1.1 Spacecraft Subsystems

An extract from “XTOS: 16.89 Final Design Report”, MIT Aeronautics and Astronautics, May 2002.

1.1.1 Introduction

Multi-Attribute Tradespace Exploration with Concurrent Engineering (MATE-CON) utilizes the working knowledge of spacecraft subsystem specialists. Through the use of a software tool that interacts with Microsoft Excel, called ICEMaker, the MATE-CON process is translated into a preliminary design tool. Each spacecraft subsystem specialist is responsible for an Excel workbook that interfaces with the other subsystem workbooks through the ICEMaker software. Each workbook has an Outputs worksheet and an Inputs worksheet. The subsystems are responsible for publishing their respective Outputs to the ICEMaker server. Publishing the Outputs to the server makes the variables available to all the subsystems, and in turn the subsystems request the published variables through their Inputs worksheet. Once an output on a single sheet is changed, it is an iterative process of publishing and requesting of all the subsystems to converge on a single design. A detailed synopsis of each subsystem follows.

1.1.2 Systems

Introduction

The Systems subsystem can be described as the “control” subsystem. Within this workbook is contained a mass summary and breakdown; a power summary and breakdown; and a sheet capturing the main items of the other subsystem sheets to verify that data is being passed correctly. These sheets are checked at each iteration of the ICE process, and the total system mass, both dry and wet, with contingency and without. Another important set of outputs are the contingency levels for each section of the satellite. The inputs far outnumber the outputs in this subsystem, as the systems chair monitors the progress of the design. The various charts (power, cost, and mass breakdowns) are projected on the video screens and verified and compared with the previous iteration. The Systems chair is responsible for calling convergence of the design.

Inputs

The input list for the Systems subsystem is fairly long; more than _ of the total inputs are taken by the chair. Inputs are drawn from every other subsystem and used for verification and calculation of the power, cost, and mass breakdowns.

Outputs

Compared to the inputs list, the outputs list is small. Yet these outputs are quite important, including the current total mass for the system (with and without contingency), average and max power per mode after adding contingencies, cost, and reliability numbers.

Assumptions

There are few assumptions made for the Systems module. Mainly, the contingency numbers are based on SMAD's contingencies for a preliminary design, but as this preliminary design is fairly detailed we have reduced the mass and power budgets to 15%. The design is fairly robust to mass margin reduction—with so much fuel, the final vehicle might come in over weight. That simply means that less fuel would be loaded thereby reducing lifetime but allowing launch.

Fidelity Assessment

As the Systems chair takes so many variables from other systems and performs a relatively small number of simple calculations, the fidelity of the chair is dependant only upon the fidelity of the other chairs' calculations.

Verification

Values in this subsystem were verified by hand calculations. Heavy communication between Systems and the other subsystems helped to point out any inconsistencies when passing variables. Errors were therefore easy to find and fix.

1.1.3 MATE-CON Chair

Introduction

The MATE-CON Chair is a new addition to the ICE process. It can be described as the link between MATE and CON. Contained within this workbook is an Excel to Matlab link that allows the running of the utility code (developed under MATE) with inputs taken from the current design in the ICE Session. The purpose of this chair is to assist the systems engineer in directing the trades in the design—as changes are made, the utility can be tracked and the proper direction for the design can be determined.

Inputs

The input list for the MATE-CON chair is short, and includes only those parameters from the ICE session that are needed for utility calculations, i.e., the parameters that are used to compute the attributes.

Outputs

The outputs list is also fairly small for this chair and includes the utility for each attribute as well as the overall multi-attribute utility.

Assumptions

The assumptions made here are the same assumptions made in the earlier MATE section of this paper.

1.1.4 Mission

Introduction

The mission subsystem can be described as the "primary" subsystem. Within this workbook is a shortened list of the design vector (from the MATE Matlab modules). These design variables are changed by hand for each iteration of the ICE process, and the outputs are then sent to every other subsystem. In addition to the design vector, the mission subsystem also calculates an assortment of mission parameters, such as orbit characteristics, launch vehicle characteristics, delta V budget, and spacecraft lifetime. The outputs far outnumber the inputs in this subsystem, as the spacecraft lifetime and delta V budget are the only two calculations requiring inputs from other subsystems.

Inputs

The input list for the Mission subsystem is significantly shorter than most. Inputs are drawn from the System, MATE-CON, Configuration, and Propulsion subsystems, yet all of these inputs are used for just the lifetime and delta V budget calculations:

- *Delta V* inputs from propulsion (for Stationkeeping, ADACS, and contingency) are combined with internally calculated values for insertion and deorbit to produce a delta V budget.
- The lifetime calculation incorporates many variables such as *coefficient of drag* and *cross sectional area* from Configuration, total wet mass from Systems, *Stationkeeping delta V per orbit, per BCD* from MATECON, and *total propellant mass, propellant mass per orbit,* and *specific impulse* from Propulsion.

Outputs

Compared to the inputs list, the outputs list is very extensive. Yet these outputs can be partitioned via the calculations that created them. The output types of each are:

- Design Variables - these outputs require no calculations and were changed many times during the iteration process: *perigee altitude, apogee altitude, inclination*, and *total delta V*.
- Launch Vehicle Selection – these outputs are directly related to the choice of launch vehicle: *launch vehicle type, payload capacity, payload dimensions, launch environment, cost, reliability, insertion error,* and *mass*.
- Orbit Determination – these outputs are directly related to the chosen orbit: *orbit parameters, eclipse time,* and *orbit period.*
- Calculations – these are simply the *lifetime* and *delta V budget* outputs calculated in the workbook.

Assumptions

When trying to find data on our primary launch vehicle, the Minotaur, we ran into a few obstacles. Because the Minotaur uses an ICBM as a lower stage, we could not find an accessible payload planner's guide or something similar. And with its first launch in the year 2000, there is very little historic data to pull from. Consequently, some of the values for the launch environment were assumed from current models of similar launch vehicles (namely the Taurus and Pegasus). These values are marked in the worksheets. In addition, J2 effects are not included in the lifetime calculation, due to the fact that a precessing orbit has negligible impact on our specific mission.

Fidelity Assessment

The largest possible source of error in the mission subsystem is in the launch environment assumptions. Yet these values are passed only to the structures subsystem, where they are used in a precautionary analysis to ensure our spacecraft survived the launch phase. Changes in these values have a minimal effect on the spacecraft as a whole. There is also uncertainty in the lifetime calculation, which is found by burning the available fuel for stationkeeping and ADACS, until there is none left. However, many of the inputs for this calculation are conservative (such as Stationkeeping delta V per orbit/BCd, specific impulse, and total usable propellant mass) and thus the calculation for lifetime is conservative as well.

Verification

Values in this subsystem were verified by hand calculations. In addition, heavy communication between Mission and directly related subsystems helped to solve any inconsistencies when passing variables.

1.1.5 Payload

Introduction

The payload sheet functioned to directly translate the specifications of the three instruments from their respective requirements documents to the ICE environment. The three instrument components of the payload were: 1) Satellite Electrostatic Triaxial Accelerometer (SETA), 2) Absolute Density Mass Spectrometer (ADMS), and 3) Composition and Density Sensor (CADS).

Inputs

The payload subsystem uses the following inputs:

- Power mode definitions, which were used to calculate the instrument power requirements for each phase of flight

Outputs

The payload subsystem output the following:

- Mass, dimension, and location and requirement for each instrument
- Combined pointing requirement
- Peak and average power requirements for each power mode
- Number of redundant instruments
- Failure rate of each instrument

Assumptions

The only assumption made was that the requirements document was accurate in its portrayal of size, shape, and other specifications of the instruments.

Fidelity/Verification

This is a function of the requirement document's accuracy.

1.1.6 Configuration

Introduction

The configuration subsystem arranges each of the subsystem components on the spacecraft. A very useful tool for this arrangement is DrawCraft.[1] The subsystem chairs publish the dimensions, mass, and locations (if applicable) of each of the components using ICEMaker. Next, these values are automatically updated to a SCMS (Shared Mechanical Control Sheet) text delimited file, which is then read by DrawCraft. DrawCraft then creates an assembly in SolidWorks which provides information on the weight distribution and surface area over the entire spacecraft.

The components need to be placed in such a way that certain criteria are met. Since this spacecraft is traveling through a significantly dense part of the atmosphere, it needs to be aerodynamically stable. In this case, the center of gravity needs to be forward of the half-chord point. For our purposes, the length of the spacecraft is approximated as its chord. Also, since the scientific sensor suite was previously chosen, the requirements of the sensors need to be met. The ADMS and CADS sensors are required to be ram-facing, and the SETA sensor is required to be within six inches of the center of gravity of the entire spacecraft. Another important requirement is that the entire vehicle needs to be able to fit inside the payload fairing for the chosen launch vehicle.

This subsystem is built so that with a small amount of human involvement, the configuration of the satellite can be dynamically changed during the ICE sessions. Human involvement is required for several different reasons. First, the updated SCMS needs to be loaded into DrawCraft. Once DrawCraft creates the model in SolidWorks, one needs to open a special window within SolidWorks to produce the weight distribution and surface area outputs. DrawCraft does provide some of these required outputs; however, SolidWorks provides all of the required outputs, and does so in a favorable manner. For example, the moments of inertia calculated by DrawCraft are about a reference axis, and the moments created by SolidWorks are both around a reference axis and the center of gravity. Since the center of gravity changes with every design iteration, SolidWorks is a more useful tool. It is conceivable that this type of program interface could be automated so that the SCMS file is automatically updated, and the information is automatically published from SolidWorks. This would aid greatly in the speediness of the ICE sessions. It was not developed in this case because of time constraints.

Even if the SCMS file could be automatically updated, and the outputs automatically published, it would only take care of parametric variations on the design. The configuration subsystem is unique in that at each iteration step in the ICE session, the configuration needs to be visually evaluated and possibly changed by the configuration chair. A good example of this is that at the start of the ICE sessions, the original design

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[1] DrawCraft - Dr. Joel C. Sercel (Caltech, Pasadena, California, USA) in the Laboratory for Spacecraft and Mission Design for the use of the DrawCraft (a spacecraft configuration tool).

for the fuel tank was a single sphere. As the fuel mass increased, the fuel tank impinged upon, then eventually exceeded, the wall of the main bus. The result was that a non-parametric change to two cylindrical tanks needed to be made, as can be seen below in figure 1. Another example that illustrates the necessity of configuration evaluation at each step concerns the scientific sensors. Once the change to cylindrical fuel tanks was made, a trade was performed in which the satellite altitude was lowered. This required more fuel to be aboard, and the fuel tanks to lengthen. Eventually the tanks, though they fit inside the main bus, encroached upon the space needed for the scientific instruments. This can only be seen when the configuration chair takes the time to visually evaluate the design. In this trade, the constraining factor happened to be the space required for the fuel tanks. If the configuration were not evaluated visually, an impossible design could be chosen.

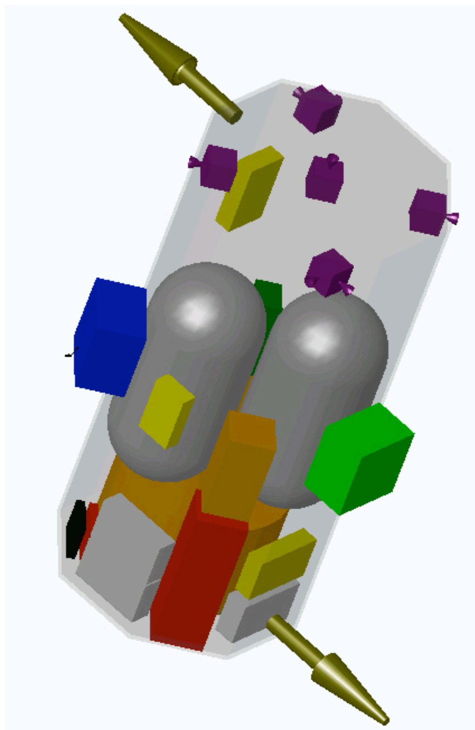

Figure Error! No text of specified style in document.**–1: Final design. Note cylindrical fuel tanks (grey)**

Inputs

Generally, the inputs to the configuration subsystem are the dimensions, mass, and location of each of the satellite components. The components that were modeled were:

- Main bus
- CADS, ADMS, and SETA sensors,
- Omni-directional antennas
- Primary and secondary batteries
- Fuel tanks
- ADACS thrusters
- Main Thruster

- Telecom boxes
- C&DH computers

Outputs

The parametric outputs of this subsystem described the weight, surface area, and volume of the total spacecraft. These are listed below:

- Basic cross-sectional shape of the main bus
- Basic shape of entire bus
- Cross sectional area
- Total surface area
- Coefficient of drag
- Distance from the c.g. to center of aerodynamic pressure
- Distance from the c.g. to center of solar radiation pressure
- Distance from the total internal torque to c.g.
- Moment of inertia, mass xx
- Moment of inertia, mass xy
- Moment of inertia, mass xz
- Moment of inertia, mass yy
- Moment of inertia, mass yz
- Panel area
- Total volume

Another important product of this subsystem is a CAD drawing that provides information on the placement of each component. SolidWorks drawings of the selected architecture can be found in Appendix A.

Assumptions

The limit of the coefficient of drag on a blunt body in the upper atmosphere, computed using free-molecule flow, is found to be 2.0.[2] Since extensive modeling would be required in order to produce a more accurate number, this value is used as a constant throughout the design. A sensitivity analysis should have been performed on this value, but was not due to time considerations.

Another important assumption is that the antennae can be folded along the main bus in order for the spacecraft to fit inside the launch vehicle payload fairing.

In order to distribute the mass of the structures and mechanisms (cabling, small struts, etc), it is contained in the mass of the main bus, which is evenly distributed along the length of the bus.

It is also assumed that solar arrays would be able to be attached to the body of the main bus.

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[2] Hoerner, Sighard, Fluid-Dynamic Drag, Sighard Hoerner, 1965

Fidelity Assessment and Verification

The fidelity of the parametric outputs is only as accurate as the inputs used to generate them. Since all of the inputs are physical parameters of subsystem components, the fidelity of the model depends on the combined fidelity of all of the contributing subsystems. Some of the outputs, however, such as the distances, are approximate values, derived from the SolidWorks configuration. These are approximated since those values do not change appreciably during the ICE session iterations, and it is costly time-wise to input these values at each iteration. The SolidWorks model directly reflects the inputs from each of the subsystems.

An electronic copy of the configuration subsystem sheet can be found on the XTOS compact disc (Configuration.xls).

1.1.7 Power and Pyrotechnics

Introduction

The Power and Pyrotechnic Subsystem (Power and Pyro) sheet selects and sizes the components of the electrical power system (EPS) for the spacecraft. The design methodology follows the steps listed in SMAD section 11.4 for both the power generation and energy storage components. A significant portion of the component sizing is carried out automatically based on the average and peak power requirements of the various subsystems. However, the user has the option of making a number of trades that can dramatically reduce (or inflate) the EPS mass and size for a given set of power requirements. These trades include solar array materials, power regulation schemes, solar array configuration (partly constrained by spacecraft configuration), battery couple, discharges per orbit, depth of discharge, and redundant components. In addition to these trades, the sheet also features variable degradation factors that allow the user to adjust the conservativeness of the design.

Inputs

The design of the EPS is affected by 135 system-level parameters. Most aspects of the mission affect the EPS simply because so many other subsystems have specific power needs. The primary drivers in the size, mass, and complexity of the EPS are as follows:

- Mission lifetime
- Spacecraft configuration
- Orbit characteristics (time in eclipse, etc…)
- Average power loads (per subsystem)
- Peak power loads (per subsystem)
- Bus voltages required (per subsystem)

Outputs

The Power and Pyro sheet passes 95 different parameters as outputs to the various subsystems. Among the largest factors in determining the overall size, mass, and cost of the spacecraft are as follows:

- Solar array mass and size
- Solar array configuration (# of panels, body-mounted vs. deployed, etc…)
- Solar array type
- Solar array power BOL/EOL
- Secondary/Primary battery mass, size, and quantity
- Secondary/Primary battery couples
- Secondary/Primary battery power capacity
- Power regulation and control mass

Assumptions

While the Power and Pyro sheet is considerably robust to various mission types and spacecraft configurations, the level and scope of the design requires that some assumptions be made to simplify the design process.

- The mission consists of only a single satellite
- Solar arrays are the predominant option for a power source.
- The solar arrays are rectangular panels, regardless of configuration.
- The battery dimensions follow a 2:1:1 ratio (length:width:height).
- Transmit power may be needed at any point during daylight or eclipse.

Fidelity Assessment

The fidelity of the Power and Pyro sheet is primarily determined by the accuracy of the information within the solar material and battery couple databases. The attributes associated with each component, such as energy density for solar materials and maximum cycle life for battery couples, play a key role in determining the overall size and mass of the EPS. Because these characteristics vary considerably between manufacturers and over time, the values in the database can be considered conservative averages at best.

The sheet also lacks fidelity in the calculation of mass and power estimates for the power regulation and control equipment (PCU). The mass of the PCU and regulators is estimated using a simple approximation from SMAD that relates total mass to the amount of power regulated. While this is a very rough approximation, a lack of better information exists without actually designing the spacecraft bus and power control systems.

As a final note, the overall conservativeness of the EPS sizing calculations remains in question. Interestingly enough, the size of the body-mounted solar panels in each iteration never actually approached a total area that would fit the spacecraft. While this discrepancy may at first seem like an obvious design conflict, it is not certain whether the solar array size is a product of overzealous power requirement estimates or a product of far too conservative efficiency calculations. Throughout the design iteration process, the power requirements were repeatedly noted as a bit high for such a small satellite. Unfortunately, the level of detail of the preliminary design is not sufficient to determine more accurate power figures.

Verification

Verification of the Power and Pyro sheet was conducted using two methods. First, sample requirements from satellite examples given in SMAD were fed through the sheet to verify that the design results matched (within a small percentage) the results listed. Once the nominal verification had been completed, several inputs were modified to ensure that moderate increases in design requirements yielded only moderate increases in EPS mass and size.

1.1.8 Structures and Mechanisms

Introduction

This subsystem module estimates the vibrational environment that the satellite will experience on the launch vehicle, determines the number of mechanisms required for operation, estimates the power required by the mechanisms, estimates the required structural mass based on a factor of safety of 1.25, and also estimates the launch carrier mass.

The vibrational environment data consists of sound pressure levels, the acoustic environment, random vibrational environment, the pyrotechnic shock environment, peak and sustained accelerations, and the power spectral density. From this data, the calculated natural frequencies of the structure are compared to the frequencies that the satellite may experience to verify that at least the first natural frequency will not be encountered while attached to the launch vehicle.

The number of mechanisms on board the spacecraft depends on the type of power source and the type of antenna. Depending on the combination of power source and antenna, the number of mechanisms required for satellite operation is determined as well as the power required.

The structural mass required is calculated based on the mass of the subsystem components, payload mass, and external component masses. Based on the selection of the primary structural material, the loads (axial and lateral) and stresses are calculated based on the spacecraft structural dimensions.

Inputs

- The structures and mechanisms subsystem uses the following inputs:
- Launch vehicle vibration data
- Subsystem component masses
- Payload mass
- External component masses (solar arrays, antennas)

Outputs

- The structures and mechanisms subsystem outputs the following:
- Average and peak power requirements
- Structural mass
- Launch carrier mass
- Structural reliability

Assumptions

This model makes use of the mass of the satellite subsystem components to estimate the structural mass required. The estimate of structural and cabling mass is based on percentages of the subsystem component masses. The estimate of the launch carrier mass is based on a percentage of the satellite mass including fuel and contingency. Each structural component is to be designed to have a reliability of 99.999%, or such that the overall structural reliability is greater than 99%.

Rationale for simplification

The use of satellite subsystem component masses for estimation of the structural mass is a good approximation in the preliminary design phase. The actual design of the structural truss and launch carrier is a detail design issue and would be nearly impossible to construct based on the dynamic state of this design tool.

Fidelity Assessment

The properties of the materials available for the structure and launch carrier material are that of well-known and documented materials. The percentage estimates of structural mass based on subsystem components are rule of thumb estimates and have an error associated with them based on the truss arrangement in the detail design phase. The vibrational and shock environments of the launch vehicle that the satellite will experience are estimated based on available launch vehicle performance data. This data is the most accurate that could be found, which may include some rounding error associated with the actual environment. The power required (peak and average) for the operation of the spacecraft mechanisms is an estimate that is dependent on the type of mechanisms used and will vary depending on the inertia of the actuated component.

Verification

The structures and mechanisms subsystem module was tested under various launch vehicles, structural material, and design parameter changes. Under all the tested conditions, the structure was not subjected to frequencies above its first natural and the structural dimensions were scaled appropriately to ensure a factor of safety of 1.25. All structural and stress calculations were based on solid mechanics equations and were verified.

1.1.9 Command Control and Data Management (CCDM)

Introduction

The Command Control and Data Management subsystem is responsible for the RF communications link and all the avionics. The system is divided into two primary segments. The Telecommunications segment manages the RF link and all associated hardware. The C&DH segment contains all the avionics, software and the data recorders.

Telecommunications

The Telecommunications segment is comprised of two Low-Gain Antennae (LGA) assemblies. Each assembly contains:

- 1 Conical Log-Spiral Antenna
- 1 Multiplexer
- 2 Filters (Bandpass)
- 1 Transmitter
- 1 Receiver
- 1 I/F Amplifier (IFA)
- 1 Low Noise Amplifier (LNA)
- 1 High Power Amplifier (HPA)
- 1 Automatic Gain Control (AGC)
- 1 Crystal Oscillator (XO)
- 2 Mixers
- 1 Set of cabling[3]

The conical log-spiral antennae allow a 0 dB gain with greater than 360o 3dB beam-width. This means the antenna does not intrinsically introduce any signal loss. A standard low-gain spiral antenna will have a negative gain requiring larger amplifiers. The X-TOS LGAs each have a 270o 3dB beam-width, as seen in Figure **Error! No text of specified style in document.**–2.

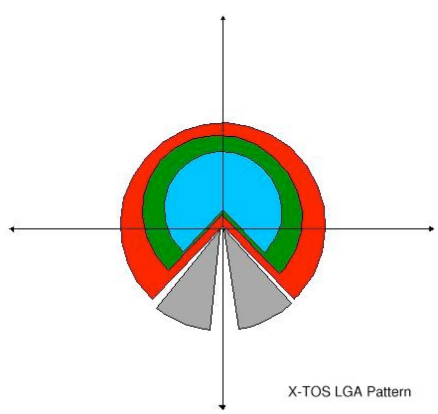

Figure Error! No text of specified style in document.–**2: Low Gain Antenna Pattern**

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[3] This was accounted for within the structures subsystem but is mentioned here for completeness

This was selected to allow for maximum overlap of the two LGAs without the spacecraft body impinging on the signal. Each antenna is made of an aluminum wire wrapped around a plastic cone. The cone is 0.1 m at the base and 0.33 m high. The spacing between the turns in the wire is determined by the frequency. We have chosen a nominal S-Band frequency of 2.2 GHz on the uplink and 2.5 GHz on the downlink. These frequencies are compatible with both AFSCN and TDRSS. These frequencies determine a spacing of 0.068 m between turns. Each cone is mounted on a 0.25 m long, 0.06 m wide plastic cylinder to bring the antenna pattern far enough from the body of the spacecraft.

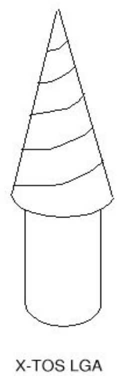

Figure Error! No text of specified style in document.**–3: Low Gain Antenna**

The antenna is connected to the accompanying hardware by coaxial cables with BNC connectors at either end. The line loss for these connectors was assumed at a length of 1 m (5dB). This is a conservative estimate that is justified because the exact location of the hardware box within the spacecraft bus is not yet known. Ideally, the hardware box should be at the base of the spacecraft cylinder. The uplink will pass from the antenna to the LNA, through the multiplexer, and on to the mixer where it is combined with the signal from XO and down-converted. After that, it will be filtered and passed through the receive IFA. It will then go through the receiver which will demodulate the data and finally be routed to the appropriate On-Board Processor.

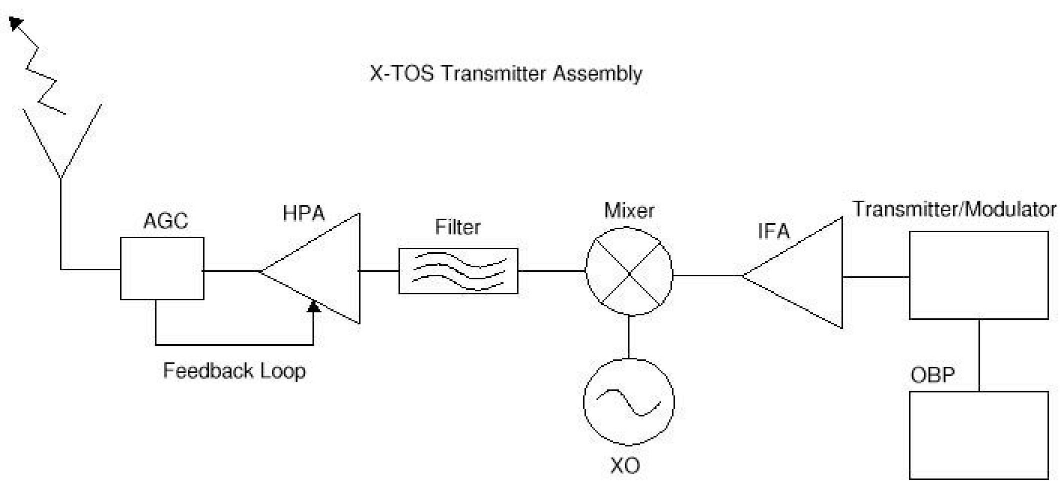

Figure Error! No text of specified style in document.**–4: Transmitter Assembly**

The structure of the command will be such that it will contain sufficient routing information for the subsystems within the spacecraft bus. The downlink will be generated by the On-Board Processors routed through the transmitter for modulation and through the other IFA. Then it will go through the mixer where it will be combined with the signal from the XO and up-converted. It will then pass through the multiplexer and HPA. The signal will then pass through the AGC and be transmitted out the antenna. The AGC will pass a gain regulation voltage back to the HPA.

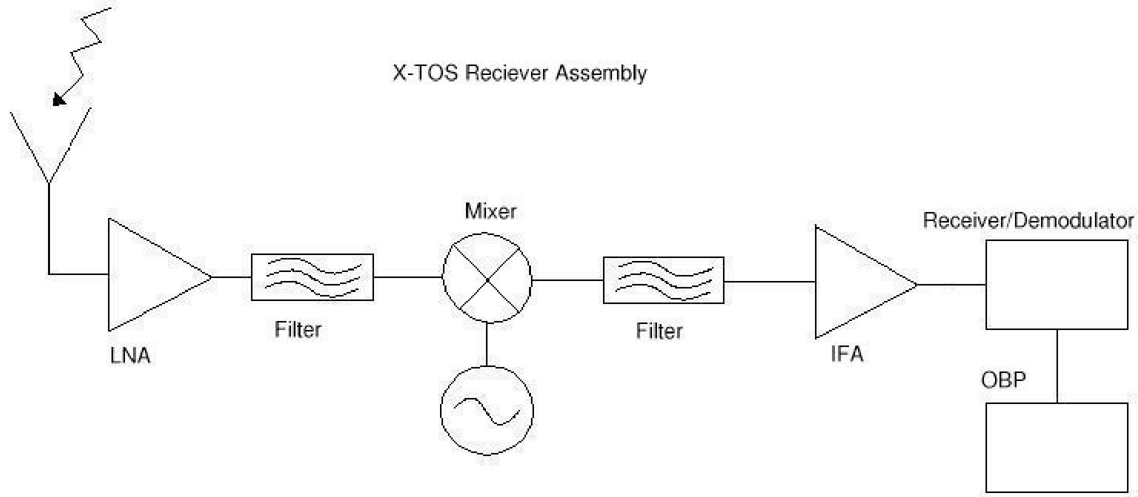

Figure Error! No text of specified style in document.**–5: Receiver Assembly**

All connections should be made with standard 50 ohm coaxial cable using BNC connectors. The VSWR of the cabling should be no greater than 2:1. Though the design states X-TOS can, from a protocol standpoint, communicate with TDRSS the system as is does not have enough power to talk to such a small aperture dish as TDRSS. Unless the power is significantly increased we must use the high gain (100-200 dB) dishes of the AFSCN. The telecommunications sub system was designed to a 4dB link margin.

Command and Data Handling

The C&DH system contains all the avionics and manages all the software. The hardware is as follows:

- 2 General Dynamics 4063RT On-Board Processors (OBP)
- 1 General Dynamics 4063RT Contingency Processor (CP)
- 5 I/O cards (2 for each OBP 1 for the CP)
- 2 Digital data bus switch.
- 2 High speed communications card (For connection to Telecommunications)
- 2 Aero-Astro S4 20Gbit data recorders
- 1 Set of cabling

The OBPs and the CP are the same model computer. The OBPs are redundant computers for nominal operations and are in operation for modes 1-6. The OBPs maintain all mission algorithms and contain all necessary boot data on an EEPROM. The OBPs should contain several memory slots for ground generated sequences. The CP is a special computer who's only use is for Safe Mode (mode 7). The CP will be off with the safe mode algorithms stored in an EEPROM and run them on power up. This should ensure the safe mode algorithms are not corrupted by any glitch in the system. The first three I/O cards connect the OBPs and CP to the spacecraft data bus. They should use IEEE standard radiation hardened 50 pin IDE cables. The cables should be connected to the digital switch which will route data to the prime OBP. The digital switches should be connected in parallel and the redundant one should remain off unless the prime one goes down or is commanded off. All the other subsystems should also be connected by 50 pin IDE cables to the digital switch this then comprises the data bus. The digital switch should allow for duplexed routing of data. The CP should also be connected to the digital switch in the same manner as the OBPs. The OBPs should also be connected via the last two I/O cards to the data recorders. Only the prime OBP should transmit data to the recorders. If one recorder is full or brought off-line, data will be routed to the alternate recorder. The OBPs will be connected to the Telecommunications System via the high speed communications cards. The data will be transferred by the prime OBP from the recorder to the telecom system and sent to the ground. The recorders can each hold five orbits worth of data. We should to attempt to nominally dump the data at most once every three orbits in order to both keep the data rate down and be able to downlink it all in one ground pass. The CP should be placed on a different power bus than the OBPs if at all possible. This would allow the CP to react to a short or other power emergencies on the main power bus.

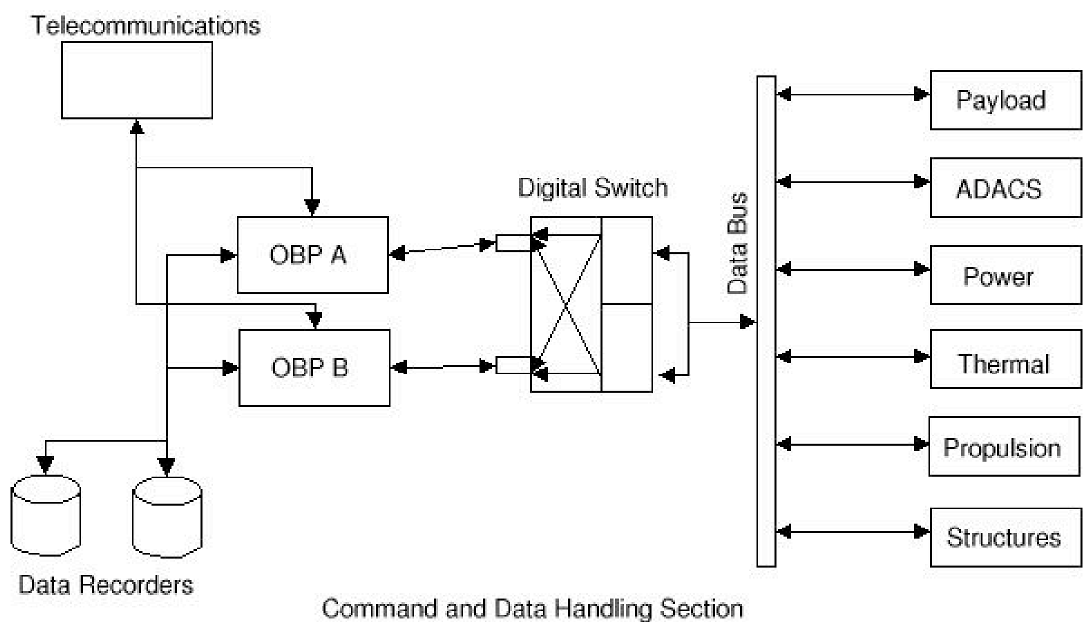

Figure Error! No text of specified style in document.**–6: Command and Data Handling Section**

Data rate, software and throughput requirements are primarily derived from the input from other subsystems. The raw data rate is 17.901 Kbps. That data rate is multiplied by 16 allowing 15:1 ratio of dump data to real-time sent on the downlink. To achieve a BER of 10-7 with a 7dB coding gain the data is Reed-Solomon encoded using a (255,223) code then Verterbi encoded. This means that first in each code word there are 223 8-bit information symbols and 32 8-bit error correction symbols. The code used is AFSCN, TDRSS and CCSDS compatible. The data should then be Verterbi encoded as the second phase of error correction coding. This code should have a rate of _ and a constraint length of 7. This means that seven bits are modulo-2 added to produce two bits, The bits are shifted by one and the process is repeated. This process results in a combined total downlink rate of 664.502 Kbps.

The software needed by the system is estimated by adding the software requirements of all the subsystems and adding a 35% margin. The total resultant code size is 900 Kwds where a word is 8 bits. The total throughput of the system is estimated in a similar fashion and is 22.3 MIPS. The hardware discussed in the previous paragraphs was sized to meet these requirements.

CCDM Subsystem Trades

No CCDM subsystem trades were performed within an ICE session. These were examined off-line in order to verify this design was the most ideal one. The trade study only made certain qualitative assessments of the impact on both configuration and structures. The other options explored for the telecommunications system were communications with TDRSS and a high gain antenna. These selections decreased the overall utility of the system in the high drag environment. TDRSS with a low gain antenna required a significant power increase and even though it allowed for longer possible communications time would have driven power requirements too high for a small satellite mission. AFSCN with a high gain antenna does not allow for a long enough pass to downlink significant amounts of data. And while TDRSS with a high gain antenna is fine from both a power and downlink perspective. A 1.9 m dish on the side of the spacecraft in a high drag environment provides a large drag force. In addition The mass of an aluminum dish and all the actuating equipment is much greater than the two plastic, wire coated antennae we currently have.

ICE Sheets

The ICE sheets for the CCDM subsystem consisted of two sets: one for Telecommunications and one for C&DH. The telecom sheets took as input each of the spacecraft modes and their duration, the overall data rates compiled by C&DH, the required link margin and the orbital data. The C&DH sheets took all the data rates code and throughput requirements from all the subsystems. It also took spacecraft mode and orbit information. The telecom sheets outputted the antenna design, link performance, mass and power requirements, a maximum downlink time per orbit and a data latency figure for utility analysis. The C&DH sheets outputted the volume and mass for each component, the mass and power for the system as a whole. It also publishes the available and required code size, the temperature output of the system, the data storage capacity, the total data rate and the encoded data rate. No functional verification or fidelity assessment was performed on these sheets independently.[SML1]

1.1.10 Thermal

Introduction

The thermal sheet accepts inputs from nearly all sub-system sheets, in order to set a maximum and minimum operating temperature for the spacecraft. With these constraints in place, the user can choose two outer materials for the spacecraft. These materials include solar panels, several different types of metal and non-metal surfaces, and a variety of painted surfaces. By choosing these surfaces accordingly, the user designs a thermally balanced satellite. After this balance is achieved, the sheet calculates the mass and power required to insulate and heat the fuel tanks and lines.

Inputs

The thermal subsystem uses the following inputs:

- Maximum and minimum operating temperatures for different spacecraft systems
- Summed power requirements for power dissipation

Outputs

The thermal subsystem outputs the following:

- Mass and power summaries for the thermal system
- Maximum and minimum allowable operating temperatures

Assumptions

The biggest assumption in the baseline model (inherited from C-TOS) is the idea of the best case/worst case scenario. The sheet calculates the equilibrium temperature for the spacecraft in full sunlight, as well as the temperature of the spacecraft in full eclipse. This assumes an isothermal satellite. For the C-TOS sheets, this means that the spacecraft would require expensive active thermal control systems in order to counteract these scenarios. For the X-TOS satellite, it is known that the orbital periods will be on the order of 90 minutes, and therefore the satellite will never actually reach such extreme thermal equilibriums. Instead, a simple dynamic thermal calculation is run to show, based on the spacecraft's mass, how the temperature will change over time. This allows the user to design a thermally balanced spacecraft that stays within the temperature constraints.

Fidelity Assessment

The single point of glaring infidelity inherited from C-TOS is the aforementioned isothermal assumption. Even in a small and simple satellite, there are obvious locations (near instrument packages, etc.) where the local temperatures will be different than those of a general satellite. The model fidelity is increased somewhat by running two separate thermal balance calculations—one for the satellite in general, and one for the ram face of the satellite, which contains two of the (fairly high powered) instruments. However, at this level of fidelity, we make no provisions for the transfer of heat, which would almost certainly be required for the final design to remain in thermal balance as predicted.

Verification

The outputs of the model seem to agree with intuition—by choosing surface materials that complement one another, one can build a satellite whose heat surplus in one case is almost exactly balanced by its heat deficit in another. The mass of the satellite (> 100 kg) means that the rate of thermal change in the two regimes will be very slow, as shown by the dynamic model tested. It has also been noted that the vast majority of short lifetime, small satellites have passive thermal control systems.

1.1.11 Attitude Determination and Control System (ADACS)

Introduction

The ADACS subsystem is in charge of the attitude determination and attitude control of the spacecraft. It estimates all the disturbances that the satellite will experience in the upper-atmospheric orbit, and given the pointing accuracy needed, will determine what precise ADACS system to use and how much fuel will be needed to control the satellite's attitude.

The disturbances that the satellite will experience are:

- Aerodynamic disturbances
- Gravity gradient torques
- Solar pressure
- Internal torques (which in this case are negligible because the satellite does not have deployable solar panels or other moving parts).

Once these disturbances are calculated, he choice of the most appropriate ADACS sensors and effectors is made as a consequence of the payload and communication subsystems pointing requirements. The total mass and power required for the selected ADACS equipment is output. An estimation of the Delta V per orbit for the attitude control is also calculated based on the disturbances, the pointing control requirements, and the type of ADACS effectors chosen.

Inputs

The ADACS subsystem primarily uses the following inputs:

- Total lifetime
- Altitude of the orbit
- Momentums of inertia of the spacecraft
- Type of thrusters (Isp, Thrust)
- Pointing requirements

Outputs

The ADACS subsystem outputs the following:

- Average and peak power requirements
- Size and mass of the ADACS components
- Delta V per orbit for the attitude control

Assumptions

The main assumption is that the center of gravity of the spacecraft is in front of the center of aerodynamic pressure. This assumption results in aerodynamic stabilization of the satellite and thus eliminates the aerodynamic disturbances from the calculation of the Delta V needed for altitude control. A more precise study of the aerodynamic stability of the spacecraft would be required to determine the additional modifications needed to achieve such stabilization.

The other assumption is that the thruster clusters used for propulsion can also be used for the attitude control as ADACS effectors.

Fidelity Assessment

The properties of the ADACS sensors (power required, mass and pointing accuracy) are well known and very precise. However, the reliability and lifetime of these instruments is not documented due to their short history.

Computationally, the calculations of the gravity gradient and solar pressure disturbances are based on precise astrophysics calculations, and the calculation of the aerodynamic disturbances does not have an influence on the ADACS subsystem calculations (once the assumption of an aerodynamically stabilized spacecraft is made).

1.1.12 Propulsion

Introduction

The Propulsion Subsystem Sheet was based extensively on the work of the C-TOS team. However, major modifications were made to incorporate the “station-keeping” thruster concept (see below for detailed explanation). The basic function of the sheet is to size both the ADACS and station-keeping thrusters and then calculate the total fuel mass required for the mission.

Originally designed for C-TOS, the sheet sizes a number of “thruster clusters” based on inputs from the ADACS sheet. Based on the assumption that the spacecraft will be 3-axis stabilized, each cluster can consist of up to four thrusters for attitude control. The rear-facing thruster will therefore be used periodically to modify the orbit via a station-keeping maneuver.

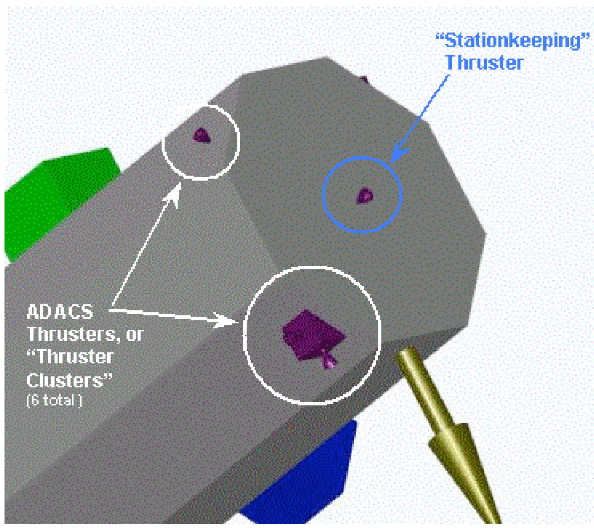

Figure Error! No text of specified style in document.**–7: Thruster Locations**

After examining the unique mission of the X-TOS spacecraft, it is evident that far fewer thrusters will be required to adequately perform the X-TOS mission compared to the C-TOS mission. Since the spacecraft will always be oriented in the same direction relative to its velocity vector, true three-axis control is no longer necessary. In addition, the unique, high-drag environment that will be encountered will partially help maintain the orientation of the vehicle once it is aligned.

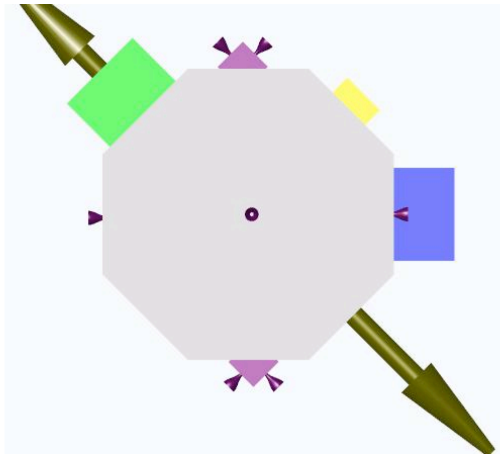

Figure Error! No text of specified style in document.**–8: ADACS Thruster Configuration**

Upon calculating the maximum drag, it was noted that a small (< 5 N) thruster could be placed at the rear of the spacecraft in order to maintain the desired altitude throughout the mission lifetime. In order to minimize mass and complexity, it was eventually decided that this single stationkeeping thruster should share fuel, tanks and other components with the ADACS thrusters.

Besides electric propulsion, other alternative methods of propulsion were only explored in discussion. Once it was determined that the fuel mass would be a significant driver in the overall mass of the satellite and the ultimate determination of mission lifetime, a high Isp was obviously desired. This ruled out cold gas systems. The next propulsion type considered was bipropellant. Although it afforded some improvement in Isp over a monopropellant, hydrazine system, quick "Rough Order of Magnitude" calculations which were based on the engine data contained in the C-TOS sheet showed that in the range of our spacecraft (100 to 400 kg containing 40 to 200 kg of fuel), the additional mass of isolated storage tanks, additional valves, tubes and regulators would essentially cancel out the decrease in mass afforded by the higher specific impulse. In addition, there were more failure modes and expense associated with the bipropellant system.

Returning to the electric propulsion issue, the sheet was set up so that the X-TOS spacecraft could be equipped with electric propulsion. It was noted, however through consultations with the Power and Pyro Chair, that the power required to drive these thrusters would significantly alter the requirements for the spacecraft in terms of solar panels and power storage (batteries). Thus, explicit trades were never performed using electric propulsion as an option.

In retrospect, the decisions to eliminate cold gas, electric and more importantly, bipropellant propulsion systems may have been made prematurely. Part of the power of the MATE and ICE processes lies in expanding the normal tradespace quickly and easily through parametric models. Often times, the assumptions we made at one stage in the process were proven wrong for our particular mission. Every engineer has longstanding beliefs about the interactions of certain subsystems, however in order to rigorously explore all of the possible design trades using MATE and ICE, these assumptions have to be suspended until a more accurate understanding of the unique aspects of the current system can be gained.

Inputs

The key Inputs to the X-TOS Propulsion Subsystem Sheet are:

- Total Mission Delta-V ("Delta V, total")
- Delta-V per orbit for attitude control ("Delta V - ADACS, per Orbit")
- Total Wet Mass of the Spacecraft ("Mass, total wet - w/ cont")

Outputs

The key Outputs of the X-TOS Propulsion Subsystem Sheet are:

- Fuel Tank Size ("Dimension 1 – Fuel Tank, Dimension 2 – Fuel Tank")
- Mass of the Fuel Tank ("Mass - Fuel tank")
- Total Mass of Fuel ("Mass, loaded – Fuel")
- Total Mass of Pressurant ("Mass, loaded – Pressurant")
- Mass of Fuel available for Stationkeeping and ADACS ("Mass, Propellant - Stationkeeping + ADACS")
- Total Mass of All Propulsion Components ("Mass, total dry – Propulsion")
- Average Power ("Power, average - Propulsion (mode 2)" etc)
- Peak Power ("Power, peak - Propulsion (mode 2)" etc)
- Thruster Exhaust Products ("Products, exhaust - Stationkeeping Thruster" etc)
- Specific Impulse ("Specific impulse, min - Stationkeeping Thruster" etc)
- Propulsion System Type ("Type - Stationkeeping Thruster" etc)
- Mass of Fuel required per orbit for ADACS ("Mass, Propellant - Integrated ADACS, per Orbit")

Assumptions

Some key assumptions drive the ultimate size and mass of the propellant tanks. The calculations of the mass and volume of the Helium pressurant are based on the Beattie-Bridgeman equation. This is carried over from the C-TOS model. In addition, there are several 'holes' in the engine database (for operating power, mass and dimensions) which have to be estimated in order to perform some trades. These do not apply to the final engine choices.

Fidelity Assessment

The most important output of the Propulsion sheet, the mass of propellant, is based on the rocket equation. This has obviously been proven to be highly accurate over the last 60 years. In addition, the final choice of thrusters (available "off the shelf"), affords us a high degree of reliability in our Isp and mass estimates. However, there are several other aspects of the sheet that can add error to the final outputs. There are some miscellaneous inputs for additional valves and mounting provisions, most merely guesses, but they account for only a small percentage of the final propulsion system mass. The size and mass of the fuel tank itself is also subject to some estimation error, but the PV/W method is based on historical data and should be accurate to within a few percent (see references on sheet).

Verification

The only real test of the sheet's output came through a true, system-level design session. Upon outputting the Isp (so that the Mission sheet could calculate the life of the spacecraft based on the periodic depletion of fuel over each orbit), it was found that the C-TOS engine database actually contained nozzle exit velocities, not specific impulse data. This discrepancy was found by examining how reasonable the calculated lifetimes were based on the fuel mass carried on a particular design. Although this was not the most technical approach, it was another example of the absolute necessity of having experienced participants in the process.

1.1.13 Cost

Introduction

This module used two Cost-Engineering Relationship (CER) models to determine the cost of the spacecraft, launch, and associated upfront operations. The first of these was based on Space Mission Analysis and Design 3rd edition (SMAD) for typical spacecraft missions weighing between 253 and 1153 kilograms (dry). The second model was also from SMAD, and was based on Earth-orbiting small satellites, weighing between 20 and 400 kilograms (dry). Both are parametric models based on historical information.

Inputs

There were 43 inputs in the module, primarily focused on mass and power from the various subsystems. The following list is of the major drivers in the cost.

- Spacecraft Bus Dry Mass
- Structures Mass
- Thermal Mass
- Thermal Average Power
- Power Subsystem Mass
- Solar Array Area
- Battery Capacity
- BOL / EOL Power
- Telemetry, Tracking & Command and Command & Data Handling Mass
- Downlink Data Rate
- Data Storage Capability
- ADCS Dry Weight
- Pointing Knowledge
- Number of Thrusters
- Launch Cost

Outputs

Again, there were a total of 26 outputs, or approximately 13 outputs for each CER. They included the following costs:

- ADACS and Propulsion
- Telemetry, Tracking & Command and Command & Data Handling
- Integration, Assembly, and Test Wraps
- Program Level Wraps
- Ground Support Equipment Wraps
- Launch & Orbital Operations Wraps
- Power and Pyro
- Structures
- Thermal
- Spacecraft
- Wraps

…as well as total initial program costs (not including extended operations).

Assumptions

The Cost-Engineering Relationships are based on historical data, and hence have ranges for which they are valid. Those are not included in this section.

- The mission will consist of only one satellite.
- There is a single launch.
- Each subsystem accounts for multiple components--the Cost Module receives the totals.
- The payload is furnished by the customer, and hence is not considered part of the cost.
- Extended mission operations are not considered due to lack of knowledge of the staffing and resources to be used.

Description

The Cost-Engineering Relationships used in this module, and available through SMAD, are based on previous missions, where subsystem information is available, and costs are known. From this data, parametric models are created, whereby relationships between known engineering values are correlated to costs. This module uses two common cost models, to provide a comparison, and estimate rough costs.

Fidelity Assessment

The fidelity of Cost-Engineering Relationships in general is questionable, and at best can be used to estimate a rough cost. In fact, error margins on these relationships can be on the same order of magnitude as the estimations themselves. Furthermore, these are based on historical data, and a very small sample set. This means that they fail account for changes in the field, such as falling costs of supplies and technology, the use of uniform spacecraft buses, and so on. As a result, they should be considered only as rough estimations, and provide relative order of magnitudes to compare spacecraft with one another.

Verification

Several methods were used to verify the outputs of the cost module. The first, and most notable one was to ensure that all values inputted were within the acceptable data range for the CERs. Secondly, two CERs were used to correlate information. It was expected that the first model, using all typical spacecraft types would yield a larger cost estimation than the other, which was based only on small spacecraft. This held true for the various spacecraft run through the ICE process. Thirdly, two outside models were used to verify the order of magnitude on the SMAD modules. The first of these was the Aerospace Small Satellite Cost Model, which is based on recent, smaller spacecraft. The second was NASA’s Space Operations Cost Model, which was used to verify that for short lifetimes (less than one year) the cost of operations was negligible compared to the program cost. Finally, operator intuition was used to make sure that the outputted numbers seemed accurate.

1.1.14 Reliability

Introduction

This module uses Markov Modeling to determine the probability of the XTOS mission being in any given operational state (full functionality, one or two failed instruments, or system failure). The probabilities are given at design mission lifetime. The module is implemented in Excel/ ICEMaker, but a Matlab function is called to perform the calculations.

Inputs

- The reliability subsystem/module uses the following inputs:
- Mission lifetime
- Failure probability of launch vehicle
- Failure rate of the instruments
- Failure rates of the different subsystems (ADACS, C&DH, Power & Pyro, Propulsion, Structure & Mechanisms, Communication, and Thermal Control)
- Number of replicates for the different subsystems (ADACS, C&DH, Power & Pyro, Propulsion, Structure & Mechanisms, Communication, and Thermal Control)

Outputs

- The reliability subsystem/module outputs the following:
- Probability of achieving target life
- Probability of achieving lifetime with full functionality
- Probability of achieving lifetime with one failed (secondary) instrument
- Probability of achieving lifetime with two failed (secondary) instruments

Assumptions

- The mission will use a single spacecraft.
- Only one launch is permitted, therefore removing the possibility for repair or replenishment.
- Only one set of instruments is available (no redundancy).
- Instrument 1 is mission critical.
- Instruments 2 and 3 are identical as far as reliability and criticality are concerned.
- The failure rates for the different subsystems are constant over time.
- The spacecraft fails if any subsystem fails.
- The spacecraft and its subsystems can only be in a failed or functional state.
- The replicates for each subsystem are placed in parallel.
- The replicates for each subsystem have the same failure rate (but not necessarily the same design).

Description

The reliability module uses a Markov Model to determine the probability of the mission being in any given state as a function of time during the mission. For the cases considered here, the module calculates reliability information at the end of the mission life period, but it has the capability for providing the reliability at any other time.

The fault tree considered to compute the different probabilities is the following:

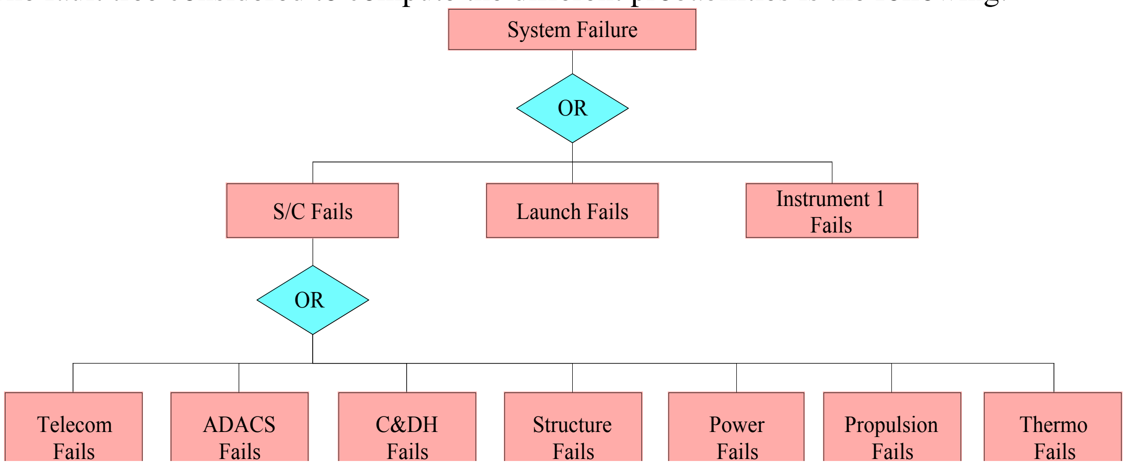

Figure Error! No text of specified style in document.**–9: Fault Tree**

Based on this fault tree, the following states are defined:

- State 1: Full functionality
- State 2: Instrument 2 or Instrument 3 fails
- State 3: Instrument 2 and Instrument 3 fail
- State 4: System failure: Launch or spacecraft or instrument 1 failure

which leads to the following state diagram:

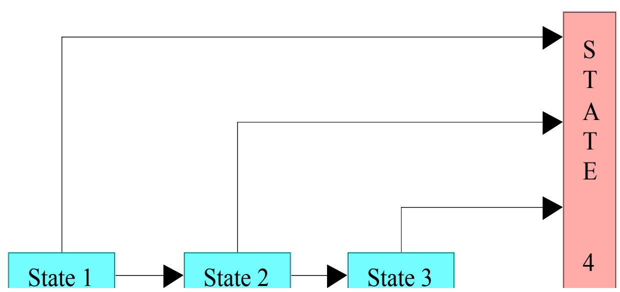

Figure Error! No text of specified style in document.**–10: State Diagram**

Using the subsystems failure rates (inverse of the Mean Time Between Failures (MTBF)) and number of replicates, the module first calculates the spacecraft reliability and MTBF. Exponential models are used for all reliabilities: $R = e^{-ft}$, where f is the failure rate and t is time.

The code then uses the state diagram, the spacecraft failure rate and the instruments' failure rates to calculate the Transition Matrix A, defined by: $\dot{P} = AP$, where P is the state vector:

$$P = \begin{bmatrix} P_1 & P_2 & P_3 & P_4 \end{bmatrix}^T, P_i = P(State\ i)$$

Note that, because launch is a single event with a probability of failure (as opposed to a MTBF), its effect is not included in the Markov model, but is added to the final state vector. The Transition Matrix and the lifetime are then used to compute the different probabilities at mission lifetime. The launch probability of failure is incorporated at the end to give the final outputs.

Fidelity Assessment

The fidelity of the reliability module suffers mostly from a lack of knowledge about the true mean time between failures of the various subsystems and instruments. Representative numbers are used, but eventually these numbers will need to be improved based on typical values used in industry.

The different subsystems should also be further refined for a more precise computation of their respective reliabilities. Because true numbers are not available, the subsystems are considered black boxes, and redundancy is applied to the subsystem as a whole. This simplification should not significantly affect the results, though, because non-redundant parts are usually highly reliable.

Verification

The reliability module was tested using various combinations of initial parameters, including a variety of mean time to failures, number of replicates for each subsystem, and mission life times. Realistic outputs were sought.