty six materials were rejected on the basis of objectionable odor (Ref. 48). In another study the general problem of the sources of contamination from a number of classes of materials was examined as well as detection and identification methods (Ref. 49).

An interesting correlation of space effects with ground testing occurred after the failure of the on-board computer during the Gemini 4 reentry maneuvers. The computer was later removed from the spacecraft and tested in the laboratory by IBM Corp. with their conclusion being that it was then working perfectly (Ref. 50). The most probable reason for the flight failure seems to be an interface problem with adjacent equipment.

1.3.8 Mariner

The Mariner series of interplanetary spacecraft has so far shown a somewhat better record of success than the closely related Ranger program. Although Mariner 1 had to be destroyed when the guidance of the launch vehicle malfunctioned, Mariner 2 was launched the following month to achieve a spectacular success in its return of scientific data about the planet Venus. Similarly for the other two Mariners to date, Mariner 3 failed to achieve its planned orbit while the closely following Mariner 4 has successfully completing its mission of photographing Mars.

The change in materials made to overcome the problem encountered with Mariner 3 shows an interesting correlation between laboratory and flight data. Mariner 3 apparently failed due to its failure to properly jettison the fiberglass fairing shroud and its consequent inability to deploy solar panels and acquire the Sun and Canopus. A laboratory test of a similar shroud showed that the inner facing of the laminated structure separated under combined stresses of aerodynamic heating and rapid pressure drop, simulating the launch environmental conditions (Ref. 51). On the basis of this evidence, the shroud material was changed back to the magnesium shroud previously used and separation occurred as planned on Mariner 4. However, no failure mode or mechanism has been discovered to account for the failure of the fiber glass shroud (Ref. 52).

For purposes of thermal control, wiring cable harnesses were wrapped with plastic aluminized tape. On earlier Ranger spacecraft, aluminized Mylar was used for this purpose. However, vacuum-UV testing of the Mylar tape showed that the Mylar degraded and flaked off so a change was made to aluminized polyvinylfluoride for Mariners 3 and 4 (Ref. 53). It was feared that the flakes of aluminum coating could give a false indication to the Canopus tracker and thus cause perturbations in the attitude control system and excessive use of control jet gas.

Still another example of a change of materials dictated by environmental experience is discussed on p. S-64 for the solar cell electrical insulation on Ranger. The partial failure of this insulating layer of Mylar on the Mariner 2 spacecraft was not sufficient to compromise the mission, due to design of ample capacity into the solar cell panels. Only one solar panel malfunctioned and then only intermittently. However, the incident led to redesign and the use of epoxy fiberglass as an insulating sheet in Mariners 3 and 4 (Ref. 54).

Mariner 4 was launched on its Mars encounter mission on 28 November 1964 and successfully carried out its mid-course correction maneuver and the re-acquisition of Canopus reference on 5 December. However, on December 7, telemetry showed that one of the eight scientific experiments had failed. The solar plasma probe instrument, designed to measure protons streaming out from the Sun, stopped transmitting intelligible information. Apparently an electronic component in the instrument had failed.

The spacecraft fix on the star Canopus, essential to its operation in the later stages of its mission but not in the early part, was temporarily lost also on December 7. It has been guessed that a small particle of dust reflected enough sunlight into the Canopus sensor to give a temporary false indication. In an attempt to shake loose any dust particles that might be causing this problem, the camera lens cover was removed by ground command on 11 February 1965 in advance of the normal sequence of removal during Mars encounter.

Encounters with meteoroids did not prove damaging to the Mariner system. Although it has passed through both the Geminid and Ursid meteoroid streams, no damage was done to the spacecraft. Leaving Earth, Mariner experienced about one interplanetary dust impact every 4 days. In the vicinity of the Martian orbit, this increased to about four impacts a day and the number of small-sized particles decreased (Ref. 55).

On 14 July 1965 the months-long Mariner 4 mission came to a brilliant conclusion with the successful pass to within 5600 sm of Mars and the acquisition and return to Earth by telemetry of the most detailed pictures of the Martian surface yet obtained. The twenty-one pictures obtained were transmitted back to Earth-based tracking stations twice in order to achieve optimum resolution of the photographs.

1.3.9 Nimbus

The Nimbus satellite is a relatively large and complex spacecraft. It represents the next generation of weather satellites following the successful Tiros program. Nimbus is Earth-stabilized in a polar orbit. The first satellite in this program was successfully orbited on 28 August 1964 and transmitted a number of excellent photographs of weather patterns before it ceased operating on 23 September 1964 because the solar panels failed to orient toward the sun with consequent catastrophic discharge of the spacecraft batteries.

The drive mechanism which oriented the solar panels to present the maximum surface to the Sun operated satisfactorily for 21 days, built-up friction for 1 to 2 days, and then stalled completely. This behavior was attributed to a bearing failure resulting from degradation of the G-300 grease (F-50 oil in a lithium soap base) because of local high temperature (estimated 250 to 300°F or higher) at the drive motor bearings. After the failure, investigations were conducted by G. E. and NASA-Goddard personnel and consultation was held with LMSC research laboratory staff. In a laboratory study at G. E. using a similar drive mechanism with space simulated thermal-vacuum and mechanical conditions, a lifetime of 300 hr at 400° F and 1300 hr at 300° F in vacuum was obtained (Ref. 56).

The solution to this problem was to change from the Kearfoot Size 8 gear-head drive motor, two phase, 400 cps, 28 V, 550 rpm, with a 0.2 in.-lb torque used in Nimbus 1 to a Size 11 motor having 0.4 in.-lb torque and similar electrical characteristics for the yet to be orbited Nimbus 2. The heat conduction paths around the motor were changed so that at full load the temperature at the bearing, lubricated with the G-300 grease,would not be above 100 to 160°F. This behavior of the new motor drive was verified by G.E. in simulated tests in vacuum. It should be noted that at these temperatures LMSC data published in 2nd edition, Space Materials Handbook (Ref. 4) show a lifetime in simulated space environment (high vacuum) at about 150°F would be in the order of 12,000 hr; this source also shows that lifetime at 300°F would be of the order of 4000 hr. Subsequent tests at LMSC show a lifetime for the G-300 grease in AISI 440 C stainless steel ball bearings R3 size double shielded with ribbon retainers in the temperature range of 145 to 175°F to be of the order of 22,000 hr when lightly loaded. Furthermore it should be pointed out that p.222 of Ref. 4 shows lifetimes of from 170 to 280 hr in vacuum of the G-300 grease, in Barden and New Departure 440 C stainless steel ball bearings at 11,500 rpm and a temperature of 325°F. Loads on the bearings in these tests at Advanced Technology Division of American Standard were ~ 15 oz on the front and ~ 8 oz on the rear radially and an axial load of 10 oz. Paper-base phenolic retainers were used in the R2 size bearings.

The solar panel orientation drive mechanism used in Nimbus 1 had a Berylco 10 (copper-based berylliumalloy) output shaft pinion mating with a Micro Mach 416 (a modified Type 301 austenitic stainless steel) nitrided gear with G-300 grease lubricant mounted on an aluminum alloy 2014-T3 tubular shaft which was connected to a backlash clutch mechanism and to the solar panels. The bearings were AISI 440 C stainless steel balls and races with crown retainers. Tests in vacuum reported in Ref. 4 have shown that lifetime of bearings with crown retainers are less than those with ribbon retainers and considerably less than those with paper base phenolic retainers.

For Nimbus 2 the output shaft pinion was changed to a AISI 4340 alloy steel hardened to 46 to 50 Rockwell C. The pinion mates with a hard anodized 2014-T4 aluminum alloy gear on the tubular aluminum shaft.

The external ring gears are now nitrided Nitralloy No. 135, a nitriding-type steel, instead of the Micro Mach 416 previously used. The R4 shaft bearings and the R2 motor bearings are AISI 440 C stainless steel races and balls with paper-base phenolic retainers, a 25% pack of G-300 grease is in the bearings and the same lubricant is used for the gears.

Several different passive thermal control material surface coatings were used on the Nimbus as well as an active thermal control system. These materials are listed in Table 1-6 together with the application on the spacecraft. Nimbus was designed to operate between 10 and 30°C; during the 23 days of reception of telemetry data from the satellite, it remained within these limits (Ref. 56).

Encapsulants and conformal coatings used for packaging of electronics modules on Nimbus were the G.E. proprietary formulations MP 49 and MP 50 (low-density epoxy resins filled with silica micro-balloons). The chief advantage of these formulations are their transparency, which permits effective inspection for voids and thus entrapped gas. This was effective in solving a problem that developed during vacuum simulation testing of Nimbus systems. The gas or air entrapped in encapsulant voids in modules and cable connectors caused high-voltage arcing in the vidicon and sequencer timer circuitry by tunneling through the voids between connector pins, circuit board terminals, and in shielded cable.

The arcing problems were solved by using a transparent encapsulant to permit positive visual inspection of modules and connectors for trapped voids or by encapsulating only one end of a shielded cable thereby providing a path for trapped gas to escape during depressurization in the launch phase of flight. The MP 49 and MP 50 encapsulant was used on those circuit elements that carried more than 200 V; no arcing problems developed during the flight lifetime of the spacecraft (Ref. 56).

1.3.10 OGO (Orbiting Geophysical Observatory)

The first OGO spacecraft, launched 4 September 1964, was successfully orbited but failed to deploy all its booms. One of the partially deployed booms obscured the Earth sensor so that the spacecraft could not be Earth-oriented. However, some useful data are being returned from all twenty experiments. On failure of the satellite to orient to the Earth, it was sent into an alternate spin-stabilized mode. Some of the experiments aboard are much more affected by the spin than are others. Measurements of the omnidirectional trapped radiation are unaffected and this experiment is functioning just as well as if the satellite were in its planned mode; whereas results from a magnetometer experiment that was planned to measure the magnitude and direction of magnetic fields become very difficult to interpret.

An extensive analysis was performed to discover the cause for nondeployment of the OGO booms (Ref. 57). However, no simple and direct a cause seems to have been found as in the case of the Nimbus failure. After a complete examination of all the possible causes of failure, including a test program to evaluate flight-type hardware under simulated normal and extreme environmental conditions, the following conclusions were reached:

- The latch spring on one of the solar array paddles failed to actuate properly and the paddle rebounded to a position where the spring torque was balanced by the electrical harness torque.

- One of the two 22-ft booms was released, but mechanical interference of the yoke and the back-up structure served to dissipate the energy of pre-load and the spring torque was insufficient to overcome the remaining resistance.

- One of the four 6-ft booms was released and partially deployed, but a combination of insufficient spring torque and high harness resistance coupled with possible interference of the omnidirectional antenna or its cabling with the structure prevented full deployment.

Several design changes are recommended in Ref. 57 to correct the OGO malfunctions. The Earth sensor should be relocated and redesigned so as to be able to track the Earth even if some booms do not deploy. Weaknesses in the deployment mechanisms should be corrected, principally by eliminating various mechanical resistances and by using more powerful springs. The springs powering the hinge joints of the OGO appendages were made of Elgiloy non-magnetic alloy, made by the Elgin Company, except for the springs in the outboard hinge of one of the 22-ft booms which were made of beryllium-copper alloy. The springs were incorporated into their respective booms and boom deployment tests made at low and ambient temperatures. It was also determined that thermal vacuum exposure had a negligible effect on spring performance (Ref. 57).

A possible materials problem may lie in the coating which was applied to the Elgiloy springs. They were coated with a baked sodium silicate base, molybdenum disulfide lubricant, Everlube 811B, to prevent cold welding in the vacuum environment. After an extended period of storage and exposure to low temperature, the appearance of a coating of hydrated sodium silicate was noted. (Ref. 57). The cause of this efflorescence has not been established; however, it is believed to be the result of a low bake temperature. A deposit such as this would be an additional source of resistance to the spring motion during deployment.

1.3.11 OSO (Orbiting Solar Observatory)

The OSO program has resulted in the successful launch of two satellites designed to measure electromagnetic radiation from the sun in the ultraviolet, x-ray, and γ-ray regions of the spectrum and to study time variations of the emissions. OSO-1 was put into orbit 7 March 1962 and transmitted data until 6 August 1963, being still in orbit but silent; its design lifetime was only from 3 to 6 months.

One of the most significant experiments carried aboard OSO-1 from a materials application standpoint was the one on thermal control materials stability. Specimens of six materials which had been proposed as coatings for spacecraft were flown on the satellite along with a reference blackbody and suitable instrumentation to monitor the solar absorptance and infrared emittance of the materials over a long period. These data are reported and compared with laboratory simulation testing in Ref. 58 furnishing an exceedingly valuable example of correlation between flight and simulation data.

The following specimens were tested:

(1) A423 Skyspar natural white enamel, a titanium dioxide pigmented epoxy formulated by Andrew Brown Paint Co., and applied as a coating 7-mil thick on a substrate of 6061 T-6 aluminum alloy.

(2) A titanium dioxide pigmented silicone base paint formulated at Marshall Space Flight Center and applied as a coating 2-mil thick to 6061 T-6 aluminum alloy.

(3) A fused white porcelain enamel consisting of alkali-titania boro-silicate glass prepared by the Ferro Corporation and baked on to 6061 T-6 aluminum as a 2.5-mil coating.

(4) A leafing aluminum paint consisting of aluminum powder in a silicone vehicle prepared by Goddard Space Flight Center and applied as a 1-mil coating to 6061 T-6 aluminum alloy.

(5) A layered construction with polished AISI 321 stainless steel substrate followed by opaque vapor-deposited aluminum, 1.1μ SiO, 110 Å germanium.

(6) A layered construction with polished AISI 321 stainless steel substrate followed by opaque vapor-deposited aluminum, 1.1μ SiO, 200 Å germanium, 500 Å SiO.

The initial radiation characteristics of the various surfaces as deduced from temperature measurements are shown in Table 1-15 for the flight data as well as laboratory measurements (Ref. 58). A close correlation between flight and laboratory measurements is apparent.

The flight data, showing increase in solar absorptance as a function of time expressed in equivalent sun hours, are shown in Fig. 1-2 (Ref. 58) for materials Specimens 1 through 4, while Fig. 1-3 (Ref. 58) indicates the change in absorptance for the two multi-layered coatings, Specimens 5 and 6, as a function of time in orbit. The latter two are compared to time in orbit rather than sun exposure since erosion by micrometeoroids was expected to be the degradation mechanism rather than solar ultraviolet.

When the flight degradation curves were further compared with those obtained by exposure to simulated ultraviolet sources in the laboratory the following results were obtained. In the case of the epoxy paint, it was found that an A-H 6 water-cooled lamp was too severe in simulation by a factor of $\sim$ 3 while a B-H 6 air-cooled lamp resulted in simulation too severe by a factor of $\sim$ 10. However, opposite results were obtained for the silicone base white paint, for which the A-H 6 lamp gave degradation only one-fifth that of the flight data while the B-H 6 lamp yielded about the same degradation as the flight data. The full explanation for these conflicting results is not yet known.

The OSO spacecraft is designed to spin for stabilization purposes while at the same time keep sensors pointed steadily toward the sun. These requirements make necessary a continuously operating bearing between the stationary and spinning parts of the satellite.

Table 1-15

THERMAL-RADIATION PROPERTIES OF OSO-1 TEST SURFACES

Surface	α_s/ϵ		ϵ		α_s	
	Laboratory	Flight	Laboratory	Flight	Laboratory	Flight
TiO_2 in epoxy	0.27	0.28	0.85	0.87	0.23	0.24
TiO_2 in silicone	0.36	0.32	0.76	0.76	0.27	0.24
White porcelain enamel	0.35	0.32	0.75	0.75	0.26	0.24
Al powder in silicone	0.96	0.96	0.26	0.25	0.25	0.24
Al-SiO-Ge	2.8	3.3	0.13[a]	0.11	0.36	0.36
Al-SiO-Ge-SiO	2.5	2.8	0.18[a]	0.17	0.45	0.47
Blackbody reference	1.04	0.98	0.93	0.96	0.97	0.94

(a) Total normal emittance.

On OSO-1 this connection was made through an aluminum alloy shaft supported by AISI 52100 ball bearings with retainers of a reinforced fluorocarbon compound, the bearings being lubricated with a molybdenum disulfide slurry. This bearing was highly successful in operating in the space environment through the unexpectedly long lifetime of the satellite.

For the second OSO satellite, launched 3 February 1965, this bearing will make use of a lubricating fluid (Ball Bros, proprietary material) impregnated into the main bearing retainer, which is not sealed from the space environment. This change was made due to manufacturing problems, not as a result of space environmental degradation.

Another change made between OSO-1 and -2 was the replacement of the filament-wound glass fiber reinforced resin gas bottles of the attitude control system of OSO-1 with titanium alloy 6Al-4V bottles on OSO-2. This change was made because of manufacturing difficulties that developed in the filament-winding for these relatively small pressure vessels.

OSO-1 carried Hoffman p-on-n solar cells with 6-mil Owens-Corning 0211 glass covers. Due to increased radiation environment prevailing at the present, OSO-2 used thin n-on-p solar cells made by Heliotek Co. and protected by 20-mil thick quartz covers.

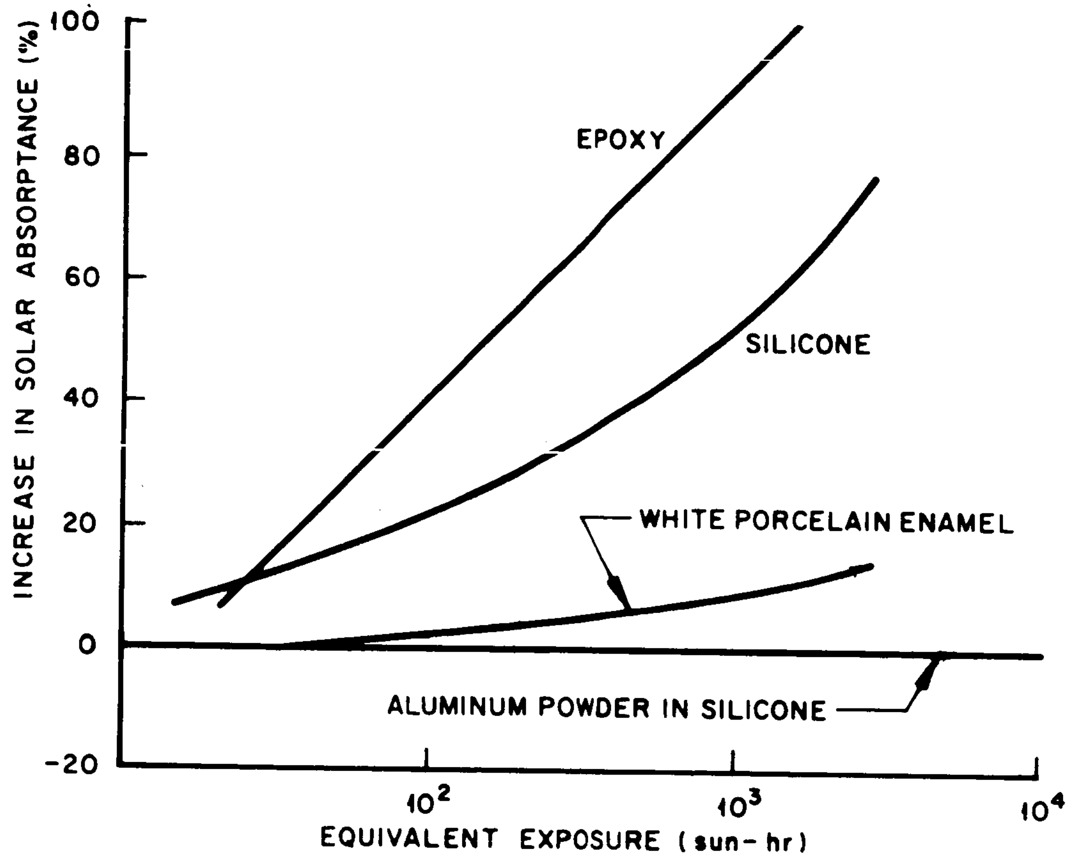

Fig. 1-2 Degradation of Thermal Control Coatings on OSO-1

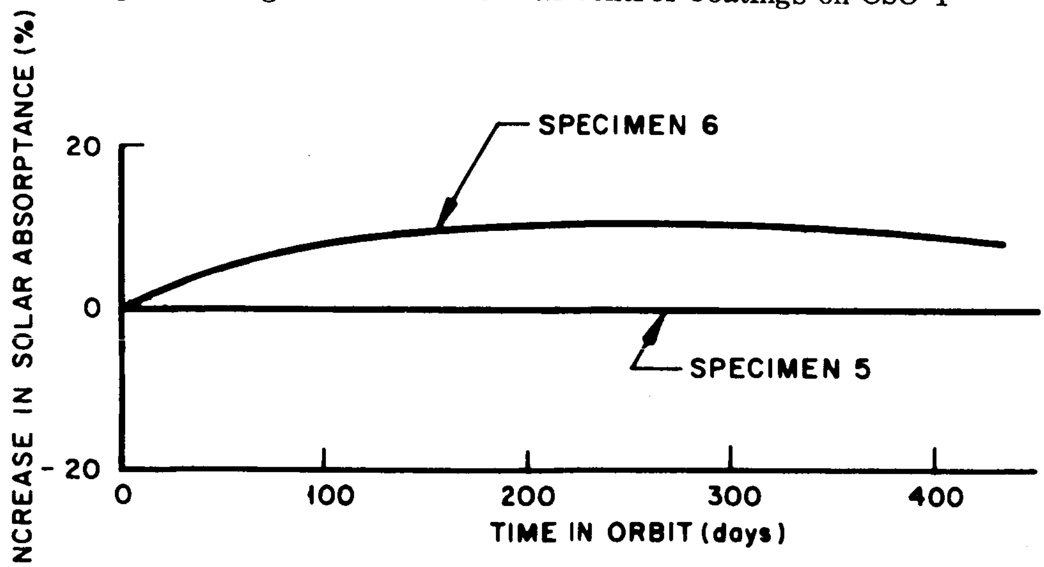

Fig. 1-3 Degradation of Multi-layered Thermal Control Materials on OSO-1

1.3.12 Pegasus

Successful launches utilizing the Saturn I booster have recently put into orbit no less than three large meteoroid detection satellites, designated as Pegasus. Launched on 16 February, 25 May, and 30 June, 1965, these three spacecraft are almost identical, their most distinguishing feature being the large "wings" which, when unfolded in orbit, expose more than 2000 ft^2 of detector surface to the near-Earth meteoroid flux.

The meteoroid detectors of Pegasus are the capacitor sandwiches first used on Explorer 23 and are described above. The materials of the Pegasus detector panels are listed in Tables 1-11 and 1-14. As noted in Sec. 1.3.6, one problem of these detectors is that the Mylar tends to build up a concentration of electrons which may discharge and cause spurious pulses to be identified as meteroid hits. Special discriminator circuitry has been built into the system to help differentiate between the two types pulses. Also ten of the panels are left without their bias voltage of 40 V; any pulses coming from these panels would be immediately recognized as spurious (Ref. 59).

No detailed reports have yet been published on the results obtained from the Pegasus spacecraft but fragmentary data indicate that Pegasus 2 experienced 73 meteoroid penetrations during the period, 25 May to 3 July (Ref. 60). The same source also reports that data from both Pegasus 1 and 2 indicate that meteoroid flux in space is less than previously expected. This is reported to point toward a saving of up to 100 lb of meteoroid shielding which can now be removed from the Apollo spacecraft.

The large surface area of the sensory panels of Pegasus is covered with an adaptation of the aluminum and chromium phosphate coating used on Echo 2. The version developed by Fairchild-Hiller Corp for Pegasus is designated MTL-3 and was developed with advice from NASA-Langley scientists. This phosphate coating aims at an α/ϵ of ~ 1.0 Table 1-6. Preliminary results from the Pegasus 1 indicate that it has been very successful in controlling temperatures within design limits (Ref. 61).

The cable harness wiring insulation for electronics (Table 1-11) used on Pegasus 1 differs from that used on the later Pegasus 2 and 3. Pegasus 1 harnesses utilized a Raychem Corp. wiring (Novathene) which is a Raychem product whose insulation is reportedly given special treatment for space applications. This insulation tended to outgas and to shrink appreciably when heated in vacuum. In addition, the relatively greater thickness of the insulation presented difficulties in maintaining desirably small harness volumes. On Pegasus 2 and 3, the harness wiring is Raychem Corp. Specification 44 Wire, based on Specification MIL-W-81044. This wire has a primary insulation of irradiated polyolefin topped with an outer jacket of polyvinylidene fluoride (Kynar). This insulation has been found to be quite abrasion resistant and not prone to significant shrinkage or outgassing when heated in vacuum (Ref. 61). In addition, the overall insulation thickness is less, permitting smaller overall harness bundle volume. For detector panel wiring see Table 1-13.

1.3.13 Ranger

The Ranger spacecraft and the closely related Mariner discussed in Sec. 1.3.8 above must be classified as relatively sophisticated systems when compared with the

Earth-orbiting satellites which have so far constituted the bulk of space research instruments. The necessity for mid-course trajectory correction coupled with relatively long-life requirements for all components constitute stringent limitations on the design of these spacecraft. On the one hand, the need for reliability calls for the use of proven materials and components, while on the other, their complex mission dictates the use of sophisticated and advanced yet conservative techniques of design and of materials selection.

The Ranger program has been plagued with perhaps more than its share of failures in the early stages with Ranger 3 through 6 failing to reach their objective of landing instruments on the Moon and/or returning pictures. The Ranger spacecraft 7, 8, and 9 have performed as planned in transmitting high-quality television pictures of the lunar surface.

As is so often the case with spacecraft failures, it becomes extremely difficult to localize the cause of the various Ranger failures to one specific subsystem, component, or material. Some reasonably informed guesses have been made on the basis of the limited data available and these failure analyses have resulted in some materials changes.

The most extensive design review of the Ranger program was made immediately following the failure of operation of the TV system in Ranger 6. Both design changes and materials changes resulted from this review.

The most likely candidate for the cause of failure of the Ranger 6 television was its premature turn-on during the first few minutes of flight of the launch vehicle. It was this brief unscheduled operation that is believed to have caused arcing between terminals in the relatively high-pressure atmosphere prevailing at that time. The circuitry was redesigned to prevent such premature operation and in addition all exposed terminals were conformal coated with a layer of Solithane 113 (an Isocyanate liquid polymer). Those terminals which had already been coated had thicker coatings installed. These changes followed from correlation with laboratory findings (Ref. 51) that corona discharge occurred at a pressure of 5×10^{-2} Torr with a 5-in. electrode spacing at 2000 V, 60 cps. In a further attack on this problem, junction connectors were filled with silicone rubber to cut down arcing.

The overall temperature history of Ranger 6 was somewhat higher than had been anticipated, and although this was not considered a direct factor in its failure, the thermal control paint pattern was changed for Ranger 7. For example, the Ranger 6 Earth sensor operated at a higher temperature than desired. A better conduction path existed between the Earth sensor and the yoke than had been believed. Changes on Ranger 7 added a shield to minimize the change in solar input with hinge angle. Shielding and changes in paint patterns resulted in performance in which, for each of the nineteen locations on the spacecraft from which the temperature was monitored, the observed flight temperatures were within the predicted temperature limits.

Other important changes which were made as a result of the Ranger 6 failure were the adding of insulation in critical areas, such as between transistor and chassis in the transmitter power supply, a review of the system assembly venting characteristics, and the spot-bonding of elements subject to vibration effects such as relays, and cable routing in assemblies (Refs. 51 and 62).

A structural change of materials made in Ranger beginning with Ranger 7 but not especially associated with the failure analysis was the change from gold-plated magnesium to 6061-T6 aluminum for the main bus support legs. This change was necessary due to the increased weight of the TV camera and electronics tower which made the magnesium design marginal as well as because of problems which had developed in gold plating the magnesium alloy previously used for the legs.

Still another correlation between laboratory simulation testing and Ranger design resulted in the change from nylon lacing cords for electronic wiring to Dacron cords. The Dacron showed better thermal-vacuum stability in laboratory testing.

Experience gained during the flight of the Mariner 2 to Venus has also resulted in a materials improvement in the Ranger spacecraft as well as in Mariner. Solar panels for the Venus Mariner were insulated with 1.5-mil Mylar and this was believed to have caused intermittent electrical shorts and partial intermittent but not disabling power failures. In Ranger, beginning with Ranger 6, a 4-mil thick sheet of more reliable fiber glass reinforced epoxy (SMP 62-63) was used to insulate the solar cells from the solar panel structure.

1.3.14 Relay

The two successfully launched Relay communications satellites of 13 December 1962 and 21 January 1964 have significantly advanced satellite communications technology. The most interesting materials information from this program is the solar cell degradation data obtained on both Relay 1 and 2. The satellites also measured radiation effects to support future satellite design (Ref. 63).

The radiation measurements included detectors for both protons and electrons and in addition included a collection of isolated solar cells with provision for measuring outputs of the short-circuited cells. One hundred of the 128 telemetry channels on the Relay 1 were reserved for monitoring cell performance.

Thirty silicon solar cells were divided into three groups of nine, plus one group of three. One of the three groups comprised three p-on-n cells protected by a 60-mil thick quartz cover, three cells with 30-mil quartz covers, and three without covers. The second of the three groups was similar to the first except that n-on-p cells were used. The third set was again similar except that special "reversed" p-on-n cells were used. These last cells had thin bases, and were especially vulnerable to radiation damage. Finally, the last three were special gallium-arsenide p-on-n cells with no covers.

Relay 1 was placed in an orbit of 820 sm perigee and ~ 4600 sm apogee; Relay 2 achieved almost the same apogee but a higher perigee of ~ 1300 sm, leading to an expectation of greater radiation exposure. Relay 2 carried some cells of the same type as Relay 1 but also included shielded gallium arsenide cells. Table 1-16 (Ref. 64) shows the degradation with time of the various groups of cells on both Relay 1 and 2. The anomalous behavior of the reversed p-on-n cells in showing an initial increase in output instead of the expected high-damage sensitivity is as yet unexplained (Ref. 64). As expected due to the higher perigee, cells on Relay 2 of the same type and with the same covers as those of Relay 1 experienced damage of slightly greater magnitude but with the same trend.

Table 1-16

RADIATION DAMAGE TO SOLAR CELLS ON RELAY 1 AND 2

Spacecraft	Cell Type	Cover[a] Thickness	Part of Initial Output Current (%)				
			After 0.1 day	After 1 day	After 10 days	After 100 days	After 300 days
Relay 1	p-on-n	0	67	41	28	21	14
	p-on-n	30	100	95	84	63	22
	p-on-n	60	100	97	87	68	58
	n-on-p	0	92	63	52	27	8
	n-on-p	30	100	100	95	81	72
	n-on-p	60	100	100	97	86	77
	GaAs	0	88	52	17	13	0
	Reversed p-on-n	0	98	36	12	7	3
	Reversed p-on-n	30	121	116	117	47	25
	Reversed p-on-n	60	115	112	120	120	78
Relay 2	Silicon p-on-n	60		96	85	70	
	Silicon	60		98	94	84	
	GaAs	60		98	94	92	
	GaAs	30		98	95	90	
	GaAs	12		98	94	82	
	GaAs	3		99	95	82	

(a) All covers were Corning 7940 silica except the 3-mil and 12-mil covers which were Corning 0211 glass.

A very interesting correlation between predicted damage to solar cells on the basis of simulation testing and the measured flight data of Relay 1 is reported in Ref. 64. Computer calculations were made of the damage expected in Relay 1 on the basis of its actual flight path when compared with tabulated values of the proton and electron fluxes in space and taking account of particle spectra, damage susceptibility of the cells as a function of particle energy, angle, and shielding. The calculations were performed separately for proton damage and for electron damage and are reported for 300 days in orbit and for silicon cells with 60-mil covers of quartz. The predicted final response for the p-on-n cells was 56% after proton damage or 47% after electron damage whereas the actual damage in orbit reduced the response to 60%. For n-on-p cells, the equivalent values were 75% residual response for either proton or electron damage as opposed to 79% measured in flight. The conclusion arrived at in Ref. 64 is that these cells were principally damaged by protons whose energy was > 17 MeV, with some contribution from electrons having energies above ~ 1 MeV.

1.3.15 Syncom

The first Syncom communications satellite was successfully orbited 14 February 1963 but all contact with the spacecraft was lost within 20 sec of firing of the apogee motor and attempts to re-establish communications were unsuccessful. Optical tracking has indicated that the satellite did go into the nominal 22,000-sm circular synchronous orbit which had been planned; however, the lack of communication made it useless.

Syncom 2 was successfully orbited 26 July 1963 and Syncom 3 on 19 August 1964. Both these vehicles have been used to carry on long distance communications experiments including television of the Tokyo Olympic Games.

The spacecraft vernier control system on Syncom 1 was nitrogen at 3850 psi in a pressure vessel made of 7 Al-4Mo titanium alloy. Post-launch analyses and testing have led to the conclusion that a failure of this vessel caused the loss of Syncom 1 (Ref. 65). On the second Syncom, the tank pressure was reduced to 2500 psi, while Syncom 3 incorporated hydrogen peroxide in all welded 1060-H12 aluminum alloy vessel for vernier control.

The electrical power system on both Syncoms 1 and 2 used silicon p-on-n type solar cells protected by 6-mil thick Corning 0211 glass covers; the cells on Syncom 3 were silicon n-on-p type. During the transfer ellipse period of ~ 6 hr when Syncom 2 was traversing the trapped radiation belts the solar cell array was degraded by 2% (Ref. 66). After 400 days in orbit at a nominal 22,000 sm, Syncom 2 retained approximately 75% of its original maximum power output from the solar cells. The predicted value had been 70% after this time (Ref. 67).

The authors of Ref. 67 make the point that although seventeen part failures occurred in ~ 1 yr of ground testing of Syncom 2, the spacecraft was subsequently operated over 1 yr in space with no failures at all. They attribute this improvement to the fact that people cannot touch or test the satellite in orbit and suggest the idea that perhaps the space environment may actually be safer for spacecraft than the ground environment from the point of view of reliability.

1.3.16 TRS (Tetrahedral Research Satellite)

This series of small research satellites are designed to carry materials research and space physics experiments and have been launched as pick-a-back operations in connection with Air Force research and development satellites. TRS 1 was launched during the last quarter of 1962 and following launches brought the total to 6 with the launch of TRS 6 on 17 July 1964.

The most significant experiment of the TRS was that to measure solar cell degradation in the space environment. Through appropriate analysis the rate of deterioration can yield a comparison between types of cells, a comparison between the effectiveness of various solar cell cover materials, damage done to various adhesives used to fasten the covers, conclusions regarding the intensity of electrons and protons, the intensity of low energy protons, and the lifetime of the artificial electron belt along the orbit of the TRS. Some of these results as obtained from TRS 2, TRS 3, and TRS 4 are reported in Ref. 68.

One rather unexpected result from these space experiments was that there was no change in transmission of either the quartz covers or the epoxy adhesive used to hold the covers to the solar cells. Based on laboratory experiments, a 15% loss in optical transmission had been predicted because of change in transmission of the adhesive.

Deterioration of 20% in power output of p-on-n silicon cells used in these observations occurred about five times as fast as the same deterioration in n-on-p cells when both types of cells were protected by 20-mil thick covers. The results also showed that when n-on-p cells were protected by covers of either 20-mil thick quartz,40-mil thick quartz, or 20-mil thick cerium glass, no difference of engineering significance could be detected in degradation rates. However, a more significant improvement in the use of 40-mil over 20-mil covers was indicated for p-on-n cells primarily because of their greater sensitivity to higher energy electrons than n-on-p cells.

A calculation was made of expected deterioration of the solar cells on the TRS satellites on the basis of the best available data on the radiation belts correlated with laboratory damage for silicon. These calculations showed that the damage to covered p-on-n solar cells was dominated by the proton exposure for 6-mil covers and dominated by the electron exposure for 20-mil covers. The n-on-p solar cells with their increased radiation resistance to electrons were always dominated by proton damage as expected.

The results of these calculations were compared with orbital data for which the 20-mil shielded cells show that about 5.5 times the p-on-n exposure is required to produce the same damage in the n-on-p cells. If the damage was produced exclusively by protons, this factor should be 3.0; if the damage resulted exclusively from 1 MeV electron bombardment, the ratio should be about 20.0. The observed ratio of 5.5 agrees with the electron and proton equivalent damage calculated from the radiation belt data thus supporting the experimental mix of charged particles as measured in other experiments.

Deterioration rates of ~ 4.5 mA/cm^2 for each decade of exposure time were expected from laboratory experiments whereas ~ 5.5 mA/cm^2-for each decade was measured on TRS flights. An analysis performed in Ref. 68 concludes that the higher rate in flight is a consequence of nonuniform defect distributions. The analysis also concluded that the flight results should be employed for spacecraft design rather than the laboratory deterioration rates, and recommends the rate of 5.5 mA/cm^2 for each decade of exposure time.

1.3.17 Tiros

Both the primary and secondary mission objectives for each of the orbited Tiros satellites have been met and in some cases, exceeded, particularly in the greater than expected lifetime of these satellites. This success is to be attributed in part to a design philosophy which has stressed system redundancy and the use of simple, rather than complex, devices. The performance of the electrical power supply, including the solar cells, batteries, circuits, and regulators has been good for each satellite. No direct failures have occurred in any of the power supplies, and the power output has always been adequate for the programming scheduled (Ref. 69). The passive thermal-control system has performed as predicted with the electronic units operating within the desired temperature ranges of 10 to 25°C during 90% of the orbits. For example, during the first 1800 orbits of Tiros 2, comparison of calculated and measured component temperatures showed deviation no more than about 5°C of the telemetered data from that calculated (Ref. 70). Exposure to the space environment for more than the design life of 3 months did result in a reduction in the thrust of the five pairs of small, solid-fuel, spin-up rockets of the first four Tiros satellites. Improved solid-fuel spin-up rockets were used in the Tiros 5 and succeeding vehicles and provided the required thrust even after being exposed to an orbital environment for 8 to 10 months (Ref. 69).

During the periods of operation of 4 months for Tiros 1, 17 months for Tiros 2, and ~ 8 months for other satellites, no detectable damage was done to camera lenses by meteoroid bombardment.

The record for space longevity of the Tiros family is held by Tiros 7 which was launched on 19 June 1963 and was still operating as of 19 May 1965. Tiros 7 had a minor telemetry problem in that a commutator became stuck on the electron temperature probe position so that no "housekeeping" data could be transmitted from that station (Ref. 71). Tiros 8 is also still functioning and transmitting pictures from its one standard Tiros 104-deg. camera. Tiros 8 also carried the first 108-deg Automatic Picture Transmission camera which was designed for only a short life and is no longer functional.

1.4 REFERENCES

1.4.1 Cited References

1. D. G. King - Hele and Eileen Quinn, "Table of the Artificial Earth Satellites Launched in 1963," Planetary and Space Science, 12, 681–701, July 1964

2. Space Log, 4, No. 4, 52–63 (1964)

3. Space Log, 5, No. 1, 34–37 (1965)

4. C. G. Goetzel, J. B. Rittenhouse, and J. B. Singletary, Space Materials Handbook, Air Force Materials Laboratory, Wright-Patterson Air Force Base, Ohio, ML-TDR-64-40, [AD-640399] [X65-14878], January 1965 [also published in a slightly abridged version by Addison-Wesley Publishing Co., Reading, Mass., 1965]

5. C. D. Florida, "Alouette's Successors," presented at 131st Annual Meeting, AAAS, Montreal, Canada, 26–31 December 1964

6. R. L. Samuels, "The Extension, Profile, and Stability of Very Long Booms in Spacecraft Experimentation," Record of the International Space Electronics Symposium, 6–4 October 1964, Las Vegas, Nev., Institute of Electrical and Electronics Engineers, 1964

7. Space Log, 2, No. 4, 28 (1962)

8. Scott A. Mills (ed) The Environmental Test Program and System Evaluation of the Canadian S-27 (Topside Sounder) Spacecraft, Goddard Space Flight Center, Greenbelt, Md., NASA-TM-X-50 194: NASA-X-326-63-82 [X63-14249] 30 April 1963

9. John H. Chapman, "Alouette Topside Sounder Satellite: Experiments, Data, and Results," J. Spacecraft and Rockets, 1, 684–686, November–December 1964

10. M. O. Robins, "The Ariel 1 Satellite Project and Some Scientific Results," presented at 9th Anglo-American Conference American Institute of Aeronautics and Astronautics, Canadian Aeronautics and Space Institute, and Royal Aeronautical Society, Cambridge, Mass., 16–18 October and Montreal, 21–24 October 1963

11. R. C. Baumann, "The Ariel 1 Satellite," Royal Society (London) Proceedings, 281A, 439–445, October 1964

12. Warner H. Hord, Jr., Environmental Test Program for Ariel 1, Goddard Space Flight Center, Greenbelt, Md., NASA-TN-D-2099 [N64-15220] February 1964

13. Courier Satellite: Final Report, Philco Corp. Western Development Labs., Palo Alto, Calif., WDL-TR-1399 [AD-403881] [N64-14702] January 1963

14. S. Montgomery, "Early Bird Going into Business," Missiles and Rockets, 16, No. 26, 24 (1965)

15. "Early Bird Speeds Chart Transmission," Missiles and Rockets, 17, No. 1, 35 (1965)

16. Richard M. Bentley, "Early Bird," Astronautics and Aeronautics, 3, 26–29, March 1965

17. G. C. Westrick and K. G. Johnson, The Orbital Behavior of the Echo I Satellite and its Rocket Casing During the First 500 Days, Langley Research Center, Hampton, Va., NASA-TN-D-1366, June 1962

18. Richard H. Emmons, Photometric Measurements of Surface Characteristics of Echo I Satellite, Goodyear Aerospace Corp., Akron, Ohio, NASA-CR-58240: GER-11648 [N64-29572] June 1964

19. W. B. Fichter, Buckling of the Echo A-12 Passive Communications Satellite, Langley Research Center, Hampton, Va., NASA-TN-D-2353, July 1964

20. Claude W. Coffee, Jr., Walter E. Bressette, and Gerald M. Keating, Design of the NASA Lightweight Inflatable Satellites for the Determination of Atmospheric Density at Extreme Altitudes, Langley Research Center, Hampton, Va., NASA-TN-D-1243, April 1962

21. Howard L. Price, Langley Research Center, Hampton, Va., personal communication, 20 May 1965

22. Chas. V. Woerner, Properties of Two White Paints for Application to Inflatable Spacecraft; Titanium Dioxide Pigmented Epoxy and Zinc Oxide Pigmented Methyl Silicone Elastomer, Langley Research Center, Hampton, Va., NASA-TN-D-2834, June 1965

23. J. A. Pilkington, "Explorer 9's Final Record," Spaceflight, 7, No. 2, 61 (1965)

24. Chas. V. Woerner and Gerald M. Keating, Temperature Control of the Explorer IX Satellite, Langley Research Center, Hampton, Va., NASA-TN-D-1369, July 1962

25. Chas. V. Woerner and Claude W. Coffee Jr., Comparison of Ground Tests and Orbital Launch Results for the Explorer IX and Explorer XIX Satellites, Langley Research Center, Hampton, Va., NASA-TN-D-24066 [N64-28233] September 1964

26. P. C. Donnelly and C. F. Palondati, Silver Zinc Batteries Power Supply for the Atmosphere Structure Satellite Explorer XVII, Goddard Space Flight Center, Greenbelt, Md., NASA-X-636-63-206: NASA-TM-X-51150 [N64-10905] October 1963

27. A. H. Smith, "Power Sources," Space Programs Summary No. 37-17, Vol. IV, pp. 103–115, Jet Propulsion Laboratory, Pasadena, Calif., October 1962

28. Space Log, 3, No. 1, 28 (1963)

29. Eugene Ehrlich, "NASA Particles and Fields Spacecraft," presented at 1st Annual Meeting, American Institute of Aeronautics and Astronautics, Washington, D.C., 29 June–2 July 1964

30. "NASA To Launch 2nd Interplanetary Explorer Satellite," News Release No. 64-242, [N64-32635] 29 September 1964

31. "IMP-2, Interplanetary Monitoring Platform," Space World, B-1-15 40-42, January 1965

32. "NASA Assesses Value of Second IMP Data," Aviation Week & Space Technology, 81, No. 15, 24, October 1964

33. M. Getler, "Ionospheric Satellite Aiding Design of Long Antenna Systems," Missiles and Rockets 16, No. 1, 34–35, January 1965

34. C. T. D'Aiutolo, The Micrometeoroid Satellite Explorer XIII (1961 Chi) Langley Research Center, Hampton, Va., NASA-TN-D-2468 [N65-12203] November 1964

35. Micrometeoroid Satellite (Explorer XIII) Stainless-Steel Penetration Rate Experiment, Lewis Research Center, Cleveland, Ohio, NASA-TN-D-1986 [N64-10182] October 1963

36. E. C. Hastings, Jr., NASA–GSFC Operations Plan 8-62, Micrometeoroid Satellite (S-55B), Goddard Space Flight Center, Greenbelt, Md., NASA-TM-X-50152: NASA-X-533-62-209 [N63-14055] November 1962

37. E. C. Hastings, Jr., The Explorer XVI Micrometeoroid Satellite. Supplement III, Preliminary Results for the Period May 27, 1963 Through July 22, 1963, Langley Research Center, Hampton, Va., NASA-TM-X-949 [N64-16692] March 1964

38. E. C. Hastings, Jr., The Explorer XVI Micrometeoroid Satellite–Description and Preliminary Results for the Period December 16, 1962 Through January 13, 1963, Langley Research Center, Hampton, Va., NASA-TM-X-810, 1963

39. E. C. Hastings, Jr., The Explorer XVI Micrometeoroid Satellite, Supplement II, Preliminary Results for the Period March 3, 1963 Through May 26, 1963, Langley Research Center, Hampton, Va., NASA-TM-X-899 [N63-20334] September 1963

40. E. H. Davison and P. C. Winslow, Jr., Micrometeoroid Satellite (Explorer XVI) Stainless-Steel Penetration Rate Experiment, Lewis Research Center, Cleveland, Ohio, NASA-TN-D-2445 [N64-27204] August 1964

41. J. R. Davidson and P. E. Sandorff, Environmental Problems of Space Flight Structures–II Meteoroid Hazard, NASA-TN-D-1493, 1963

42. J. L. Patterson and W. E. Miller, "Protecting Explorer XVI Solar Cells," Astronautics and Aerospace Engineering, 1, No. 9, 48–51 (1963)

43. D. E. Fink, "Explorer 23 Furnishing Data for Pegasus," Aviation Week & Space Technology, 81, No. 22, 53–55 (1964)

44. C. L. Hemengway and I. Linscott, "Preliminary Studies of Possible Cosmic Dust Impacts on Project Mercury Vehicle Periscope Lenses," Annals of the New York Academy of Sciences, 119, Art. 1, 106–115 (1964)

45. H. Dotts, "Materials and Fabrication Methods Used in the Gemini Spacecraft," Metal Progress, 83, No. 3, 66–70 (1963)

46. E. C. McCandlish, McDonnell Aircraft Co., St. Louis, Mo., personal communication, 12 February 1965

47. E. K. Welhart, McDonnell Aircraft Co., St. Louis, Mo., personal communication, 12 February 1965

48. G. Salisbury, Investigation of Non-Metallic Materials for Manned Space Vehicles, McDonnell Aircraft Co., St. Louis, Mo., Report A081 [AD 417302] August 1960

49. W. J. Olewinski, G. Rapier, T. K. Slawecki, and H. Warner, Investigation of Toxic Properties of Materials Used in Space Vehicles, General Electric Co., Philadelphia, Pa., Report 63SD795: AMRL TDR 63-99 [AD-431149] January 1963

50. "No Defects Located in Gemini-4 Computer," Aviation Week & Space Technology, 82, No. 26, 29 (1965)

51. Robert Boundy, Jet Propulsion Laboratory, Pasadena, Calif., personal communication, 23 November 1964

52. Mariner C Shroud Failure Evaluation, Lockheed Missiles & Space Co., Sunnyvale, Calif., LMSC-A730531, January 1965

53. Robert Freeman, Jet Propulsion Laboratory, Pasadena, Calif., personal communication, 23 November 1964

54. J. A. Zoutendyk, R. J. Vondra and A. H. Smith, Mariner 2 Solar Panel Design and Flight Performance, Jet Propulsion Laboratory, Pasadena, Calif., NASA-CR-51514: JPL TR-32-455 [N63-22166] June 1963

55. "Close Tracking of Mariner Continues," Missiles and Rockets, 17, No. 3, 14–15 (1965)

56. T. Neville, General Electric Co., Valley Forge, Pa., personal communication, 15 June 1965

57. Final Report, Investigation of OGO-A Malfunctions, Space Technology Laboratories, Redondo Beach, Calif., Report 2311-6094-SU-000, December 1964

58. C. B. Neel, Research on the Stability of Thermal-Control Coatings for Spacecraft, Ames Research Center, Moffett Field, Calif., NASA-TM-X-51196 [X64-10410] 1963

59. M. Getler, "Hope Grows for Follow-on Pegasus," Missiles and Rockets, 16, No. 8, 14–15 (1965)

60. "Apollo Weight May be Cut," Missiles and Rockets, 17, No. 3, 10 (1965)

61. R. C. Downey, Fairchild-Hiller Corp., Hagerstown, Md., personal communication, 19 May 1965

62. H. M. Schurmeier, Ranger VII: Mission Description and Performance, Part I, Jet Propulsion Laboratory, Pasadena, Calif., JPL-TR-32-700: NASA CR-59904 [N65-13280] December 1964

63. J. D. Kiesling, "The NASA Relay 1 Experimental Communication Satellite, RCA Review, 25, 232–261 (1964)

64. R. C. Waddel, "Radiation Damage to Solar Cells on Relay I and Relay II," IEEE Trans. Nucl. Science, NS-11, No. 5, 60–68 (1964)

65. R. M. Bentley and A. T. Owens, SYNCOM Satellite Program," J. Spacecraft, 1, No. 4, 395–399 (1964)

66. P. S. DuPont, "Syncom 2 Electrical Power System," paper 64–456, presented at 1st Annual Meeting, AIAA, Washington, D.C., June 29–2 July 1964

67. E. J. Althaus and J. C. Meyer, "Syncom Reliability," Proceedings Eleventh National Symposium on Reliability & Quality Control, Miami Beach, Florida, Jan. 12–14, 1965, Institute of Electrical and Electronic Engineers, New York, 1965

68. Final Flight Report, Tetrahedral Research Satellites 1963-14B, 1963-14C, 1963-30B, Vol. I, II, TRW Space Technology Laboratories, Redondo Beach, Calif., Report 8685-6006-RU-000, February 1964

69. A. Schnapf, "Tiros-The Television and Infra-Red Observation Satellite," J. British Interplanetary Society, 19, 386–408, May–June 1964

70. E. A. Goldberg, "Tiros Preflight Testing and Postlaunch Evaluation, J. Spacecraft, 1, No. 4, 374–380 (1964)

71. R. Rados, Goddard Space Flight Center, Greenbelt, Md., personal communication, 17 May 1965

1.4.2 References Used in Preparing Tables 1-6 Through 1-14

"Air Density Injun Explorer B (Explorers 24-25)," Space World, B-4-18, 10–12, April 1965

Althaus, E. J. and Meyer, J. C., "Syncom Reliability," Proceedings Eleventh National Symposium on Reliability & Quality Control, Miami Beach, Florida, 12-14 January, 1965, Institute of Electrical and Electronic Engineers, New York, 1965

Baranovskaya, I. V. and Gyurdzhian, A. A., "Containers for Certain Biological Materials Used in Experiments in Spaceships," in Problems of Space Biology, National Aeronautics and Space Administration, Washington, D.C., NASA-TT-F-174 [N64-11702] November 1963

Baumann, Robt. C., The Ariel I Satellite, Goddard Space Flight Center, Greenbelt, Md., NASA-TM-X-51621: X-673-63-75 [N64-16700] April 1963

Baumann, Robt. C. Ariel I, The First International Satellite: Project Summary, Goddard Space Flight Center, Greenbelt, Md., NASA SP-43 [N64-10332] 1963

Bartko, F. Kunde, V., Catoe, C., and Halev, M., The Tiros Low Resolution Radiometer, Goddard Space Flight Center, Greenbelt, Md., NASA-TN-D-614, September 1964

Barath, F. T., A. H. Barrett, J. Copeland, D. E Jones, and A. E. Lilley, "Mariner II Microwave Radiometer Experiment and Results," Astron. J. 69, No. 1, 49–58 (1964)

"Beacon Explorer," Space World, B-1-15, 17–19, January 1965

Bentley, Richard M., "Early Bird," Astronautics and Aeronautics, 3, 26–29, March 1963

Bourdeau, R. E., "Description of S-66 Spacecraft," COSPAR Information Bulletin, No. 15, 45–55, May 1963

Boundy, Robt., Jet Propulsion Laboratory, Pasadena, Calif., personal communication, 23 November 1964

Bush, E. G., The Use of Solid Circuits in Satellite Instrumentation, Goddard Space Flight Center, Greenbelt, Md., NASA-TN-D-1758, July 1964

Burdett, Jerry, Goddard Space Flight Center, Greenbelt, Md., personal communication, 15 October 1964

Carpenter, J. F., McDonnell Aircraft Co., St. Louis, Mo., personal communication, 12 February 1965

Carr, Frank A., The Environmental Test Program and Systems Evaluation of the S-3 Energetic Particles Satellite, Goddard Space Flight Center, Greenbelt, Md., NASA-TM-X-50853: NASA-X-326-63-10 [N63-23034] January 1963

Chafey, J., Fairchild-Hiller Corp., Hagerstown, Md., personal communication, 19 May 1965

Chapman, J. H., "The Alouette Topside Sounder Satellite," COSPAR Information Bulletin, No. 14, 67–76, March 1963

Chapman, J. H., "Topside Sounding of the Ionosphere," Advances in the Astronautical Sciences, Vol. 12, American Astronautical Society, 1962

Clemmons, D. L. Jr., and F. D. Camp, "Amorphous Phosphate Coatings for Thermal Control of Echo II," Electrochemical Technology, 2, No. 7–8, 221–232 (1964)

Cherry, W. R., "Solar Energy Systems for Space Application," pp. 12–15, Proceedings, 17th Annual Power Sources Conference, Ft. Monmouth, N. J., 21–23 May 1963

Coffee, Claude W. Jr., Walter E. Bressette, and Gerald M. Keating, Design of the NASA Lightweight Inflatable Satellites for the Determination of Atmospheric Density at Extreme Altitudes, Langley Research Center, Hampton, Va., NASA-TN-D-1243, April 1962

Colner, W. H., Hughes Aircraft Co., Culver City, Calif., personal communication, 29 March 1965

Courier Satellite: Final Report, Philco Corp. Western Development Labs., Palo Alto, Calif., WDL-TR 1399 [AD-463881] (N64-14702] January 1963

Crumpton, B., Ball Bros. Research Corp., Boulder, Colo., personal communication, 9 February 1965

D'Aiutolo, Chas. T., The Micrometeoroid Satellite Explorer XIII (1961 Chi), Langley Research Center, Hampton, Va., NASA-TN-D-2468 [N65-12203] November 1964

Davison, Elmer H. and Paul Winslow Jr., Micrometeoroid Satellite (Explorer XVI) Stainless-Steel Penetration Rate Experiment, Lewis Research Center, Cleveland, Ohio, NASA-TN-D-2445 [N64-27204] August 1964

Denney, Joseph M., TRW Systems, Inc., Redondo Beach, Calif., personal communication, 25 November 1964

Di Girolamo, R. and J. P. Strong, Jr., "Special Aspects of Nimbus Spacecraft Integration," Proceedings of the Nimbus Program Review, Valley Forge, Pa., 14–16 November 1962, Goddard Space Flight Center, Greenbelt, Md., NASA-TM-X-50151: NASA-X-650-62-226 [N63-18601] 1962

"Design and Development," pp. 37–44 in Space Programs Summary No. 37-29, Vol. 6, Jet Propulsion Laboratory, Pasadena, Calif., 30 September 1964

Donnelly, P. C. and C. F. Palandati, Silver Zinc Batteries, Power Supply for the Atmosphere Structure Satellite Explorer XVII, Goddard Space Flight Center, Greenbelt, Md., NASA-X-636-63-206: NASA-TM-X-51150 [N64-10905] October 1965

Dotts, H., "Materials and Fabrication Methods Used in the Gemini Spacecraft," Metal Progress, 83, No. 3, 66–7, March 1963

Downey, R. C., Fairchild-Hiller Corp., Hagerstown, Md., personal communication, 19 May 1965

Drummond, F. and P. Crossfield, "Thermal Control of the Nimbus Spacecraft," Proceedings of the Nimbus Program Review, Valley Forge, Pa., 14–16 November 1962, Goddard Space Flight Center, Greenbelt, Md., NASA-TM-X-50151: NASA-X-650-62-626 [N63-18601] 1962

Dunipace, D. W., Ball Bros. Research Corp., Boulder, Colo., personal communication, 9 February 1965

DuPont, P. S., "Symcom 2 Electrical Power System," paper 64-456, 1st Annual Meeting, American Institute of Aeronautics and Astronautics, Washington, D.C., 29 June–2 July 1964

Ehrlich, Eugene, "NASA Particles and Fields Spacecraft," paper 64-337, 1st Annual Meeting, American Institute of Aeronautics and Astronautics, Washington, D.C., 29 June–2 July 1964

Ely, R., Hughes Aircraft Co., Culver City, Calif., personal communication, 29 March 1965

Evans, H. E., "Space Lubrication Problems and Solutions: Scientific Satellites EGO, POGO, OAO,NIMBUS, TIROS, ARENTS," in Proceedings of Space Lubrication Conference, Office of the Director of Defense Research and Engineering, Washington, D.C., FL 210/53 [AD 439844] September 1963

Farnsworth, D., "Passive Thermal Systems for Spacecraft Temperature Control Using Chromium Phosphate Immersion Coatings," presented to American Ceramic Society, Philadelphia, Pa., 5 May 1965

Final Flight Report, Tetrahedral Research Satellites, Space Technology Laboratories, Redondo Beach, Calif., Report 8685-6006-RU-000 Vols. 1 and 2, February 1964

Final Report, Investigation of OGO-A Malfunctions, Space Technology Laboratories, Redondo Beach, Calif., Report 2311-6094-SU-00, December 1964

First in the World Group Flight in Cosmic Space (Basic Results) Foreign Technology Div., Air Force Systems Command, Wright-Patterson Air Force Base, Ohio, FTD-TT-62-1578/1+2 [AD 292620] [N64-12055] November 1962

Fixed Frequency Ionosphere Topside Sounder Satellite, S-48, National Aeronautics and Space Adminstration, Washington, D.C., NASA-TM-X-50833, [X63-15846] February 1962

Freeman, Robt., Jet Propulsion Laboratory, Pasadena, Calif., personal communication, 23 November 1964

"Friendship 7's Voice Cable: A Story of Reliability," J. Teflon, 3, No. 7, 3 (1962)

Fulk, M., Ball Bros. Research Corp., Boulder, Colo., personal communication, 9 February 1965

Farnsworth, D., Fairchild-Hiller Corp., Hagerstown, Md., personal communication, 19 May 1965

Fink, D. E., "Explorer 23 Furnishing Data for Pegasus," Aviation Week & Space Technology, 81, No. 22, 53–55 (1964)

Forney, R. G., Mariner II–Attitude Control System, Jet Propulsion Laboratory, Pasadena, Calif., NASA-CR-52612, 1963

Ganssle, E. R., "Structural Design of the Meteoroid Technology Satellite," pp. 35–42 in AIAA Unmanned Spacecraft Meeting, Los Angeles, Calif., March 1–4, 1965, American Institute of Aeronautics and Astronautics, New York, 1965

Gatzek, L. E., "Design Factors for Materials Application in Spacecraft," lecture notes presented at University of California at Los Angeles, 22–26 June 1964

Getler, M., "Hope Grows for Follow-on Pegasus," Missiles and Rockets, 16, No. 8, 14–15 (1965)

Getler, M., "Ionospheric Satellites Aiding Design of Long Antenna Systems," Missiles and Rockets, 16, No. 1, 34–35 (1965)

Ginner, L., Lockheed Missiles & Space Co., Sunnyvale, Calif., personal communication, 4 August 1965

Goldberg, Jerry, Goddard Space Flight Center, Greenbelt, Md., personal communication, 15 October 1964

Goldberg, E. A., "Tiros Preflight Testing and Postlaunch Evaluation," J. Spacecraft, 1, No. 4, 374–380, July–August 1964

Hammel, R. L., TRW Systems, Inc., Redondo Beach, Calif., personal communication, 25 November 1964

Harris, M., Report on BBRC Radiation Resistant Sensors (S-17 Mod III Sensors), Ball Bros. Research Corp., Boulder, Colo., TN 64-22, May 1964

Hastings, Earl C., Jr., "First Results from the Explorer XVI Micrometeoroid Satellite," American Geophysical Union, Transactions, 44, 648–652 (1963)

Hastings, Earl C., The Explorer XVI Micrometeoroid Satellite: Supplement II, Preliminary Results for the Period March 3, 1963 Through May 26, 1963, Langley Research Center, Hampton, Va., NASA-TM-X-899 [N63-20334] September 1963

Heindl, J. C. and Belanger, R. J., OGO Solar Array Drive and Shaft Support Bearing Tests, Space Technology Laboratories, Redondo Beach, Calif., TR-2311-6026-RU-000, September 1962

Hord, Warner, H. Jr., Environmental Test Program for Ariel I, Goddard Space Flight Center, Greenbelt, Md., NASA-TN-D-2099 [N64-15220] February 1964

Huston, W., Goddard Space Flight Center, Greenbelt, Md., personal communication, 15 October 1964

Islinger, J. S., "Sealing Spacecraft Against Vacuum," Research/Development, 14, 62–66, October 1963

James, T. G., Effects of Electron Irradiation on Some Properties of the Echo II Laminate, Langley Research Center, Hampton, Va., NASA-TN-D-2207, October 1964

Jones, D. P., "Epoxies for the Tiros Satellite," Adhesives Age, 4, No. 5, 28–29, (1961)

Kershner, R. B., "The Transit Program," Astronautics, 5, No. 6, 30–31 (1960)

Kidwell, R. E., Goddard Space Flight Center, Greenbelt, Md., personal communication, 17 May 1965

Kiesling, J. D., "The NASA Relay 1 Experimental Communication Satellite," RCA Review, 25, 232–261 (1964)

Krausz, A., and Robinson, R. L., "The Electric Power Supply of the Orbiting Geophysical Observatory," AIAA Unamnned Spacecraft Meeting, Pub. CP-12, March 1965

Kunde, V. G., Tiros IV Radiation Data Catalog and Users Manual, Goddard Space Flight Center, Greenbelt, Md., NASA-TM-X-51826 [N64-27404] December 1963

Logan, Wm., Goddard Space Flight Center, Greenbelt, Md., personal communication, 17 May 1965

London, A., "Thermal Control of the Nimbus Satellite Control Systems," AIAA Unmanned Spacecraft Meeting, Los Angeles, Calif., March 1–4, 1965, American Institute of Aeronautics and Astronautics, New York, CP-12, 1965

Ludwig, G. H., The Orbiting Geophysical Observatories, Goddard Space Flight Center, Greenbelt, Md., NASA-TN-D-2646 [N65-18616] March 1965

Mackie, G. H. C., Construction Techniques and Inspection Procedures for the S-27 Alouette Satellite Payload, Defense Research Telecommunications Establishment (Canada), DRTE 1119, July 1963

Marcotte, Paul G., IMP D&E Feasibility Study, Goddard Space Flight Center, Greenbelt, Md., NASA-TM-X-55166: X-672-64-4 [N65-19844] January 1964

"Mariner 2 Subsystem Performance," pp. 22–27, in Space Programs Summary No. 37-20, Vol. 6, Jet Propulsion Laboratory, Pasadena, Calif., 30 April 1963

"Mariner C and D," Space World, B-1-15, 5–10, January 1965

"Mariner Mars 1964 Design," pp. 27–34, in Space Programs Summary No. 37-20, Vol. 6, Jet Propulsion Laboratory, Pasadena, Calif., 30 April 1963

Markowitz, L., Radio Corp. of America, Princeton, N. J., personal communication, 14 June 1965

Mesmeringer, J. C., "Electrical Solar Power Supply System Considerations for UK-2/S-52 International Satellite," Proceedings, 10th Annual East Coast Conference on Aerospace and Navigational Electronics, Baltimore, Md., 21–23 October 1963, Western Periodicals Co., North Hollywood, Calif., 1963

Michelson, L., Nimbus Meteorological Satellite–Integration and Testing Materials Report, General Electric, Missile and Space Vehicle Dept., Philadelphia, Pa., Report 61SD4325, December 1961

Michelson, L., Nimbus Meteorological Satellite–Integration and Testing Materials Report No. 3, General Electric, Missile and Space Division, Philadelphia, Pa., Report 63SD4206, February 1963

Michelson, L., Nimbus Meteorological Satellite–Integration and Testing Materials Report No. 2, General Electric, Missile and Space Division, Philadelphia, Pa., Report 62SD4287, July 1962

Mills, Scott A. (ed.) The Environmental Test Program and System Evaluation of the Canadian S-27 (Topside Sounder) Spacecraft, Goddard Space Flight Center, Greenbelt, Md., NASA-TM-X-50194: NASA-X-326-63-82, 30 April 1963

McCandlish, E. C., McDonnell Aircraft Co., St. Louis, Mo., personal communication, 12 February 1963

McKenzie, H., Ball Bros. Research Corp., Boulder, Colo., personal communication, 9 February 1965

McClellan, D. C., H. Sherman, and P. Waldron, The Lincoln Experimental Satellite, Massachusetts Institute of Technology, MIT-LL-GR-1964-55, October 1964

"NASA Set to Launch Second Interplanetary Monitoring Platform," Missiles and Rockets, 14, No. 14, 18 (1964)

Neel, C. B., Research on the Stability of Thermal-Control Coating for Spacecraft, Ames Research Center, Moffett Field, Calif., NASA-TM-X-51196 [X64-10410] 1963

Neville, T., Nimbus Meteorological Satellite—Integration and Testing Materials Report No. 7, General Electric, Spacecraft Dept., Philadelphia, Pa., Report 65SD4215, January 1965

Neville, T., Nimbus Meteorological Satellite—Integration and Testing Materials Report No. 4, General Electric, Missile and Space Division, Philadelphia, Pa., Report 63SD4266, July 1963

Neville, T., Nimbus Meteorological Satellite—Integration and Testing Materials Report No. 6, General Electric, Spacecraft Dept., Philadelphia, Pa., Report 64SD4318, July 1964

Neville, T., Nimbus Meteorological Satellite—Integration and Testing Materials Report No. 5, General Electric, Spacecraft Dept., Philadelphia, Pa., Report 64SD4213, January 1964

Neville, T., General Electric, Spacecraft Dept., Philadelphia, Pa., personal communication, 9 June 1965

News Release No. 65-38, "Saturn I to Launch Pegasus Meteoroid Detection Satellite," [N65-16639] 15 February 1965

News Release No. 64-242, "NASA To Launch 2nd Interplanetary Explorer Satellite," [N64-32635] 29 September 1964

Nichols, Harry, Goddard Space Flight Center, Greenbelt, Md., personal communication, 17 May 1965

Nuclear Detection Satellite, Space Technology Laboratories, Redondo Beach, Calif., Promotional Brochure 35-1976-7-64 (1964)

Orbiting Geophysical Observatories S-49, S-50 Payload Description Document, National Aeronautics and Space Administration, Washington, D.C., NASA-TM-X-50488 [X63-15678] May 1962

Osgood, C. C., "The Dynamical Design of the Relay Satellite," pp. 21–25 in Systems Engineering and Space Technology, Radio Corporation of America, New York, 1964

Patterson, John L. and Wm. E. Miller, Protecting Explorer XVI Solar Cells, Langley Research Center, Hampton, Va., NASA-RP-115, October 1963

Press, Harry, "The Nimbus Spacecraft System," Conference on Artificial Satellites, Virginia Polytechnical Institute, Blacksburg, Va., August 1964, Goddard Space Flight Center, Greenbelt, Md., [N65-15494] August 1964

Price, Howard L., Langley Research Center, Hampton, Va., personal communication, 20 May 1965

Price, Howard L. and Geo. F. Pezdirtz, Mechanical Properties of Echo II Laminate, Langley Research Center, Hampton, Va., NASA-TN-D-2367 [N64-25543] August 1964

"Radiation Detection Satellite (Explorer 26)," Space World, B-4-18, 13–16, April 1965

Rados, R. M., "Tiros – The First Meteorological Satellite," in Virginia Polytechnical Institute Conference on Artificial Satellites, Pt. B [N65-15493] August 1964

"Ranger Design Review," pp. 5–10 in Space Programs Summary No. 37-20, Vol. 6, Jet Propulsion Laboratory, Pasadena, Calif., 30 April 1963

Ranger 1964, Jet Propulsion Laboratory, Pasadena, Calif., NASA-CR-56289 [N64-22461] (1964)

"Ranger VII, Moon Photo Flight," Space World, A-12, 5–9, October 1964

Richard, V., Fairchild-Hiller Corp., Bayshore, L.I., N. Y., personal communication, 9 June 1965

Robinson, M., Ball Bros. Research Corp., Boulder, Corp., personal communication, 9 February 1965

"S-55C (Explorer 23)," Space World, B-4-18, 7–9, April 1965

Satellite 1963 22A Attitude Stabilization and Damping, Applied Physics Laboratory, Johns Hopkins University, Silver Spring, Md., Report TG508 [AD-429538] July 1963

Scannapieco, J., Radio Corp. of America, Princeton, N. J., personal communication, 14 June 1965

Schach, M. and R. E. Kidwell, Jr., "Thermodynamics of Space Flights," lecture notes presented at Pennsylvania State University, March 1963

Schnapf, A., "Tiros – The Television and Infrared Observation Satellite," J. British Interplanetary Society 19, 386–408, May–June 1964

Schurmeier, H. M., Ranger VII: Mission Description and Performance Part I, Jet Propulsion Laboratory, Pasadena, Calif., JPL-TR-32-700: NASA-CR-59904 [N65-13280] December 1964

Sheetz, H., Radio Corp. of America, Princeton, N. J., personal communication, 14 June 1965

Siglin, P. W. and G. Senn, "The Courier Satellite," Communication Satellites, L. J. Carter, ed., Academic Press, New York, 1962

Slater, R. J., "Design and Fabrication of Inflatable and Rigidizable Passive Communications Satellites (Echo I and Echo II)," Transactions, Aerospace Expandable Structures Conference, Dayton, Ohio, 23–25 October 1963

Smith, A. H., "Power Sources," Space Programs Summary No. 37-17, Vol. IV, pp. 107–115, Jet Propulsion Laboratory, Pasadena, Calif., 30 October 1962

"Soviets Describe Equipment for Elektron," Space Technology International, 7, No. 7, 41–42, July 1964

"Spacecraft Design and Development," pp. 11–15 in Space Programs Summary No. 37-27, Vol. 6, Jet Propulsion Laboratory, Pasadena, Calif., 31 May 1964

Space Log, 2, No. 4 (1962)

Space Log, 4, No. 2 (1964)

Space Log, 4, No. 3 (1964)

Space Log, 4, No. 4, 21, 25–26, (1964)

Stambler, Irwin, "The OGO Satellites," Space/Aeronautics, 39, No. 2, 70–77, February 1963

Stambler, Irwin, "The Explorers," Space/Aeronautics, 41, 98–46, February 1964

Stampfl, Rudolf A., The Nimbus Spacecraft and its Communication System as of September 1961, Goddard Space Flight Center, Greenbelt, Md., NASA-TN-D-1422 [N63-12125] January 1963

Stampfl, Rudolph A., "The Nimbus Satellite Communication System," Space Communications, A. V. Balakrishnan, ed., McGraw-Hill Book Co., 1963

Stations in Space (Selected Articles), Foreign Technology Div., Air Force Systems Command, Wright-Patterson Air Force Base, Ohio, FTD-TT-63-1041/1+2 [AD-435503] [N64-23634] February 1964

Stein, K. J., "Avionics Changes Included in Tiros Wheel," Aviation Week & Space Technology, 81, No. 5, 44–47 (1965)

Stengard, E. O., "Development of the Nimbus Structure," Proceedings of the Nimbus Program Review, Valley Forge, Pa., 14–16 November 1962, Goddard Space Flight Center, Greenbelt, Md., NASA-TM-X-50151: NASA-X-650-62-226 [N63-18601] 1962

"Syncom in Stationary Orbit; Wired with Teflon FEP-ML," J. Teflon, 5, 8, January 1964

"Teflon to Venus; Aids Mariner II Space Success," J. Teflon, 4, No. 2, 1–2, February 1963

Thienel, C., Goddard Space Flight Center, Greenbelt, Md., personal communication, 17 May 1965

Turner, H. C., McDonnell Aircraft Co., St. Louis, Mo., personal communication, 12 February 1965

Turner, H. C., "The Structural Application of Beryllium to the Gemini System," presented at the Second International Conference on Beryllium Technology, Franklin Institute, Philadelphia, Pa., October 1964

Turk, W., Ranger Block III Attitude Control System, Jet Propulsion Laboratory, Pasadena, Calif., JPL-TR-32-663: NASA-CR-59903 [N65-13278] November 1964

Turtil, J. F. and K. Hoepfner, "Stabilization and Control Subsystem," Proceedings of the Nimbus Program Review, Valley Forge, Pa., 14–16 November 1962, Goddard Space Flight Center, Greenbelt, Md., NASA-TM-X-50151: NASA-X-650-62-226 [N63-18601] 1962

Van Allen, James A., Survival of Thin Films in Space, State University of Iowa, Iowa City, Ia., NASA-CR-58122: SUI-64-22 [N64-26600] July 1964

Waddel, R. C., "Radiation Damage to Solar Cells on Relay I and Relay II," IEEE Trans. Nucl. Science, NS-11, No. 5, 60–68, November 1964

Wagner, Carl L. Jr., Ariel I – Evolution of its Structure, Goddard Space Flight Center, Greenbelt, Md., NASA-TN-D-1903 [N63-20605] September 1963

Wakelyn, N. T., Langley Research Center, Hampton, Va., personal communication, 20 May 1965

Warren, H. R. and J. Mar., "Structural and Thermal Design of the Topside Sounder Satellite," Canadian Aeronautics and Space Journal, 8, 161–164, September 1962

Welhart, E. K., McDonnell Aircraft Co., St. Louis, Mo., personal communication, 12 February 1965

Wilks, W. E., "GD/A's Satar will Serve as Space Probe, Reentry Vehicle or Satellite," Missiles and Rockets, 14, No. 24 (1964)

Wittenberg, A. M., "Thermal Data From the Telstar Satellites," pp. 513–521 in Institute of Environmental Sciences, Annual Technical Meeting, Philadelphia, Pa., April 13–15, 1964, Proceedings, Institute of Environmental Sciences, Mt. Prospect, Ill., 1964

Woerner, Chas. V. and Claude W. Coffee, Jr., Comparison of Ground Tests and Orbital Launch Results for the Explorer IX and Explorer XIX Satellites, Langley Research Center, Hampton, Va., NASA-TN-D-2466 [N64-28233] September 1964

Woerner, Chas. V. and Gerald M. Keating, Temperature Control of the Explorer IX Satellite, National Aeronautics and Space Adminstration, NASA-TN-D-1369, July 1962

Woerner, Chas. V., Properties of Two White Paints for Application to Inflatable Spacecraft: Titanium Dioxide Pigmented Epoxy and Zinc Oxide Pigmented Methyl Silicone Elastomer, Langley Research Center, Hampton, Va., NASA-TN-D-2834, June 1965

Zoutendyk, John A., Robt. J. Vondra, and Arvin H. Smith, Mariner II Solar Panel Design and Flight Performance, Jet Propulsion Laboratory, Pasadena, Calif., NASA-CR-51514: JPL-TR-32-455 [N63-22166] June 1963

Section 2

SELECTION OF MATERIALS FOR OPTIMUM PERFORMANCE

2.1 INTRODUCTION

This section discusses selection of materials for spacecraft construction and supplements Chapter 21 of the 2nd edition of the Space Materials Handbook (Ref. 1). Although this section summarizes the material contained in Sec. 1 of this Supplement and Chapters 10 through 20 of Ref. 1, it must not be regarded as a substitute for the detailed presentation of space effects given in those sources. Selection of a particular material for spacecraft application can only be tentatively decided from the information in this section; the final decision must rely on the data given in Ref. 1 and the original literature cited there. The designer must satisfy himself that the material he selects is suitable for his intended application.

Materials are recommended for the performance of several hypothetical space missions. The space environments upon which these recommendations are based vary in a different manner with the various mission parameters. Atmospheric pressure is a simple function of altitude at least to the accuracy desired here; pressures vary with altitude as shown in Tables 4-2 and 4-3 in Ref. 1. The micrometeorite flux is also weakly dependent upon the altitude as shown in Table 8-2 in Ref. 1. The radiation environments are more complicated. A treatment of non-penetrating radiation from the solar, albedo, and earth emission sources requires that ~~one consider not only altitude and~~ orbit but also vehicle geometry to arrive at expected vehicle temperature values. These specialized calculations will not be presented here. On the other hand, the very important penetrating radiation environment permits a somewhat more generalized treatment.

An orbiting vehicle will experience the sum of the effects of several radiation sources because of the spatial relationships of a particular trajectory with the characteristic distribution of radiation sources in space. For example at a 230-sm altitude, a polar orbiting vehicle will go through the trapped radiation (Van Allen) zones twice and will traverse the auroral regions four times per orbit. On the contrary an equatorial orbit at the same altitude will miss the auroral zones altogether. Solar flare activity occurs sporadically and the usual major flares seem to have a statistical distribution which follows the 11-yr solar cycle. Because of the deflecting properties of the earth's magnetic field there is a concentration of solar flare protons in the polar regions. Thus, an equatorial orbit will experience considerably less solar flare radiation (on the average) than a polar orbit. Finally, there is a more uniform distribution of primary cosmic rays although they are also affected somewhat by the earth's magnetic field. The contributions from each of the principal known radiation sources in space for a vehicle in various orbits are tabulated in Table 2-1. The corresponding maximum dose rates are given in Table 2-2.

Table 2-1

ANNUAL INTEGRATED RADIATION DOSES RECEIVED IN VARIOUS ORBITS

Sources of Radiation	Radiation Dose[a] (R)											
	90-deg Inclination Orbits (Polar)								30-deg Inclination Orbits			
	200 nm (230 sm)		500 nm (576 sm)		1000 nm (1152 sm)		2000 nm (2304 sm)		200 nm (230 sm)		500 nm (576 sm)	
	Surface	Internal	Surface	Internal	Surface	Internal	Surface	Internal	Surface	Internal	Surface	Internal
Cosmic Primary Protons	4	4	4	4	5	5	7	7	(b)	(b)	(b)	(b)
Trapped Protons (Van Allen)	30	10	3×10^3	1.5×10^3	1×10^5	5×10^4	5×10^5	2×10^5	50	15	8×10^3	4×10^3
Trapped[c] Electrons (Van Allen & Artificial)	1.5×10^5	1×10^4	1.5×10^7	1.5×10^6	1.5×10^8	1.5×10^7	8×10^7	4×10^6	2×10^5	1.5×10^4	3.5×10^7	3.5×10^6
Auroral Protons	300	0	300	0	300	0	300	0	(b)	(b)	(b)	(b)
Auroral Electrons	5×10^6	1	5×10^6	1	5×10^6	1	5×10^6	1	(b)	(b)	(b)	(b)
Solar Flare[d] Protons	3×10^3	1×10^3	3×10^3	1×10^3	3×10^3	1×10^3	3×10^3	1×10^3	(b)	(b)	(b)	(b)
Approximate Total	5×10^6	1×10^4	2×10^7	1.5×10^6	1.5×10^8	1.5×10^7	9×10^7	4×10^6	2×10^5	1.5×10^4	3.5×10^7	3.5×10^6

(a) Determined for vehicle surfaces and interiors (through 0.1-in. Al skin); the radiation dose is given in roentgens (R).

(b) These radiation sources have smaller intensities than those due to trapped electrons and will therefore not appreciably affect the total dose. When compared to doses for polar orbits, those for 30-deg inclination would be smaller approximately by a factor of 2 for cosmic rays, a factor of 10 for solar flares, and would be essentially zero for auroral protons and electrons.

(c) Includes Van Allen electron and artificial electron belt data as of 1/1/63; relies principally on the data of McIlwain from Explorer 15 (see discussion at end of Sec. 7.2.4, Ref. 1).

(d) Includes statistical maximum of one major flare per month and model flare data of Bailey (see discussion in Sec. 7.4, Ref. 1).

Table 2-2

MAXIMUM RADIATION DOSE RATES RECEIVED IN VARIOUS ORBITS

Sources of Radiation	Radiation Dose Rate[a] (R/hr)											
	90-deg Inclination Orbits (Polar								30-deg Inclination Orbits			
	200 nm (230 sm)		500 nm (576 sm)		1000 nm (1152 sm)		2000 nm (2304 sm)		200 nm (230 sm)		500 nm (576 sm)	
	Surface	Internal	Surface	Internal	Surface	Internal	Surface	Internal	Surface	Internal	Surface	Internal
Cosmic Primary Protons	4×10^{-4}	4×10^{-4}	4×10^{-4}	4×10^{-4}	5×10^{-4}	5×10^{-4}	7×10^{-4}	7×10^{-4}	(b)	(b)	(b)	(b)
Trapped (Van Allen) Protons	1.5	7×10^{-1}	2.5×10^{1}	1.5×10^{1}	1.5×10^{2}	8×10^{1}	5×10^{2}	2×10^{2}	2.5	1	70	40
Trapped[c] Electrons (Van Allen & Artificial	6×10^{3}	5×10^{2}	7×10^{4}	8×10^{3}	1×10^{5}	1.5×10^{4}	9×10^{4}	5×10^{3}	9×10^{3}	7×10^{2}	1.5×10^{5}	2×10^{4}
Auroral Protons	700	0	700	0	700	0	700	0	(b)	(b)	(b)	(b)
Auroral Electrons	1×10^{7}	2	1×10^{7}	2	1×10^{7}	2	1×10^{7}	2	(b)	(b)	(b)	(b)
Solar Flare[d] Protons	300	100	300	100	300	100	300	100	(b)	(b)	(b)	(b)
Highest Dose Rate	1×10^{7}	5×10^{2}	1×10^{7}	8×10^{3}	1×10^{7}	1.5×10^{4}	1×10^{7}	5×10^{3}	9×10^{3}	7×10^{2}	1.5×10^{5}	2×10^{4}

(a) Determined for vehicle surfaces and interiors (through 0.1-in. Al skin).

(b) These radiation sources have smaller intensities than those due to trapped electrons and will therefore not appreciably affect the total dose. When compared to doses for polar orbits, those for 30-deg inclination would be smaller approximately by a factor of 2 for cosmic rays, a factor of 10 for solar flares, and would be essentially zero for auroral protons and electrons.

(c) Includes Van Allen electron and artificial electron belt data as of 1/1/63; relies principally on the data of McIlwain from Explorer 15 (see discussion at end of Sec. 7.2.4, Ref. 1).

(d) Includes statistical maximum of one major flare per month and model flare data of Bailey (see discussion in Sec. 7.4, Ref. 1).

The radiation dose is given in roentgens (R) which is a measure of the ionization produced by a given amount of energy. Conversion of these units to equivalent fluxes of penetrating particles is inexact because interactions of particles with matter vary with the particle energy, mass, and flux. Nevertheless, the following statements may be used to give a rough estimation of the relationship between the dose and fluxes of penetrating particles:

1 roentgen	=	0.88 rad
1 rad	=	100 erg/gm
1 rad	$\approx$	10^6 protons/cm^2 (this may be inaccurate by a factor of 4, i.e., it may be 4×10^6 or 2.5×10^5 protons/cm^2.)
1 rad	$\approx$	3×10^7 electrons/cm^2 (this may be inaccurate by a factor of 2, i.e., it may be 6×10^7 or 1.5×10^7 electrons/cm^2.)

The accuracy and precision of these contributions vary widely. Primary cosmic ray fluxes are known quite accurately while Van Allen proton doses are still subject to change although a number of measurements have been made on them. The trapped electron dose includes contributions from both natural Van Allen electrons and artificial electrons injected by high-altitude nuclear explosions since July 1962. Thus, there is the experimental uncertainty at any one time as well as the decay of the artificially injected component. Still further down the scale of precision of uncertainty are the estimates of the auroral and solar flare contributions. These rely on long-time averages and assumptions as to rates of activity and model flare data.

2.2 THERMAL CONTROL MATERIALS

2.2.1 Environmental Considerations

The thermal design of a specific spacecraft begins with information on the operating temperature limits and power dissipation levels of its components, together with a definition of the vehicle and trajectory geometries. From such information, the thermal design parameters required to obtain the necessary temperature environment are determined. The predominant parameters are the α_s/ϵ ratio of the external surfaces and the emittance of internal surfaces.

Predictions of ascent conditions are required in connection with the structural and aerodynamic design of some spacecraft. The general nature of these conditions was discussed in Chapter 2, pp. 7–9 of Ref. 1. Materials are tested in conditions simulating the ascent environment; having determined the behavior of candidate thermal control surfaces, one then selects those with radiation characteristics needed for orbital conditions and with the necessary resistance to the particular ascent environment. In general, materials are expected to suffer more serious degradation for ascent heating histories with a given peak temperature the longer that temperature is maintained.

Materials such as potting compounds, sealants, torque paints, or wire insulations will, upon being heated in the ascent environment, liberate gaseous products which may recondense on and contaminate nearby thermal control materials. Tables 10-4 through 10-10, pp. 141–149 in Ref. 1, lists some of these materials with their temperature limits.

In manned spacecraft a system of active thermal control is used to maintain proper temperatures of the cabin in orbit. During ascent, the thermal insulation and active thermal control system maintains the proper temperature in the spacecraft. Although exterior coatings of the spacecraft are used mainly for corrosion protection and assistance in visual tracking, they must be chosen to withstand the ascent heating environment. During reentry, the heat shield is designed to supply ablative and thermal protection to the spacecraft. In the Gemini, the pressurized cabin walls were double spaced and made of ~ 0.010-in. thick commercially pure titanium sheet (AMS 4901). The adapter section was HK31A magnesium alloy sheet for the skin, stiffener rings, and stringers. The skin was coated with a zirconium dioxide pigmented paint with absorptance to emittance ratio of 0.24 to supply some passive thermal control to the relatively large area of the adapter section. In Gemini 5 the interior of the adapter section will have a coating of Lockspray gold over a white epoxy coating on Dow 17-treated Mg alloy. HK31A-H24.

In unmanned spacecraft a nose fairing or shroud usually protects the spacecraft during the ascent phase of the mission. Temperatures on and under the shroud can get quite high. For example, simulation tests of the Mariner 3 shroud material showed a peak temperature at the surface of ~ 675° F after about 120 sec, and ~ 250° F at the back face of the shroud. Therefore, the thermal control coatings used should be able to withstand fairly high temperatures (at least short time stability to 500° F) and thermal cycling in vacuum (should withstand more than three hundred 15-min cycles from - 150 to 70° F). Certain portions of the spacecraft will experience low temperatures (- 250 to - 350° F); therefore, the thermal control materials should be stable at low temperatures also.

Some selected thermal control coating materials are listed in Tables 2-3 through 2-6 where the thermal absorptance and emittance properties and resistance to ascent heating, thermal cycling in vacuum, and simulated solar ultraviolet radiation are given for solar absorbers, solar reflectors, flat absorbers, and flat reflectors. Table 2-7 lists some miscellaneous thermal control materials.

Temperature. Within the temperature limits shown, the behavior of the material as a thermal control surface is not affected. The minimum temperature limit for paints is often the temperature below which coating failure due to differential thermal contraction of coating and substrate occurs; in such cases, this limit must be related to a specific coating (substrate) geometry.

Ultraviolet resistance. The resistance to simulated solar ultraviolet radiation of normal incidence at the Earth's distance from the Sun for up to an equivalent of 5000 hr exposure (2 to 6 suns with appropriately reduced exposure time) is evaluated in the table. Metallic surfaces are generally not degraded by ultraviolet radiation. Such radiation will generally not affect the infrared emittance of materials.

Table 2-3

THERMAL CONTROL MATERIALS FOR SOLAR ABSORBERS

Material	Substrate	α_s/ϵ	Absorptance and Emittance, 70° F	Ascent Temperature Limits (° F)	Ultraviolet Resistance	Thermal Cycling Resistance	Remarks
6061 Al Chemically Cleaned	**As rolled**	2.7 ± 0.05	$\alpha_s = 0.16 \pm 0.04$ $\epsilon = 0.07 \pm 0.03$	Structural limits only	No effect	No effect	The surface characteristics of the sheet materials are subject to variations depending on fabrication operations.
6061 Al Chemically Cleaned	**Sheet sanded before processing**	2.7	$\alpha_s = 0.16 \pm 0.05$ $\epsilon = 0.06 \pm 0.03$	Structural limits only	No effect	No effect	Subject to degradation from prelaunch environment including personnel handling.
6061 Al Chemically Cleaned	**Forging**	3.2 ± 0.08	$\alpha_s = 0.29 \pm 0.06$ $\epsilon = 0.09 \pm 0.06$	Structural limits only	No effect	No effect	
	weld area	2.6 ± 0.08	$\alpha_s = 0.26 \pm 0.06$ $\epsilon = 0.10 \pm 0.06$				
2024 Al Chemically Cleaned	**As rolled hand sanded**	3.7 ± 0.06	$\alpha_s = 0.20 \pm 0.05$ $\epsilon = 0.06 \pm 0.03$	Structural limits only	No effect	No effect	The surface characteristics of the sheet materials are subject to variations depending on fabrication operations. These values are given for the as-rolled condition.
Al Foil Plain (MIL-A-148)	**Not applicable**	$3.0^{+0.05}_{-0.04}$	$\alpha_s = 0.12 \pm 0.04$ $\epsilon = 0.05 \pm 0.02$	Structural limits only	No effect	No effect	Subject to degradation from prelaunch environment including personnel handling; adhesive is limiting factor in space environment.

Al Foil Rubber-Based Adhesive Backed (Fasson Foil)	Any clean rigid surface	$3.0^{+0.05}_{-0.04}$	$\alpha_S = 0.12 \pm 0.04$ $\epsilon = 0.05 \pm 0.02$	375	No effect	No information	Clean foil surface with lacquer thinner, followed by methyl-ethyl-ketone followed by isopropyl alcohol. May not be external during ascent. Foil should be perforated ($\approx$ 1/32-in. diam. on 1/2-in. centers) to prevent lifting due to adhesive gas evolution in vacuum.
Al Foil Silicone-Based Adhesive Backed (Mystik 7402)	Any clean rigid surface	$3.0^{+0.05}_{-0.04}$	$\alpha_S = 0.12 \pm 0.04$ $\epsilon = 0.04^{+0.02}_{-0.01}$	750	No effect	No information	If applied externally, the tape should have mechanical fastening on both ends to prevent ascent forces from peeling the tape from the substrate; subject to handling degradation.
Quilted Inconel Foil (H.I. Thompson Spec. No. TPS 0101B) MIL-N-6840	Not applicable	3.17 ± 0.07	$\alpha_S = 0.38 \pm 0.05$ $\epsilon = 0.12 \pm 0.05$	2200	No effect	No effect	Very susceptible to increase in α_S and ϵ by fingerprints and oxidation in prelaunch environment; primarily for engine heat shield usage.
Inconel X Foil MIL-N-7786	Not applicable	4.40 ± 0.10	$\alpha_S = 0.66 \pm 0.09$ $\epsilon = 0.15 \pm 0.05$	1500	No effect	No effect	Subject to handling degradation.

Table 2-3 (concl'd)

Material	Substrate	α_s/ϵ	Absorptance and Emittance, 70°F	Ascent Temperature Limits (°F)	Ultraviolet Resistance	Thermal Cycling Resistance	Remarks
QMV Be Chemically Polished	Not applicable	5.00 ± 0.08	$\alpha_s = 0.50 \pm 0.06$ $\epsilon = 0.10 \pm 0.06$	1700 (Test Maximum)	No effect	No effect	High ascent temperature has no effect on α_s or ϵ if at pressure of 0.05 Torr or less.
Hanovia Gold No. 6518 on René 41	René 41	6.0 ± 0.08	$\alpha_s = 0.53 \pm 0.06$ $\epsilon = 0.09 \pm 0.06$	900 No change	No effect	No effect	This coating may be suitable for substrates other than René 41. The noted values apply only to that substrate however. Any other application must be evaluated prior to use. At 1700°F, the values changed to $\alpha = 0.81 \pm 0.06$ and $\epsilon = 0.40 \pm 0.10$ (as measured at 70°F after exposure to ascent temperature.)

Table 2-4

THERMAL CONTROL MATERIALS FOR SOLAR REFLECTORS

Material	Substrate	α_s/ϵ	Absorptance and Emittance, 70°F	Ascent Temperature Limits (°F)	Ultraviolet Resistance	Thermal Cycling Resistance	Remarks
White Acrylic Paint (Sherwin Williams M49WC17 Kemacryl)	Any clean rigid surface primer required	0.33 ± 0.05	$\epsilon = 0.89^{+0.03}_{-0.06}$ $\alpha_s = 0.28 \pm 0.04$	450 200 to 450°F α_s increases by 0.04 (constant) maximum allowed	$\Delta\alpha_s = 0.18 \pm 0.04$ after 2000 sun-hr	No failure in 335 cycles -150 to 70°F 18-min cycles	5-mil dry film thickness required for opacity to solar; 1-mil thickness sufficient for opacity in infrared.
White Epoxy Paint (A. Brown Skyspar SA9185)	Any rigid surface	$0.24^{+0.05}_{-0.07}$	$\epsilon = 0.91^{+0.03}_{-0.06}$ $\alpha_s = 0.22 \pm 0.04$	450 200 to 450°F α_s increases by 0.04 (constant) maximum allowed	$\Delta\alpha_s = 0.35 \pm 0.06$ after 2000 sun-hr	No failure in 385 cycles -150 to 70°F 18-min cycles	α_s highly susceptible to change from prelaunch sunlight and fluorescent lights. Otherwise quite resistant to prelaunch environment. Not recommended where α_s is critical. 4-mil dry film

Table 2-4 (concl'd)

Material	Substrate	α_s/ϵ	Absorptance and Emittance, 70°F	Ascent Temperature Limits (°F)	Ultraviolet Resistance	Thermal Cycling Resistance	Remarks
White Silicone Paint (W.P. Fuller 517-W-1)	HM21A-T8 Mg, Hm21A-0 Mg, Al, Ti, stainless steels, super-alloys, and other rigid substrate capable of withstanding cure cycle	$0.28^{+0.04}_{-0.07}$	$\epsilon = 0.90^{+0.03}_{-0.06}$ $\alpha_s = 0.25 \pm 0.03$	650	$\Delta\alpha_s = 0.09 \pm 0.05$ after 2000 sun-hr	Cracking and loss of adhesion in 170 cycles -240 to 70°F, 18-min cycles	5-mil dry film thickness required for opacity to solar; 1-mil thickness sufficient for opacity in infrared; peak cure cycle temperature, 465°F.
White Silicate Paint (LMSC)	1100-Al	0.17	$\alpha_s = 0.13 \pm 0.03$ $\epsilon = 0.85 \pm 0.04$	600	$\Delta\alpha = 0.04$ after 2000 sun-hr	No Effect	At present, this material may be applied to only 1100 Al and certain other substrates. High-temperature cure required ($\approx$400°F)
White Silicone Air Dry Paint (LMSC)	Any rigid Substrate	0.16	$\alpha_s = 0.14$ $\epsilon = 0.86$	700	$\Delta\alpha_s = 0.03$ after 2000 sun-hr	No Effect	

Table 2-5

THERMAL CONTROL MATERIALS FOR FLAT ABSORBERS

Material	Substrate	α_S/ϵ	Absorptance and Emittance, 70°F	Ascent Temperature Limits (°F)	Ultraviolet Resistance	Thermal Cycling Resistance	Remarks
Black Acrylic Paint (Sherwin Williams M49BC12)	Any clean rigid substrate; primer required	1.06 ± 0.04	$\alpha_S = 0.93 \pm 0.03$ $\epsilon = 0.88 \pm 0.03$	No effect at 450	$\Delta\alpha_S < 0.05$ after 600 sun-hr UV	No failure in 385 cycles -150 to 70°F 18-min cycles	1.5-mil dry film thickness required for solar and infrared opacity.
Black Silicone Paint (W. P. Fuller 517-B-2)	Same as white silicone	1.01 ± 0.07	$\alpha_S = 0.89 \pm 0.05$ $\epsilon = 0.88 \pm 0.05$	No effect at 1070	$\Delta\alpha_S < 0.05$ after 600 sun-hr UV	Cracking and loss of adhesion in 170 cycles -240 to 70°F 18-min cycles	1-mil dry film thickness required for solar and infrared opacity; peak cure cycle temperature, 465°F.
Oxide Flame Sprayed (Norton Co. Rokide C, 85% Cr_2O_3)	René 41 with a 2-mil coating of nichrome	1.06 ± 0.06	$\alpha_S = 0.90 \pm 0.04$ $\epsilon = 0.85 \pm 0.04$	No effect at 1660	No effect	No failure 70 to 1600°F in 5 min	All values have been obtained with René 41 as a substrate. It is possible to use this coating with other substrates. The bonding between Rokide C and the substrate is purely mechanical and thermal shock is a potential problem.
QMV Be, Pt-black coated	QMV Be	1.11 ± 0.08	$\alpha_S = 0.94 \pm 0.03$ $\epsilon = 0.85 \pm 0.07$	No effect at 1200	No effect	No information; probably no effect	Possesses stable high-temperature emittance.
Mg Dow 17 Coated	HM21A Mg alloy	1.11 ± 0.10	$\alpha_S = 0.78 \pm 0.08$ $\epsilon = 0.70 \pm 0.06$	No effect at 500	No effect	No effect	Proprietary process of Dow Chem. Co.; thermal stability > 500° doubtful.

Table 2-6

THERMAL CONTROL MATERIALS FOR FLAT REFLECTORS

Material	Substrate	α_S/ϵ	Absorptance and Emittance, 70°F	Ascent Temperature Limits (°F)	Ultraviolet Resistance	Thermal Cycling Resistance	Remarks
Silicone Paint (W. P. Fuller 171-A-152)	**Same as white silicone**	0.92 ± 0.08	$\alpha_S = 0.22 \pm 0.04$ $\epsilon = 0.24 \pm 0.04$	No effect at 800	$\Delta\alpha_S = 0.09 \pm 0.04$ after 600 sun-hr UV	Cracking and loss of adhesion in 170 cycles -240 to 70°F 18-min cycles	3-mil dry film thickness required for solar opacity, 1-mil thickness required for infrared opacity; peak cure cycle temperature, 456°F.
Al Silicone Paint (W. P. Fuller 172-A-1)	**Same as white silicone**	0.89 ± 0.10	$\alpha_S = 0.25 \pm 0.07$ $\epsilon = 0.28 \pm 0.07$	No effect at 880	$\Delta\alpha_S = 0.09 \pm 0.04$ after 600 sun-hr UV	Cracking and loss of adhesion in 170 cycles -240 to 70°F 18-min cycles	See above.
Al Acrylic Paint (Sherwin-Williams Clear Kemacryl With Non-leafing Al Pigment)	**Same as white silicone**	0.85 ± 0.08	$\alpha_S = 0.41 \pm 0.03$ $\epsilon = 0.48 \pm 0.05$	650 α_S increases by < 0.03 from 240 to 650°F; maximum allowable, 650.	$\Delta\alpha_S < 0.05$ after 600 sun-hr UV	No failure in 385 cycles -150 to 70°F 18-min cycles	α_S and ϵ are subject to wide variation from spraying technique; control application with optical surface comparator.

Table 2-7

MISCELLANEOUS THERMAL CONTROL MATERIALS

Material	Substrate	α_S/ϵ	Absorptance and Emittance, 70°F	Ascent Temperature Limits (°F)	Ultraviolet Resistance	Thermal Cycling Resistance	Remarks
LMSC Silicone Tape (1A48)	Any rigid substrate	0.18	$\alpha_S = 0.16$ $\epsilon = 0.86$	700	$\Delta\alpha_S = 0.04$ after 2000 sun-hr	No effect	
Rokide A (Al Oxide Flame Sprayed by Norton Abrasives Co., San Jose, Calif.)	Any metallic substrate	0.36 ± 0.05	$\alpha_S = 0.27 \pm 0.04$ $\epsilon = 0.75 \pm 0.03$				This material was used on Explorers 1, 3, and 7 and Tiros 2. Total area covered by this material small; actual performance of material cannot be evaluated. Tiros 2 transmitted 1 yr, Explorer 7 transmitted about 2 yr.
Sandblasted Stainless Steel AISI 410	Not applicable	0.88	$\alpha_S = 0.75$ $\epsilon = 0.85$		No Information		This material with Rokide A stripes was primary thermal control surface of Explorers 1, 3, and 4.
Sandblasted Al (2024)	Al alloy (2024	2.0	$\alpha_S = 0.42$ $\epsilon = 0.21$		No Information		Material used in Explorer 7 as support for solar cells and as stiffener ring between glass reinforced polyester conical sections of spacecraft structure. Thermal design was 0 to 60°C. While in orbit instruments in spacecraft were never lower than 16°C or higher than 41°C. Transmitted from 10/13/59 to 8/24/61.

Table 2-7 (concl'd)

Material	Substrate	α_s/ϵ	Absorptance and Emittance, 70°F	Ascent Temperature Limits (°F)	Ultraviolet Resistance	Thermal Cycling Resistance	Remarks
LMSC White Silicone Air Dry Paint	Any rigid substrate	0.16	$\alpha_s = 0.14$ $\epsilon = 0.86$	700	$\Delta\alpha_s = 0.03$ after 2000 sun-hr	No effect	
Black Mico-bond (L6X962) Midland Industrial Finishes Co., Waukegan, Ill.	Any metal surface	1.11 ± 0.05	$\alpha_s = 0.93 \pm 0.04$ $\epsilon = 0.84 \pm 0.03$	No information	No effect after 500 sun-hr	No information	
Reynolds Wrap Foil Smooth	Not applicable	Dull side 5.0 Shiny side 6.3	$\alpha_s = 0.20$ $\epsilon = 0.04$ $\alpha_s = 0.19$ $\epsilon = 0.03$	Structural limits only	No effect	No effect	
Lockspray Gold	Anodized Mg or Al alloys coated with clear or white, glossy or matte epoxy	7.3 to 4.8	$\alpha_s = 0.22$ to 0.24 $\epsilon = 0.03$ to 0.05	No effect to 400° F	No information	No effect	Material used as coating on visor of face plate on Astronaut White's helmet during extra-vehicular activity in the Gemini 4 mission; used as coating on interior of Gemini 5 adapter section with substrate of white epoxy on Dow 17 treated Mg alloy HK31A-H24

Emissivity is substantially unaffected by ultraviolet exposure. Solar absorptance will increase in approximately the manner shown in Fig. 2-1. These values are based on simulated exposures in ultraviolet radiation from an A-H6 lamp at six times the intensity of solar radiation in space and in vacuum at $\sim 10^{-7}$ Torr.

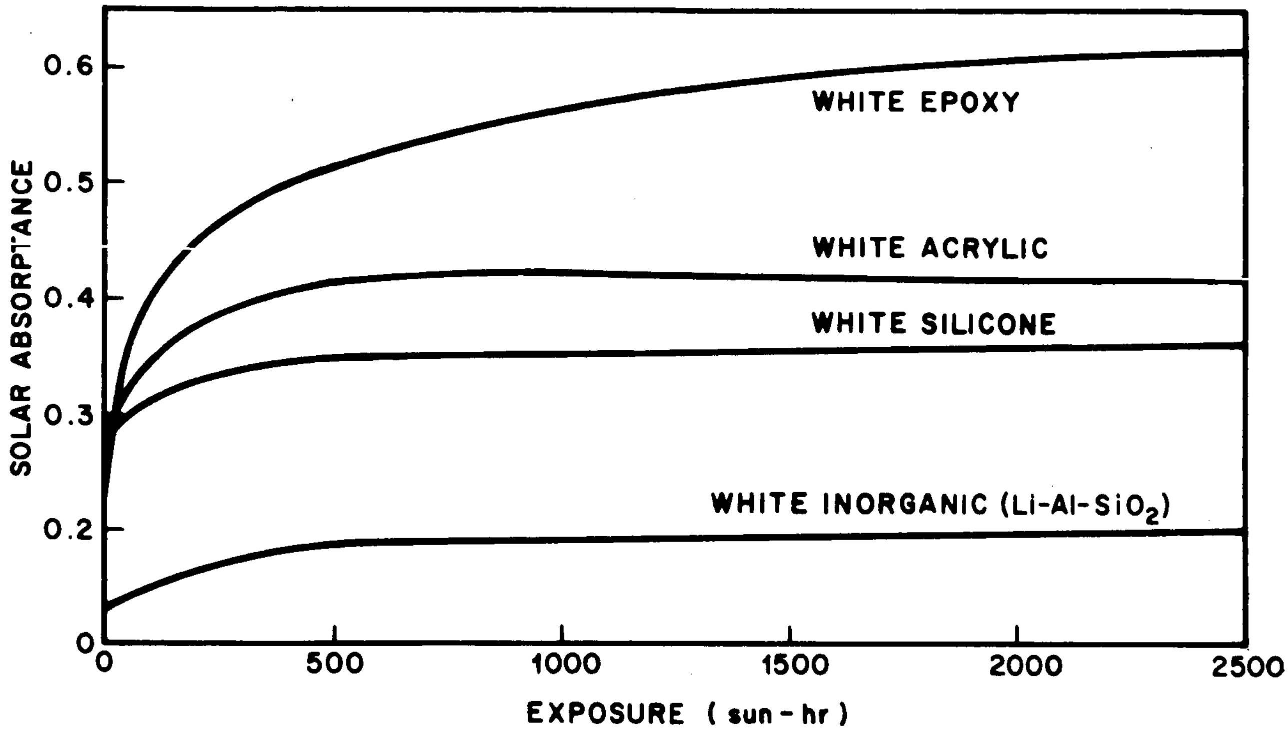

Fig. 2-1 Change in Solar Absorptance, α_s, of Some Thermal Control Materials in Simulated Solar Ultraviolet Radiation (Ref. 2)

Penetrating radiation environment. The data in Table 2-8 show the changes in solar absorptance of some thermal control materials upon exposure to reactor radiation in vacuum (for induced radiation, see Chapter 10, p. 157, Ref. 1). The infrared emittance of materials is not expected to be affected by the penetrating radiation environments at room temperature. Metals are unaffected at the dose levels studied at the normal temperature of spacecraft application.

Simulated environmental tests have served mainly to screen potential materials, rather than to ascertain their reliability. A limited, and by no means sufficient, number of materials have been tentatively accepted as suitable thermal control materials. Among these are organic paints, in spite of their inferiority to inorganic materials. The chief reasons for using them are their immediate availability, ease of application, and low cost. Similarly, other materials have been tentatively approved because superior materials are still in research, development, and testing stages.

Table 2-8

EFFECT OF PENETRATING NUCLEAR RADIATION IN VACUUM ON SOME THERMAL CONTROL MATERIALS(a)

Material	Substrate	α_s/ϵ	α_s	ϵ at 70°F	Nuclear Radiation Stability at	
					0.45×10^8 R 0.45×10^{14} nvt $\Delta\alpha_s$	1.8×10^8 R 1.8×10^{14} nvt $\Delta\alpha_s$
LMSC Silicate Paint	1100 Al alloy	0.17	0.15	0.90	0.02	0.05
LMSC Silicone Tape (1A48)	Any rigid substrate	0.18	0.16	0.86	> 0.01	> 0.01
White Skyspar Enamel (A. Brown A423 Color SA9185)	Any rigid surface	$0.24^{+0.05}_{-0.07}$	0.22 ± 0.04	$0.91^{+0.03}_{-0.06}$	0.07	0.12
Fuller Gloss White Silicone Paint (517-W-1)	HM21A-T8 Mg alloy, Al alloys, Ti alloys, stainless steels, high-strength alloys, and any rigid substrate which can withstand 465°F cure cycle	$0.28^{+0.04}_{-0.07}$	0.25 ± 0.03	$0.90^{+0.03}_{-0.06}$	0.06	0.09
Tinted White Kemacryl Paint (Sherwin Williams M49+C17)	Any clear rigid surface; primer required	0.33 ± 0.05	0.28 ± 0.04	0.86 ± 0.03	0.06	0.09

(a) The material listed in Tables 2-3 through 2-7 show no significant change in absorptance or emittance in penetrating nuclear radiation in vacuum.

In general, the thermal control materials used in successful satellites and space probes have performed reasonably close to the design limits expected of them (pp. S-9, S-10, and S-35, Sec. 1). A few exceptions were discussed in Ref. 1, p. 579; e.g., Explorer 10 and 12 which was designed to operate at a maximum of 40°C actually operated at 60°C. This performance may have contributed to the malfunction of the magnetometer. In Mariner 2, several units of the spacecraft were hotter at the Venus encounter than had been predicted (Ref. 1, p. 583). This behavior was attributed to lack of suitable solar simulation in the prototype system thermal-vacuum tests. This situation was corrected for the Mariner 3 and 4 spacecraft where the 25-ft diameter vacuum chamber with arc lamp solar ultraviolet radiation simulator was used for qualification at the subsystem and system level. Flight results from Mariner 4 indicate that the spacecraft temperatures at eighteen interior and six exterior locations compare favorably with Type Approval and Flight Acceptance test results (Ref. 3).

2.2.2 Example of Selection of Thermal Control Materials

As an example of the selection of thermal control materials consider an earth satellite to operate in a 300-sm circular polar noon orbit where the angle β between the orbit plane and the Sun is zero. The satellite is a right cylinder 5 ft in diameter and 5 ft long fabricated of HM21A magnesium alloy sheet for the skin. This section contains an electronic component with an optimum operating temperature of 70°F and must be controlled for 1 month in space between 0 and 140°F.

This section of the spacecraft does not view any other section and is insulated from other parts of the vehicle. During ascent the maximum peak temperature is 600°F for a duration of 1 min (see Ref. 1, p. 9, Fig. 2-1).

The electronic component dissipates 200 W of power continuously. The emittance for the component is assumed to be 0.85 and the internal power dissipation factor $P/A\epsilon$ (P = electrical power in watts, A = area of component, ϵ = infrared emittance of the component surface material) is 10 Btu/ft^2-hr. An interpolation in Fig. 10-8, p. 118, Ref. 1 results in a value of $\alpha_s/\epsilon = 0.6$ to produce a space-time average of 0°F for the external spacecraft skin. If the conductance between the electronic component and the vehicle skin is 9.5 Btu/°F-hr then the selected $\alpha_s/\epsilon = 0.6$ will result in an average temperature for the component of 70°F. Now, since Tables 2-3 through 2-7 indicate that no single material has an $\alpha_s/\epsilon = 0.6$, it is therefore necessary to design a mosaic. Actually this procedure results in a more flexible design approach which permits adjustment of the thermal control mosaic pattern in response to results of thermal-vacuum solar simulation tests at the subsystem level which can later be verified at the system test level. In selecting materials for a mosaic, Fuller Gloss White Silicone paint (517-W-1), a solar reflector with these properties $\alpha_s/\epsilon = 0.28$, $\alpha_s = 0.25$, and $\epsilon = 0.90$, and Fuller Black Silicone paint (517-B-2) a flat absorber with these properties $\alpha_s = 1.01$, $\alpha_s = 0.89$, and $\epsilon = 0.88$, are chosen to bracket the selected α_s/ϵ value. By properly adjusting the respective areas in the mosaic an approximation of the selected α_s/ϵ can be attained.

This represents a preliminary choice of materials. The final choice of materials and striping pattern depends upon a detailed analysis which includes the effects of circumferential variations of temperature around the cylinder and the orbital temperature fluctuations. From the data on ultraviolet resistance for the Fuller White silicone shown in Fig. 2-1, it is evident that the striping pattern should allow for the increase in average α_s/ϵ resulting from ultraviolet exposure. The detailed analysis would very likely result in a choice of the same materials with perhaps a modification of the striping pattern.

2.3 OPTICAL MATERIALS

Spacecraft optical materials are selected on the same basis as those chosen for ground applications where the optical design parameters with respect to transmittance and reflectance for a specified spectral range of electromagnetic radiation govern the choice. In general, the optical design has to be strong enough to withstand flight qualification tests in shock, vibration, and the like. Protection of precision parts and surfaces must be provided to obviate damage during ground handling of optical components during systems tests.

Temperature stability of materials and minimizing temperature gradients within the spacecraft structure and the optical structure are prime requirements. For example, the orbits for the Orbiting Astronomical Observatory (OAO) satellites are such that the spacecraft skin will fluctuate between -200 and 40°F (Ref. 4). For this reason, Alzak (3003 Al alloy base sheet clad with 99.9% Al brightened and clear anodized $\alpha_s = 0.15$, $\epsilon = 0.77$) skin panels are attached to the structure with pin-connector nylon fasteners to minimize heat transfer. Alternate thermal designs for two proposed OAO satellite structures call for temperature ranges of -22 ±18°F or 32 ±18°F, depending upon the orbital parameters and astronomical instrumentation needed for the mission. The central telescope tube is insulated from the space environment and must be maintained within 3°F of the spacecraft structure. The structural tube of the telescope is made of Ti6Al-4V for strength. This material also provides compensating thermal expansion characteristics compatible with the aluminum alloys used for other parts of the structure.

Although there are three sets of astronomical instruments each with its own specific design for the two OAO satellites, fused quartz generally is used for mirrors, lenses, and spectrometer gratings. In some cases, the quartz is coated with aluminum and overcoated with magnesium fluoride to provide better ultraviolet reflectance for mirrors. Beryllium is used for mirrors in one of the telescopes. The polished beryllium surface is coated with aluminum and overcoated with magnesium fluoride.

In most cases, since a shroud protects the spacecraft during the ascent phase of the mission, it is not necessary to protect the optics from direct contact with the ascent heating environment. However, measures must be taken to insure stability of the construction materials for the shroud and the spacecraft so that materials such as paints and polymers do not volatilize under ascent heating conditions. The volatile constituents of the less stable materials can condense on sensitive surfaces of optical materials and damage them. In cases where a shroud is not used, a cover that can be actuated for removal in space after completion of the ascent phase protects sensitive optical components.

In space, erosion by micrometeoroids and interplanetary dust can be detrimental to optical materials. Sputtering erosion by impingement of ions or solar plasma is a related phenomenon. Estimates indicate a removal rate from polished aluminum of about 100 Å/yr by interplanetary dust erosion. Erosion by impingement of O_2, N_2, O, or N atoms, and ions in lower orbits is estimated to remove about 600 Å/yr from polished aluminum; solar plasma sputtering is estimated to remove 300 Å/yr. About 300 Å would probably be removed in a single solar flare. These rates can cause significant damage to optical surfaces in ~1 yr.

Some of the effects of sputtering experiments with electropolished metals are summarized below (Ref. 1).

Metal	Flux (ions/cm^2)	Remarks
Commercial pure titanium	10^{21}	No change in hemispherical emittance
Titanium alloy	10^{21}	No change
Pure aluminum	7×10^{20}	Hemispherical emittance increased 60%
Aluminum alloy 2024	10^{21}	Hemispherical emittance decreased, then increased 50%
Copper	10^{20}	Hemispherical emittance increased 13 to 58%

Flint glass or other optical glasses such as cerium stabilized glass or fused quartz can be used for lens systems. Absorption of radiation causes a decrease in transmission by the formation of color centers. Ordinary glass is most susceptible to damage, fused silica and synthetic quartz next, followed by crystalline quartz and sapphire as the least sensitive.

In simulation tests fused silica, synthetic quartz and crystalline quartz were discolored with a 10^7 rad dose of 50 kV x-rays or 2 MeV electrons. The infrared transmission in the 1- to 5-μ region was unchanged when GE 105 quartz, sapphire, silicon and germanium were irradiated with 10^7 rad of Co 60; 10^{15} e/cm^2 at 1 MeV and 1.5×10^{12} p/cm^2 at 12 MeV for the quartz. Additional data on optical materials can be found in Chapter 11, pp. 187–207 in Ref. 1.

The Mariner 3 and 4 TV camera telescope used light flint glass and synthetic quartz lenses, the mirrors in the Cassegrainian system were beryllium overcoated with aluminum and magnesium fluoride. A protective cover over the lens systems of the

TV telescopes protected them from sputtering or micrometeoroid impingement. This cover was removed by ground command to the spacecraft on 11 February 1965, 76 days after launch 28 November 1964. Twenty-one successful TV pictures of the Martian surface were taken and telemetered to Earth by Mariner 4 spacecraft on 14 July 1965, 154 days after command removal of the lens cover. Some fogging or spotting was detected on the early photographs. The exact nature and cause of the fogging and whether the phenomenon could be attributed to damage to the camera optics by the space environment was not completely analyzed at the time of this report, but did not repeat in the later photographs which were quite clear.

The optical materials used in the Nimbus 1 weather observation satellite are listed in Table 1-7. One item of interest is that the vidicon camera lens system was of conventional optical glass protected by flat plates of fused silica.

Fused silica (Vycor) or aluminum silicate glass appear the best choice for the viewing windows of manned spacecraft. These materials coated with magnesium fluoride were used in the Gemini missions. The window consisted of two sections: the outer one was unsealed and made of Vycor; the sealed inner window of the pressurized cabin was aluminum silicate glass.

Solar cell covers probably account for the greatest use of optical materials on spacecraft. The covers serve three major purposes. They provide protection for the sensitive silicon photovoltaic modules from damage in mishandling; although the cover material is susceptible to damage by rough usage it is less so than the silicon semiconductor material. They also provide temperature control to the solar array because the emittance of the cover surface can be controlled by the use of antireflectance coatings. In some cases the cover sheets are attached directly to the solar cell silicon element with a compatible adhesive. This is the design most generally used. In some cases the cover sheet is held in a metal framework and is spaced away from the silicon element. In this case the "green-house" effect has to be considered in the design and selection of materials with correct thermal radiation and conduction properties to compensate for solar heating, to insure efficient operating temperatures for the solar array. This type of construction was used in some Vanguard, Explorer, and Telstar satellites. The third function of the cover sheet is to protect the sensitive solar cell element from cosmic dust, micrometeorite damage, and radiation damage. In the case of the cemented cover sheet, filter coatings are applied which also protect the adhesive from radiation damage, particularly UV radiation damage to the adhesive.

The lightest and most economical solar cell cover sheet material is the 6-mil thick fused silica microsheet. The material in common usage is Corning 0211 with a blue filter (OCL 207) and a magnesium fluoride antireflectance coating. This material is susceptible to radiation damage. Laboratory tests have shown that 500 hr in simulated sunlight (ultraviolet light from a xenon source) of 5 times the intensity of normal sunlight caused the glass to turn brown. A flux of 5×10^{10} e/cm^2 and 1 MeV energy caused a change of 4% in the solar cell output of a module covered with microsheet. Protons at 19 MeV and a flux of 5×10^{11} p/cm^2 caused darkening of microsheet (Ref. 1).

Some orbit conditions and planned mission lifetimes will permit the selection of microsheet as a solar cell cover material. The Orbiting Geophysical Observatory (OGO) series of satellites designed by Space Technology Laboratories is an example of this selection of optical materials where the designers are estimating an allowable 20 to 25% degradation of solar cell output during the mission duration.

A fused silica (quartz) cover sheet 0.04-in. thick will shield against 400 keV electrons, 0.07 in. of quartz will shield against 800 keV electrons. Electron fluxes of 10^{14} to 10^{16} e/cm^2 cause slight coloration in quartz. At 22 MeV quartz cover sheets darkened with 2×10^{13} p/cm^2. However, protons cannot be completely shielded with quartz, 0.12 in. of quartz will shield out all protons less than 20 MeV. These results are from laboratory simulation studies (Ref. 1). Up to 500 hr and five times the sun intensity in simulated solar ultraviolet radiation caused no change in the transmittance of quartz.

Synthetic sapphire which was used in Telstar 1 and 2 (Chapter 20, Ref. 1) is the most radiation resistant solar cell cover sheet material. Simulation tests have shown that there was no detectable change in transmittance after 500 hr in five times the sun's intensity in simulated solar ultraviolet radiation. A thickness of 0.04 in. of sapphire showed no detectable change when irradiated with 3×10^{12} p/cm^2 at 19 MeV. A thickness of 0.030 in. of sapphire has a stopping power of 18 MeV for protons.

At LMSC fused silica, Corning 7940, has been recommended for most satellite solar cell cover sheet applications. The thicknesses recommended are 0.006 in. for close in orbits and 0.020 in. for higher orbits where electron and proton radiation damage are expected. The adhesive is General Electric Silicone RTV 602, the magnesium fluoride antireflectance coating and the OCL 207 (Optical Coatings Laboratory, Inc.) blue filter are also recommended (Ref. 2).

The Ranger and Mariner series of space probes designed by Jet Propulsion Laboratory used the 0.006-in. thick Corning 0211 glass with antireflectance and UV filter coatings. The adhesive was the GE RTV 615 silicone.

Radiation damage to solar cell cover plate materials has been the subject of many laboratory simulation experiments. A number of spacecraft have carried solar cell materials test panels. The TRS (Tetrahedral Research Satellite) designed and built at Space Technology Laboratories (now TRW systems) was used entirely for testing of solar cell materials in space.

The complication in simulation testing is the equal energy-equal damage concept with respect to radiation damage. Some tests have indicated that this criterion does not hold for electrons and gamma radiation but that flux (i.e., number of particles of a given energy per unit area and energy) are the governing criteria. For protons of energies below 20 MeV the damage is related to the proton energy. Above 20 MeV the damage is related to flux and energy (Ref. 1).

Another consideration in evaluating solar cell materials is the separation of damage to the cover sheet material, the adhesive, the sensitive silicon element, and the contribution each factor makes to the overall operational behavior of the entire module. Tests seem to indicate that penetrating radiation that is not stopped by the cover sheet can cause darkening of the adhesive and radiation damage to the base material (n layer of p-on-n or p layer of n-on-p cells) of the silicon. Darkening (loss of transmission) of the cover sheet material itself also contributes to the loss of efficiency of solar cell modules in the space environment.

2.4 MATERIALS FOR LUBRICATED SYSTEMS

Table 2-9 summarizes lubricant recommendations for equipment exposed to the space environment. Recommendations are made separately for different applications, since there is no one type of lubricant that is best for all. The discussion amplifies the information in the table.

Design factors that influence the choice of lubricant or self-lubricating material include:

- Definition of lifetime

  Based on torque. Gyro bearings must operate with a very low and constant torque; a slight increase in the viscosity of the lubricant and the bearing torque means failure of the gyro to operate properly. Bearings for gears and motors can operate until actual freezing of the bearings.

  Based on wear. Lifetime may be limited by the accumulation of wear debris or by the actual consumption of the material, as in the operation of motor brushes or ball bearings with special plastic retainers, thin film lubricants, self-lubricating materials, or laminar solids.

  Based on noise level. Lifetime of electrical contacts is limited by the electrical noise generated.

- Environment

  Vacuum
  Radiation
  Reactive materials

- Operating conditions

  Speed
  Load
  Temperature
  (1) High-temperature operation
  (2) Low-temperature operation
  (3) Operation over an extended temperature range

Table 2-9

RECOMMENDED LUBRICANTS

Application	Conditions	Recommendation	Specification	Notes
Small ball bearings (3/4-in. bore, 2-in. OD or less)	Temperature: - 65 to 225°F Speed: to 8000 rpm Pressure: ambient to 10^{-9} Torr Operating life: 1 yr continuous operation Radiation: 1×10^{6} rad	Versilube G-300 grease		Use double metal shields on bearing; use at 200,000 DN[a] or below. Thermal-vacuum tests and flight experience of Nimbus 1 solar array orientation mechanism demonstrated that the lifetime of this material in vacuum is drastically reduced when bearing operating temperature was above 300°F. See also Table 2-10 for lifetime data of R3 ball bearings lubricated with G-300 grease. See also pp. 221-222 of Ref. 1.
		Aeroshell 15 grease	MIL-G-25013D	Use double metal shields on bearing; use at 200,000 DN or below
	Same as above with additional requirement of low starting torque at low temperature (-30° F or lower)	Versilube F-50 oil		Vacuum impregnate into phenolic retainer; use double metal shields
	Temperature: -65 to 400°F Speed: to 8000 rpm Pressure: 10^{-9} Torr Operating life: limited Other: nearby surfaces such as camera windows sensitive to clouding by volatiles from oil lubricants Radiation: 1×10^{8} rad	Hi-T-Lube, dry-film lubricant		Dry film lubricant. Apply to races, retainers, and balls; use unshielded bearings; limited to 8×10^{6} revolutions bidirectionally based on R3 size bearing; bearing should be run in at low speed and excess lubricant removed by blowing out with clean, dry gas
	Same as immediately above except Radiation: 1×10^{6} rad Temperature: -65 to 300°F	Unlubricated bearing with retainer of Rulon C		Use unshielded bearing; good to 3×10^{7} revolutions, based on R3 size bearing
	Temperature: -65 to 400°F Speed: to 8000 rpm Pressure: 10^{-9} Torr Operating life: limited Radiation: 1×10^{9} rad Other: required to carry electric current	Bearing with full complement (i.e., no retainer) of silver-plated balls of tool steel (e.g., M-2, M-50) and races of 52100 steel		Limited to 1×10^{7} revolutions, based on R3 size bearing

Table 2-9 (cont'd)

Application	Conditions	Recommendation	Specification	Notes
All bearings at atmospheric pressure (760 Torr) in a sealed unit	-65 to 250°F continuous with peak temperature to 300°F	MIL G-23827	MIL-G-23827	Synthetic oil based grease, best for general application, corrosion inhibited; good to 400,000 DN
	-20 to 300°F	MIL-G-3545	MIL-G-3545B	Sodium soap thickened petroleum oil base, good high-speed performance; good to 500,000 DN
	-100 to 450°F	MIL-G-25013	MIL-G-35013D	Synthetic oil based grease, wide temperature range; good to 200,000 DN
	-65 to 250°F	MIL-L-6085A	MIL-L-2085A Amendment 2	Synthetic oil, corrosion inhibited, good general purpose oil
	-65 to 350°F	MIL-L-7808	MIL-L-7808E	Synthetic oil, wide temperature range; for high temperature use should be supplied by spray or oil mist
Gyroscope bearings	Sealed unit Temperature: -45 to 165°F Require lubricant with minimum change in viscosity in operating temperature range	Teresstic V-78		Oil should be vacuum impregnated into linen- or paper-based phenolic retainer, bearing double shielded
Larger, low-speed bearings	Temperature: -65 to 150°F Speed: to 1800 rpm Pressure: 10^{-9} Torr Radiation: 1×10^6 rad	Unlubricated bearings with retainers of Rulon C or Duroid 5813		Tests to 12,000-hr operating time have been conducted in simulated space environment tests with Rulon[b]
	Temperature: -65 to 400°F Speed: to 1800 rpm Pressure: 10^{-9} Torr Radiation: 1×10^8 rad	Unlubricated bearings with retainers of metal composites such as Sinitex or Westinghouse silver or copper matrix based material		

All bearings	Exposure to the temperatures of cryogenic liquids Radiation: 1×10^6 rad	Unlubricated bearings with retainers of Rulon C		Races of 440C are preferred over 52100 for corrosion resistance
	Exposure to strong oxidizers (liquid oxygen, nitrogen tetroxide) Radiation: 1×10^6 rad	Unlubricated bearings with retainers of Rulon C		Races of 440C are preferred over 52100 for corrosion resistance
Lightly-loaded sliding surface	Exposed to vacuum Temperature: -65 to 165° F	Dry lubricant such as MoS_2 or Teflon on hard metal surfaces; plastics such as nylon, phenolic laminates, and Delrin; porous metals impregnated with oil and/or MoS_2		Radiation effects on plastics should be considered before they are used. Otherwise satisfactory for 1-yr orbit intermittent operation
Heavily-loaded sliding surfaces	Exposed to vacuum Temperature: -65 to 1200° F	For lower temperatures, bonded solid film lubricant which is compatible with a grease or oil and a low volatility grease or oil; for example, Electrofilm 4856 and 4896, with Versilube F50 or Aeroshell 15; bonded solid film lubricant selected for use with hard metals and high loads such as Electrofilm 77-S; dispersion of MoS_2 in grease such as Molykote Type G; for higher temperatures, sodium silicate or ceramic bonded MoS_2 or CaF_2		Satisfactory for 1-yr orbit, intermittent operation
Fine-pitch gears	Sealed unit Temperature: -65 to 250° F Radiation: 1×10^7 rad	Aeroshell 7, Aeroshell 15, G-300 greases on Nitrided Nitralloy 135, heat-treated AISI 4340 (46-50RC), hard anodized aluminum alloy 2024T3, heat-treated Berylco 25, beryllium-copper alloy	MIL-G-25013D MIL-G-23824 MIL-L-7808	

Table 2-9 (concl'd)

Application	Conditions	Recommendations	Specification	Notes
Fine-pitch gears	Exposed to vacuum Temperature: -65 to 250°F	Gears of composite materials such as Sinitex and sintered materials such as the Westinghouse copper and silver base compositions		Service tests required to establish lifetime and reliability
	Temperature: -45 to 165°F	Plastic gears such as Teflon, polyformaldehyde or polyamide		Service tests required to establish lifetime and reliability; consult Chap. 12 of Ref. 1 for effects of radiation on plastics
High-speed, moderate heavily loaded gears	Sealed unit Temperature: -45 to 250°F Speed: to 30,000 rpm		MIL-L-7808E MIL-G-25336	MIL-G-25336 is modification of MIL-L-7808 and should be used for heavy loads
Fittings on oxidizer or oxidizer pressurization lines		Non-adhesive backed Teflon tape Permacel 412 or Kel-F tape; inert highly halogenated lubricant such as Fluorolube oils and greases		

(a) DN equals bearing bore diameter (mm) times speed (rpm).
(b) "Lubrication Evaluation," LMSC-A707712, Oct 1964.

Duty cycle (continuous or intermittent)
Type of motion (sliding, reciprocating, rotating, oscillating, make and break contact)
Current density (with electrical contacts)
Brush force (with electrical contacts)

- Parts

  Geometry and type of part (e.g., bearing, gear, cam, and brush)
  Materials and material combinations
  Hardness
  Surface finish
  Tolerances
  Effect on other parts (e.g., condensation of oil on nearby optical surfaces)

- Installation and handling variables

  Alignment
  Balancing
  Mounting rigidity
  Run-in
  Cleanliness
  Sealing and protection against corrosion and contamination

- Lubrication (other than type of lubricant)

  Amount of lubricant
  Manner of application
  Replenishment

The recommendations made in Table 2-9 should serve as guides in determining types of lubrication that may be most suitable. They are based on limited tests and experience with satellites and space probes launched to date. Equipment qualification tests should include tests to confirm the usefulness of the lubricant or self-lubricating material selected by the designer and to establish their lifetime and reliability for specific applications.

The radiation stability of the lubricants recommended in the following sections should not be a limiting factor for lifetimes up to 1 yr for all the altitudes considered here. Based on information now available, the problems of vacuum operation would not result in different recommendations at different altitudes from 100 to 2300 nm (115 to 2650 sm).

2.4.1 Small, Lightly Loaded Ball Bearings

Typical examples of small, lightly loaded ball bearings are those in drive motors for servomechanisms. These are generally fractional horsepower motors operating at speeds up to 24,000 rpm, with 8000 rpm being an average speed for induction motors.

The shaft bearings in gear trains of the drive motors range from high to very low speeds. Bearings used in cam followers will generally operate at much slower speeds than the motor bearings. Examples of these are the sealed drive of the Ranger 1 through 9 and the Mariner 2 high gain antenna, the Nimbus solar panel drive and the Pegasus meteoroid detection array erection mechanism. The loads on these bearings are small. There are no gravitational loads once the device is placed in orbit (though there may be heavy loads during the launch phases). The principal loads in orbit are the equivalent radial loads due to the transmission of power (on the order of several ounces to several pounds in typical cases). Dynamic loads arise from centrifugal forces in the bearings and from unbalance or misalignment. The latter forces should be kept small by proper design and fabrication.

The passive temperature control techniques should maintain the temperatures of the mechanisms in which the bearings are used within a range from -45 to 165° F. Higher temperatures (200 to 300° F) are reached on motor bearings, due to power losses within the motor windings. Temperatures of motor windings and bearings can be significantly higher under vacuum conditions than in air, because of the absence of convection cooling; and this factor should be considered in designing proper conductive paths for dissipation of the heat to obviate regions of high temperature. High temperatures promote evaporation of oils and greases, under vacuum conditions, and lead to early lubrication failures. Low temperatures, on the other hand, increase the viscosity of the lubricant and can result in stalling drive motors if the temperature becomes very low and the lubricant becomes very viscous. The recent failure of the Nimbus discussed in Sec. 1 shows that the lifetime of grease- or oil-lubricated high speed ball bearings is significantly affected by the operating temperature in vacuum. The operation of the ball bearings may be continuous or intermittent, depending upon the duty cycle required. Bearing rotation can be unidirectional, bidirectional, or oscillating back and forth through small angles. Bidirectional or oscillating motion can reduce the lifetime, particularly with bonded films of MoS_2.

Recommended lubricants. Small, lightly loaded ball bearings should preferably be lubricated with a low volatility grease, except where special considerations (see below) make oil or MoS_2 films more desirable. Based on information available on the performance of ball bearings lubricated with greases and operated under vacuum conditions, GE Versilube G-300 or Aeroshell 15 silicone greases are recommended for general applications and for most orbital conditions.

Where the ball bearings are required to operate at low temperatures (e.g., below -30° F), a low volatility oil is preferable to grease. Based on information available on the performance of ball bearings lubricated with oils and operated under vacuum conditions, GE Versilube F-50 silicone oil is recommended for such applications and for most orbital conditions. The oil should be vacuum impregnated into a linen- or paper-reinforced phenolic retainer.

Where the ball bearings are required to operate in vacuum at high temperatures (e.g., above 225° F), some means of relubrication (see below) or MoS_2 films should be considered. The use of MoS_2 films should be restricted to speeds below 8000 rpm and for lifetimes below 8×10^6 revolutions of unidirectional operation or 5×10^6 revolutions of bidirectional operation. Based on information available on the performance of ball bearings lubricated with MoS_2 films and operated under vacuum conditions, Hi-T-Lube is recommended for applications where MoS_2 appears suitable.

Where nearby or adjacent parts are best lubricated with oil (e.g., gears in a partially enclosed gear train, where grease lubrication would create excessive torque and cause stalling), the ball bearings can also be lubricated with the same oil as used for the other parts. Based on information available on the performance of ball bearings lubricated with oils and operated under vacuum conditions, GE Versilube F-50 silicone oil is recommended for such applications.

Where nearby or adjacent parts are sensitive to the condensation of oil films (e.g., a horizon sensor can be affected by infrared absorption in condensed oil films on its optical surfaces), bonded films of MoS_2 can be used.

Where the operation of the device requires that there be no outgassing from the lubricant (e.g., evacuated and sealed electronic tubes with rotating parts), films of MoS_2 such as Hi-T-Lube, or thin films of silver or gold are recommended. The use of thin films of silver or gold requires special bearing design, handling, processing, and care in installation; their lifetime and reliability are limited. This behavior has been shown by service tests conducted on some typical applications.

Where the ball bearings must be able to conduct an electrical current through them (from inner rings to outer rings), thin films of silver are recommended.

Additional comments. Periodic relubrication of ball bearings with fresh lubricant will, of course, prolong the lifetimes of the ball bearings indefinitely. Conventional relubrication from oil and grease reservoirs is often impractical with these bearings on spacecraft mechanisms. The use of porous retainer rings impregnated with the oil and placed near the bearings permits the replacement of oil that evaporates from the bearing itself, and this procedure can extend operating lifetimes.

Bearings lubricated with grease or oil should be double shielded (both to reduce lubricant evaporation and to minimize contamination). Machined linen- or paper-based phenolic (e.g., Synthane) retainers are preferred for operation at high speeds with oil lubricants. Ribbon-like retainers have been less satisfactory than machined Synthane retainers with oil lubrication. The oil should be vacuum impregnated into the retainer; if grease is used, a retainer, which is vacuum impregnated with the same oil that is used in the grease, may be used. For example, if G-300 silicone grease is used, the retainer may be impregnated with F-50 oil, since the G-300 grease is a combination of the F-50 oil in a lithium-base soap. Table 2-10 (Ref. 5) shows some of the most recent data on lifetimes of R-3 ball bearings in simulated space environment tests.

On ball bearings lubricated with grease or oil, the running torque may decrease slightly at first, due to channeling of the grease or removal of excess oil. This is

Table 2-10

LONG-TERM SIMULATED SPACE ENVIRONMENT TESTS OF LUBRICANTS

Materials		Temperature (°F)	Pressure (Torr)	Radiation[b] Dose From Co 60 Source (R)	Results
Lubricant	Bearings[a]				
Versilube F50 Oil	AISI 52100 Synthane retainer	185	9×10^{-9}		Failed at 13,652 hr
Versilube F50 Oil	AISI 440C Synthane retainer	250 max	4×10^{-9}		Still running at 10,408 hr
Versilube F50 Oil	AISI 440C Synthane retainer	160	3×10^{-9}	3.4×10^{7} (8057 hr)	Still running at 10,936 hr[c]
Apiezon K Oil	AISI 440C Synthane retainer	215 max	1×10^{-8}		Test discontinued at 20,019 hr
Teresstic V78 Oil	AISI 440C Synthane retainer	220 max	3×10^{-8}		Failed at 14,340 hr
Teresstic V78 Oil	AISI 52100 linen-based phenolic retainer	170	3×10^{-9}	3.4×10^{7} (8857 hr)	Still running at 11,140 hr
USAF Oil ML061-97(DC7024)	AISI 440C Synthane retainer	205 max	4×10^{-7}		Still running at 16,422 hr
Versilube G-300 Grease	AISI 440C ribbon retainer	175 max	2×10^{-7}		Still running at 22,411 hr
Versilube G-300 Grease	AISI 440C ribbon retainer	230 max	1×10^{-7}		Still running at 22,385 hr
Versilube G-300 Grease	AISI 440C ribbon retainer	300 max	4×10^{-7}		Failed at 4359 hr (grease hardened)

Versilube G-300 Grease	AISI 440C ribbon retainer	165	3×10^{-9}	3.47×10^{7} (8542 hr)	Still running at 10,635 hr[c]
Aeroshell 15	AISI 440C ribbon retainer	250 max	2×10^{-7}		Still running at 15,193 hr
Aeroshell 15	AISI 440C ribbon retainer	195	7×10^{-9}	1.7×10^{7} (4097 hr)	Failed at 5518 hr
Martin-Rockwell EG 509	AISI 440C ribbon retainer	200	1×10^{-8}		Still running at 23,309 hr
USAF MLG-61-92	AISI 440C ribbon retainer	200	1×10^{-9}		Still running at 18,170 hr
USAF MLG-62-142	AISI 440C ribbon retainer	165	1×10^{-9}		Still running at 18,192 hr

(a) Deep groove, double shielded R3-size ball bearings (3/16-in. bore, 1/2-in. OD) 8000 rpm.
(b) Average energy of γ-radiation is 1.25 MeV.
(c) Total running time in vacuum; value in parentheses is time of exposure to vacuum and Co 60 radiation.

followed by a nearly constant or slightly rising torque for perhaps 90% of the lubricant lifetime. During this time, the torque should be well within the operating specifications. Following this period, the torque increases at an accelerating rate, and lubrication failure finally results in excessive wear and rapid seizure of the high-speed ball bearings themselves. Ball bearings lubricated with bonded MoS_2 films retain a more nearly constant torque during operation (following an initial reduction in torque due to wear in), and the final increase in torque leading to failure is much more rapid than with oil and grease lubrication.

Ball bearings lubricated with bonded films of MoS_2 should be carefully run-in and blown free of loosened debris before operation. Excess MoS_2 in powdered or slurry form should also be removed by run-in and careful blasting with dry air.

2.4.2 Fine-Pitch Gears and Gear Trains for Servomechanisms

Operating characteristics. Speeds of gear trains vary from a high of about 24,000 rpm down to several revolutions per day. The loads due to the transmission of torque are generally small, increasing as the speed is reduced. Temperatures in typical applications cover a range of from -45 to 165° F. The primary consideration in sealed gear trains lubricated with oil is the effect of temperature on oil viscosity and lubricating qualities. At low temperatures, excessive oil viscosity can result in large increases in torque and cause the device to stall, whereas at high temperatures excessive thinning of the oil can occur with resultant poor lubricity. The operation of the gears may be continuous or intermittent; gear rotation can be unidirectional, bidirectional, or oscillating back and forth through small angles.

Recommended lubricants. Very little data are available on which to base recommendations for attaining a specified lifetime for gears operating under specified conditions. Where the gear trains are sealed by labyrinth or rubbing seals so that there is some protection against the vacuum environment; oils and greases such as those covered by MIL-L-7808 or MIL-G-23827, Versilube F-50 oil, or G-300 grease appear most useful. The gear material itself is also important. Where metal gears are used, hard anodized aluminum 2024T3 appears satisfactory, and has the advantage of lighter weight as compared to stainless steel gears. Low alloy high-strength steel gears and pinions of AISI 4340 heat-treated to Rockwell C 46 to 50 have been successfully used in Ranger and Mariner 2 sealed antenna actuators. Worm gears and pinions of Berylco alloy 25 (Be-Cu) or AISI 303 stainless steel or worms lubricated with Electrofilm 4396, a proprietary MoS_2 dryfilm lubricant, operated successfully on Rangers 6–9 and Mariner 2. Sintered D-10S (oil impregnated bronze plus MoS_2) has given good results. Plastic gears, especially sintered nylon (e.g., Nylasint 64 and Nylasint 66), molded nylon (e.g., Zytel), phenolic laminates, Delrin, and Lexan should also be considered, particularly for gear trains exposed to vacuum. Plastic gears are suitable for lightly loaded applications and should be initially lubricated and run-in with oil or grease.

2.4.3 Large Low-Speed Support Bearings

Operating characteristics. Typical examples of large, low-speed support bearings are those used for rotating satellite payloads and other devices. These may be either ball or roller bearings that vary in size up to several feet in diameter. Speeds are generally low, below ~ 100 rpm.

These bearings may be subjected to heavy shock and vibration loads during the launch phases. Once in orbit, loads are low. The principal loads are the equivalent radial loads due to the transmission of power, and may be on the order of several pounds.

Temperatures in typical applications cover a range from - 45 to 165° F.

The operation of these bearings can be continuous or intermittent; rotation can be unidirectional, bidirectional, or oscillating back and forth through small angles.

In most cases it is difficult or impractical to enclose these bearings, so that they are required to operate under high vacuum conditions.

Recommended materials. Experience with OSO and other satellites indicate that conventional bearings can be used. If design analysis indicates a problem that arises from shock or vibration, the load may be supported by blocking which is removed after launch. The large bearings in OSO-1 which permitted the "sail" to remain fixed while the "wheel" rotated for spin stabilization were AISI 52100 steel with a reinforced fluorocarbon retainer lubricated with a MoS_2 slurry.

2.4.4 High-Speed Gears and Bearings Transmitting Moderate to Heavy Loads

Operating characteristics. Typical examples of high-speed gears and bearings transmitting moderate to heavy loads are those used on turbine drive units for liquid propellant pumps. Speeds on such units vary upward from 10,000 rpm; 30,000 rpm is typical. The gears and bearings are subjected to loads that arise from transmitting several hundred horsepower. Ambient temperatures in typical applications cover a range from - 45 to 165° F. In typical applications, the gears and bearings are enclosed and sealed from the vacuum environment of space by labyrinth seals.

The total running time of these gears and bearings is generally limited to less than a few hours. This running time may be intermittent in engines that have restart capability and spread over several much shorter operating periods separated by several months in deep space planetary missions.

Recommended lubricants. The use of a low volatility oil and labyrinth seals minimizes evaporation. Under the usual conditions, the main emphasis should be placed on gear lubrication rather than the space environment. Metal gears and bearings are needed because of the speeds and loads. The recommended lubricants are those that are presently used, namely, MIL-L-7808 type oils and, for higher loads, MIL-G-25336. Radiation stability should be satisfactory for a 1-yr orbit at all altitudes.

2.4.5 Lightly Loaded Sliding Surfaces

Operating characteristics. There are many applications, such as cam operated switches and other mechanisms, reset and positioning mechanisms, and guides that involve sliding contact between two surfaces that are lightly loaded and that move at relatively slow speeds. Such parts may be difficult to enclose and seal, so that they are often required to operate while exposed to the hard vacuum of space. Temperatures are nominal, with a usual operating range from - 65 to 165° F. Operation is continuous or intermittent, and may also be unidirectional or bidirectional. Sometimes such a device is only required to operate once.

Recommended lubricants. The following materials and lubricants are useful in such applications.

- Surface coatings of solid lubricants, such as MoS_2 and Teflon, on hard metal substrates – Burnishing MoS_2 on to the surface is adequate in some cases; in more severe applications, a bonded film should be used.
- Plastics such as phenolic laminates, nylon, and Delrin – These have sufficient strength and wear resistance for many applications in this category. They have less weight than similar parts of steel or other metals, and they eliminate the need for lubrication to prevent cold welding in hard vacuum.
- Powder metallurgy compacts, impregnated with MoS_2 and/or oil.

Additional comments. Contacting pressures should be kept to a minimum for least frictional drag and wear. Where contacting pressures are high, hardened materials should be used. Austenitic stainless steel rubbing against austenitic stainless steel is a poor choice, in that there is a maximum tendency for cold welding. Contact between similar metals should be avoided and wherever possible dissimilar metal pairs should be used.

When surface coated, the base metal should be pretreated before applying the coating (e.g., phosphating treatment for steel) and the coatings should be run-in after application. These precautions are discussed in Chapter 12 of Ref. 1.

2.4.6 Heavily Loaded Sliding Surfaces

Operating characteristics. Applications that involve sliding surfaces under heavy loads include such mechanisms as the gimbal sleeve bearings and actuator rod spherical bearings in propulsion systems. Such applications may also involve heavy shock and vibration loads during the launch phase.

The normal operating conditions may call for long idle periods interrupted by short-time operation, with the parts oscillating back and forth through small angles rather than through complete revolutions. Temperatures are nominal, within the range - 65 to 200° F. Excessive wear and galling can cause the parts to "freeze" and become inoperable. The geometry of the parts usually prevents the sliding surfaces from being directly exposed to the vacuum of space.

Recommended lubricants. For such applications, the following types of lubrication should be satisfactory and adequate for all orbital conditions considered:

- Bonded solid film lubricant in conjunction with an oil or grease – The solid film lubricant must be compatible with the oil or grease. Electrofilm 4856 plus MIL-G-23827 grease is an acceptable combination; for temperatures below -30° F, an oil conforming to MIL-G-25336 may be preferable to the grease.

- Bonded solid film lubricant, selected for use at high loads and with hard metals, such as Electrofilm 77-S (MoS_2 plus graphite in a thermosetting resin).

- Dispersion of MoS_2 in grease, such as Molykote Type G (MoS_2 in a mineral oil + lithium soap base grease)

2.4.7 Gyroscope Bearings

Operating characteristics. Gyroscope bearings must operate with the lowest possible torque and with a minimum change in torque with temperature. Very small increases in torque, due to wear, lubricant breakdown, or any other cause result in large errors in the operation of the gyroscope and are considered as failures. Consequently, lubrication is more critical then in most other applications, where the bearing torque is only a small part of the drive requirements, and relatively large increases in bearing torque can be tolerated.

Gyroscope bearings usually operate in a hermetically sealed unit that is often filled with an inert gas, such as helium. As a result, the bearings are not exposed either to the hard vacuum of space or to a reactive atmosphere.

Gyroscope bearings are usually of the oil-lubricated ball bearing type or the self-acting gas type. Speeds are upward of 10,000 rpm, continuous and unidirectional. Loads are small. Sizes vary up to perhaps 1-in. in diameter with the ball bearing type. Temperatures are usually within the range -45 to 165° F.

Recommended lubricants (for ball bearing type). Because gyroscope bearings are protected against exposure both to vacuum and to reactive atmospheric constituents, the choice of lubricant is based primarily upon the service conditions of load, speed, and lifetime. The lubricating oils normally selected for this service have sufficient radiation resistance for 1-yr service in the orbits considered.

Lubricants generally used are highly refined mineral or synthetic oils (e.g., Teresstic V-78). These are vacuum impregnated into the bearing material, which is made of a porous material such as Synthane (a paper-base phenolic laminate), from which the oil is fed during bearing operation.

2.4.8 Lubricants in Oxidizers

Operating conditions. Liquid rocket engines employ strong oxidizers such as liquid oxygen. Materials in contact with these must be inert to prevent explosions.

Recommended materials. Fittings on oxidizer and oxidizer pressurization lines may be lubricated with highly halogenated materials such as non-adhesive backed Teflon tape, and Kel-F, or Fluorolube oils and greases.

2.4.9 Slip-Ring Assemblies

Operating characteristics. Slip-ring assemblies are used for transmitting power and signals in spacecraft devices. Lifetimes are limited by wear rates and excessive electrical noise, which interferes with the transmission of intelligible signals. Metal-to-metal contact is desirable from the standpoint of reducing noise but, on the other hand, results in more rapid wear than if lubricant films could be used and eventually result in excessive noise.

Recommended materials. Metal contacts impregnated with molybdenum disulfide should be considered for slip rings or brushes operating in vacuum. Graphite and metals impregnated with graphite should not be used unless the assembly can be pressurized with air and hermetically sealed. For miniature slip ring assemblies with brushes and rings of precious metals, low noise and wear can be achieved by providing a partial pressure on the order of 10^{-6} Torr of silicone oil; a porous material impregnated with the oil and mounted adjacent to the slip ring assembly provides a suitable reservoir for maintaining an oil vapor atmosphere, even if the unit is not sealed tightly. Studies at LMSC (Ref. 6) have shown that the slip-ring mechanism average peak electrical noise was significantly reduced when a low partial pressure of the nonhalogenated silicone oil DC 704 was introduced into the vacuum test chamber. Graphite brushes impregnated with molybdenum disulfide should be considered for d-c motors that must operate in vacuum.

A new brush material (CLB alloy) for slip-ring assemblies for conduction of high currents (75A at 430 V) was developed at LMSC (Ref. 7) under contract to Arnold Engineering Development Center (Contract AF 40(600)-1070). This composite brush material nominally 82.5–85% Ag, 2.5% Cu, and 12.5%–15% MoS_2 operated for 700 hr in a vacuum of 10^{-9} Torr (300 A/in.^{2} at 424 in./min) with peak electrical noise of 4 mV. There was no appreciable wear of the silver ring, and 0.015 in. total wear to the powder metallurgically produced brush material was observed.

2.5 POLYMERIC MATERIALS FOR ADHESIVES, SEALS, STRUCTURAL, AND RELATED APPLICATIONS

The principal environmental effects that must be considered in selecting optimum polymeric materials for adhesive, seal, structural, and related applications have been discussed in Ref. 1, Chapters 13, 14, and 15. This section will present relative ratings of these materials for space use based on environmental effects.

Explanation of the rating system for each of the classes of materials is given in Tables 2-11, 2-15, and 2-19 while Tables 2-12, 2-16, and 2-20 are materials selectors covering several typical spacecraft missions. Except for a few metallic seal materials, required for the more severe applications, all the materials presented are polymers.

These materials are rated on the basis of their suitability to perform a given function and on their stability to the environments encountered in spacecraft service. The recommended materials belonging to the several rated categories are listed in Tables 2-13, 2-14, 2-17, 2-18, and 2-21 along with summarized information on space environmental effects bearing on their selection (Refs. 1 and 2).

Only representative commercially available material types widely used by the aerospace industry are rated. No attempt has been made to rate all possible commercial materials or those that are considered experimental. Most polymeric materials are identified by their generic chemical names. However, adhesives because of their complex formulations and proprietary nature are identified by trade names. It is therefore necessary to rate examples of a specific manufacturer's product. It should not be construed that endorsement is given or implied to a given product or its manufacturer. Other comparable and competitive formulations having properties similar to those of the types recommended may be equally suited for the intended application, but are not included in order to keep the list at a manageable level and because of the lack of supporting test data available.

The relative ratings presented were made for materials applications on space vehicles for mission durations of 0.1 and 1 yr. Each orbital altitude and mission duration establish the natural environment associated with materials service. In particular, the total incident dose of penetrating radiation varies greatly with these parameters and also with the degree of shielding afforded a material by the vehicle skin or component housing.

The use of the tables may be illustrated by randomly selecting a specific material application such as a high-energy propellant fuel seal for use in a vehicle to orbit at, say, an altitude of 1152 sm. If the seal is required to perform reliably for 1 yr in such an environment, reference to Table 2-16 shows that materials of relative rating A or A* are recommended. Table 2-15 gives the relative ratings and Table 2-17 identifies the materials. Since the application was that of a seal for fuel service, the choice is limited to A and A* materials. Further qualifications and limitations regarding the use of the materials are given under Remarks. Note that there may be some limitations to the use of some A materials, although considered satisfactory for propellant seal service from the compatibility standpoint, because of the high internal radiation dose encountered even with the shielding afforded by a 0.1-in. thick aluminum alloy skin (or equivalent). This dose approaches the limiting radiation dose for retention of useful properties of Teflon. Incorporation of additional shielding, however, such as the use of a sufficiently thick seal housing, would permit the use of A-rated materials for this mission.

As another example of the use of these tables consider an application involving the use of a polymer film for wrapping electrical cable harnesses on the external portions of an unmanned instrumented spacecraft. The purpose of the material is for thermal control, and analysis has shown that an aluminized polymer film will provide the required thermal properties. The orbit is beyond 2000 sm, the duration is 1 yr.

Table 2-11

RELATIVE RATING OF ADHESIVE MATERIALS

Relative Rating	Description
I	Recommended structural adhesives for high-strength metal-to-metal bonding applications
II	For moderate elevated and reduced temperature, high-strength, general purpose, metal-to-metal bonding applications
III	For general and specific non-structural bonding applications at moderate temperature and radiation
IV	For general and specific non-structural bonding applications at moderate temperature and low radiation

Table 2-12

MATERIALS SELECTOR FOR ADHESIVES IN SPACE MISSIONS

Mission Duration (yr)	Application	Materials Rating[a]			
		230-sm orbit	576-sm orbit	1152-sm orbit	2304-sm orbit
0.1	Surface	I, II, III	I, II, III	I, II, III	I, II, III
	Internal	I, II, III, IV	I, II, III, IV	I, II, III	I, II, III
1	Surface	I, II, III	I, II, III	I	I, II
	Internal	I, II, III, IV	I, II, III	I, II, III	I, II, III

(a) Relative ratings are given in Table 2-11 and lists of adhesives in Tables 2-13 and 2-14.

Table 2-13

RECOMMENDED ADHESIVES FOR SPACECRAFT BONDING APPLICATIONS FOR SIGNIFICANT SPACE ENVIRONMENTS

Relative Material Rating	Chemical Types	Coded Representative Commercial Adhesives(a)	Radiation Dose Limit (rad)	Temperature (° F)		
				Maximum		Minimum
				Short Time (1/2 hr)	Continuous	
I	Epoxy-phenolic	A1, B1, C1	10^9	650	500	-423
	Vinyl-phenolic	D1, B2		300	250	-423
II	Nitrile-phenolic	D2, D3, E1	5×10^8	250	200	-423
	Epoxy (filled)	B3, F1		250	200	-320
III	Nylon-epoxy	C2, B4	10^8	250	200	-423
	Neoprene-phenolic	G1		250	200	-100
	Nylon-phenolic	C3, G2		250	200	-100
	Modified phenolic	D4, B5		600	450	-100
IV	Silicones	H1	5×10^6	650	500	-65
	General Purpose Thermoplastic and Elastomeric Non-structural (low-strength) Adhesives	D5, D6	5×10^6	225	180	-40

(a) Code identification is given in Table 2-14.

Table 2-14

CODE IDENTIFICATION FOR REPRESENTATIVE COMMERCIAL ADHESIVES[a]

Code	Adhesive Commercial Designation	Manufacturer
A1	422J	Shell Chemical Co.
B1	HT-424	Bloomingdale Rubber Co.
B2	FM-47	Bloomingdale Rubber Co.
B3	Epon 8	Bloomingdale Rubber Co.
B4	FM-1000	Bloomingdale Rubber Co.
B5	HT-20	Bloomingdale Rubber Co.
C1	25-1	Narmco, Inc.
C2	Metlbond 406	Narmco, Inc.
C3	Metlbond MN3C	Narmco, Inc.
D1	EC-1469	Minnesota Mining & Mfg. Co.
D2	AF-6	Minnesota Mining & Mfg. Co.
D3	EC-1245	Minnesota Mining & Mfg. Co.
D4	EC-1639	Minnesota Mining & Mfg. Co.
D5	EC-847	Minnesota Mining & Mfg. Co.
D6	EC-776	Minnesota Mining & Mfg. Co.
E1	Plastilock 620	B. F. Goodrich Chemical Co.
F1	A-1	Armstrong Products Co.
G1	Cycleweld C-3	Chrysler Corp., Cycleweld Div.
G2	Cycleweld C-14	Chrysler Corp., Cycleweld Div.
H1	DC-A-4000	Dow-Corning Corp.

(a) Not necessarily limited to these formulations; other products may be satisfactory but test data on radiation stability are lacking; consult manufacturers and/or conduct experimental evaluation if other materials besides those listed are to be used in structural applications where radiation dose exceeds 5×10^7 rad; use chemical types listed wherever practical. See Table 2-13.

Table 2-15

RELATIVE RATINGS OF MATERIALS FOR PROPELLANT, HYDRAULIC, AND PNEUMATIC SEALING APPLICATIONS

Relative Rating	Applications and Qualifications
A	Preferred material for most applications. Suitable for long-term operation under the most severe environmental and service conditions.
A*	Preferred metallic seal materials but use should be restricted to those applications where the most severe environmental and service conditions are encountered and where operational requirements cannot be met with conventional elastomeric seals.
B	Satisfactory. Use wherever processing or design considerations do not permit use of A-rated material or when environmental or service conditions are less severe.
C	Not recommended. Use only in specialized, limited service applications.
D	Not to be used as seals under these conditions.

Table 2-16

MATERIALS SELECTOR FOR SEALS IN SPECIFIC SPACE MISSIONS

Mission Duration (yr)	Materials Rating[a]									
	Service			Vacuum Sealing		Service			Vacuum Sealing	
	Propellant	Hydraulic	Pneumatic	External	Internal	Propellant	Hydraulic	Pneumatic	External	Internal
	230-sm Orbit					576-sm Orbit				
0.1	AA*BC	AA*BC	AA*BC	SS*M	SS*MU	AA*BC	AA*BC	AA*BC	SS*M	SS*MU
1	AA*BC	AA*BC	AA*BC	SS*M	SS*M	AA*	AA*	AA*	N	SS*M
	1152-sm Orbit					2304-sm Orbit				
0.1	AA*BC	AA*	AA*	N	SS*M	AA*BC	AA*BC	AA*BC	N	SS*M
1	AA*	AA*	AA*	N	SS*	AA*	AA*	AA*	N	SS*

(a) Relative ratings are given in Table 2-15 and materials in Table 2-17; materials and relative ratings for vacuum seals are given in Table 2-18.

Table 2-17

SEAL MATERIALS FOR PROPELLANT, HYDRAULIC, AND PNEUMATIC APPLICATIONS

Material	Application	Continuous Operating Temperature Range (°F)	Penetrating Radiation Damage (rad)	Relative Ratings[a]			Remarks
				Propellant Service	Hydraulic	Pneumatic Service	
Metal Seals							
Aluminum (1100)	Static crush and wedge type seals (primarily for long-term service under severe environmental usage)	-300 to 700	$> 10^{12}$ (Not a limitation for spacecraft)	A*	A*	A*	Preferred seal materials for long-term service with high-energy fuels and oxidizers (e.g., N_2O_4, IRFNA, UDMH, hydrazine). Avoid dynamic applications (e.g., reciprocating and rotating shaft seals) Limited use in hydraulic and pneumatic seal applications where extreme environmental resistance is required.
Stainless Steel (301)		-300 to 1500		A*	A*	A*	
Nickel (Inconel)		-300 to 1800		A*	A*	A*	
Copper		-300 to 700		D	A*	A*	Same as above except unsatisfactory for oxidizer and fuel service. May be used for hydraulic and pneumatic seal applications where conditions warrant.
Carbon Steel		-300 to 1500		D	A*	A*	
Elastomeric Seals							
Butadiene-Acrylonitrile Rubber (Buna-N)	Static and dynamic seals (e.g., O-rings gaskets, boots) for applications such as access doors, air locks, pneumatic, hydraulic, and propellant seals for tanks, lines, and valves	-40 to 250		D	A (Except silicate and phosphate esters)	A	Preferred seal material for hydraulic and pneumatic service under static and dynamic conditions. Do not use with MIL-L-7808 diester oils, phosphate, and silicate esters. Depending on type and nitrile content, fair to good resistance to hydrocarbon fuels, oils, alkalis, and acids. Fair heat resistance with low permeability to gases.

(a) Relative ratings are given in Table 2-15.

Table 2-17 (cont'd)

Material	Application	Continuous Operating Temperature Range (°F)	Penetrating Radiation Damage (rad)	Relative Ratings[a]			Remarks
				Pro-pellant Service	Hy-draulic	Pneu matic Service	
Neoprene		-65 to 250	10^7 to 2×10^8	D	D (Except silicate esters)	B	Recommended for pneumatic service under static and dynamic conditions. Similar to Buna-N in radiation and temperature resistance. Wide range of mechanical properties depending on formulation type. Not recommended for propellant or hydraulic service. Good resistance to non-aromatic fuels, solvents, and fatty oils; good water and alkali resistance.
Natural Rubber		-40 to 225	10^6 to 3×10^8	D	D	B	Not to be used for propellant or hydraulic seal service but can be used for low pressure pneumatic seals. Poor weathering and aging resistance. Stress cracks in ozone and ultraviolet. Highest radiation resistance in absence of O_2. Fair gas permeability. Resistance to water, acids, and alkalis. Poor aromatic and aliphatic fuel, oil, and solvent resistance.

Butyl Rubber		-65 to 225	10^6 to 10^7	C (Except oxidizers)	D	A	Preferred seal material for pneumatic service under static and dynamic conditions. Limited service to hydrazine type fuels for short periods. Not compatible with oxidizers (e.g., N_2O_4, IRFNA) except for momentary (splash) contact. Good water and alkali resistance. Poor resistance to oils, aromatic and aliphatic gasolines, and all but oxygenated solvents. Least radiation-resistant elastomer. Gradually softens and at 6×10^7 rad becomes useless tarry fluid. Extremely low permeability to gases.
Fluoro-silicone Rubber LS-53		-100 to 400	10^6	D	D	B	Not to be used for propellant or hydraulic seal service but can be used for low pressure pneumatic seals. Static sealing applications only.
Styrene-Butadiene Rubber (Buna-S, GRS, SBR)	Static and dynamic seals (e.g., O-rings, gaskets, boots) for applications such as access doors, air locks, pneumatic, hydraulic, and propellant seals for tanks, lines, and valves	-40 to 250	5×10^6 to 6×10^8	D	D	B	Not recommended for propellant or hydraulic service although compatible for short period with hydrazine type fuels. May be used for low-pressure pneumatic seals. Poor resistance to aromatic and aliphatic gasolines, oils, and solvents. Fair to good resistance to water, acids, and alkalis. Better weathering and ultraviolet resistance than natural rubber, which it otherwise closely resembles in properties.

(a) Relative ratings are given in Table 2-15.

Table 2-17 (cont'd)

Material	Application	Continuous Operating Temperature Range (°F)	Penetrating Radiation Damage (rad)	Relative Ratings[a]			Remarks
				Propellant Service	Hydraulic	Pneumatic Service	
Silicone Rubber		-70 to 480	5×10^6 to 5×10^7	D	D (Except phosphate esters)	B	Not recommended for propellant or hydraulic service. Satisfactory for high temperature pneumatic seals but has highest gas permeability of any elastomer. Among highest heat resistance of all elastomers (smokes at 550°F). Fair water resistance; poor resistance to oils, gasoline, and solvents. Poor radiation stability. Becomes brittle and crumbly at 5×10^8 rad.
Viton (Fluororubber)		-65 to 450	5×10^6 to 5×10^7	C (Short term only)	A (Except phosphate esters)	A	Preferred seal material for pneumatic and hydraulic applications for high temperature, high pressure operation. Extremely low permeability to gases, and outstanding temperature resistance. May be used for limited short-term service with UDMH, but incompatible with oxidizers for more than momentary (splash) contact. Poor radiation stability, especially when irradiated in air at temperatures above 250° F. Can be used at high temperature with many types of fuels, lubricants, hydraulic fluids, and solvents, and is highly resistant to ozone and weathering. Good mechanical properties.

Elastomeric Sealants							
Polysulfide Rubber (RTV)	Sealant applied as a trowelable caulking to adherend surfaces. Intended for fluid pressure sealing of non-removable doors, bulkheads, windows, etc., to structure.	-65 to 180	5×10^5 to 5×10^6	D	C	C	Very good water resistance. Good alkali resistance. Fair acid resistance. Excellent hydrocarbon fuel and oil resistance. Used as fuel tank sealant. Poor vacuum and temperature resistance. Low radiation stability. Gradually softens to tarry material. Satisfactory for pneumatic and hydraulic sealing at low pressures (except silicate and phosphate esters). Fair to poor gas permeability. Poor mechanical properties. Room temperature curing.
Silicone Rubber (RTV		-70 to 480	5×10^6 to 5×10^7	D	D (Except phosphate esters)	C	Room temperature vulcanizing. Smokes at 210°F. See silicone rubber. Good vacuum stability but high permeability to gases. Incompatible with propellants and hydraulic fluids (except phosphate types). Limited use for sealing pneumatic lines. Poor adhesion to substrates – requires metal priming. Poor radiation stability. Excellent thermal resistance.
Thread Sealant	Thread sealant and retainer (nuts and bolts)	-75 to 250	10^8 est.	C	B	A	"Loctite A," Class 10 smokes at 450°F. Backup thread seal for low order pressure differential.

(a) Relative ratings are given in Table 2-15.

Table 2-17 (concl'd)

Material	Application	Continuous Operating Temperature Range (°F)	Penetrating Radiation Damage (rad)	Relative Ratings[a]			Remarks
				Pro-pellant Service	Hy-draulic	Pneu-matic Service	
<u>Plastic Seals and Retainers</u>							
Nylon (Hexamethylene-diamine Adipimide)	Back-up rings and low pressure moving seals and gaskets	-100 to 300	5×10^7 to 10^9	C	B	B	Recommended seal or retainer material for UDMH service only. Not for use with hydrazine or oxidizers. May be used for hydraulic or pneumatic seals, gaskets, and retainers. Good radiation and temperature resistance.
Teflon (TFE) Unfilled and filled with inorganic fibers (Polytetrafluoro-ethylene)		-400 to 500	10^5 to 10^6	A	C	C	Preferred seal or retainer material for propellant service for all but the longest exposures. Inert to practically all materials but lacks resiliency and subject to "cold-flow" under compressive stress. Very poor radiation resistance, particularly in air. At 10^7 rad it becomes a useless crumbly solid. In vacuum or inert atmosphere, radiation resistance is up to two orders of magnitude higher. Most heat-resistant polymer for seal use. Fillers improve radiation resistance slightly but lower chemical inertness. Filled Teflons hold seal better than solids.

Teflon (FEP) (Fluorinated-ethylene-propylene) (Filled and unfilled)	B	-400 to 450	10^5 to 10^6	A	C	C	Similar properties to TFE except more limited temperature range. Slightly higher radiation resistance. Preferred for seal service with propellants under all but longest exposures. Limited use in hydraulic, pneumatic seal application.
Kel-F (Polytrifluoro-chloroethylene)		-400 to 350	10^6 to 10^7	B	C	C	Similar properties to Teflon. Slightly more permeable to N_2O_4. Radiation stability slightly better than Teflon, above 10^8 rads becomes very brittle.

(a) Relative ratings are given in Table 2-15.

Table 2-18

MATERIALS FOR VACUUM SEALING

Relative Rating	Material	Application	Penetrating Radiation Damage (rad)	Continuous Operating Temperature Range (° F)	Remarks
	<u>Metal Seals</u>				
S*	Aluminum (1100 series alloy)	Crush and wedge seal one time application only	$> 10^{12}$	-300 to 700	Limited to vacuum applications whose seal will not be broken.
S*	Copper		$> 10^{12}$	-300 to 700	One time application only, when design does not permit use of preferred elastomeric material
	<u>Elastomeric Seals</u>				
S	Viton Fluoro-rubber	Cabin seals, hatches, air locks, access doors	5×10^6 to 5×10^7	-65 to 450	Preferred material for internal and external vacuum seals, low permeability to gases, good vacuum stability, good temperature resistance, moderate radiation stability
S	Butyl Rubber		10^6 to 10^7	-65 to 225	Preferred material for internal and external vacuum seals. Low permeability to gases, good vacuum stability, temperature resistance moderate, least radiation resistant elastomer, softens at 6×10^7 rad and becomes tarry liquid
M	Buna-N Butadiene-Acrylonitrile Rubber		10^7 to 10^8	-40 to 250	Marginal for internal and external vacuum seals, more permeable to gases than class S materials. Moderate temperature resistance, good radiation resistance

M	Neoprene		10^7 to 2×10^8	-65 to 250	Marginal for internal and external vacuum seals. Properties similar to Buna N
M	Styrene-Butadiene Rubber Buna-S(SBR)		5×10^6 to 6×10^8	-40 to 250	Marginal for internal and external vacuum seals. Properties similar to Buna-N
U	Silicone Rubber		5×10^6 to 5×10^7	-70 to 480	Not generally recommended for internal or external vacuum sealing applications because of high gas permeability. Has limited application because of high temperature stability moderate radiation resistance
U	Fluorosilicone		10^6	-100 to 400	Not generally recommended for internal or external vacuum seals. Has limited application because of high temperature stability, has high gas permeability and low radiation resistance.
M	<u>Plastic Seals</u> Teflon (TFE) (Polytetrafluoro-ethylene) Filled and Unfilled	Can be used as O-rings and seals in some vacuum applications	10^5 to 10^6	-400 to 500	Marginal for internal and external vacuum seals. Has low resiliency and subject to cold flow under compressive stress. Has poor radiation resistance in air, in vacuum or inert atmosphere radiation resistance is two orders of magnitude higher. Has good temperature stability. Has higher permeability to gases, than viton or butyl elastomers.

Table 2-18 (concl'd)

Relative Rating	Materials	Application	Penetrating Radiation Damage (rad)	Continuous Operating Temperature Range (°F)	Remarks
M	**<u>Plastic Seals</u>** **Teflon (FEP) Fluorinated ethylene-Propylene) Filled or Unfilled**		10^5 to 10^6	-400 to 500	Properties similar to Teflon (TFE)
M	**Kel-F Polytrifluoro-Chloroethylene**		10^6 to 10^7	-400 to 350	Slightly better radiation resistance than Teflon, slightly less temperature resistant than Teflon, otherwise properties similar to Teflon
U	**Nylon Hexamethylene-diamine Adipimide**		5×10^7 to 10^9	-100 to 300	Not generally recommended for internal or external vacuum seal applications. Has better radiation resistance than Teflon or Kel-F. Has lower temperature resistance. Is more permeable to gases and has retained moisture which can outgas in vacuum.
N	—		—	—	Radiation dose in these orbits exceeds limit of resistance of available elastomeric or plastic sealing materials.

Table 2-19

RELATIVE RATING OF STRUCTURAL PLASTICS

Relative Rating	Application
A	Recommended for general use in interior and external structural applications, particularly involving high energy penetrating radiation and elevated temperatures.
B	Not preferred for primary structural use but may be used in place of A-rated materials if electrical or processing requirements dictate. Recommended for internal or external secondary structure and electrical or thermal applications.
C	Not recommended for high-load structural applications but satisfactory for all internal and external low-load or nonstructural uses and where electrical requirements dictate.
D	For structural and non-structural sandwich construction, internal insulation or electrical harness lacing applications.

Table 2-20

MATERIALS SELECTOR FOR POLYMERS FOR STRUCTURAL AND RELATED APPLICATIONS

Mission Duration (yr)	Application	Materials Rating[a]							
		230-sm Orbit		576-sm Orbit		1152-sm Orbit		2304-sm Orbit	
		Reinforced Plastics – Laminates & Moldings	Polymer Films, Foams & Fibers	Reinforced Plastics – Laminates & Moldings	Polymer Films, Foams & Fibers	Reinforced Plastics – Laminates & Moldings	Polymer Films, Foams & Fibers	Reinforced Plastics – Laminates & Moldings	Polymer Films, Foams & Fibers
0.1	Surface	ABC	BC	ABC	BC	ABC	BC	ABC	BC
	Internal	ABCD	BCD	ABCD	BCD	ABCD	BCD	ABCD	BCD
1	Surface	AB	BC	AB	BC	AB	B	AB	B
	Internal	ABCD	BCD	ABCD	BCD	ABCD	BCD	ABCD	BCD

(a) Relative ratings are listed in Table 2-19 and materials in Table 2-21.

Table 2-21

STRUCTURAL PLASTICS

Material	Continuous Material Temperature Range (°F)	Radiation Damage Threshold (rad)	Ultraviolet and Vacuum Stability	Relative Rating	Design Remarks
Phenolic Resin, Glass Fabric Laminate	-300 to 500	8×10^8 to 8×10^9	Good	A	Preferred structural plastic. Thermosetting. Excellent resistance to temperatures > 1000°F for short periods. Combined heat (to 900° F) and irradiation at doses up to 10^{10} rad produce no more degradation than does high temperature alone. Excellent mechanical properties throughout operating temperature range if properly postcured.
Epoxy Resin, Glass Fabric Laminate (Low Pressure) and Sheet (High Pressure)	-300 to 250	2×10^9 to 5×10^9	Good	A	Preferred structural plastic. Thermosetting. Undamaged in significant mechanical and electrical properties at dose of 10^9 rad, except impact strength which decreases approximately 30%. Best mechanical properties of any reinforced plastic type at ambient temperatures but strength decreases rapidly at elevated temperatures.
Silicone Resin, Glass Fabric Laminate (Low Pressure) and Laminate (Low Pressure Pre-Preg)	-300 to 500	10^9 to 5×10^9	Good	B	Excellent elevated temperature electrical properties. Good mechanical properties at temperatures of 750° F for short exposures.

Table 2-21 (cont'd)

Material	Continuous Material Temperature Range (°F)	Radiation Damage Threshold (rad)	Ultraviolet and Vacuum Stability	Relative Rating	Design Remarks
Polyester Resin, Glass Fiber Laminate, Low Pressure	-100 to 200 -100 to 300	2×10^9 to 5×10^9 8×10^8 to 2×10^9	Good	B	Thermosetting. Laminate has good electrical and structural properties. General purpose structural laminate.
Phenolic Resin, Glass Fiber Molding	-300 to 450	8×10^8 to 8×10^9	Good	C	General purpose moldings. Good strength and temperature resistance. Not for primary structural applications.
Silicone Resin, Glass Fiber Molding	-300 to 500	10^9 to 5×10^9	Good	C	Not for primary structural applications.
Polyester Resin, Glass Fiber Molding	-100 to 180	2×10^9 to 5×10^9	Good	C	Not for primary structural applications. General purpose nonstructural moldings. Thermosetting. Good physical and electrical properties.
Melamine Resin, Glass Fiber Molding	-65 to 350	10^8 to 10^9	Good	C	Not for primary structural applications. Excellent electrical properties, moderate temperature resistance. Thermosetting.
Modified epoxy, Phenolic-Glass Fiber (heat resistant epoxy)	-300 to 500	5×10^9	Good	B	For general use in interior and exterior structural applications particularly where moderate to high temperatures and radiation exposures are encountered; excellent mechanical strength properties and good stability to vacuum and UV radiation.

Films					
Polyimide (H-Film)	-423 to 600	5×10^8	Good	B	Preferred materials for internal and external semi-structural and related applications (e.g., inflatable space structures) particularly where long-term resistance to UV radiation is required and moderate temperatures and radiation doses are encountered; good vacuum stability and mechanical strength properties. Polymer films are generally aluminized on one side in most spacecraft applications.
Polyvinylfluoride (Teslar or Tedlar)	-100 to 250	10^8	Good	B	
Polyester Film (Mylar)	-100 to 300 (Low load applications)	5×10^7 to 1×10^8	Fair to poor UV stability unless metallized. Good in vacuum.	C	Inflatable semistructural film. Thermosetting type. For spacecraft, antennas, etc. Low vapor permeability. Can be rigidized in space with foam-in-place materials and techniques. Aluminizing surface improves UV stability in space.
Polyethylene (High Density)	-320 to 225	10^8	Poor	D	Not preferred for long-term exposures to UV radiation unless surface is metallized but may be used for shorter missions internally or externally at low to moderate exposures and moderate temperatures; good vacuum stability. Note that radiation resistance of Teflon (FEP and TFE) is improved about two orders of magnitude in vacuum or inert atmosphere.
Polypropylene	-423 to 275	10^8	Fair	D	
Kel-F	-423 to 350	10^6	Good	C	
Teflon (TFE)	-423 to 500	5×10^4	Good	C	
Teflon (FEP)	-423 to 450	5×10^5	Good	C	
Polyethylene (Low Density)	-320 to 180	10^8	Poor	D	Not recommended for external applications because of UV radiation susceptibility; may be used internally in applications where low-to-moderate temperatures are encountered.
Polyvinyl Chloride (PVC)	-100 to 165	10^8	Poor	D	

Table 2-21 (concl'd)

Material	Continuous Material Temperature Range (°F)	Radiation Damage Threshold (rad)	Ultraviolet and and Vacuum	Relative Rating	Design Remarks
Polyurethane, Rigid Foam	-300 to 350	10^8 to 10^9	Good	D	Foamed sandwiches do not degrade in flexure and compression up to 10^9 rad. Vibration damping. Good electrical properties. Density range: 2 to 20 lb/cu ft (closed cell). Thermoplastic.
Polystyrene, Rigid Foam	-100 to 165	8×10^8 to 4×10^9	Good	D	Thermal insulation. Sandwich construction for low load applications. Density range: 2 to 6 lb/cu ft. Thermoplastic.
Foams (Closed Cell, Rigid Thermoset Types)					
Polyether	-423 to 300	10^9	Not applicable; protected by sandwich skin	D	General purpose foamed sandwich and honeycomb materials for internal spacecraft applications.
Phenolic	-423 tp 450	10^9		D	
Epoxy	-423 to 250	10^9		D	
Silicone	-423 to 500	10^9		D	
Fibers					
HT-1	-423 to 600	5×10^8	Good	D	General purpose use for electrical harness lacing and other nonstructural applications. Dacron has slightly better thermal-vacuum stability than nylon.
Dacron (Polyester)	-423 to 300	10^8	Good	D	
Nylon	-320 to 300	5×10^7	Fair	D	

Table 2-20 shows that B-rated materials are the only ones suitable for this orbit or duration. Table 2-19 shows that B-rated materials are recommended for external electrical or thermal applications. Table 2-21 shows two B-rated films, a polyimide (H-film) or a polyvinylfluoride (Teslar or Tedlar). Since the application does not require the wide range of temperature stability of H-film (-423 to 600° F) the choice will be Teslar or Tedlar (both trade names for essentially the same material) for this application. This choice is further substantiated by the use of this material for this application on the very successful Mariner 4 mission as discussed in Sec. 1.3.

In this application, the film is wrapped around the cable harness bundle and secured with a pressure sensitive adhesive aluminized tape. In the process of wrapping many creases are developed in the film. It was believed that these creases would crack the aluminized layer thus allowing the passage of unfiltered solar ultraviolet radiation to the substrate polymeric film. If the film were subject to embrittlement by UV degradation the creases and cracks could cause flaking off of the aluminized layer. Vacuum-ultraviolet simulation tests of aluminized Mylar (polyester film) showed that this was indeed the case but that aluminized polyvinyl fluoride did not flake. This behavior prompted spacecraft materials and process engineers at Jet Propulsion Laboratory, project managers for Mariner 4, to recommend the use of aluminized polyvinyl fluoride for wrapping exposed cable harnesses on that spacecraft. The primary concern was that flakes of aluminum could cause a spurious indication in the Canopus star sensor which controlled the spacecraft orientation on the correct Earth-Sun line. This orientation was required for the high-gain antenna to properly maintain telemetry and command communication with the Deep Space Instrumentation Facilities on Earth and for the solar panel array to be pointed at the Sun for most efficient spacecraft power production. Here then is another example of the correlation of simulation testing with actual performance of materials on successfully orbited spacecraft and probes.

There are many examples of the successful application of structural plastics for external and internal primary and secondary structure in successfully orbited spacecraft discussed in Sec. 1. Many other polymeric materials such as seals, and adhesives have been successfully used in many exacting application. Proper selection of these materials by designers and engineers for the correct application have resulted from examination of these materials properties where advantageous weight savings would be achieved through trade-offs for higher density materials.

It should not be inferred that all the materials having the same relative rating will perform equally well for a given application. The best material for one design is not necessarily the best for another, and the designer must make the final selection based on his particular requirements. This section is intended only as a general guide to the designer and is a broad generalization of the material properties and environmental effects presented in detail in Chapters 13, 14, and 15 and the experience with materials related in Chapter 20 of Ref. 1 and in Sec. 1 of this supplement.

2.6 ELECTRONIC AND MISCELLANEOUS COMPONENTS AND MATERIALS

The space environment of overriding importance to the performance of electronic components is penetrating radiation. Except for the case of solar cell arrays most electronic equipment is usually located within the interior of a vehicle so as to be shielded by the structure from UV radiation and meteoroid bombardment; individual components are sealed against the vacuum environment where necessary.

Recommended components and materials for orbits of 230, 576, 1115, and 2304 sm are given in Tables 2-22 through 2-25.

At doses above 10^{12} fast neutrons/cm^2, 10^{12} protons/cm^2, 10^{13} electrons/cm^2, or 10^6 R of gamma radiation, the radiation environment becomes a dominant factor in choice of electronic materials and the design of electronic circuits. Semiconductor devices are among the electronic components most sensitive to radiation. To insure successful operation in the radiation environment, the degradation of the electronic components should be known, and the design should take it into account.

Tunnel diodes, zener diodes, and high-speed thin-base switching diodes are relatively radiation resistant. Silicon controlled rectifiers, unijunction transistors, and low-speed transistors are easily destroyed by radiation and should be used with caution. High-speed and field effect transistors are moderately resistant to radiation and may be used if their degradation is taken into account. Thin-base transistors are more radiation resistant than thick-base ones. High-frequency transistors are more resistant than low-frequency ones and n-p-n silicon transistors are better than p-n-p ones of similar characteristics.

Table 2-23 (Ref. 2) shows some of the materials used for insulation and dielectric applications. Most of these materials will be used in internal locations in the spacecraft so that some shielding of the space penetrating radiation and most if not all of the unfiltered UV radiation is affected by the spacecraft skin or structure. In most instances, electronic equipment is housed in some sort of a metallic container of either aluminum or magnesium alloys (see Sec. 1). For example, most of the electronic equipment in the Ranger-Mariner series were housed in Mg alloy AZ31B boxes. In Telstar 1 and 2, most electronic equipment was housed in a sealed 6061-T6 Al alloy canister.

The relative ratings of these insulation and dielectric materials are as follows:

A — Preferred materials for insulation and dielectric applications with stable properties in severe environments

B — Satisfactory material with application limited by initial properties or property degradation in radiation, or thermal environments

C — Use of such material limited to specific application unless shielded by protection of vehicle skin or container or some loss of certain properties can be tolerated

Table 2-22

RECOMMENDED ELECTRONIC COMPONENTS FOR 1-YR CIRCULAR ORBIT SPACE MISSIONS

Component	90-deg Inclination Orbit (Polar)								30-deg Inclination Orbit			
	230 sm		576 sm		1115 sm		2304 sm		230 sm		576 sm	
	E[a]	I	E	I	E	I	E	I	E	I	E	I
Capacitors	All paper, ceramic, mica, plastic, tantalum, and oil-impregnated capacitors tested are suitable for all orbits											
Resistors	All wire-wound, carbon, and composition resistors and potentiometers tested are suitable for all orbits											
Electron Tubes												
Vacuum tubes	M	S	U	S	U	U	U	M	M	S	U	M
Gas-filled tubes	S[b]	S	M	S	U	S	U	S	S	S	S	S
Photomultipliers	U	M	U	U	U	U	U	U	U	M	U	U
Traveling wave tubes	U	S	U	S	U	U	U	M	U	S	U	M
Camera tubes	U	S	U	U	U	U	U	U	U	S	U	U
Infrared Detector Cells	S	S	U	S	U	U	U	S	S	S	U	S
Diodes												
Silicon	S	S	S	S	S	S	S	S	S	S	S	S
Germanium	S	S	S	S	S	S	S	S	S	S	S	S
Transistors												
Silicon - thick base	M	S	U	M	U	U	U	M	M	S	U	M
Silicon - medium base	S	S	M	S	U	U	U	S	S	S	U	S
Silicon - thin base	S	S	S	S	S	S	S	S	S	S	S	S
Germanium - thick base	M	S	U	M	U	U	U	M	M	S	U	M
Germanium - medium base	S	S	M	S	U	U	U	S	S	S	U	S
Germanium - thin base	S	S	S	S	S	S	S	S	S	S	S	S
Miscellaneous												
Quartz crystals	S	S	S	S	S	S	S	S	S	S	S	S
Differential transformers	S	S	S	S	S	S	S	S	S	S	S	S
Magnetic cores	S	S	S	S	S	S	S	S	S	S	S	S

(a) E, external; I internal (refer to components located outside, or inside, the shielding of a satellite skin of at least 0.1-in. Al alloy or equivalent.

(b) S, satisfactory for use; U, unsatisfactory; M, marginal value, more information needed.

Table 2-23

ELECTRICAL INSULATION AND DIELECTRIC MATERIALS

Material	Continuous Material Temperature Range (°F)	Radiation Damage Threshold (rads)	Stability-Space UV and Vacuum	Relative Rating	Remarks
<u>Electrical Insulation</u>					
Polyethylene Expanded Tubing, Heat Shrinkable Types 1, 2, and 3	to 215	2×10^7 to 10^8	Fair	A	Type 1 – Heat shrinkable jacket insulation for equipment Type 2 – Material for repair of jacket defects Type 3 – Dual-wall shrinkable material for embedding and parts (~ 75%) Excellent electrical properties, for wire connector insulation and membrane insulation
Modified Neoprene Rubber Tubing Heat Shrinkable, Flexible	-65 to 250	3×10^6 to 3×10^8	Fair	C	For repair of jackets where oil resistance is needed; softens in benzene group solvents and in ketones
Polytetrafluorethylene Coated Tape, Glass Braid Lacing	to 500	10^6	Good	B	Teflon coating limits radiation resistance, excellent electrical properties at elevated temperature
Vinyl Coated Tape, Glass Braid Lacing and Tying	to 215	2×10^7 to 5×10^8			Excellent electrical properties

Irradiated Modified Polyolefin, Insulation for Vehicle Wiring	-65 to 275	5×10^8	Fair	A	Used for vehicle wiring and all flight items, thermally stabilized, and irradiated modified polyolefin insulated wire
Polyamide (Nylon)	-65 to 300	5×10^6 to 10^8	Good	C	Electrical insulation
Polyester (Mylar), Glass Filled	-70 to 300	2×10^9 to 5×10^9	Good	B	Electrical insulation
Polytetrafluorethylene (Teflon)	to 400	10^5 (air) to 10^6(vacuum)	Good	B	Special applications requiring unique electrical properties or heat resistance, where radiation total dose is low
Natural Rubber (Anti-Rad Type)	-40 to 225	2×10^7 to 2×10^8	Poor	B	Elastomer for vibration damping; wire insulation and sheathing
Neoprene Rubber (Polychloroprene)	-65 to 250	3×10^6 to 2×10^8	Fair	C	Elastomeric cable sheathing, molded parts, bumpers, and shock absorbers, jacketing and sheathing only, not insulation

Table 2-23 (cont'd)

Material	Continuous Material Temperature Range (°F)	Radiation Damage Threshold (rads)	Stability-Space UV and Vacuum	Relative Rating	Remarks
Synthetic Rubber (Buna-N)	-40 to 250	10^7	Fair	C	Cable jacket and tubing material; wire insulation
Neoprene Sheath, Buna-N insulated	-40 to 250	6×10^6 to 3×10^7	Fair	C	Cable sheath; wire insulation
Butyl Rubber	-65 to 250	10^6 to 10^7	Good	C	Elastomer for vibration damping; wire insulation or sheathing
Polyvinyl Formal (Formvar)	-75 to 325	2×10^7 to 3×10^8	Good	B	Magnet wire insulation; solar array application
Polyethylene	-70 to 250	2×10^7 to 2×10^8	Fair	A	Embrittled in UV radiation
Polyester (Mylar)	-100 to 300	5×10^7 to 5×10^8	Fair in UV unless metallized Good in vacuum	B	Electrical insulation in manufacture of cables. Thermosetting
Polyvinyl Chloride	-500 to 200	2×10^7 to 2×10^8	Poor	C	Electrical insulation and vibration protection. Plasticized types outgas and embrittle above 10^8 rad. Liberates HCl above 10^7 rad.

Kel-F (Monochlorotrifluorethylene) AMS 3650	-80 to 300	3×10^6 to 2×10^7	Fair	C	Electrical insulation, thermoplastic elastomer; embrittled in UV radiation; may evolve corrosive gases
Tetrafluorethylene (Teflon)	-150 to 500	10^5 air to 10^6 (vacuum)	Good	B	Electrical insulation, thermoplastic; low radiation resistance, releasing corrosive fluoride gases at higher doses
Polymethyl Methacrylates	-80 to 190	1×10^7	Good in vacuum; fair in UV	C	Magnet wire insulation; thin coating
<u>Circuit Boards and Other Electrical Laminates</u>					
Copper-Clad Epoxy, Glass Laminate	-65 to 300	5×10^9	Good	A	Printed circuit board
Epoxy-Glass Laminate	-65 to 300	5×10^9	Good	A	Material in thickness of 7 mils up to 0.25 in.
Epoxy-Glass Laminate	-65 to 250	5×10^9	Good	A	Satisfactory where used within temperature limits; antenna window and related uses

Table 2-23 (concl'd)

Material	Continuous Material Temperature Range (°F)	Radiation Damage Threshold (rads)	Stability-Space UV and Vacuum	Relative Rating	Remarks
Phenolic Glass Laminate	-65 to 450	8×10^9	Excellent	A	Elevated temperature resistance to 700°F for short time less than 10 min; brackets, insulating blocks, and strips
Silicone Glass Laminated Sheet	-65 to 500	3×10^8	Fair	A	Use only where continuous temperature is over 300°F
Silicone Glass Laminate	-65 to 500	3×10^8	Fair	A	Good electrical properties and temperature resistance; antenna application
Diallyl Phthalate	-100 to 400	10^8		B	Used for molded potting containers; specify type and size
Copper-Clad Teflon Glass Laminate	-65 to 400	10^5 (air) 10^6(vacuum)	Good	C	Restricted Use: Do not use in general applications; use only for VHF or UHF where other material is not satisfactory or a very precise or low dielectric constant is required. Specify laminate and copper thickness

Table 2-24

ELECTRICAL ENCAPSULANT MATERIALS

Material	Application	Continuous Material Temperature Range (°F)	Radiation Damage Threshold (rad)	Remarks
Polyurethane Encapsulants				
Type I – Eccofoam FP (Class A, B, and C) Emerson and Cuming, Inc.; Hitco R (Class A, B, and C) H. I. Thompson Co.	Foam-in-place	-65 to 165	10^8	Rigid foam encapsulant for mechanical shock and vibration damping; type is determined by maximum temperature of use, class is determined by density: Class A – 2 lb/ft^2, Class B – 10 lb/ft^2, Class C – 20 lb/ft^3
Type II – Eccofoam FP (Class A, B, and C) Emerson and Cuming, Inc.; Hitco MT (Class A, B, and C) H. I. Thompson Co.	Foam-in-place	-65 to 250	10^9	
Type III – Eccofoam FPH (Class A and B) Emerson and Cuming, Inc.; Hitco HT (Class B and C) H. I. Thompson	Foam-in-place	-65 to 350	10^9	
PE-102 American Latex Co. Type I Type II	Foamed slab Foam-in-place	-65 to 165		Flexible foam packing material; specify density and softness; not to be used for structural applications

Table 2-24 (cont'd)

Material	Application	Continuous Material Temperature Range (°F)	Radiation Damage Threshold (rad)	Remarks
<u>Silicone Rubber</u>				
Dow-Corning Corporation Class 1 Q-9-0031-1/2 Class 2 Q-9-3003-1/2 Class 3 Q-9-0006-1/2 Class 4 RTV-503 Products Research Company Class 1 PR-1930-1/2 Class 2 PR-1920-1/2 Class 3 PR-1910-1/2	Potting	-85 to 500	5×10^6	Requires rigid mold for mechanical shock and vibration; class determined by viscosity; should not be used at 400 – 500° F unless given a progressive cure in 100° steps to 500° F; use for elevated temperature where limited to room temperature to cure.
<u>Silicone Rubber Primer</u>				
General Electric Co. SS-4004 Products Research Company 1902	Primer to RTV rubber	-85 to 500	5×10^6	Primer used silicone rubber; 1-hr cure at room temperature
<u>Polysulfide</u>				
3C-737, Class A 3C-747T, Class B Churchill Chemical Co.	Potting	-70 to 300	10^6	Use for cable connectors; not for general purpose potting or embedding of electric modules

<u>Epoxy Encapsulants</u>				
Insulating Lacquer 1162 Dennis Chemical Company	Moisture resistant coating (thin)	-65 to 200	10^7	Spray or dip coat (< 0.005 in.); printed wiring board coating and pre-encapsulation check for non-welds in welded modules.
Coating, IRFNA and UDMH Resistant Furane Plastics, Inc., H2C-101	Coating, missile fuel and acid	-65 to 200	10^7	Packaged in aerospray bomb; acid and missile-fuel-resistant overspray coating.
Epocast H2E-011 Furane Plastics, Inc. Scotchcast No. 3 3M Co.	Impregnating	-65 to 300	5×10^8	Low-viscosity coil impregnant for inductors; not recommended for welded modules; rigid, unfilled epoxy.
Epocast H2E-012 Furane Plastics, Inc.	Potting	-65 to 300	5×10^8	Rigid, filled epoxy.
Hysol 4238/3475 Class A Hysol 4239/3487 Class B Eccobond 70C Class C	Conformal coating	-65 to 250		Electrically conductive; r-f interference applications
Stycast 2762/Cat 17 Emerson and Cuming, Inc.	Potting and embedding	-65 to 500		Require elevated temperature cure for 500°F use

Table 2-24 (concl'd)

Material	Application	Continuous Material Temperature Range (°F)	Radiation Damage Threshold (rad)	Remarks
Epoxy Encapsulants				
Epocast 202/9615, Class A and B Epocast 202/9647, Class C and D Epocast 202-11/9615A, Class E Furane Plastics, Inc.	Potting, embedding, conformal coating	-65 to 200	10^7	Preferred material in its type for radiation protected applications in electronic enclosures in vehicles; clear amber-liquid resin; class is determined by hardness; hardener/resin ratio can be varied to produce Shore D hardness from 3 to 95; low Shore D will have decreased resistance to radiation and vacuum; shore D range of 50−85 is recommended for encapsulation of soldered and welded modules.
Epocast H2E-037 Furane Plastics, Inc.	Embedding	-65 to 250	10^8	Filled opaque compound for electrical insulation; semirigid, good thermal shock resistance.

Table 2-25
RECOMMENDED MISCELLANEOUS MATERIALS FOR
1-YR CIRCULAR ORBIT SPACE MISSIONS

Component	90-deg Inclination Orbit (Polar)								30-deg Inclination Orbit			
	230 sm		576 sm		1152 sm		2304 sm		220 sm		576 sm	
	E[a]	I	E	I	E	I	E	I	E	I	E	I
Refrigerants												
Freon	U[b]	S	U	U	U	U	U	U	U	S	U	U
Carbon dioxide	S	S	S	S	S	S	S	S	S	S	S	S
Explosives and Propellants												
RDX	M	S	U	S	U	U	U	M	M	S	U	M
TNT	S	S	S	S	S	S	S	S	S	S	S	S
Tetryl	S	S	S	S	S	S	S	S	S	S	S	S
Lead Styphnate	S	S	S	S	S	S	S	S	S	S	S	S

(a) E, external; I, internal (refer to components located outside, or inside, the shielding of a satellite skin of at least 0.1-in. Al alloy or equivalent).
(b) S, satisfactory for use; U, unsatisfactory; M, marginal value, more information needed.

Encapsulating materials are listed in Table 2-23 (Ref. 2). The criteria for selection of the best material for each application depend on the properties required and processing methods. The list of acceptable materials has been reduced to epoxy and polyurethanes, with some use of silicone rubber and polysulfides to keep the list of materials to a manageable level. Ultraviolet radiation stability of these materials is applicable only when they are used externally to the spacecraft structure.

Encapsulants for electronic devices should be selected on the basis of analysis of the electrical, mechanical, and thermal requirements of the particular design. Encapsulating resins have considerably higher coefficients of thermal expansion than embedded component materials such as metals, ceramics, and glass. These differences in coefficient of thermal expansion may cause severe internal stresses in electronic packages during thermal cycling and can cause cracking of the encapsulant or voids which can be sources of short circuits or high voltage breakdown or arcing (Sec. 1).

One of the suspected causes of the failure of the TV electronics in Ranger 6 was high voltage arcing. A complete review of the TV circuitry resulted in the use of transparent conformal coating formulations (modified Solithane 113) for the encapsulation of the electronic components. This resulted in more reliable inspection to assure that all protruding corners and connections were adequately covered with sound encapsulant. The procedure also permitted inspection of connectors after encapsulation to assure that the pins were not displaced and that there were no voids for paths of high-voltage currents. The success of Rangers 7 through 9 attest to the success of this very thorough program.

The recommendations listed in Tables 2-22 through 2-25 must be regarded as guideposts rather than as firm specifications because of uncertainties regarding the space environment, the statistical accuracy of tolerance data, and the exact applicability of the tolerance data. Within any single category, specific items may be better or worse than the tables indicate. More details on the behavior of electronic components are to be found in Ref. 1.

It is expected that new product developments and improvements will add rapidly to the list of preferred parts since the technology of radiation resistance is being accelerated in many areas. It is also noted that there is no strong discrimination between preferences for 230 and 576 sm orbits. This occurs because the altitude dependence is probably no stronger than normal variations in mission lifetime.

2.7 MATERIALS FOR SPACECRAFT ANTENNAS

The most frequently used metals for spacecraft antennas are aluminum and magnesium. The several alloys of these metals have good physical characteristics, light weight, and good stability in the space environment. Titanium also has excellent properties for lightweight structural applications. Its use is presently limited chiefly because of cost. Table 2-26 lists metals recommended for spacecraft antennas. The selection of the various metals will be based upon the required mechanical properties consistent with reliable fabrication processes and good engineering practice.

Recent developments in successfully orbited satellites made use of flat strips of thin metal deployed in space into long tubular antennas. As discussed in Sec. 1, Alouette used strips of 0.004-in. thick, 4-in. wide spring steel which were deployed into one antenna 150 ft long and another 75 ft long. Explorer 20 used Be-Cu strip (Berylco Alloy 25) to form a 0.5-in. diameter tube for its antenna.

On the other hand the S-band antenna of Nimbus 1 was a fiberglass reinforced resin cone containing on its surface two logarithmic spirals of copper. This device operated satisfactorily until the satellite prematurely ceased transmission of excellent cloud cover pictures because the solar panel orientation mechanism failed.

The antennas of the Ranger and Mariner 2 series of space probes were constructed of Al alloy 5052 screening on an Al alloy 2024-T4 sheet metal parabolic frame. This high-gain antenna system was Earth-oriented by an antenna actuator system and by the spacecraft attitude control system throughout the trajectory of Rangers to the Moon and Mariner 2 to Venus. The antenna was deployed after injection into proper trajectory and was moved out of the way during mid-course trajectory rocket firings.

Table 2-26

RECOMMENDED METALS FOR SPACECRAFT ANTENNAS

	Sheet Metal	Castings	Extrusions	Tubes	Weldments	Foil	Wire and Screen	Plating Material	Remarks
Aluminum Alloys Magnesium Alloys Titanium	S[(a)] S S	S S S	S S S	S S S	S S S	S U O	S U S	U U U	Aluminum, magnesium, titanium all have good weight, strength, and stiffness; readily formed, machined and welded, although titanium somewhat less workable than aluminum or magnesium
Steel Stainless Steel	S S	S U	S U	S S	S S	O S	O S	U U	Steel and stainless steels used in structural applications where greater strength is required or high temperature prevails
Beryllium Copper	S	O	U	U	U	S	O	O	Fatigue resistant; excellent spring forming qualities
Molybdenum	U	U	U	S	U	S	S	O	Excellent high temperature material
Invar Kovar	S S	U U	U U	S S	U U	U S	S S	U U	Extremely low thermal expansion of both Invar and Kovar may be used to advantage in structure requiring thermal stability
Silver	O	O	O	S	O	O	O	S	Used principally for high microwave frequency wave-guides
Gold	O	O	O	O	O	O	O	S	Excellent plating material
Rhodium	O	O	O	O	O	O	O	S	

(a) S – Satisfactory, recommended for use
U – Unsatisfactory, not recommended in this form for spacecraft antennas
O – Not commonly used in these forms for spacecraft antennas

In the case of Mariner 4 a fixed antenna was mounted on top of the spacecraft and consisted of aluminum honeycomb with 4-mil foil facings and 1/4-in. core cells of 0.7-mil foil. The proper orientation was maintained by the Canopus star tracker.

Another interesting application of antenna materials was in the Tetrahedral Research Satellites where steel measuring tape was used. A coiled tape was enclosed in small housings at two corners of the tetrahedron. The cover of the housing was removed by command after the satellite was injected into orbit thereby permitting the coil tape to deploy. These antennas performed satisfactorily in some instances (one was reported to be slow in deployment) and permitted the satellites to transmit data about the performance of solar cell materials in space (Refs. 8 and 9).

Table 2-27 lists recommended dielectric materials for spacecraft antennas. In all these materials, the dissipation factors are low and the physical properties are stable in the space environment except where noted. Polystyrene is one of the most stable in penetrating radiation. Teflon, polyurethane, and Kel-F are susceptible to radiation damage and should not be used in exposed situations where the radiation dose may exceed 7×10^6 rad. Micalex may be used for antenna windows where good structural strength is required and for dielectric structural components. Beryllium oxide has good dielectric properties and its thermal conductivity is similar to many metals, so it can be used where heat flow is important. Disadvantages of beryllium oxide are its toxicity, poor machinability, and variations of dielectric constant from batch to batch. "LOCKHEAT," a glass-fiber inorganic-binder composite, is a new dielectric developed at Lockheed for use primarily in high-temperature radomes, but other uses may be made of this filament-wound, high strength material. Fiberglass laminates have been utilized both with epoxy, phenolic, or silicone resins for structures as well as for good dielectric properties. High strength and good thermal properties in these materials make them useful for many antenna applications. Windows have been painted with metallic paint or laminated with metal foil to form complicated slot apertures and printed radio frequency circuits. The foamed dielectrics, polystyrene, and polyurethane, are used to load or pressurize cavities and to strengthen cantilevered components.

Table 2-28 lists materials used for inflatable and/or pressure-erected antenna designs.

All vehicle environments must be considered in the selection of cable insulation for spacecraft application. The vehicle environment will differ from the external space environment, due to temperature at the skin, temperature gradients, and the internal equipment rack environment. Table 2-29 summarizes the comparative temperature and r-f attentuation characteristics of the Teflon, polyethylene, and polyolefin cable materials.

Polyethylene has the greatest radiation resistance but the lowest softening point. It cannot be used where the temperature will exceed 185° F. Polyolefin shows a tendency to outgas at high vacuum and should therefore be used with caution. When all environments are considered, Teflon-insulated cable is the best compromise among present commercially available cables. Small diameter Teflon insulated cables show severe radiation damage at high radiation dose rates (in excess of 5×10^8 R).

Table 2-27

RECOMMENDED DIELECTRIC MATERIALS FOR SPACECRAFT ANTENNAS

	Molded Forms	Machined Forms	Structural Components	Radomes	Foam	Film	Cable Core Material	Remarks
Polystyrene	S[a]	S	O	U	S	U	O	
Polyurethane	U	S	O	U	S	U	U	Flexible or rigid foam
Mylar	O	O	O	O	O	S	U	Resistance to UV low where not aluminized
Teflon	S	S	U	U	U	S	S	Subject to radiation damage, can be used in thin films in certain specific applications
Kel-F	S	S	U	U	U	S	U	
Micalex	S	S	S	U	O	O	O	Good structural dielectric material
Aluminum Oxide	S	O	O	S	U	U	U	Good high temperature dielectric material
Beryllium Oxide	U	O	O	S	U	U	U	Excellent high temperature material; ϵ varies from batch to batch
LOCKHEAT	O	S	S	S	O	O	O	A glass-inorganic or organic composite material for high-temperature radomes
Epoxy-Glass Laminates	S	U	S	S	O	O	O	
Polyester-Glass Laminates	S	U	S	S	O	O	O	
Silicone-Glass Laminates	S	U	S	S	O	O	O	Excellent high temperature
Phenolic-Glass Laminates	S	U	S	S	O	O	O	

(a) S – Satisfactory, recommended for use
U – Unsatisfactory, not recommended in this form for antenna applications
O – Not commonly used in these forms for antenna applications

Table 2-28

MATERIALS FOR PRESSURE-ERECTED ANTENNAS

Mylar-Aluminum Foil Laminates Tedlar (Teslar)-Aluminum Foil Laminates	Polymer-foil-polymer laminates are flexible, have good thermal properties, are resistant to radiation, and possess adequate strength. They are predominant materials used for inflatable and pressure-erected antenna designs. Foil is usually pure aluminum or alloy 1080, 1145, or 3003.
Fiberglass cloth reinforced resin binder material	Plated with gold, woven glass fiber reinforced materials make excellent flexible reflecting surfaces when held in tension by erected frame structures.
Polyurethane Foam	An excellent flexible foam.
Epoxy Cements	Strong, stable cements used to "iron-on" polymer-foil laminates

Table 2-29

TEMPERATURE AND r-f ATTENUATION CHARACTERISTICS OF CABLE MATERIALS

	Teflon	Polyethylene	Polyolefin
Maximum temperature	420° F	185° F	224° F
Minimum temperature	-120° F	-65° F	-65° F
Attenuation	least	acceptable	acceptable

Semi-flex outer conductor cables are the preferred engineering material for spacecraft application. Type TNC connectors are recommended for frequencies below 2 Gc and Type N or Type SC about 2 Gc. Above 4 Gc, waveguide is recommended where possible.

Gold plating is recommended for all contact surfaces.

Active r-f components, such as diodes, varactors, and ferrites should be used in a protected environment. The selection of component types and materials will be based on the application and the complete physical environment. Additional information on materials for antennas can be found in Chapter 16, Ref. 1.

2.8 STRUCTURAL MATERIALS – INORGANIC

Spacecraft usually are designed for specific mission requirements. Unmanned spacecraft are generally instrumented vehicles and require structure and housings for the support of guidance, telemetry, command, control, and power systems with mechanisms and actuators for solar panel arrays, horizon scanners, TV cameras, antennas, and instrument deployment booms. Manned spacecraft require life support systems, cabin pressurization, communications, guidance, and attitude control. All spacecraft need surfaces for passive temperature control and mechanisms for active thermal control. Thus, the primary factors governing selection of inorganic structural materials are based upon optimum strength-to-weight ratios and required mechanical properties that are consistent with reliable manufacturing processes. In general, the effects of the space environment related to a specific orbit or trajectory do not govern the selection of spacecraft inorganic structural materials.

The inorganic structural materials that have proved successful in aircraft construction are readily applicable to spacecraft. Therefore, the aluminum alloys (2014, 2024, 2219, 5052, 5524, 6061, and 7075), the magnesium alloys (HM21A, HM31A, HK31A, AZ31B, ZH62A, and ZK60A), titanium alloys (commercially pure, Ti-6Al-4V and Ti-5Al-2.5Sn) some stainless steels (AISI 302, 304, 410, 440C) and some low alloy high-strength steels (AISI 4340) in sheet, plate, structural shapes, extrusions, forgings, and some castings have been applied in spacecraft construction. Some of the high alloy high-strength steels have been used in special applications (120-in. diameter solid propellant rocket motor chambers). For specialized applications such as rocket engines and rocket motor combustion chambers and nozzles the refractory metals and alloys such as high alloy stainless steels, nickel based alloys (René 41), molybdenum and tungsten are being used where resistance to high temperature erosive exhaust gases is encountered.

No appreciable effects on any of the inorganic structural materials are expected from the space environmental factors of vacuum, temperature, and radiation in earth satellite orbits or space probe trajectories. Vacuum sublimation of inorganic materials for lifetimes on the order of 1 yr will not affect the structural properties of inorganic materials (i.e., metals and alloys, refractory metals, or ceramic materials). Loss of material by direct evaporation of magnesium and its alloys could be a problem above about 400° F (0.004 in./yr). But magnesium alloys would not be a wise choice for continuous use at 400° F because of unfavorable mechanical properties in this temperature range rather than because of evaporation tendencies. Furthermore magnesium and its alloys are usually coated for corrosion protection during preflight operations. These coatings will tend to minimize evaporation losses in high vacuum. Simulation testing studies of the evaporation of magnesium alloys in high vacuum have shown in general that calculated evaporation rates tend to be greater than those determined experimentally. Designs based upon calculated rates would be conservative.

In the case of cadmium an evaporation rate of 0.040 in./yr can be calculated (see Ref. 1 for data and details) at 250° F. This rate may be significant structurally; however, cadmium is not used as a structural material. Cadmium is most generally used as a plating on a mild steel part. At the calculated rate of evaporation, deposition of the cadmium vapor by condensation on surfaces could detrimentally affect thermal control surfaces, electrical contacts and optical surfaces. However, the evaporation would have to take place at a critical flux density at temperature or in direct line of sight of the cadmium part with the surface upon which the condensation were to occur. Proper design to restrict direct line of sight view angles of components, to vent containers and for proper temperature control can minimize the surface contamination problem. Simulation tests have shown that plated cadmium has an evaporation rate of only 2% of that calculated for the bulk metal (Ref. 10). Although condensation of cadmium vapor on surfaces was not reproducible in the studies of Ref. 10, a condensation did occur on aluminum and copper surfaces from 23 to 60° F with a flux of cadmium vapor of about 10^{-8} gm/cm^2-sec. It is probably conservative design nevertheless to minimize the use of cadmium plated fasteners or electrical connectors in spacecraft applications. In manned spacecraft the presence of cadmium vapor inside the cabin could be a problem because of its high toxicity.

Erosion damage by sputtering of ions or solar plasma will be negligible for all structural inorganic materials.

Penetration and spalling of spacecraft skins by meteoroids is a potential problem. The probability of hits by meteoroids of a certain mass was discussed in Chapters 8 and 18 of Ref. 1. Explorer and Pegasus satellites and the Mariner 2 and 4 space probes have returned data on meteoroid encounters. From this data and the known properties of materials, survival probabilities can be estimated and material selection can be based upon structural requirements which will permit the calculation of required skin thicknesses.

Simulation experiments in hypervelocity particle impact have been limited in velocity or the mass particle which could be projected. However, recent improvements in equipment have permitted studies with velocities of 15,000 ft/sec (Ref. 11) with a 0.062 in. dia copper sphere on aluminum alloy 2024-T3 and -T4 target material 0.01 to 0.25 in. thick. These tests were to evaluate the material as a meteoroid protective shield ("bumper") against damage of the main spacecraft structure. The results showed that the optimum bumper thickness was between 0.5 and 2.0 times the diameter of the impacting projectile while a bumper standoff distance of eight projectile diameters was required to limit penetration of the vehicle structure.

Results from the meteoroid detection satellites have given estimates of the meteoroid particle density to be about 0.2 gm/cm^3 for sporadic meteors. The results of Explorer 16 (see Sec. 1 and Ref. 12) have given the best estimate of the penetration in beryllium-copper (Berylco Alloy 25) to that of related thicknesses of aluminum (2024-T3 or -T4). An analysis and correlation of these results with a conversion to penetration in the Al alloy and Mylar has been prepared (Refs. 13 and 14) and are shown in Fig. 2-2.

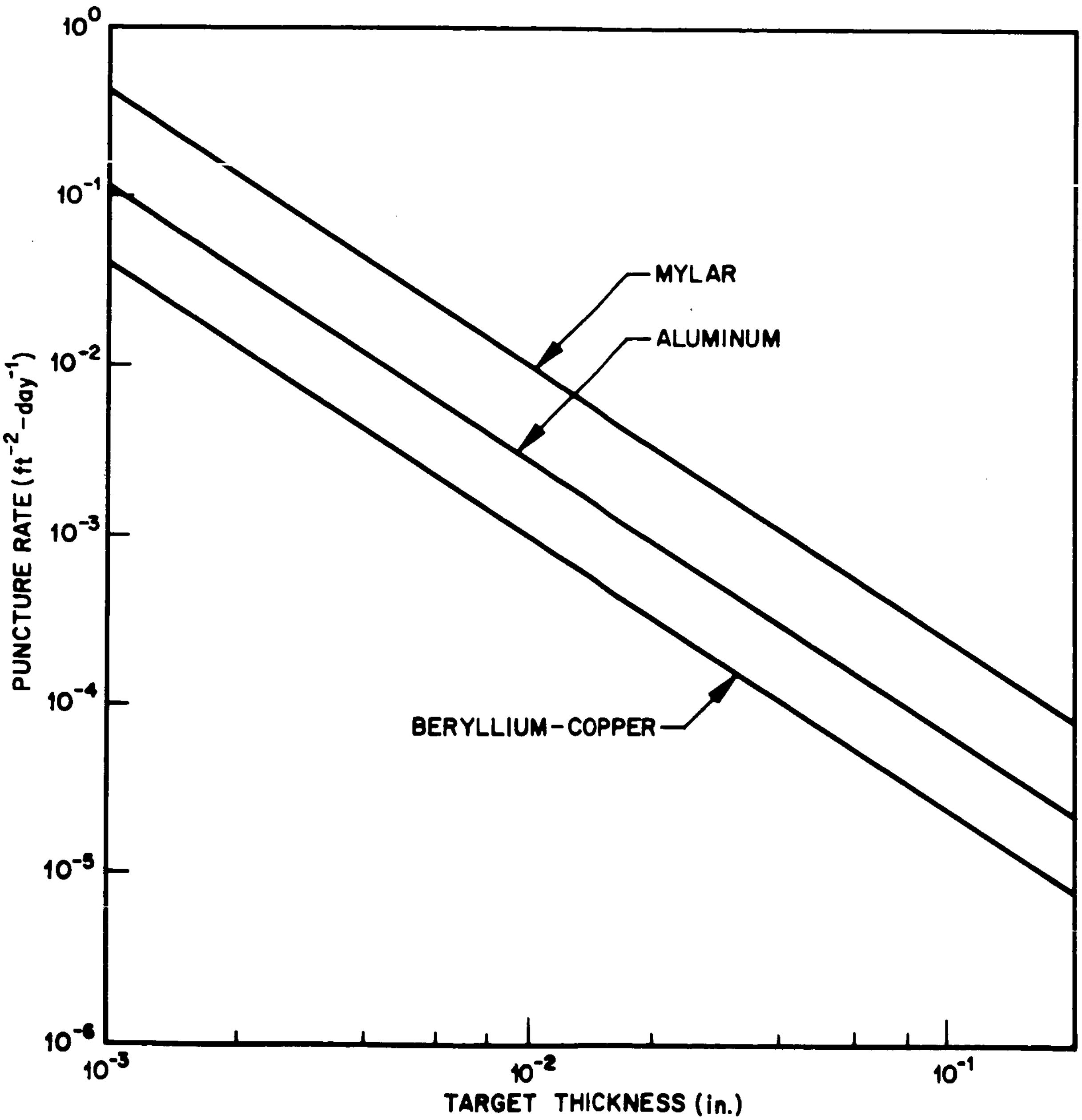

Fig. 2-2 Micrometeoroid Penetration in Some Structural Materials

Beryllium prepared by powder metallurgy techniques and cross-rolled into sheet has been used in compressively loaded applications in the Agena second stage booster (Ref. 15). Because of their favorable strength-to-weight ratio beryllium and beryllium-aluminum alloys offer significant weight savings over aluminum and magnesium alloys in compressively loaded structures. However, the beryllium particularly has a limited ability to absorb impact energy which can limit its usefulness in many structural applications.

In summary, the selection of inorganic structural materials for spacecraft is not usually determined by the effects of the space environment on them. The selection and design can be based upon strength-to-weight ratios and the required mechanical properties consistent with reliable manufacturing processes. The base metals or alloys of aluminum, beryllium, magnesium, titanium and high-strength steels are all candidate materials.

2.9 REFERENCES

1. C. G. Goetzel, J. B. Rittenhouse, and J. B. Singletary, Space Materials Handbook, Air Force Materials Laboratory, Wright-Patterson Air Force Base, Ohio, ML-TDR-64-40,[AD-646399][X65-14878] January 1965 [also published in a slightly abridged version by Addison-Wesley Publishing Co., Reading, Mass. 1965]

2. Program 461 Spacecraft Materials Handbook, Lockheed Missiles & Space Company, Sunnyvale, Calif., LMSC-A327227, May 1964

3. W. S. Shipley and J. E. Mackay, "Mariner 4 Environmental Testing," Astronautics and Aeronautics, 3, No. 8, 42–48 (1965)

4. L. H. Hemmerdinger, "Thermal Design of the Orbiting Astronomical Observatory," J. Spacecraft and Rockets, 1, 477–483, September–October 1964

5. Lubrication Evaluation, Lockheed Missiles & Space Company, Sunnyvale, Calif., LMSC-A707712, October 1964

6. Final Report, Program 461 Reliability, Materials Research and Application Behavior of Slip Rings in a [Simulated] Space Environment, Lockheed Missiles & Space Company, Sunnyvale, Calif., LMSC-A311368-A, May 1963

7. F. J. Clauss, Electrical Transmission Components for a Large Aerospace Environmental Chamber, Arnold Engineering Development Center, Tullahoma, Tenn., AEDC TR-65-40, February 1965

8. Joseph Denny, Space Technology Laboratories, Redondo Beach, Calif., personal communication, 25 November 1964

9. Final Flight Report, Tetrahedral Research Satellites 1963-14B, 1963-14C, 1963-30B, Vols. I, II, TRW Space Technology Laboratories, Redondo Beach, Calif., Report 8685-6006-RU-000, February 1964

10. R. Wales and M. Leubben, "Evaporation of Electroplated Cadmium in High Vacuum," J. Spacecraft and Rockets, 2, No. 3, 467–469, May–June 1965

11. D. A. Humes, An Experimental Investigation of the Effectiveness of Single Aluminum Meteoroid Bumpers, NASA TND-1784, May 1963

12. C. T. D'Aiutolo, "Satellite Measurement of the Meteoroid Environment," Annals of New York Academy of Science, 119, Art. 1, 82–97 (1964)

13. E. C. Hastings, The Explorer 16 Micrometeoroid Satellite, NASA TM X-810 February 1963

14. J. Neiger, Lockheed Palo Alto Research Laboratory, Palo Alto, Calif., personal communication, July 1965

15. C. G. Goetzel and J. B. Rittenhouse, "Powder Metallurgy Applications in Space Vehicle Systems," presented at International Powder Metallurgy Conference, New York, 14–17 June 1965

GLOSSARY

GENERAL TERMS

Ablation – The removal of surface material from a body by vaporization, melting, or other process, due to aerodynamic effects while moving at high speed through a planetary atmosphere

Absorptance – The ratio of quantity of absorbed radiation to the quantity incident upon it

Absorptivity – The absorptance of an opaque, optically smooth, clean portion of material

Aerodynamic heating – Surface heating of a body caused by air friction and compression processes on passage of air or other gases over the body; significant chiefly at high speeds

Aeronomy – Study of the upper regions of the atmosphere where physical and chemical reactions due to solar radiation take place

Albedo – The reflecting power expressed as the ratio of light reflected from an object to the total amount falling on it

Ambient – Condition of the environment surrounding an aircraft or other body in motion, but undisturbed or unaffected by it, as in "ambient air," or "ambient temperature"

Aphelion – That point in the orbit of a planet or comet which is farthest from the sun

Apogee – That point in the orbit of an earth satellite which is farthest from the earth

Apsides (sing. Apsis) – Points in the orbit of a satellite or planet nearest to or farthest from the center of attraction

Asphyxia – The consequence of interference with respiration, whether by mechanical means or by the inhalation of gases containing insufficient oxygen

Astronomical unit, AU – The mean distance from the Earth to the sun

Balanced circulation – System of winds resulting when the horizontal pressure gradient force is just balanced by the Coriolis and centrifugal forces associated with the circulation

Ballistic trajectory – The trajectory followed by a body under the action of gravitational forces and the resistance of the medium through which it passes. (A rocket vehicle without lifting surfaces will describe a ballistic trajectory after its engines are shut off) ·

Biconical antenna – An antenna consisting of two conical conducting surfaces, excited at the vertices of the cones

Blackbody – The blackbody or ideal radiator absorbs all incident radiant energy and radiates energy at the maximum rate possible per unit area at each wavelength for any given temperature

Blackout – A fadeout of radio communications due to environmental factors such as ionospheric disturbances or a plasma sheath surrounding a reentry vehicle

Bremsstrahlung – Radiation due to acceleration of fast charged particles by nuclei

Chain scission – Cleavage of primary skeletal bonds in a polymer chain causing degradation to form low molecular weight fragments: see depolymerization

Coriolis force – A deflecting force due to the earth's rotation, diverting moving objects to the right in the northern hemisphere, and to the left in the southern hemisphere

COSPAR – Acronym for Committee on Space Research, international council of scientific unions for coordinating space data of international significance

Covalent linkage – Chemical bond formed by the sharing of electrons between elements; electron pairing is the most common type of chemical linkage existing in organic materials (i. e., compounds of carbon) in contrast to the electrostatic or Van der Waal forces encountered in the ionic bond of inorganic compounds

Cross-linking – Mechanism whereby adjacent molecular chains of certain high polymers are interlocked by means of a bridge, thus forming a three-dimensional network; polymers undergoing cross-linking are converted from soft, thermoplastic to hard, thermosetting types

Cryogenics – The science of producing and maintaining very low temperatures; for example, from -50° F to absolute zero

Cyanosis – An effect of asphyxia, among other causes, in which there occurs an abnormally large amount of un-oxygenated blood, resulting in a blue color of the skin and mucous membranes

Depolymerization – Process whereby high molecular weight polymers are degraded into low molecular weight fragments or monomers; depolymerization involves cleavage of the primary skeletal bonds in the polymer chain

Discone antenna – An antenna consisting of a conical conducting surface and a concentric circular image plane, excited at the apex of the cone, usually by a coaxial transmission line through the circular image plane

Dissipation factor – A quantity derived from the complex inductive capacity of a dielectric material that specifies the relative attenuation of an electromagnetic wave in the dielectric; often expressed as $\tan \delta$, where δ is the phase angle of the total current density J with respect to the applied electric field strength

Diurnal – Happening each day

Doppler effect – An apparent change in wavelength of the radiation emitted by a source when there is relative motion between source and observer

Dose (Dosage) – The radiation delivered to a specified area or to the whole body. Units of dose specification are roentgens for x- or γ-rays, reps for beta rays and protons, and alternatively rads for all types of radiation

Dose rate – Radiation dose delivered per unit time

Ecliptic – The plane of the Earth's orbit around the Sun, inclined to the Earth's equator by about 23 deg 27 min

Ecology – The biological science dealing with the relations of organisms and their environment, including relations with other organisms

Elastomer – Type of rubbery high polymer having elastic properties manifested by a high degree of elongation and compressibility

Electron volt, eV – The kinetic energy acquired by a particle of electronic charge when the particle has fallen through a potential drop of 1 volt, V; 1.602×10^{-12} erg, 1.602×10^{-19} joule, J, 1.52×10^{-22} Btu

Emissivity – The emittance of a specimen of material with an optically smooth, clean surface, and sufficient thickness to be opaque

Emittance – The ratio of the radiant flux intensity from a given body to that of a blackbody, at the same temperature

Encephalopathy – Any disease of the brain

Faying surfaces – The adherend surfaces in contact with each other in a bonding operation or bonded structure

Flux – The rate of flow of energy through a surface

Gamma ray, γ-ray – Penetrating short wavelength electromagnetic radiation of nuclear origin

Gyro – A device utilizing the angular momentum of a spinning rotor to sense angular motion of its base about one or two axes at right angles to the spin axis. Also called "gyroscope"

Heat sink – A material capable of absorbing heat; a device utilizing such material for the thermal protection of a spacecraft or reentry vehicle

Hohlraum – An apparatus for determining infrared reflectance in which a sample forms a part of one wall of a uniformly heated cavity

Honeycomb material – Composite material laminate consisting of a central core layer made up of open cells (usually hexagonal, hence the name) and outside facing layers of thin solid material

IMP – Acronym for Interplanetary Monitoring Platform, an Explorer series satellite measuring fields prevailing in space

Injection – The process of putting an artificial satellite into orbit. Also the time of such action

Ionization – The process or the result of any process by which a neutral atom or molecule acquires either a positive or a negative charge

Ionizing radiation – Any electromagnetic or particulate radiation capable of producing ions, directly or indirectly, in its passage through matter

Laser – (From Light Amplification by Stimulated Emission of Radiation.) A device for producing light by emission of energy stored in a molecular or atomic system when stimulated by an input signal

Maser – An amplifier utilizing the principle of Microwave Amplification by Stimulated Emission of Radiation, or emission of energy stored in a molecular or atomic system by a microwave power supply stimulated by the input signal.

Meteor – The light flash resulting from the entrance of a meteoric particle into the Earth's atmosphere at high speed

Meteoric particle – An intra-solar-system body or particle of any size smaller than a planet and generally of cometary or asteroidal origin

Meteorite – A meteoric particle reaching the Earth's surface with a mass greater than 10^{-4} gm

Meteoroid – A meteoric particle in space of any size greater than 10^{-4} gm

Micrometeorite – A meteoric particle reaching the Earth's surface with a mass equal to or less than 10^{-4} gm

Micrometeoroid – A meteoric particle in space of mass equal to or less than 10^{-4} gm

Monocoque – A construction technique, extensively used in the aircraft industry, wherein the vehicle skin is designed as the load bearing member

Monopole – Usually a quarter-wavelength long radiator perpendicular to a conducting image plane, excited by means of a coaxial transmission line through the image plane

Micron, μ – 10^{-6} meter, m; 10^{4} angstroms, Å; 10^{-4} cm; 3.94×10^{-5} in.

NaK – An alloy of sodium and potassium used as a reactor coolant

Nodal regression – The slow westward motion of the nodes (those points at which the orbit passes through the plane of the ecliptic) of the moon's orbit, a cycle being completed in about 18.6 yr; this effect is caused by solar attraction

n-on-p solar cells – Photovoltaic energy conversion cells in which a base of p-type silicon (having fixed electrons in a silicon lattice and positive holes which are free to move) is overlaid with a surface layer of n-type silicon (having fixed positive holes in a silicon lattice with electrons which are free to move)

OGO – Acronym for Orbiting Geophysical Observatory

OSO – Acronym for Orbiting Solar Observatory, a satellite designed to make measurements of solar parameters

Ozone – The triatomic form of oxygen. A blue gas with a characteristic pungent odor and of high toxicity

Perigee – That point in the orbit of an earth satellite which is nearest to the earth

Perihelion – That point of the orbit of a planet or comet at which it is nearest to the sun

Photon – According to the quantum theory of radiation, the elementary quantity or quantum of radiant energy

Polymer – High molecular weight resinous composition consisting of a repeating chemical unit

p-on-n solar cells – Photovoltaic energy conversion cells in which a base of n-type silicon (having fixed positive holes in a silicon lattice and electrons which are free to move) is overlaid with a surface layer of p-type silicon (having fixed electrons in a silicon lattice and positive holes which are free to move)

Rad – An absorbed radiation unit equivalent to 100 ergs/gm of absorber

Radome – Housing for an antenna, essentially transparent to radio frequency radiation

Rayleigh scattering – The law, discovered by Lord Rayleigh, which related the scattering of light to the fourth power of its wavelength

Reflectance – The fraction of radiation incident on an object which is reflected

Reflectivity – The reflectance of an opaque, optically smooth, clean portion of material

Restrahlen bands – Spectral regions (usually in the infrared region) in which certain materials exhibit extraordinarily high reflectivity

Roentgen, R – The quantity of x- or γ-radiation, the associated corpuscular emission of which, per 0.001293 gm of air, produces, in air, ions carrying 1 electrostatic unit of electricity of either sign

Scale height – A measure of the altitude variation between density and/or temperature in an atmosphere

Solar constant – The rate at which solar radiation is received in a surface perpendicular to the incident radiation and at the Earth's mean distance from the Sun, but outside the Earth's atmosphere. $G = 442$ Btu/hr-ft^2 or 1.94 cal/min-cm^2

Solar cycle – Sunspot cycle of approximately 11-yr duration

Solar flare – A sudden disturbance on the surface of the sun in the course of which high-energy particles are emitted

Spallation – A type of nuclear reaction in which several particles are ejected from the nucleus. Also the ejection of secondary material from the rear surface of a sheet of material bombarded by a meteoric particle

Spectral irradiance – Radiation falling on unit area of a body and having wavelengths lying in some range between λ and $\lambda + \Delta\lambda$

Spacecraft – Devices, manned and unmanned, that are designed to be placed into an orbit about the earth or into a trajectory to another celestial body

Synchronous satellite – A satellite orbiting the earth at periods equal to, or multiples of, the Earth's rotational period; i.e., making one, two, three, etc., orbits in a 24-hr period

Tabor black – A type of thermal control surface having a high absorptance at wavelengths shorter than 2 μ and a low absorptance at longer wavelengths

Telemetry – The science of measuring a quantity or quantities, transmitting the measured value to a distant station, and there interpreting, indicating, or recording the quantities measured.

Terminator – The line dividing the illuminated and the dark parts of the disk of a satellite or planet

Thermistor – A resistance element made of a semiconducting material which exhibits a high negative temperature coefficient of resistivity

Torr – A pressure unit equivalent to that required to support a column of mercury 1 mm high

Total hemispherical emittance – This is generally taken to be equivalent to the unmodified term, emittance. Total indicates that emitted energy of all wavelengths is accounted for; hemispherical means that all energy emitted into the hemispherical space above the surface element of interest is accounted for

Transmittance – The fraction of radiation incident on an object which is transmitted through the object

TRS – Acronym for Tetrahedral Research Satellite

Twilight orbit – An orbit in which the satellite orbital plane is perpendicular to the earth-sun line; the satellite always passes over those parts of the earth experiencing twilight conditions

Type N connector, Type TNC connector, Type SC connector – Units for joining coaxial radio frequency cables and transmission lines, fabricated according to standardized military specifications

Ullage – The amount that a container lacks of being full

Whip antenna – A monopole antenna consisting of a thin conductor, such as wire or a metal rod, usually flexible

X-rays – Penetrating electromagnetic radiations having wavelengths shorter than the ultraviolet. It is customary to refer to photons originating as a result of some nuclear change as γ-rays and to reserve the term x-ray to those photons originating as a result of a change in only the extranuclear part of the atom

Yagi – An antenna array consisting of a driven element and one or more subsidiary elements which radiate due to currents induced in them by the primary field of the driven element

Zodiacal light – Light bands extending on either side of the sun approximately in the plane of the ecliptic, and believed to be part of its outer atmosphere

Generic or Trade Name	Chemical Name or Composition	Commercial Products Referred to in This Handbook
PLASTICS		
Epoxy, Epon	Epoxide, Epoxy	Epon 828, Epon 1001, Epon 815, Epon 8
Phenolic	Phenol-formaldehyde	Conolon 506, Tre-varno F-130, CTL-91LD
Silicone	Siloxane	DC 2104, DC 2106
Kel-F	Monochlorotrifluoroethylene	Kel-F-K25, -300, -500, -81
Teflon (TFE)	Polytetrafluoroethylene	
Teflon (FEP)	Fluorinated ethylene propylene	
Tenite	Cellulose acetate butyrate	Tenite II
Marlex, Hi-Fax	Polyethylene (linear or high density)	Marlex-50
Geon, PVC, Tygon	Polyvinyl chloride	Geon 2046, 8630, 8640
Teslar, Tedlar, PVF	Polyvinyl fluoride	
Kynar	Polyvinylidene fluoride	
Acrylic, Plexi-glass, Lucite	Polymethyl methacrylate	Plexiglass II
Lexan	Polycarbonate	
Irrathene	Irradiated polyethylene	
Saran	Polyvinylidene chloride poly-vinyl chloride (copolymers)	
Mylar	Polyethylene terephthalate	
Dacron fiber	Polyethylene terephthalate	
Orlon fiber, Acrilan	Polyacrylonitrile	Acrilan 16
Nylon (6, 6)	Hexamethylene adipamide	Zytel 101

Generic or Trade Name	Chemical Name or Composition	Commercial Products Referred to in This Handbook
H-Film, HT-1 Fiber	Polyimide	
Delrin	Polyacetal	Delrin 500, 507
TAC	Triallylcyanurate	Vibrin 135, Laminac 4232
DAP	Diallylphthalate	
ELASTOMERS		
Thiokol	Polysulfide	
Nitrile, Buna-N	Polybutadiene-acrylonitrile	Hycar OR-15, Hycar OS-10
Neoprene	Polychloroprene	Neoprene W, Neoprene GN
Natural rubber	Polyisoprene	
Viton	Hexafluoropropylene and vinylidene fluoride (copolymers)	Viton A, Viton B
Butyl	Polyisobutylene-isoprene	
Buna-S, SBR, GR-S	Polybutadiene-styrene	
Philprene	Polyisoprene (synthetic)	
PFBA	Polyperfluorobutyl acrylate	IF4
Fluorosilicone	Fluorosilicone	LS-53, LS-55
Hypalon	Chlorosulfonated polyethylene	
Kel-F	Monochlorotrifluoroethylene	Kel-F-3700, Kel-F-550
Silastic, Silicone	Siloxane, methyl	Silastic 7-170, RTV
Adiprene	Polyurethane (isocyanate)	Adiprene C
Acrylic, Acrylate	Polyethylacrylate	Vyram

CONVERSION FACTORS

LENGTH

1 angstrom, Å	= 10^{-8} cm
1 micron, μ	= 10^{-4} cm
1 mil	= 0.001 in.
1 cm	= 0.3937 in.
1 meter, m	= 3.281 ft
1 kilometer, km	= 0.5396 sm
1 statute mile, sm	= 1.609 km
1 nautical mile, nm	= 1.853 km = 1.1516 sm
1 Earth radius	= 3963 sm = 3440 nm
1 astronomical unit, AU	= 9.29×10^{7} sm = 1.495×10^{8} km

MASS

1 gram, gm = 2.205×10^{-3} lb
1 pound, lb = 453.6 gm

PRESSURE

1 Torr = 1 mm of Hg = 1.934×10^{-2} psi
1 psi = 51.71 mm of Hg = 6.805×10^{-2} atm
1 atm = 14.696 psi

TEMPERATURE

To obtain °F from °C, multiply by 9/5 and add 32
To obtain °K from °C, add 273.16
To obtain °R from °K, multiply by 9/5
To obtain °R from °F, add 459.69

ENERGY

1 electron volt, eV	= 1.602×10^{-12} erg = 1.52×10^{-22} Btu
1 calorie, cal	= 4.184×10^{7} erg = 3.968×10^{-3} Btu
1 British thermal unit, Btu	= 1.0549×10^{10} erg
1 kilowatt-hour, kW-hr	= 3413 Btu

Temperature corresponding to 1 eV = 11606 °K

POWER

1 watt, W	=	10^7 erg/sec	=	1.341×10^{-3} hp
1 horsepower, hp	=	745.65 W	=	0.70696 Btu/sec
1 British thermal unit per second, Btu/sec	=	1.0549 W	=	1.4145 hp

RADIATION

1 rad	=	100 ergs/gm
1 roentgen, R	=	87.7 ergs/gm (approximately)
1 rad	=	10^{16} protons/cm^2 (to within factor of 4)
1 rad	=	3×10^7 electrons/cm^2 (to within factor of 2)

MISCELLANEOUS

1 cal/cm^2-sec-° C/cm	=	0.8062 Btu/ft^2-sec-° F/in.
	=	241.8 Btu/ft^2-hr-° F/ft
1 mean solar year, yr	=	8766 hours, hr = 3.16×10^7 sec

ERRATA FOR SPACE MATERIALS HANDBOOK, 2nd EDITION, ML-TDR-64-40

Location	Instead of	Read
p. 33, Col. 1		add λ to Col. 1 heading
p. 76, Ref. 11	p. 1572	p. 1573
p. 142, Col. 1 last item	Ecobond 70	Eccobond 70
p. 152, graph scale label	abosrptance	absorptance
p. 192, Line 37	plexiglass	Plexiglass
p. 193, Line 13	concent	consent
p. 197, Line 14	clevite	Clevite
p. 207, Line 2	Lamb	Lomb
p. 226, Col. 9 Item 1	4.574	4,574
p. 267, Line 8	reinforces	reinforced
p. 294, Ref. 90	Design,	Machine Design, p. 22
p. 297, Line 7	as low as -420° F	as low as -423° F
p. 297, Line 8	5000 psi	8000 psi
p. 302, Fig. 12-2		omit line "Epon VIII stopped here"
p. 319, Ref. 8	Interact	Interaction
p. 325, Line 30	resistance among polymers	resistance among common polymers
p. 330, Cols. 3 and 4, heading	Leak Rate (ft^3/yr)	Leak Rate (lb/inch-year)
p. 330, Cols. 5–8, heading	Leak Rate (ft^3/yr)	Leak Rate (lb/year)
p. 332, Col. 2, Item 9	11.33	11-33
p. 332, Cols. 2–6 heading	Permeability (10^{-7} $cm^3/sec/cm^2/cm$)	Permeability (cm^3/sec-cm^2-cm-atm)
p. 333, Col. 1, Item 3	Polytetrafluoroethylene	fluorinated ethylene-propylene
p. 338, Col. 1, Item 2	1F4	1F4
p. 365, Ref. 24	M. M. Fulk	M. M. Fulk and K S. Horr

Location	Instead of	Read
p. 382, Col. 4, Item 1	8.3×10^5	8.3×10^{-5}
p. 383, Line 18 and Line 20	polybenzimidazoles	polyimides
p. 384, Line 23	of O (Ref. 14).	of O_2 (Ref. 14).
p. 384, Line 28	polybenzimidazole	polyimide
p. 385, Table 15-12, Title	polybenzimidazole	polyimide
p. 386	(blank page)	(see p. S-182)
p. 390, Line 8	polybenzimidazole	polyamide
p. 392, last running head	polybenzimidazole	polyimide
p. 395, Line 4	polybenzimidazole	polyimide
p. 395, Line 7	polybenzimidazole	polyamide
p. 400, Table 15-23 Col. 1, Item 6	(Plexiglass	Plexiglass
p. 400, Table 15-23 Col. 1, Item 10	(Tenite II)	Tenite II
p. 407, Line 40	plexiglass	Plexiglass
p. 410, Ref. 23	Chem Eng. Prog. (A.I.Ch.E.) 59 (40) 103–117, 1963	Chem. Eng. Prog. Symposium Series 40, Vol. 59, p. 103, 1963
p. 497, Fig. 18-1 vertical scale	lower five indices, 10^9, 10^8, 10^6, 10^4, 10^2	10^{-10}, 10^{-8}, 10^{-6}, 10^{-4}, 10^{-2}
p. 501, Lines 10 and 11	$\rho t^{2/3}$ as in Eq. (18.1) or $\rho t^{1/2}$	$\rho_t^{2/3}$ as in Eq. (18.1) or $\rho_t^{1/2}$
p. 501, Line 17	22 km/sec[1]	22 km/sec
p. 504, Fig. 18-5, Title	...in Space vs Lifetime Product	...in Space vs Surface Area-Lifetime Product
p. 530, Col. 3, Item 4	4/29/61	6/29/61
p. 530, Col. 3, Item 5	4/7/61	7/7/61
p. 532, Col. 3, Item 5	6/26/63	7/26/63
p. 539, Col. 6, last item	about	about 10,160

Location	Instead of	Read
p. 541, Col. 2, Item 20	4/7/61	7/7/61
p. 541, Col. 2, Item 25	4/29/61	6/29/61
p. 547, Last two lines	Cadillac epoxy	Cat-a-lac epoxy
p. 548, Line 11	unshiedled	unshielded
p. 554, Line 20	upper out skin – 22 KF	upper out skin – 22° F
p. 559, Line 4	Cadillac	Cat-a-lac
p. 560, Line 7	diallylpthalate	diallylphthalate
p. 565, Line 28	44-in. diameter "wheel"; 9-sided "sail"...	44-in. diameter 9-sided "wheel"; "sail"...
p. 567, Lines 9, 10	...Aluminum steel spring brass bellows stainless steel piston copper ring Mg shutter activated by expansion of penetane;...	...Aluminum, steel spring, brass bellows, stainless steel piston, copper ring, Mg shutter activated by expansion of pentane...
p. 568, Line 32	Cadillac	Cat-a-lac
p. 569, Line 29	diallylpthalate	diallylphthalate
p. 603, Cols. 10 and 11 sub-heading	200 nm (2304 sm)	200 nm (230 sm)
p. 607, Line 13	...in vacuum for induced radiation,...	...in vacuum (for induced radiation,...
p. 607, Line 40	asecnt	ascent
p. 624, Col. 2, Item 2	IIIB(c)	IIIB
p. 631, Col. 1, Item 10	Nylon	Nylon, HT-1
p. 635, Line 12	Tables 21-15, 21-16, and 21-17	Tables 21-14, 21-15, and 21-16
p. 666, Ref. 21	Opon 828	Epon 828
p. 677, Col. 2, Item 15	544, 547, ...	544, 546, ...
p. 680, Col. 2, Item 33	Cadillac black... ...549, 560, 569	Cat-a-lac black... ...547, 559, 568
p. 682, Col. 1, Last item	Ecobond 70	Eccobond 70

Location	Instead of	Read
p. 684, Col. 1, Item 7	Diallylpthalate, . . .	Diallylphthalate, . . . also add pp. 559, 560, 569
p. 684, Col. 2, Item 20	interglactic	intergalactic
p. 687, Col. 1, Item 1	Epo-1368	Epocast H-1368
p. 688, Col. 2, Item 7	silicone	silicon
p. 689, Col. 2, Item 12	. . . silicate paint, . . .	. . . silicone paint, . . .
p. 693, Col. 2 Item 23	Interglactic	Intergalactic
p. 696, Col. 1, Item 37	microbond	micobond
p. 696, Col. 2, Item 26	aluminated	aluminized
p. 699, Col. 2, Item 37	279–282, 385, 613	279–282, 385–387, 390, 613
p. 699, Col. 2, Item 38	Polybenzimidazole, etc.	(omit entire entry)
p. 705, Col. 1, Item 8	Silicone	Silicon
p. 711, Col. 1, Items 18 through 23	Volatization	Volatilization

Figures 15-2 and 15-3 (Ref. 16) summarize the results of tests at elevated temperature and radiation on du Pont HT-1 fabric which is polyamide woven from drawn fibers into fabric form. These data show that the material has much better elevated temperature strength retention at 400 and 600° F after simultaneous exposure to a radiation dose of 3.5×10^6 rad and aging at elevated temperatures than the strength retention at these elevated temperatures but without irradiation. Lesser improvements in elevated temperature strength retention were also noted by irradiation at ambient temperature. The absolute ambient temperature strength value did not significantly change as a result of this radiation exposure (3.5×10^6 rad) with or without simultaneous elevated temperature.

The effects of penetrating radiation on mechanical properties of two additional film materials, polyvinyl chloride and polyvinyl fluoride, are presented in Tables 15-13 and 15-14, (Ref. 17) respectively. Table 15-13 shows that the Geon 8630 was the more radiation resistant of the two types tested at the highest doese, particularly so in the case of the measured tensile strengths of the specimens having the larger film thickness (0.020 in.). More degradation in tensile strength and elongation was apparent in the thinner specimens of each material type. Also, the results show that irradiation in vacuum produces less degradation than irradiation in air for an equivalent dose. This conclusion is generally valid for most polymeric materials and is explained as the elimination of oxidative attack of the polymer chain during irradiation. The data in Table 15-13 show that Geon 8630 retains useful mechanical properties at radiation doses in excess of 10^8 rad. Table 15-14 presents data on the effects of nuclear radiation on polyvinylfluoride film exposed to temperatures from -65 to 260° F and various immersion media (air, MIL-L-7808, and Oronite 8515 diester hydraulic fluids. Good strength retention at these temperatures is evident at doses up to 1.4×10^8 rad (gamma) and 8×10^{15} nvt (neutrons) in air. There is, however, approximately a 65% decrease in total elongation at this dose on the specimens exposed to elevated temperature (260° F). There was no significant protection afforded the polyvinylfluoride by the immersion fluids at the above dose or lower radiation levels, indicating that oxygen attack is not the predominant mechanism in the polymer degradation. See Tables 14-9, p. 337 and 14-18, p. 348.

15.2.3.2 Ultraviolet Radiation

Unshielded organic polymers in space will have a high absorbance in the extreme ultraviolet and will be damaged to some extent. All damage is expected to occur at the surface. If the damaged surface is able to maintain some structural integrity and not gas, flake, or erode off, then the bulk polymer beneath the surface should have some measure of protection.

INDEX SECTION

NOTE ON INDEX USAGE

Attention of the user is called to the presence of two indexes in this volume.

The INDEX, pp. S-185 through S-205, indexes only the information in this volume, ML-TDR-64-40, Supplement 1.

The CUMULATIVE INDEX, pp. S-207 through S-263, indexes information in both this Supplement and in ML-TDR-64-40, Space Materials Handbook, 2nd Edition, which was issued January 1965. Page numbers having an S-prefix refer to the present volume; page numbers with no prefix, refer to the Space Materials Handbook, 2nd Edition.

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